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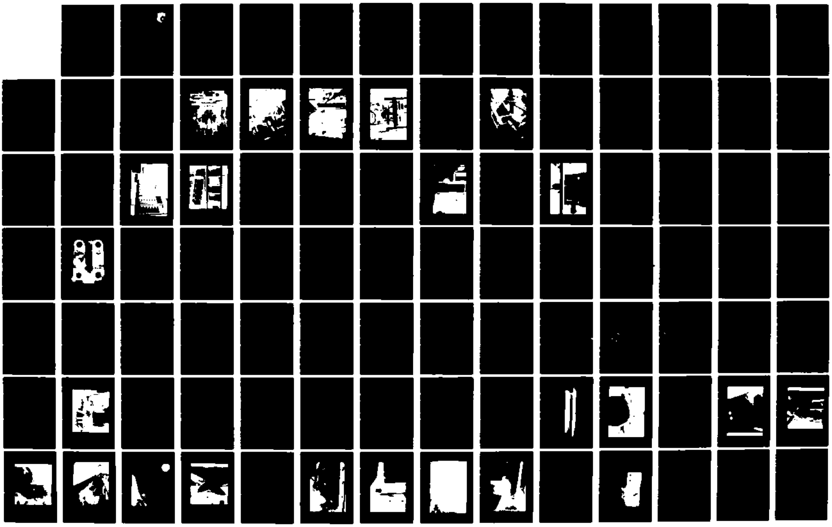

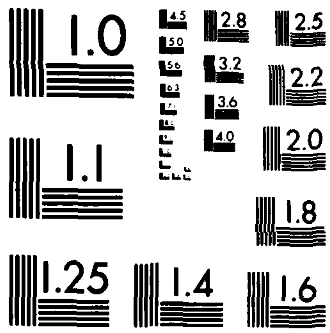

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F-4C/D LIFE EXTENSION PROGRAM

Robert L. Schneider

Structures Test Branch
Structures and Dynamics Division

October 1982

Final Report for Period October 1974 - May 1981

Approved for public release; distribution unlimited.

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

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The F-4 airframe was designed over 20 years ago for a service life of 4000 flight hours projected to be sufficient to complete the era of its usefulness in the inventory in both the USN and the USAF. However, the F-4 aircraft was slated to continue service well beyond the time when it was to have been replaced. Some 4000 F-4 aircraft are currently retained in service by the USN, USAF, and allied nations. With fatigue problems occurring in the fleet and the need for the aircraft throughout the 1980's time period, a structural integrity program

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20. Abstract (Continued)

was initiated to extend the life of the F-4 aircraft from 4000 hours to 8000 hours.

The most comprehensive airframe fatigue test program ever conducted by the Air Force, sponsored by the Aeronautical Systems Division and the Ogden Air Logistic Center, was initiated in January 1976 by the Structures Test Branch Structures and Dynamics Division, of the Flight Dynamics Laboratory. This test program was successfully completed in April 1981 after the F-4 airframe has sustained 24,000 simulated flight hours. Actual testing and inspection accounted for only 21 months of this 63 month period, most of which was spent awaiting the design, fabrication and installation of structural modifications required to extend the service life of the F-4 aircraft. A major achievement for the test program was the physical proof that the modified structure did indeed provide the required life extension.

FOREWORD

This report was prepared by the Flight Dynamics Laboratory as a formal record of the F-4C/D Aircraft Structural Integrity Program conducted under Project No. 327A5001, by the Structures Test Branch. The program was directed by Mr. Robert L. Schneider, Project Engineer and assisted by Capt. Gerald K. Boman and Mr. Timothy P. Sikora. Mr. James L. Mullineaux and Mr. John E. Pappas were responsible for the 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 program equipment respectively.

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SECTION I

INTRODUCTION

The McDonnell Douglas (McAir) F-4 aircraft has been a component of the USAF inventory since 1958. The structural service life of many of the USAF fleet was quickly approaching the service limit of 4,000 flight hours. In February 1972, coordinated efforts of McAir Aeronautical Systems Division and the Flight Dynamics Laboratory resulted in a program designed to extend the service life of the F-4 aircraft from 4,000 to 8,000 flight hours.

The program was to focus on the results of a durability test of a newly assembled F-4 aircraft. Fatigue improvements, in the form of planned structural modifications, were incorporated at different points as the test article underwent repeated load testing. Spectrum loads were applied to the test article in a flight-by-flight sequence with each simulated flight terminated by a landing cycle.

The testing was performed in the Structures Test Facility, at Wright-Patterson Air Force Base, Ohio. The fatigue test took approximately five years to complete once the test fixture and aircraft were assembled together. The majority of this time (43 months) was due to aircraft modifications and repairs. Actual set-up, cycling, inspections, down time for equipment malfunctions, and final tear-down required only 21 months (Figure 1).

1. DESCRIPTION OF THE TEST ARTICLE

The airframe had no prior test or flight history and was of the configuration of an 1974 Japanese RF-4E fuselage and an RF-4C wing, less fatigue improvements which have evolved during the service life of the F-4C/D aircraft. The (JA) RF-4E fuselage and RF-4C wing were demodified to the maximum practical extent to simulate the F-4C/D airframe. This procedure was necessary because C/D models were no longer in production.

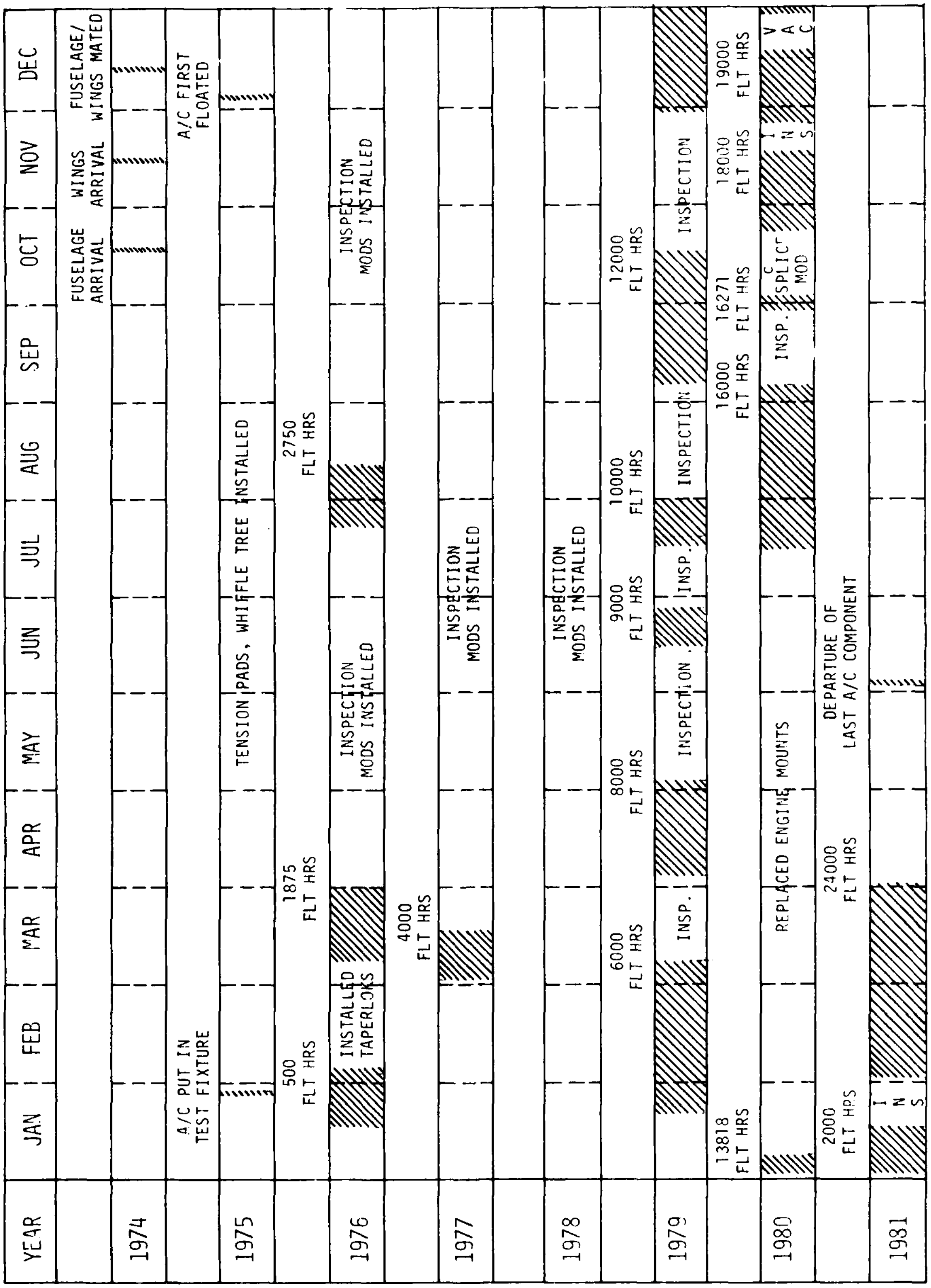


Figure 1. Test Schedule

Items that were excluded from the test article were:

a. Stabilator

b. Rudder

c. Nose landing gear

d. Main landing gear wheels and brakes

e. Canopies and all other items that did not contribute to the structural integrity of the airframe.

Laboratory loading fixtures were added to the wings and fuselage to simulate main landing gear uplock fittings and engines. All control surfaces were locked in the retracted or neutral position.

2. *SPECTRUM LOAD CONDITIONS*

All flight conditions for the Aircraft Structural Integrity Program (ASIP) test spectrum were derived from three basic flight profiles, defined as air-to-ground, air-to-air and non-tactical missions. The flight conditions were further defined by categorizing the missions into flights with variations of airspeed, altitude, and vertical acceleration within each flight. Aircraft weight is also a variable parameter, being assigned a different value for each of the three missions.

Spectrum loads were applied to the aircraft in a flight-by-flight sequence with each simulated flight terminated by a landing load cycle. The spectrum was divided into 1,000 flight hour segments with each 1,000 hours containing a total of 730 flights and 139 combinations of airspeed and altitude. The 139 airspeed/altitude combinations are distributed within each 1,000 hour segment as 46 air-to-ground missions and 45 nontactical missions. Random load cycles were scattered through ten of the 1,000 hour segments, making the spectrum repetitive in 10,000 hour increments.

The spectrum included both positive and negative vertical loads, but did not include longitudinal or lateral loadings. All wing loads were symmetrical.

Additional information can be obtained in the McDonnell Douglas Report on Load Spectrum Development (Reference 5).

SECTION II

TEST SET-UP

The F-4C/D test article was supported within a fixture having the capability of reacting all test loads and maintaining an interference-free geometry with respect to the deflected airframe and loading systems (See Figures 2 and 3). Positive or negative wing loads were balanced by controlled reactions on the fuselage and the aircraft was longitudinally and laterally restrained at the Fuselage Station (FS) 515.0 arresting hook pin (Figure 4). The aircraft was also laterally restrained at the FS 82.0 nose gear (NLG) (Figure 5). All equipment used in the test set-up was maintained in an updated, calibrated state. All testing was conducted at ambient room temperature.

1. LOADING SYSTEMS

All test and balancing loads were applied by hydraulic cylinders with mechanically integrated load cells. All loading cylinders, except those at FS's 82.0 and 515.0 were servo-controlled with servo feedback signals supplied by auxiliary load sensing circuits from the load cells. The total loading system had the capability of automatic shut-down in the event of any load being out of tolerance by a pre-determined amount. Inputs for test loads were computer programmed and were linear within the limits of the computer digital format when proceeding from one load level to the next load level. A separate load system was provided to counterbalance the dead weight of the fuselage, dummy engines, main landing gears (LG), and wing tension pads through the same set of whiffletrees used for load application.

a. Wing

Loads were applied to the upper surface of each wing of the aircraft through a series of nine separate tension pad arrays, whiffletrees, load cells, and hydraulic cyclinders. Similiarly, loads were applied to the lower surface of each wing by four separate loading bays making a total of 13 load systems per wing. The geometry of the 13 load systems are

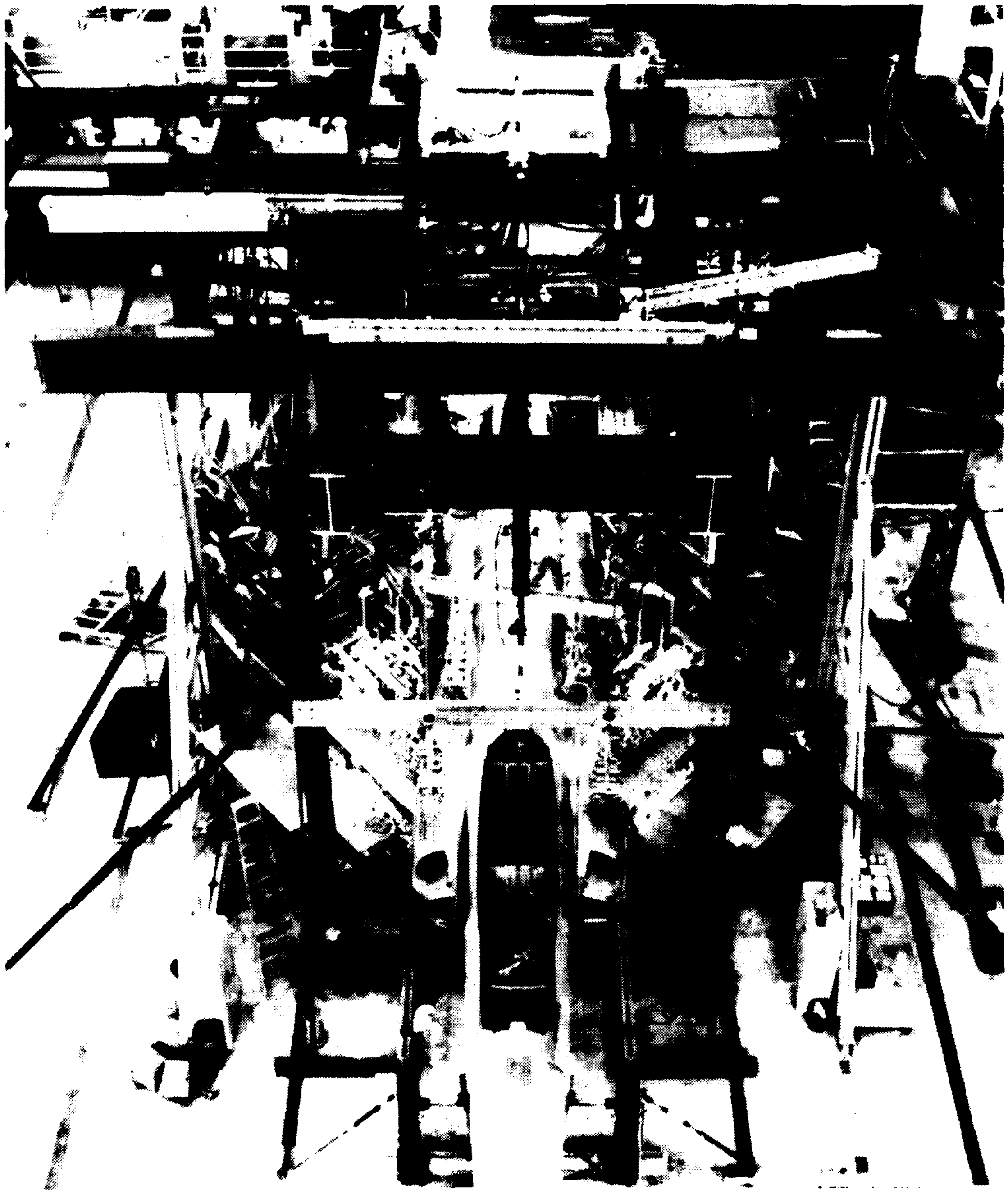


Figure 2. Test Set-up, Front View

Figure 3. Tet Set-up, Side View

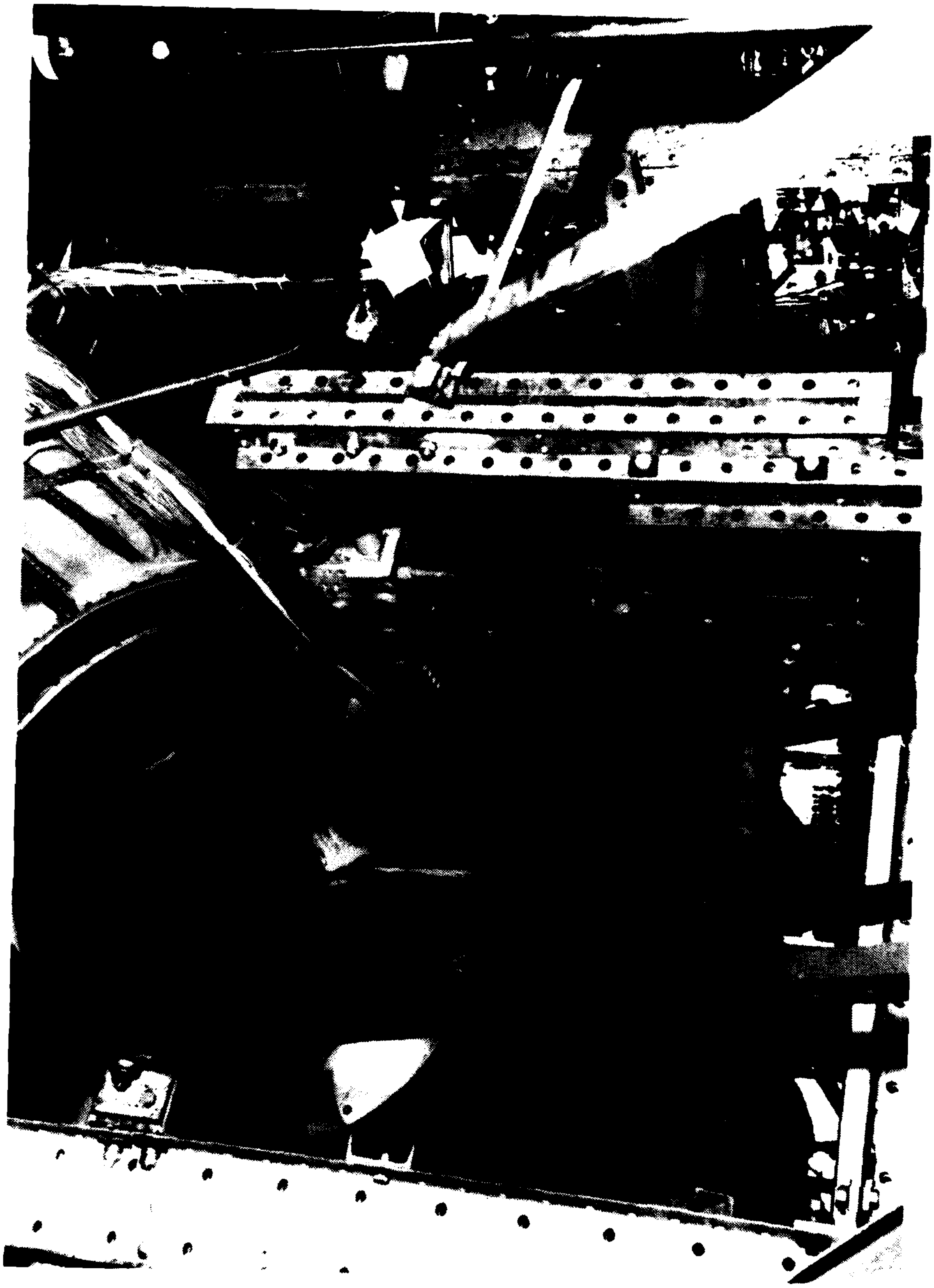


Figure 4. FS-515 Arresting Hook Restraint

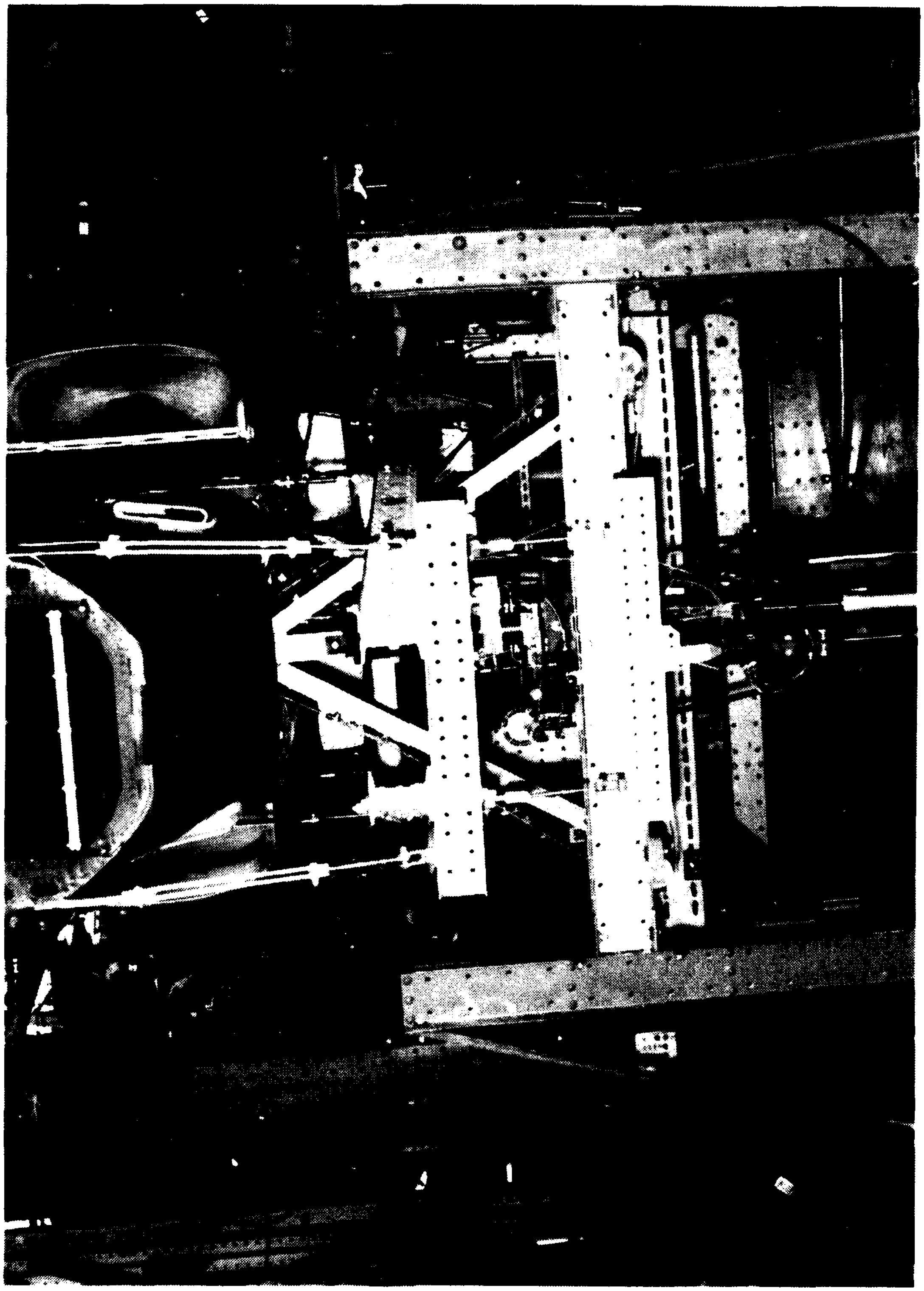


Figure 5. FS-82 NLG Restraint

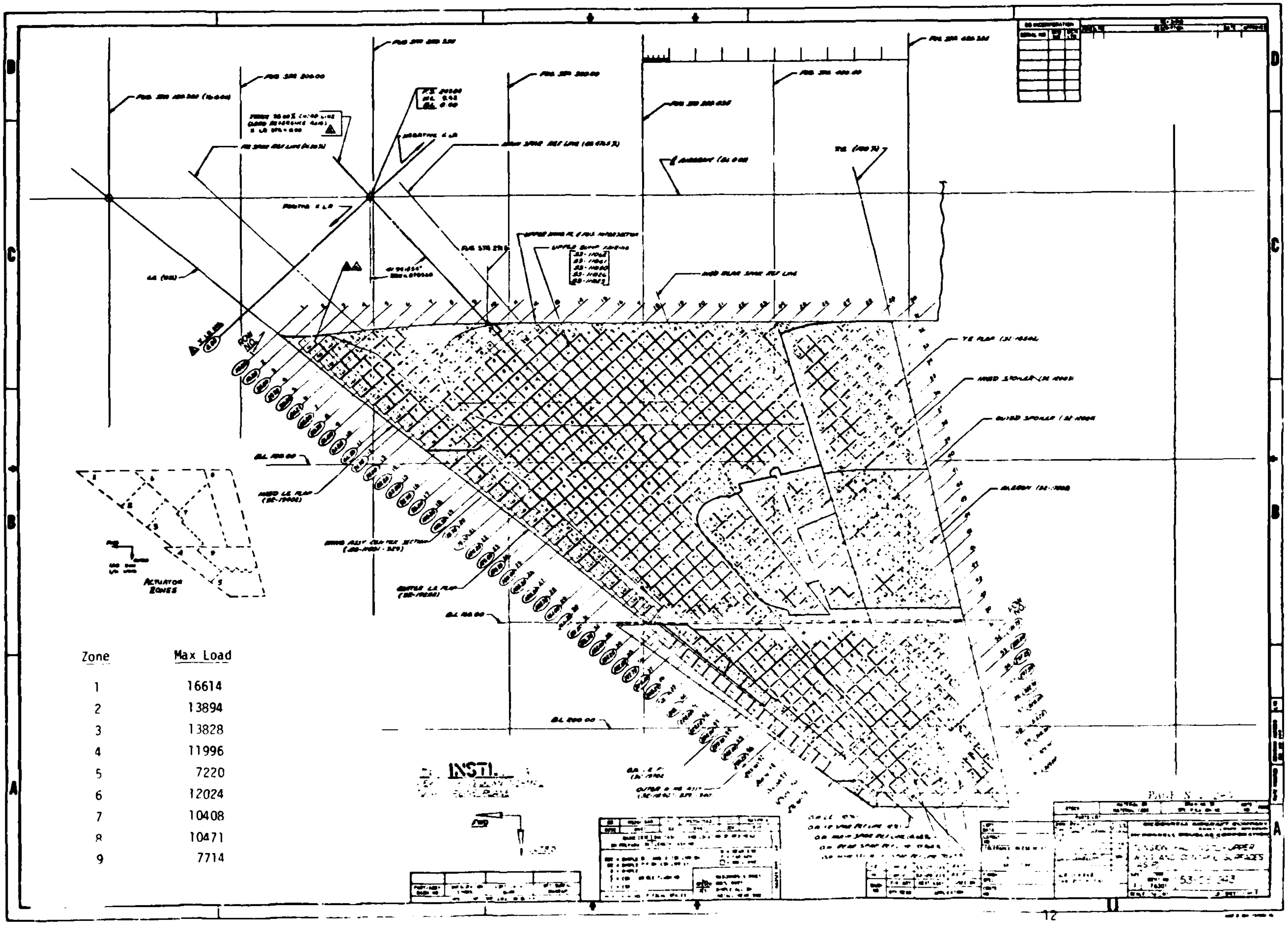
the same for the left and right wings. Loads were applied to the wings through 2 X 2 inch tension pads bonded to the wing surfaces and located per the coordinates defined by MacAir (Figure 6). The general arrangement of the tension pads, whiffletree center of pressures, and maximum actuator loads are shown in Figures 7 and 8.

The lines of action for the nine loading systems on each of the wing upper surfaces were canted inboard from the vertical. The angles at which the load systems are inclined were selected to maintain the loads near normal to a nominal deflected wing chord plane. The angles and deflected wing geometry represent an average condition based on loads which cause stresses greater than 10,000 psi in the wing lower skins. The line of action for the four loading systems on each of the wing lower surfaces was vertical because of the negligible downward wing deflections.

Inertia loads for a retracted main landing gear due to positive maneuvers were applied to the wing at the location of the left and right gear uplatch fittings. The line of action for the main landing gear inertia load was canted and derived using the same rationale as used for the direction of wing up-loads. Negative inertia loads for the main landing gear were accounted for by local tension pad load relief on the wing lower surface.

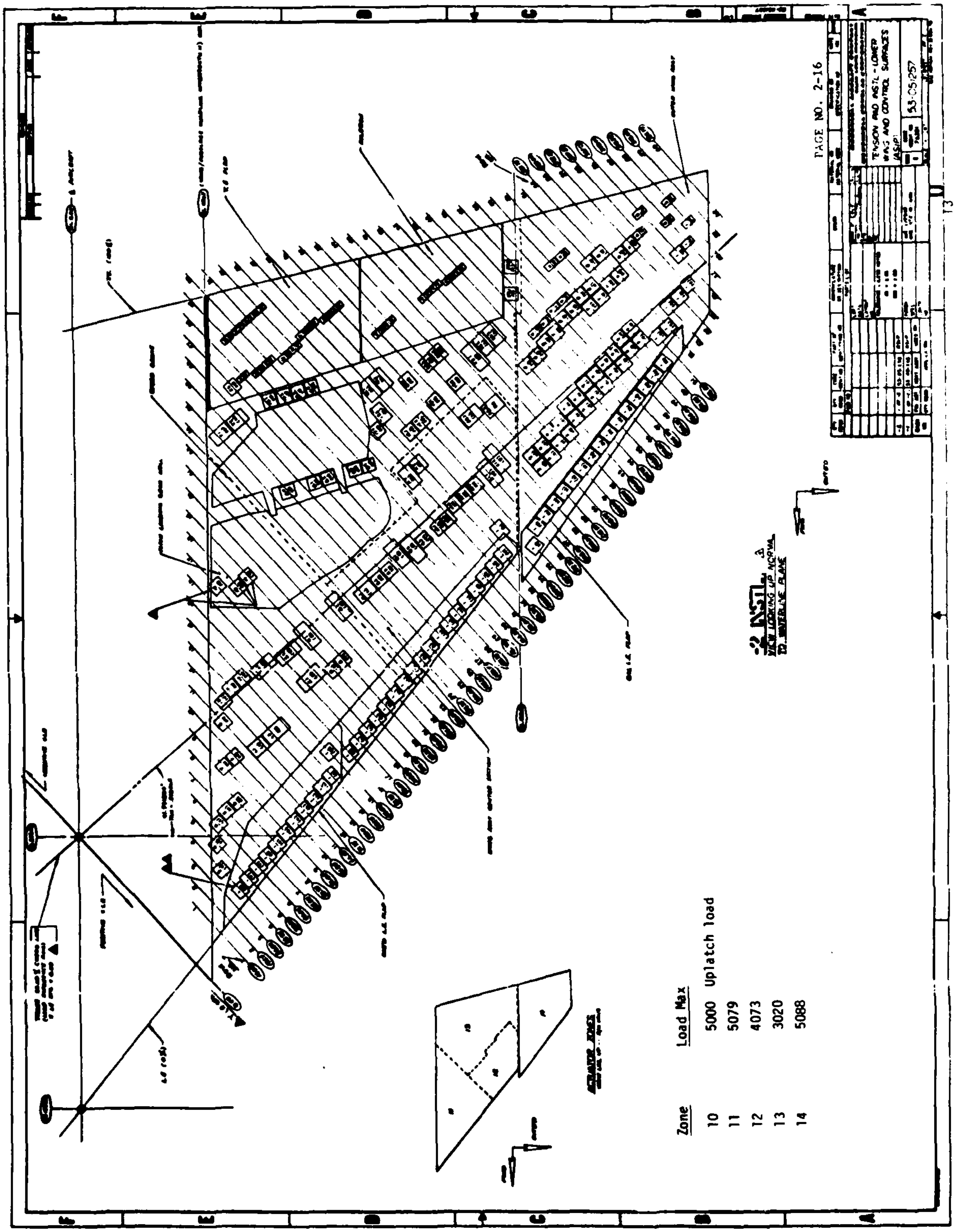
Landing loads were applied to the wing through the main landing gear and consisted of a forward acting drag, simulating gear springback, and a vertical, upward acting load. Both loads were introduced at the gear axle. The ratio of the drag-to-vertical load was equal to 0.066 and constant for all landing conditions. The drag load was induced by canting the gear load actuator in a forward direction. Loads were applied with the stroke of each main landing gear adjusted to 4.3 inches from fully extended and with the oleo struts hydraulically blocked to maintain the desired strut extension.

Figure 6. 2 X 2 Pads (Over-all View)



Zone	Max Load
1	16614
2	13894
3	13828
4	11996
5	7220
6	12024
7	10408
8	10471
9	7714

Upper Surface Loads and Distributions



Zone	Load Max	
10	5000	Uplatch load
11	5079	
12	4073	
13	3020	
14	5088	

Figure 8. Lower Surface Loads and Distributions

b. Fuselage

Wing loads were reacted through the fuselage by vertical loads (Figure 9). The hydraulic cylinders at FS 2.0 (nose gear trunnions) and 515.0 (arresting hook hinge pin) which have a tension/compression capability, were position controlled by using a mechanical servo and ratio changer. Any out of balance on the aircraft caused a change of position, which was sensed by the ratio changer which caused the application of a counter balance load that maintained the vertical position of the aircraft within the desired 3/4 inch deflection. This system reduced the possibility of undesirable loads being induced when dumping due to failure occurred.

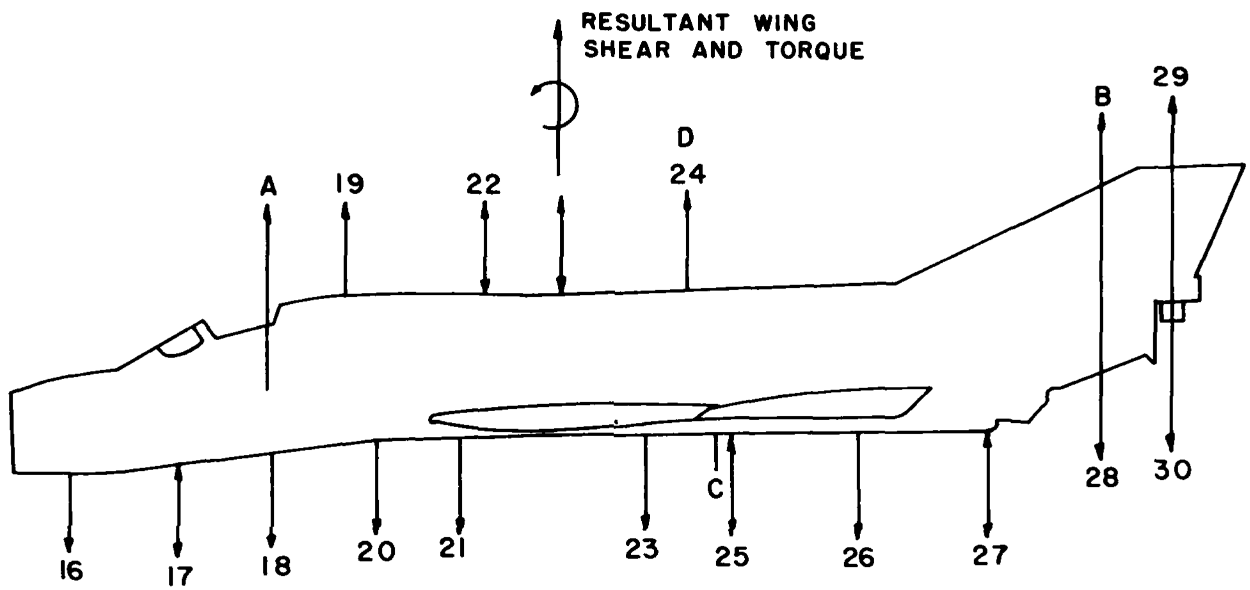
The FS-82 and FS-515 stations were also used to restrict movement in the side direction.

c. Roll Control

A roll control system was required to react or counter any roll unbalance that could occur during the F-4 fatigue testing. Two roll control systems were used. The first system utilized an inclinometer for sensing roll movement and hydraulic struts to load particular wing bays to compensate for the roll. The secondary means of countering roll was through the use of a dead weight roll control system. This system was primarily used throughout the test to maintain position when in a dumped or inactive loading condition.

The inclionometer system was a solid state, dc, closed loop, force-balance accelerometer ideally suited to the precise requirements of control systems. The inclinometer had a $\pm$30MV output for a range of $\pm 2^{\circ}$ inclination. The inclinometer was mounted in a horizontal plane at FS 350 with the sensitive axis aligned to measure roll inclination about the fuselage horizontal axis.

The output of the inclinometer was utilized as a preload program to a load controller operating in the normal program mode. Any roll of the fuselage generated a proportional output of the inclinometer. A



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CONTROL CHANNEL	FUSELAGE STATION	LOAD	CONTROL CHANNEL	FUSELAGE STATION	LOAD	CONTROL CHANNEL	FUSELAGE STATION	LOAD
16	25	8044	23	333	19860	30	613.4	39872
17	82	15058 - 1307	24	358.5	3184	A	132.8	3093
18	133	6300	25	378.5	32000 - 9000	B	573.6	3010
19	174.5	9730	26	446.5	13564	C	372.5	3238
20	189	14882	27	515	4330 - 9143	D	358.5	2063
21	230.3	20247	28	573.6	6879			
22	250.6	6450	29	613.4	3179			

Figure 9. Fuselage Loads

corrective output by the Controller then generated a force couple opposing the fuselage movement. The force couple was achieved by decreasing or increasing the load that was being applied to wing bays 3, 4 and 12. The design was insensitive to vertical movement and allowed free motion in that direction.

The dead weight roll system was composed of cables attached to loading bays of each wing. The cables were looped around pulleys and fastened to a roll control dead weight as shown in Figure 10. The dead weight of the bays and the peripheral linkage weight were considered in determining the roll control dead weight value.

In application, a tendency of the F-4 to roll would cause the downward wing to hoist the dead weight and, in turn, allow a release of the load on the opposite upward moving wing, thus causing a force couple.

A hydraulic cylinder was attached between the cables and the dead weight to facilitate ease of handling the dead weight. During downtimes the roll control dead weight was lowered to the floor by releasing the fluid pressure in the cylinder.

d. Counterbalance

The fuselage structure and load fixture weight were counterbalanced by hydraulic cylinders located at Fuselage Stations 123.8, 358.5, and 573.6. The two dummy engines were counterbalanced by a hydraulic cylinder at FS 372.5 and Buttock Line (BL) 0.0. The wing structure and both upper and lower surface wing pads and whiffletrees were counterbalanced by hydraulic cylinders attached to the wing upper whiffletrees. The counterbalance cylinders were located adjacent to the test load cylinders at each upper whiffletree. The main landing gear was counterbalanced independently of the gear load cylinders.

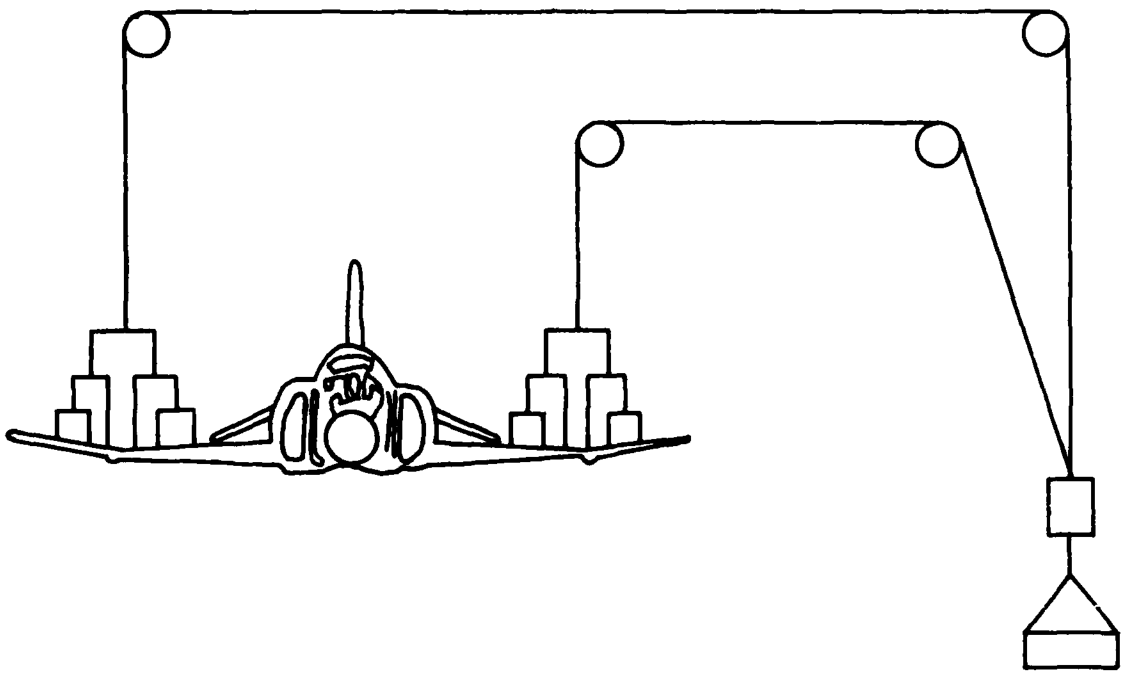


Figure 10. Dead Weight Roll System

2. SAFETY SYSTEMS

a. Load Controllers

Each Controller had an error detector circuit (load cell feedback vs program) subsystem (Figure 11). Each circuit was independently adjustable from 0 to 100% full scale in both positive and negative direction. An overload limit detector initiated action when a test load was above pre-selected values (normally maximum load plus tolerance) in both tension and compression directions. The output of this error and overload detector circuit permitted actuation of the hydraulic dump circuit on the master control panel. Loss of controller power also actuated the dump circuit.

b. Back-up Overload

The load magnitude and load error safeties incorporated in the commercially available load controllers covered most modes of possible equipment failure, but not all. Loss of load cell feedback signal, caused by:

(1) Power supply failures

(2) Open circuits

(3) Feedback amplifier failures was not protected by available load controllers.

A back-up overload subsystem (Figure 12) was integrated into the overall load system and was employed in parallel with the basic load controller (Figure 13).

Five units (50 channels) were utilized as a load error device (load cells vs program).

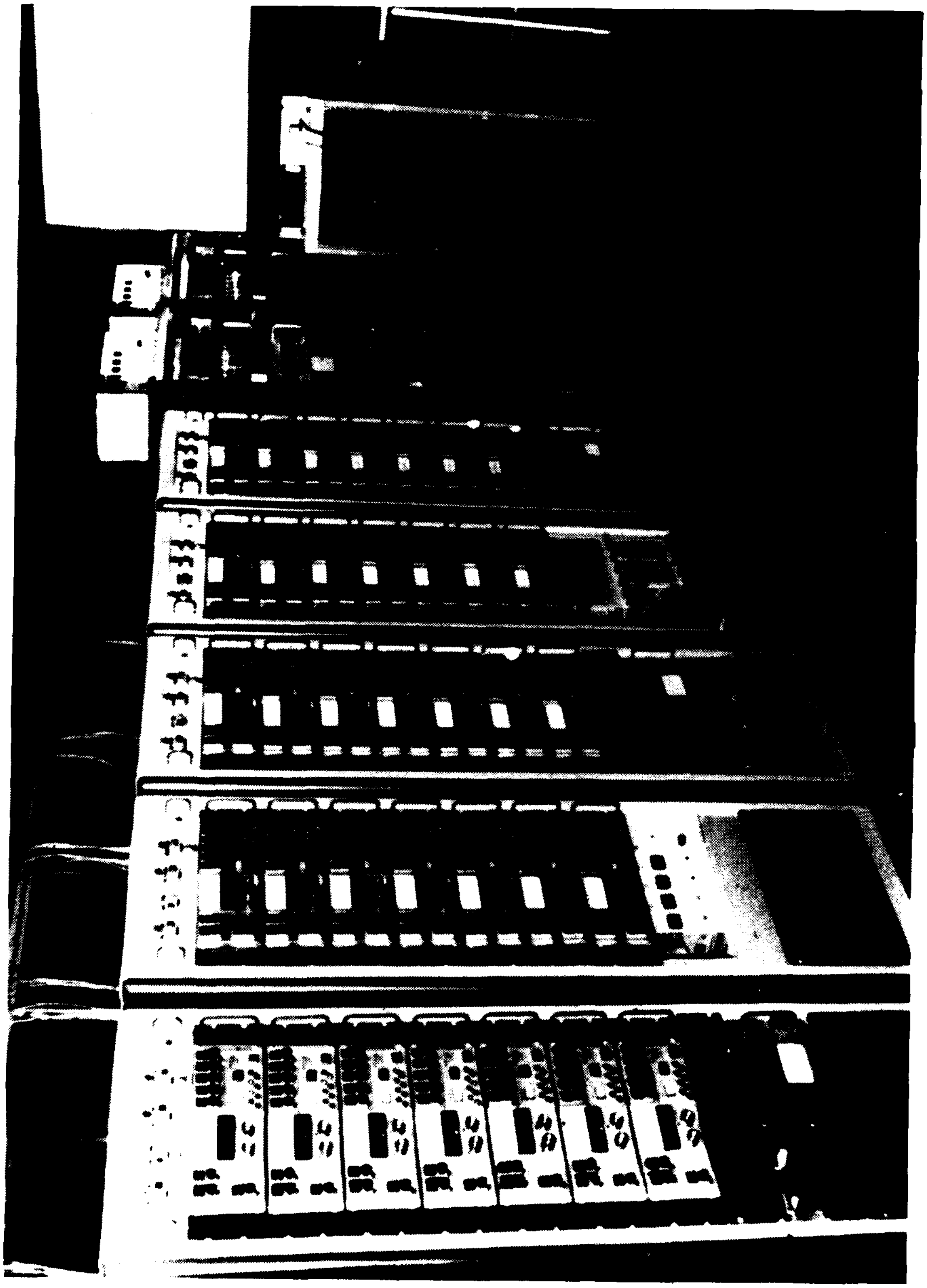


Figure 11. Load Controllers

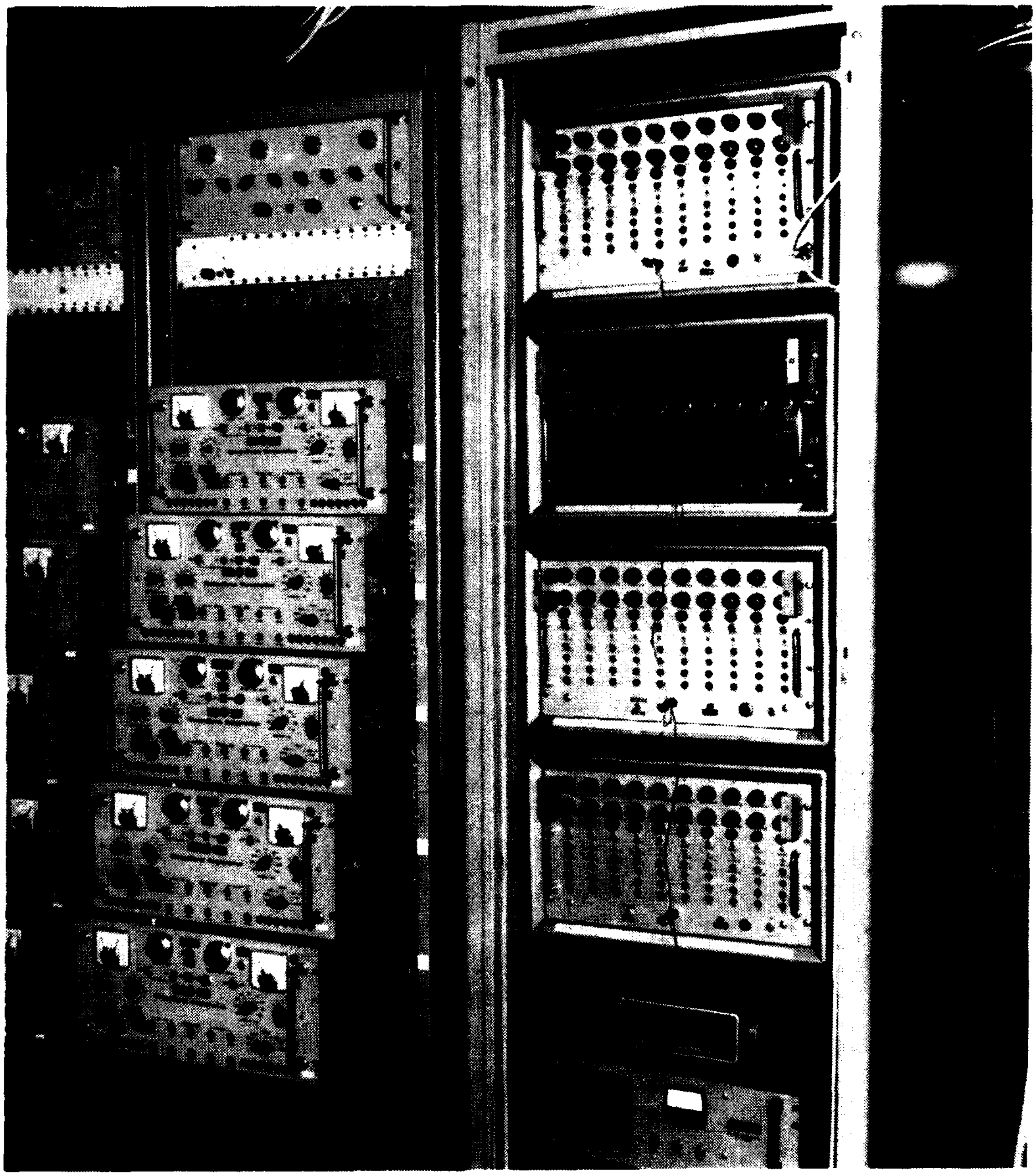


Figure 12. Back-up Overload Subsystem

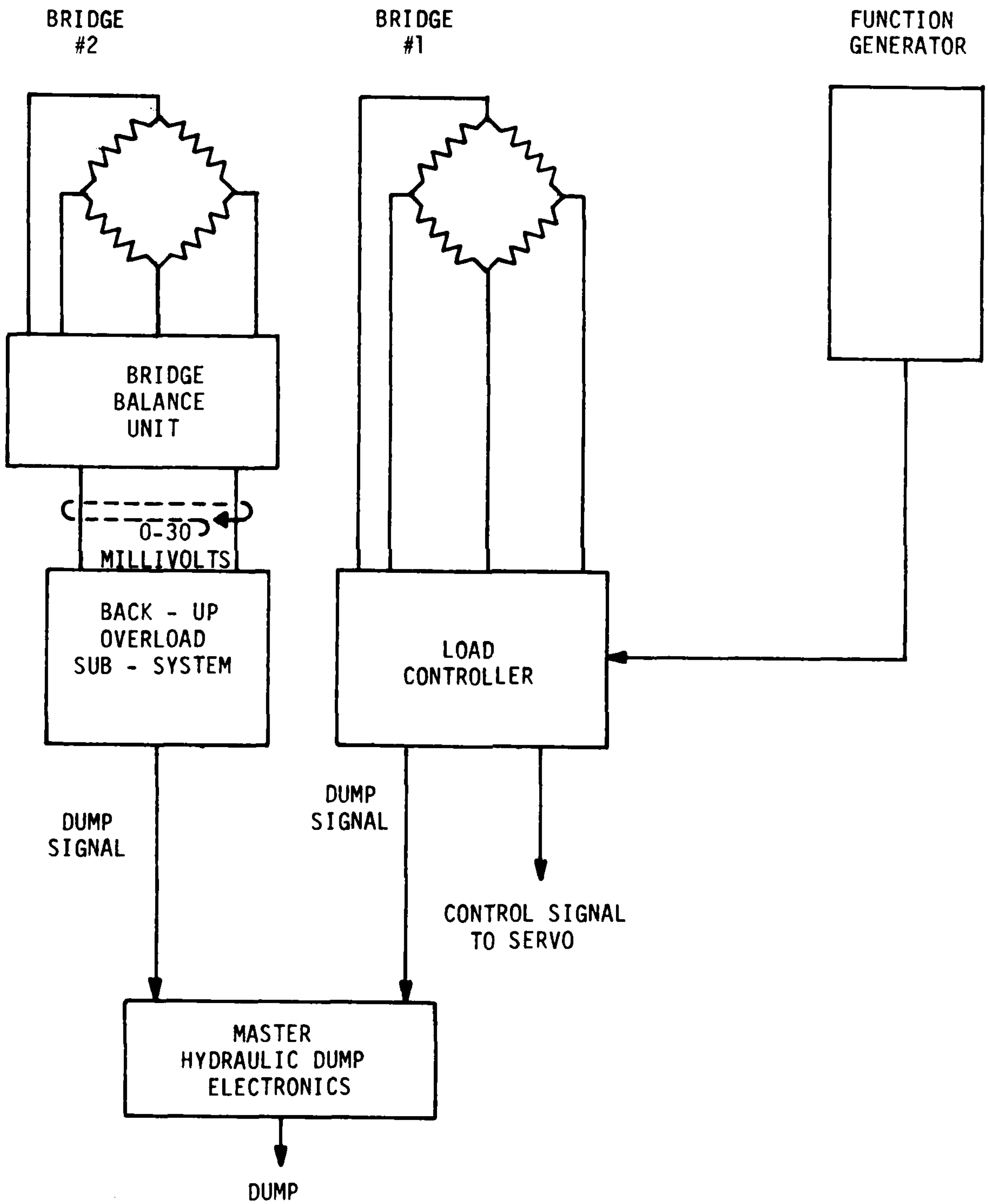


Figure 13. Wiring Diagram, Back-up Overload Subsystem

All units consisted of 10 channels each of the same basic protection electronics. The subsystem consisted of a low level amplification input stage, a dual detector network, level setting components, indicating components (one for positive signals and one for negative signals), and two fine resolution potentiometers on the front panel for individual adjustment of the positive and negative signal trigger levels. Low level sensor signals from a totally independent load cell bridge were transmitted to the subsystem. These signals, after amplification, were compared internally to the program. If any of the load cell to program errors exceeded the preset positive or negative allowable percentage levels, the output dump relay for that channel was activated and held in position till reset. This activated the dump circuit on the master control panel. Indicator lamps on the front panel indicated which channel or channels had exceeded the present limits.

c. Program Check

In addition to the load programming computer the desired program, containing maximum and minimum of each individual cycle in the proper sequence, was stored in an independent computer.

As the program computer generates the program function, the Check Mini Computer compared the desired cycle number and maximum and minimum program levels to that being generated.

A hydraulic dump resulted, when any one of the 30 control voltages exceeded the maximum programmed level for that channel.

d. Load System and Aircraft Repeatability Checks

During the strain surveys, data were collected at various load levels on all strain gages. Data collected on the designated critical sensors were stored in the data (host) computer.

These data were identified by mission number, "g" level, and airspeed. This allowed the computer to compare the previously stored data (strain survey) to the identical mission that was being applied during a typical flight. These data were not necessarily the maximum load, but loads that were applied in a greater number of the flights. With this type of check any load system or aircraft deviation from normal would be spotted for investigation. A complete explanation of this system is in the Instrumentation Section. (Section III-7).

e. Break Wires

Instrumentation wires of long durability were bonded around the entire periphery of the tangential links connecting the dummy engines to the aft engine mount. The wires were connected to the load control circuit so that a break in any wire would immediately cause a hydraulic dump. This break wire system was installed to minimize damage to the fuselage in case of failure of the links.

3. CONTROL SYSTEM

a. Load Programming

Load programming of the flight-by-flight spectrum was accomplished on a 48,000 word 16 bit PDP - 11/40 Mini Computer (Figure 14).

This computer generated 30 simultaneous real time load programs for the analog load (servo) controllers.

The flight-by-flight test spectrum for the F-4C/D fatigue test represented the most complex test spectrum run to date. The programming involved in conducting this test was too complex and of such great magnitude that an additional report was written pertaining to this effort. Therefore, this report will only refer to programming as published in Reference 6.

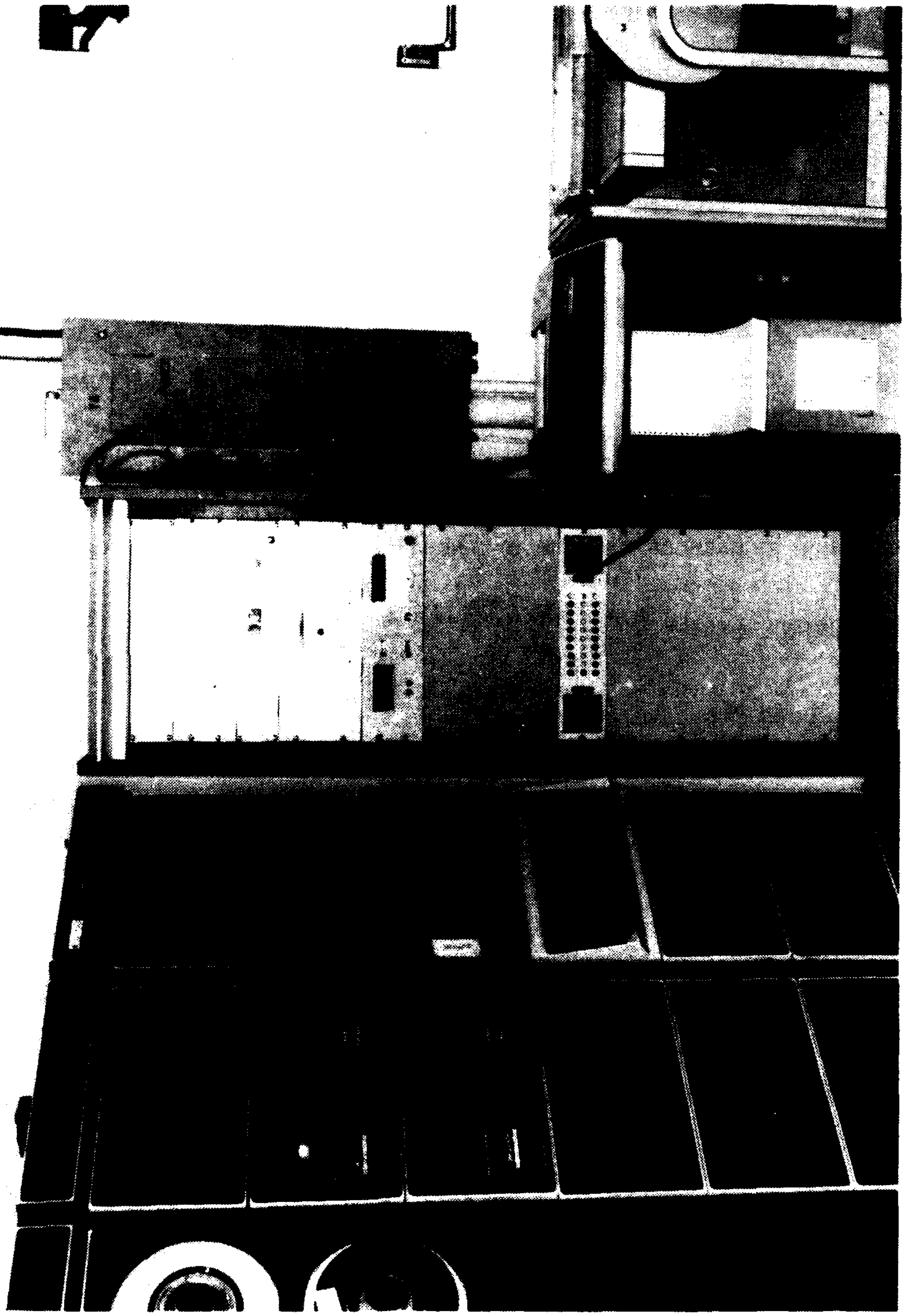


Figure 14. Program Computer

b. Analog Servo Load Controller

The BAFCO Model No. 840 analog servo load controllers (Figure 11) were used as part of the electrohydraulic closed loop loading system. The BAFCO is a completely transistorized electronic control module specifically designed to control test loads via a hydraulic servo-valve, actuator, and load cell combination according to a specified external program.

The controller accepted the external signal programming (0 to ± 6 VDC) generated by a PDP 11/40 Mini Computer via a multiplier circuit and external load cell feedback of 0 to ± 30 mv. d.c. with 10V excitation voltage. The 0 to ± 6 volt program full scale signal corresponds to the 0 to ± 30 mv full scale feedback signal. Minus (-) program signals generated tension loads in the actuator and plus (+) signals generated compression loads in the actuator. Output to the servo-valves was adjustable to ± 40 ma at 10 VDC.

Each controller included: (a) An error detector circuit which detected errors (load cell feedback vs. program) in both the negative and positive direction. The circuits were independently adjustable from 0 to 100% full scale in both directions (7.5% was used during this program). (b) An overload limit detector which initiated action when any test load was above a preselected value (normally maximum loads plus 5% tolerance) in both tension and compression directions. The output of the error and overload detector circuits permitted actuation of the dump circuit on the master control panel. Loss of controller power also actuated the dump circuit.

Each controller had the following features and controls:

(1) A manual control for applying calibrated loading signals from 0 to 100% in 0.1 percent increments in either tension or compression modes.

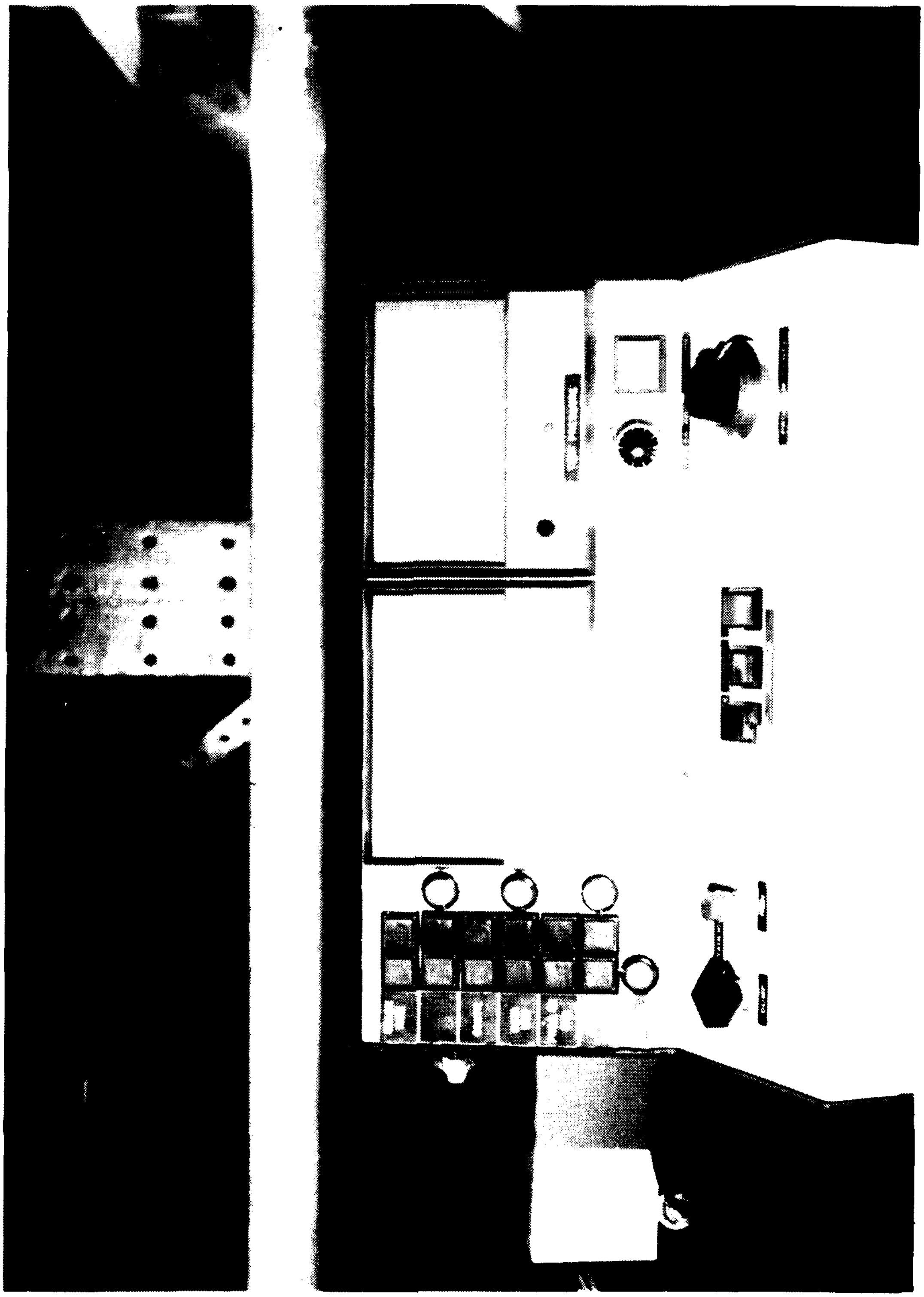


Figure 15. Control Console

(2) A load range control for adjusting the program level to a desired percent of full scale load from 0 to 100% in 0.1 percent increments.

(3) A gain control for adjusting the proportional band (system stability) from 2 percent to 40 percent.

(4) A preload control to furnish a calibrated signal of up to 10 percent of full load in either tension or compression mode by proper switch selection.

(5) A panel meter to monitor various controller parameters including:

(a) Load cell feedback
(b) External program input signal
(c) Manual program input signal
(d) Percent error
(e) Preload input program
(f) Voltage and current to servo-error amplifier.

(6) Each controller had the capability for selecting any one of four external programs.

(7) Each error and overload circuit utilized ten turn potentiometers with digital type locking dials and had actuated indication lights and reset controls.

(8) Each controller contained an adjustable gain plus integral action network which eliminated the proportional offset in one-mode controllers necessary to maintain a constant rate of load change in normal electrohydraulic load control systems.

(9) The linearity deviation of each controller was less than $\pm$ 0.25 percent full scale and overall drift was less than $\pm$ 2 microvolts/degree C.

4. HYDRAULIC SYSTEM

The hydraulic system consisted of two pump assemblies rated at 87 GPM and 5000 PSI.

Controls for this system consisted of a run, dump button, a high-low pressure switch, and a panel relief valve by which the low pressure could be varied between 150 PSI and 3000 PSI.

All components of the control system were wired so that in the event of an electrical power failure all valves go into a dump position.

Emergency dumping of hydraulic supply was accomplished by dropping the secondary breaker supplying electrical power to the pumps. Buttons to operate the emergency shut-down were located at the pump control panel (outside the pump house door) and the test control console.

Counterbalance pressure was controlled by pressure reducing and pressure relief valves installed in pairs.

Two auxiliary hydraulic pump units were connected to the counter-balance system to maintain pressure automatically, if the main system pressure was lost. This system was a low volume system and was only to sustain pressure for the time required to shut down the test safely.

5. TEST MONITOR AND CONTROL

The testing was monitored by the Project Engineer through the use of the IMLAC System, Sanborn Strip Chart Recorders and exceedance printout. These systems are explained in detail in the Instrumentation Section.

Control was accomplished through the Control Console (Figure 15). The console incorporated the following:

a. Aircraft positioning energize and dump
b. Crack wire continuity light

c. Check Mini-ready and dump light

d. Controller-ready and dump light

e. Hydraulic pressure monitor (both high and low)

f. Run, stop, hold, dump, hold peak, and end of cycle switches

g. Manual proqram potentiometer - varied program from "0" to 100% on all program channels.

SECTION III

INSTRUMENTATION AND DATA HANDLING

1. GENERAL

The data acquisition portion of the test program was designed not only to provide the classic data necessary to determine stress distributions,load paths and impending structural failures but also to verify and monitor the application of load during the complete fatigue test program.

Successful implementation of the design was required to prevent undesirable structural failures and to provide the measured parameters necessary for comparison with theoretical predictions and for verification that the life of the aircraft had been extended to 8,000 hours by the implementation of the Engineering Change Proposals (ECP), and Time Compliance Technical Orders (TCTO).

2. TRANSDUCERS

a. Load Cells

Forty-nine load cells for monitoring, controlling, and recording the primary hydraulic loading struts were purchased from BLH Electronics in ranges from 2,500 lbs to 50,000 lbs. These universal type load cells contained three strain gage bridges, each with 350 ohm impedance, 3 MV/V full scale sensitivity, and ±.01% full scale linearity. The load cells were calibrated by the manufacturer against a National Bureau of Standards (NBS) certified secondary standard load cell. The load cells were also checked in-house against a secondary load cell traceable to NBS. The calibration data were recorded and the best fit slope (sensitivity) in pounds per microvolt output was obtained with a linear regression analysis computer program.

In addition, two strain-gate-based force sensitive transducers (Figure 16) were manufactured, calibrated, and installed by McDonnell Douglas. These load links were located on the left front engine mount. Each load link had a single bridge with 350 ohm impedance, nominal 5,000 pound capacity and 3200 micro-inch per inch full scale output. A 75,000 ohm shunt calibration yielded a nominal 4,072 pound output. These load links were not part of the load control system. They were used for data monitoring and recording only.

b. Strain Gages

198 axial and 95 rosette (3 element) strain gages were initially installed on the test aircraft. The strain gages were mainly constantan foil in an encapsulating polymide carrier bonded to the structure with methyl-2 cyano-acrylate adhesive (M-Bond 200). The gages had an electrical resistance of 350 ohms and were connected in Wheatstone bridge configurations with a single active element. Since each rosette installation had three elements, there were a total of 483 active elements; each requiring one data channel for recording (Reference 4).

The strain gage installations were water proofed with a two part polyurethane and modified epoxy protective coating developed by McAir. A five-conductor shielded cable was used for all strain gage wiring between the aircraft and an interface panel used for the bridge completion network. Strain gage cable tiedowns were bonded to the structure at convenient locations which did not interfere with inspections.

c. Deflections

Eighteen electrical deflection transducers were used to determine total deflections of the test airplane. Deflection transducer locations are shown in Figure 17.

The commercial transducers were basically 1000 ohm, wire-wound, multiple-turn potentiometers. The transducers used in this test program

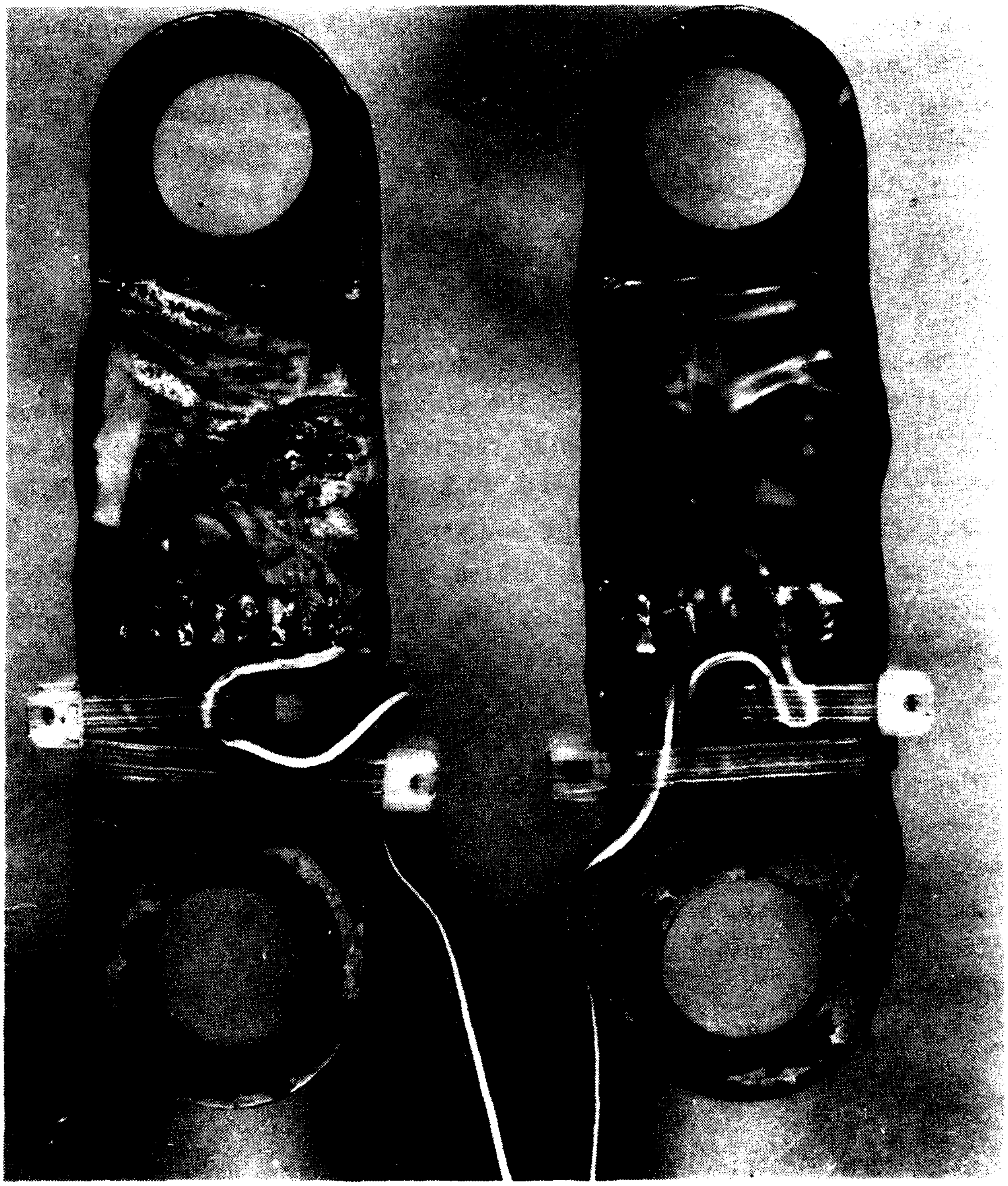


Figure 16. Engine Load Links

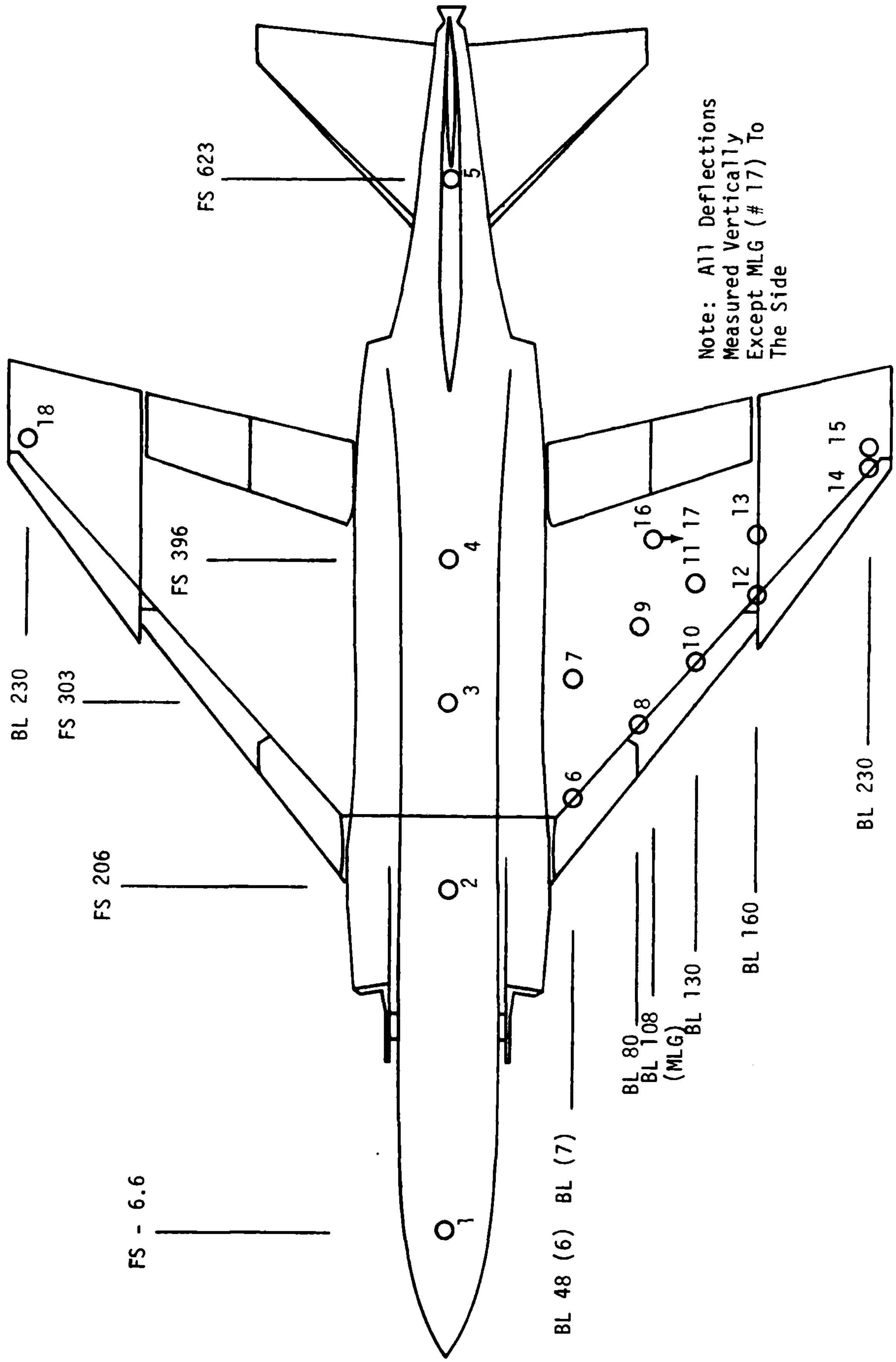


Figure 17. Deflection Transducer Locations

ranged from 2 inches to 50 inches full scale. The linearity of the transducers was 0.4% of full scale and the resoulution was 0.08%. Constant force springs within the transducer maintained a uniform static tension on the attachment cable.

d. Fatigue

Fatigue transducers (fatigue sensor, strain gage and mechanical multiplier device) were installed at three critical structural areas of both the right and left wings and were monitored periodically for permanent response (R_p) to the test spectrum during the F-4C/D fatigue test program. Response of transducers at each location gave a reference calibration of the airplane response to the baseline test spectra derived from the average of various past usage fleet experiences.

This baseline calibration could subsequently be used on similarly instrumented fleet aircraft as a measure of the relative usage severity for individual aircraft assessment of the Aircraft Structural Integrity Program (ASIP) fleet tracking requirements. These data may assist in the determination of damage indices for each monitoring area as related to the index control point. The calibration response data also have application as a reference "gauge" of the fleet use economic and safety limits as may be determined by the fatigue test results.

The transducers, in general, performed with repeatable and predictable responses as shown in Figures 18, 19 and 20 for Butt Line 110" main spar location, one of three areas selected in this effort. The responses shown are typical of all locations investigated.

Additional fatigue sensors incorporating the crack growth techniques was also evaluated (References 2 and 3).

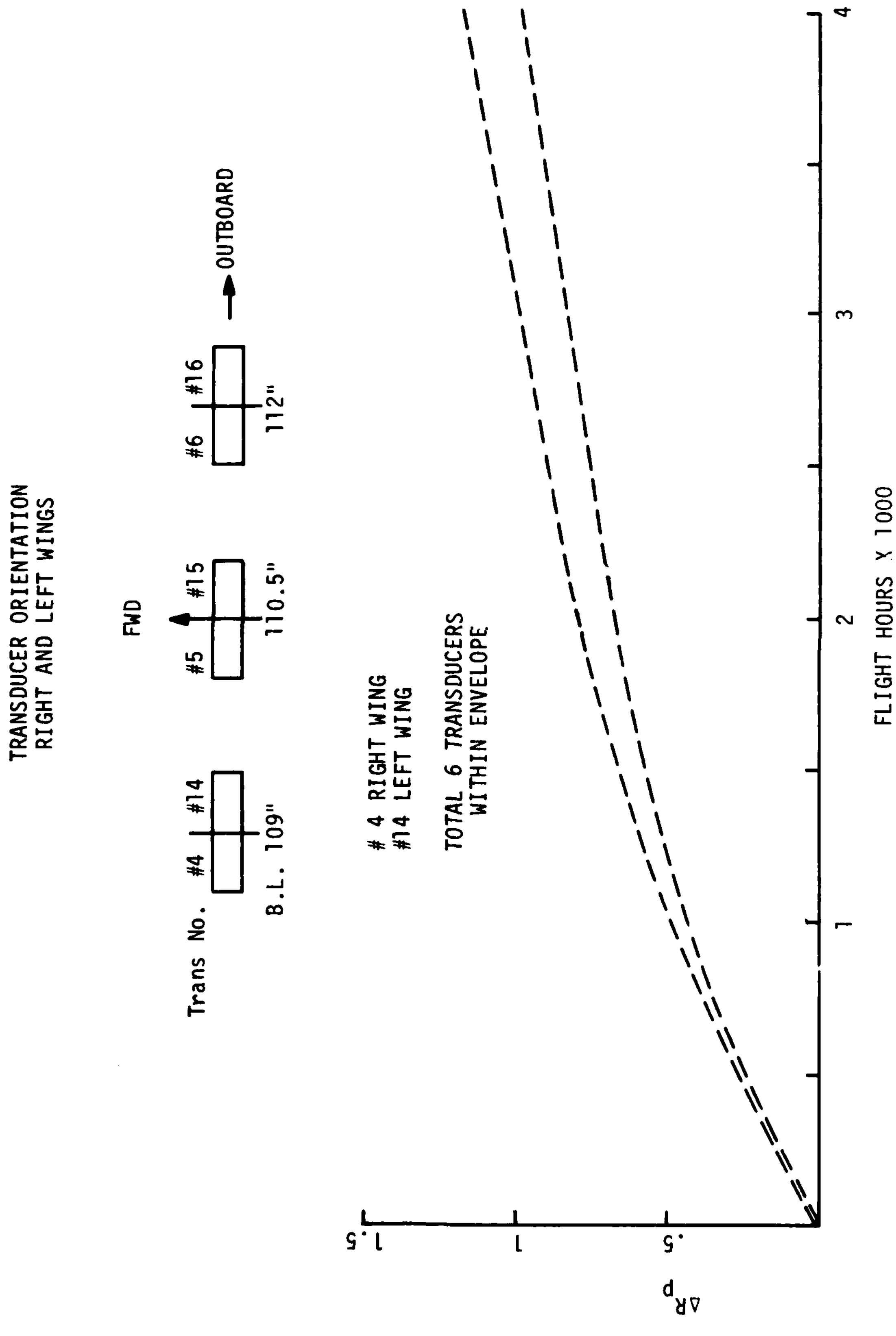


Figure 18. Fatigue Gage Orientation and Gage Response, Numbers 4R & 14L

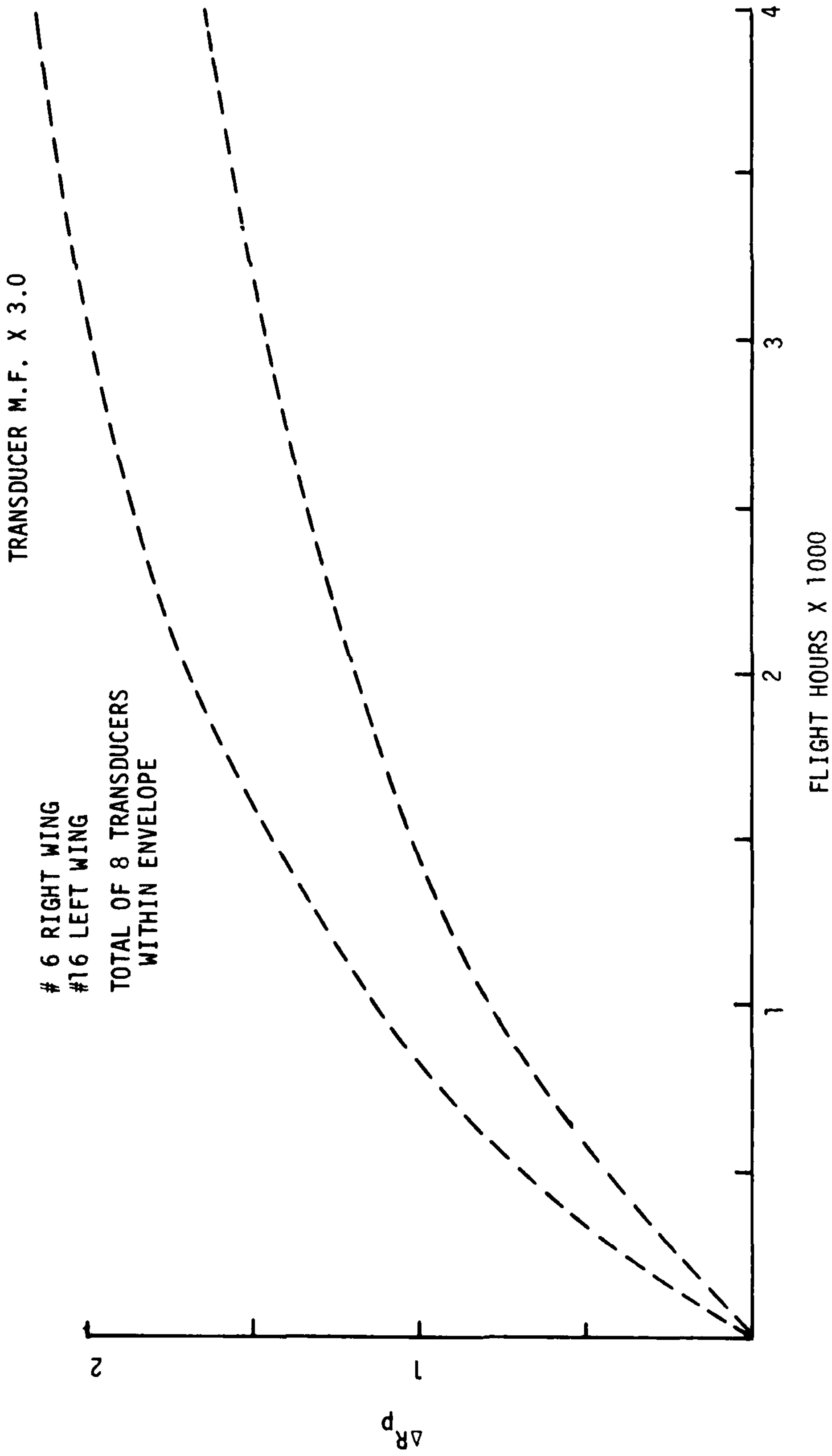


Figure 19. Fatigue Gage Response, Numbers 6R & 16L

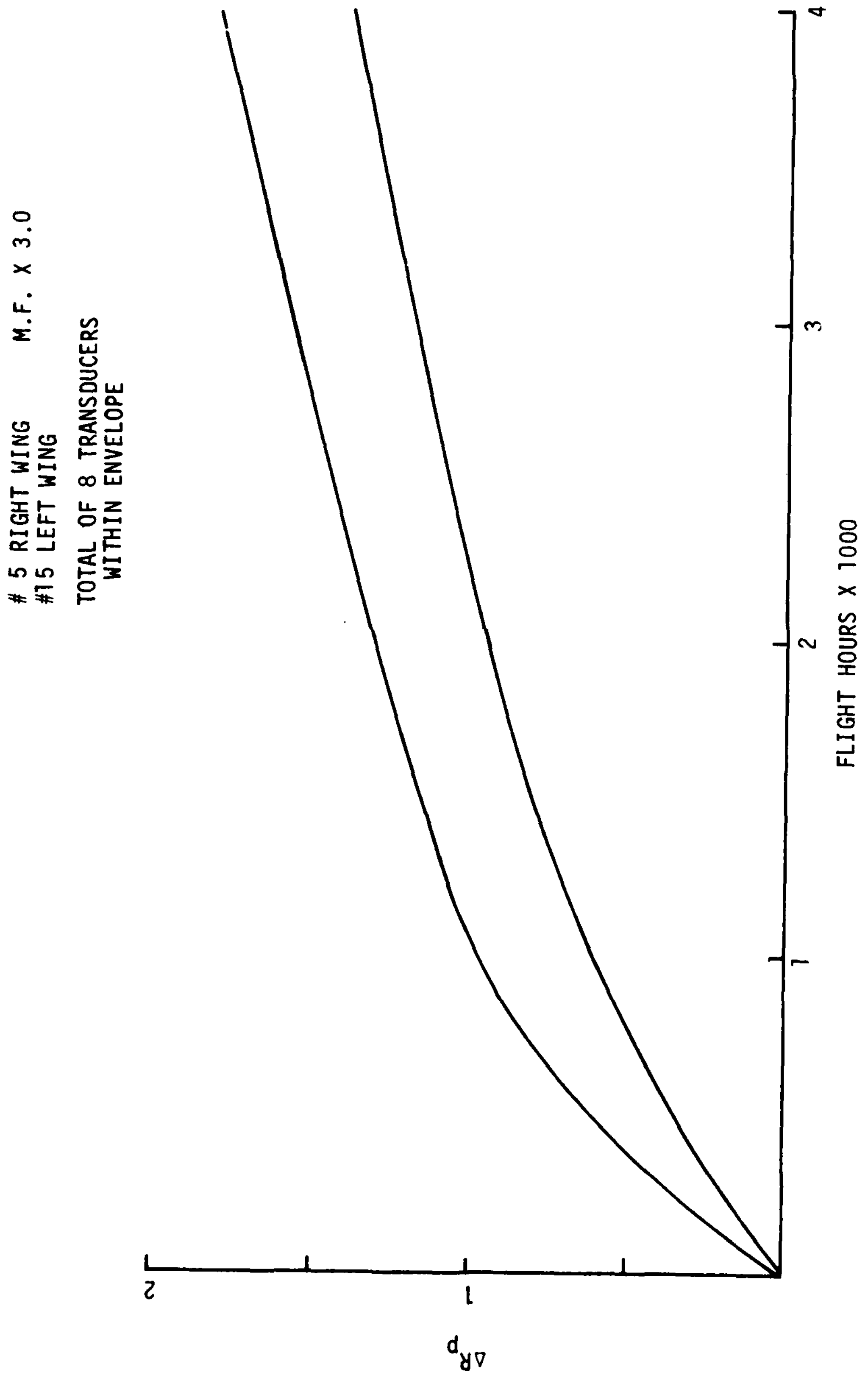


Figure 20. Fatigue Gage Response, Numbers 5R & 15L

3. DATA ACQUISITION SYSTEM

The digital acquisition system provided a media for enabling three interrelated procedures:

a. The logging of test data from all channels for subsequent off-line processing, analysis, and reporting.

b. The on-line, continuously up-dated display of selected channels on a demand basis.

c. Automatic computer monitoring of 100 channels for magnitude exceedence (Reference 1).

The data system consisted of nine signal conditioning units (one for load cells, one for deflection transducers, and seven for strain gages), four Real Time Peripheral (RTP) units, one Digital Equipment Company (DEC) PDP-11 minicomputer, and a Systems Engineering Laboratory (SEL) Systems 86 digital computer for raw data collection, magnetic tape storage, and on-line data processing for test information graphic displays. Figure 21 illustrates the data system in a block diagram. The system modular size was based on the maximum number of RTP units that could be controlled by the PDP-11 minicomputer. Signal conditioning was provided for 49 load cells, two load links, 18 deflection transducers and, 407 strain gage elements.

Strain data acquisition provided a convenient example to explain the data system component utilization and performance. The strain gages terminated in cables approximately 60 feet from the structure. These cables were routed to strain gage interface panels which were the initial connection to the data acquisition system.

The strain gage interface panels provided terminals for connection of the strain gage cables and the bridge completion resistor. The 407 strain gage elements which were monitored and recorded for each strain

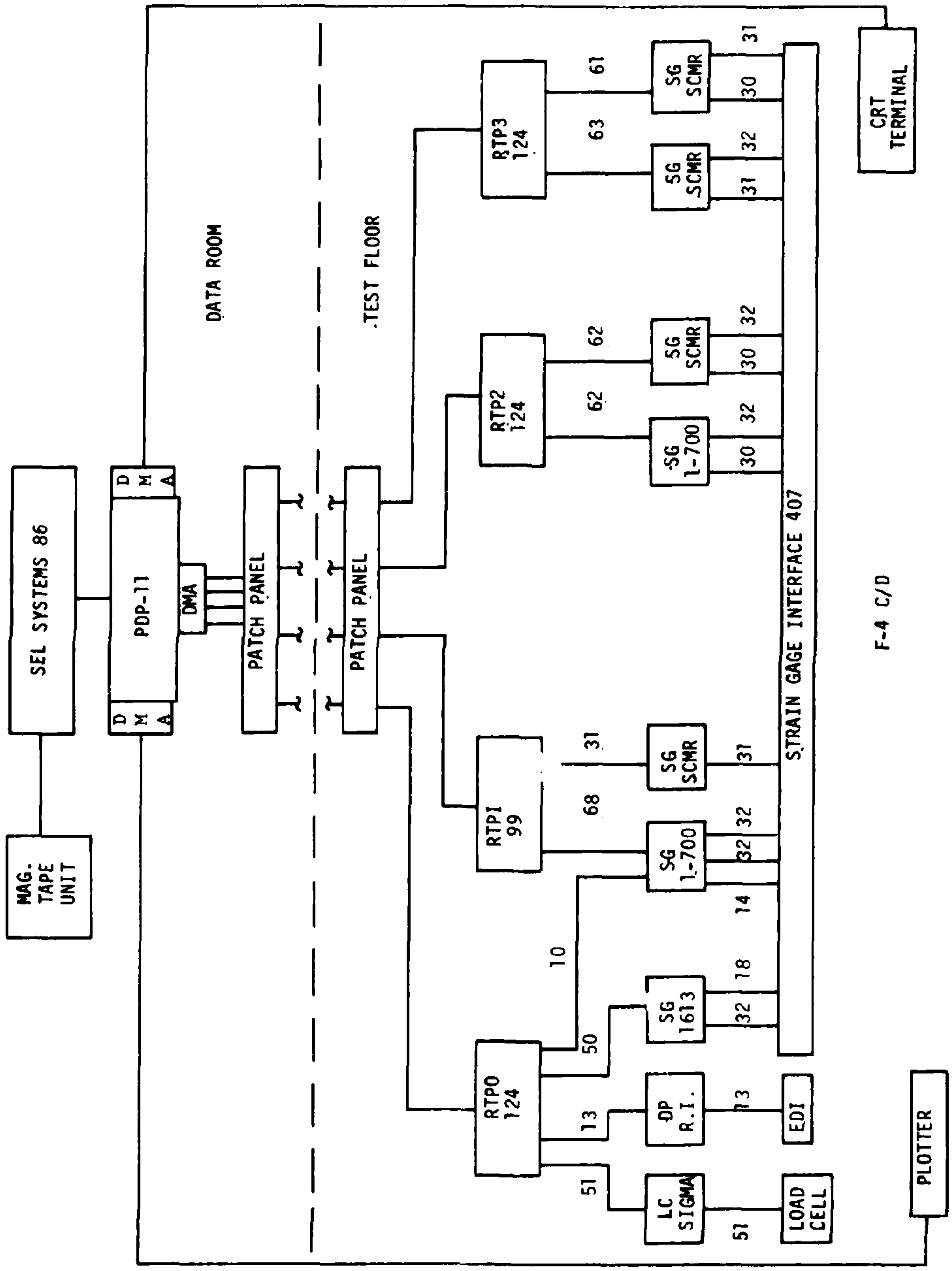


Figure 21. Block Diagram of Data System

survey condition and for the fatigue test were selected from the 483 sensor cables and were connected into the appropriate interface panel.

The strain gage half-bridges formed at the interface panels were connected by 407 six-conductor shielded cables to signal conditioning units. The signal conditioning units were of the constant voltage type and provided individual channel excitation level setting, initial bridge balance, and shunt calibration. Two conductor output cables carried the conditioned low-level signals to an RTP unit. Each TRP unit was a low-level, analog-to-digital coverting and multiplexing system with random access and remotely selectable gain. The all solid-state electronics unit was completely self-contained including all power supplies, cooling fans, and equipment necessary for complete operation of the unit.

Referred to the output, the system noise was less than 0.5 millivolt RMS, the zero stability was less than one millivolt for 48 hours, the linearity of the system was $\pm 0.01\%$ of full scale, and the system accuracy was $\pm 0.05\%$ of full scale. The RTP units operated at a maximum rate of 60 samples per second per channel. Thus, a complete 128 channel data scan was achieved in only 17 milliseconds.

The PDP-11 minicomputer, located in the third floor data room, controlled the operation of the RTP units through a direct memory access (DMA) channel. The data blocks obtained from the RTP units by the PDP-11 were immediately passed on to the SEL 86 processor where the raw data were recorded on magnetic tape for processing. The processed information was sent back to the on-line display devices located on the test floor. Due to the large amount of continuous processing and scanning involved in the program, the effective data sampling rate for recording purposes was reduced to approximately one scan of all 128 channels every 125 milliseconds in the automatic sampling mode. On-line data displays and computer controlled exceedance monitoring were also automatically up-dated at this same rate. The load cells, load links, and deflection transducers were recorded in a manner similiar to the strain gages.

The SEL 86 host computer also provided monitoring functions as well as data processing. The computer provided for on-line load program retrieval for use by the program checking mini as explained in Section II-2c. Figure 22 is a pictorial diagram of the load control and data system interaction. For 100 critical channels, the computer compared the transducer output during specific cycles of the fatigue test to the output obtained in the strain survey cycles and provided a printout of the data for channels that exceeded the strain survey data. The exceedance monitoring function is fully explained in Section III-3f.

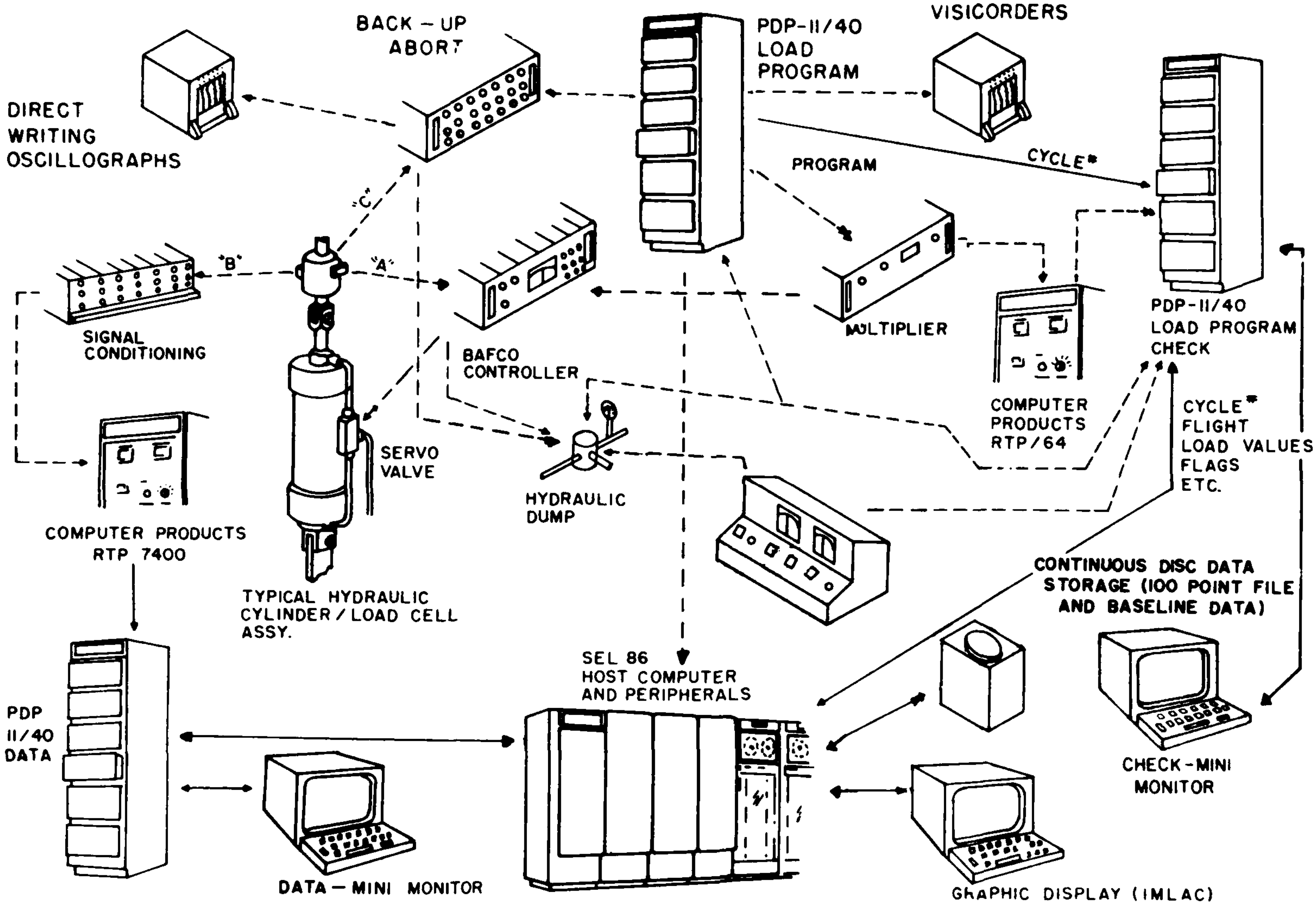
4. DATA MONITORING SYSTEMS

The on-line display unit was an IMLAC alphanumeric cathode-ray tube (CRT) terminal which displayed sensor outputs on demand. This unit enabled the test engineer to monitor the quality of input load application. The codes for this system are shown in Table 1. The terminal also was utilized as the display device for the last 100 data points which were continuously being stored in the computer as the two cycle file explained in Section III-8.

A parallel bridge output for each of the load cells was monitored with analog direct-writing oscillographs. These oscillographs ran continuously when the aircraft was being loaded. The charts, therefore, showed all the load cycles applied to the structure and served as a back-up recording and diagnostic system.

5. DATA SYSTEMS OPERATIONAL CHECK-OUT

Prior to the first strain survey, the data systems operational checkout was accomplished. This included checking all connecting cables to the strain gage interface panels and RTP units. All data system channels were balanced initially at a floating, zero load condition. This floating condition was recorded on magnetic tape as a system zero followed by a calibration/standardization output for all channels. Several random strain gage, load cell, and deflection channels,



F-4 FATIGUE TEST LOAD CONTROL & MEASUREMENT SYSTEM

Figure 22. Pictorial Diagram of Load Control and Data System

TABLE 1

F-4 C/D IMLAC CODES

SK	SKIP MODE: Mode in which program starts, data are available for update, no data are recorded on disk, i.e. 100 point file, no baseline comparisons are made. Also terminates continuous record.
RU	RUN MODE: 100 Point file is continuously updated, baseline comparisons are made.
HO	HOLD MODE: Suspends data gathering. Computer must be in this mode in order to call for IMLAC PLOT.
IPXXXX	IMLAC PLOT RTP CHANNEL NUMBER: Displays latest 100 points recorded on channel designated (System must be in HOLD mode).
RPXXXX	REAL-TIME PLOT: Gives point-by-point plot on IMLAC of channel designated. (System must be in RUN MODE).
CHXXXX	CHANNEL RTP CHANNEL NUMBER: Initiates a continuously updated tabular display of 60 channels (if 60 are available to end of listing) starting with channel designated. (System is in SKIP MODE unless RUN command is given.)
CR	CONTINUOUS RECORD: Records on tape at the same rate as the sampling rate. Use SKIP MODE to terminate.
WR	WRITE TWO CYCLE FILE: Writes contents of 100 points file onto magnetic tape as permanent record.
DLXXXX	DELETE PORT CHANNEL NUMBER: Deletes designated channel from further baseline comparison checking.
CU	CANCEL UPDATE: Of channel or loads.

representing each of four RTP units, were unbalanced in equal increments by adjusting the balance control of the signal conditioning modules. These simulated load increments were recorded with the PDP-11 mini-computer using the SEL 86 digital computer for processing data. This simulated load data, recorded on magnetic tape, was processed and printed out in final form to check the data format and the accuracy of the system. During the test simulation, the random data channels were keyed into the IMLAC terminal to observe the simulation display.

6. TEST DATA PROCESSING

The test data for processing and analysis included strain, deflection, and load cell measurements. Sensor numbers, data system channels and computer constants, for all the transducers used on each test condition were logged on a Data System Check Sheet.

Since the total number of transducers installed on the test article exceeded the capacity of the four RTP units, two strain surveys were conducted. The transducers not recorded on the first survey were connected in the system for the second survey. Analysis of the strain survey data determined which transducers were to be used for the fatigue test. The processed data format for all transducers except strain gage rosettes was printed in three columns:

a. Test Time
b. Raw data output in microvolts
c. Measured data in engineering units

Strain gage rosette data was printed out in seven columns:

a. Test time
b. "A" leg raw data output in microvolts
c. "A" leg strain
d. Maximum principal strain
e. Minimum principal strain
f. The maximum shear strain
g. Direction of maximum principal strain.

7. BASELINE DEVIATION DATA (EXCEEDANCE PRINT-OUTS)

During the strain surveys, the transducer output with the aircraft floating at zero, applied load was recorded for all data channels. An electrical shunt calibration/standardization record was then obtained. This information was used for computing loads, strains, and deflections throughout the survey. After all loads were adjusted, stabilized, and verified by the IMLAC on-line display at 1 G, blocks of data were recorded on magnetic tape with the SEL 86 computer. As the load was applied in 1/2 G increments, data from all channels were recorded at each increment, up to maximum G for the condition. At the maximum G point, the data from the 100 critical transducers, Table 2, were stored on a magnetic disc and identified as baseline value. This was done for all flight conditions noted in Table 3. Whenever the identical flight cycle occurred during the fatigue program the data from the 100 critical transducers were automatically compared with the maximum G data obtained in the original strain survey. The actual data deviated from the baseline, data by the test value and baseline value were typed on the SEL 86 printer.

8. 100 POINT FILE

As the cyclic loads were applied, the last 100 samples of data per channel were automatically recorded on a magnetic disc. The disc was continuously updated, i.e. after 100 samples, the earliest sample was erased as the latest sample was recorded. Therefore, the last 100 samples of data from all the transducers were always available for review by the test personnel with the IMLAC on-line terminal, for printing as a hard-copy or for recording on magnetic tape as a permanent record. At a data sampling rate of one sample per second, the 100 point file represented 100 seconds of real time or slightly less than six load cycles. The 100 point file recording system was also connected to the load control system so that recording was automatically stopped if there was a hydraulic dump. Since any test hardware or structural failure or load system error would initiate a dump, the last 100 samples of data were always preserved and could be reviewed to assure that the dump was not generated by an overload.

TABLE 2

CRITICAL TRANSDUCERS USED FOR BASELINE DEVIATION

Load Cells		Strain Gages	Deflection
W1L	W1R	11009 A,B,C	15
W2L	W2R	11043 A,B,C	
W3L	W3R	11075	
W4L	W4R	11115 A,B,C	
W5L	W5R	11153	
W6L	W6R	11175 A,B,C	
W7L	W7R	11207 A,B,C	
W8L	W8R	11215 A,B,C	
W9L	W9R	11225 A,B,C	
W10L	W10R	11233 A,B,C	
W11L	W11R	11239	
W13L	W12R	15003	
W13L	W13R	31005	
W14L	W14R	32025	
15L	15R	32043	
16	17	32067	
17-1	18	32091	
19	20	32117 A,B,C	
21	22	32141	
23	24	32161	
25L	25R	32175	
26	27	32203	
27-1	27-2	32249	
28	29	32263 A,B,C	
30	OUTBD		

TABLE 3

STRAIN SURVEY CONDITIONS

Mission Code	Max G Level	
237*	+4.0	
143*	+5.5	
237	-1.5	
335	-2.0	
299	landing	
335*	4.0	
557*	7.5	
647*	7.0	

* Maximum G data from critical transducers recorded on disc for baseline deviation. Maximum G data for missions 557 and 647 were recorded as cycle was first applied during spectrum loading.

SECTION IV

TEST PROCEDURES

1. STRAIN SURVEY

Prior to repeated loads testing, a series of strain surveys were conducted to establish a baseline of data for all future testing. Additional strain surveys were conducted during the test program for structural investigations due to modification and repairs. All strain surveys were credited to the total number of cycles of fatigue loading to be sustained by the airframe. Strain survey loads were applied in .5G increments through the complete load cycle. Eight flight conditions were used for the strain survey program and are defined in Table 3. Two of the eight strains surveys, which included large loads (7 and 7.5 G's), were conducted as they occurred during the fatigue program to avoid imposing potentially damaging load on the aircraft out of sequence with the spectrum. Wing, fuselage, and main landing gear deflection data were recorded during each strain survey.

After completion of the initial strain survey program, repeated loads test commenced.

Once the fatigue test had begun, all additional strain surveys requested by McDonnell Douglas were conducted when that particular cycle arose in the normal test sequence.

2. FATIGUE TEST

The fatigue test was started using an average cycling of 1.5 per minute. The cyclic speed was increased within a few weeks to the 3.5 cycles per minute which was used for the remainder of the test. The cyclic speed was limited to this rate to assure that all loads were within the desired test tolerances throughout the load cycle. Failures, repairs, and modification extended the test program to such a degree that the cyclic speed had no appreciable effect on the total duration of the test program. (Figure 1).

3. OPERATION

The test aircraft was tested in a floating set-up so as to assure specimen safety. The following sequence of events were used to obtain zero reference, start up, and shut down:

a. Engaged roll control dead weight system

b. The wing dead weight struts activated one by one as whiffletrees were straightened

c. Fuselage and engine dead weight struts activated

d. Positioning struts FS - 515 (Arresting Hook) FS - 82 (Nose Landing Gear) energized thereby stabilizing the aircraft in pitch and vertical position

e. Lateral restraining cable strut arrangement at FS - 82 and FS - 515 pressurized and jacks lowered from the aircraft.

f. Main landing gear loading strut pressurized through dead weight strut to alleviate the dead weight of the main landing gear

g. The test aircraft in a zero gravity condition and all load cells and strains recorded for the zero reading.

With all zero readings recorded, program mini in the ready state and the check mini synchronized, start up began.

a. Disengaged dead weight pressure on main landing gear and open active load servo loop to strut

b. Pushed Run Button - allowed low pressure oil to the active loading system

c. With all struts feeling load and in control, increased hydraulic pressure to the 3000 PSI system pressure

d. Turned program multiplier pot up slowly to obtain 100% of the program voltage sent down by the program mini, which at start up was at a one gravity condition

e. Looked over all loads to see if all loads had responded as programmed

f. Pushed run button to start program and pushed peak button to have the program stop on the peak of the load cycle

g. When program and load reached the peak, set all loads to within test accuracy desired

h. With the load accuracy satisfied, pushed peak button again - this allowed the program to continue down the ramp and continue throughout the spectrum sequence

Upon completion of cycling the standard shut down sequence was as follows:

a. Pushed end of cycle button - this allowed the program to stop on the one gravity condition following a load cycle

b. Turned the program pot to zero which placed the aircraft in a zero gravity condition except for the small preloads applied by the controllers

c. Pushed low pressure to approximately 500 PSI, then pushed dump button

d. The dump button deactivated the active control and the aircraft was in a zero gravity and floating with the dead weight system only

e. Raised all jacks that supported the aircraft which were the fuselage and wing jacks

f. After they were secured, removed pressure from the fuselage and engine dead weight system and de-energized the position struts, both vertical and lateral

g. Removal of pressure on the wing dead weight system completed the shut down pertaining to the test structure

4. INSPECTION TECHNIQUES

Visual inspection and dye penetrant were used throughout the test program and constituted approximately 80 percent of the time. Eddy current was used when fasteners were removed. Removal of fasteners were kept to a minimum so as not to generate flaws and induce cracks that would not normally occur in the fleet aircraft. Ultrasonic inspection was used on the critical taper locks installed in the main spar to lower skin attachment. Acoustic emission was used in an attempt to verify suspected cracks as interpreted by the ultrasonic device.

SECTION V

STRUCTURAL TEST FAILURES

The test aircraft sustained many fatigue discrepancies (Figure 23) during the test. Discrepancies occur in the speed brakes, flap up-stops, wing/fuselage attachment fittings, fuselage bulkheads, engine mounts, door hinges, intercostal assemblies, stiffeners, etc. Several rivets and other fasteners also failed. Many of these discrepancies were discovered at approximately the same time (spectrum hours) that similar failures were encountered in the fleet aircraft. A complete list of failures and their disposition is provided in Appendix A. The discrepancies marked with an asterisk (*) required immediate repair in order to continue the testing.

AFWAL-TR-82-3047

ERRATA - February 1983

The following correction is applicable to AFWAL-TR-82-3047, "F-4C/D Life Extension Program," UNCLASSIFIED report dated October 1982:

Page 53

Substitute the attached Figure 23 for the present Figure 23.

FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

[illegible] UNCLASSIFIED [illegible]

875 HOURS

1 SHROUD CRACKS AT F.S. 312.90 (32-[illegible]208) SHT 5 SECT [illegible] H41
2 B.L. 34 FORWARD FUSELAGE SPLICE AD RIVETS MISSING ON R.H. SIDE
3 SPEED BRAKE AFT CLOSURE ANGLES (53-11040-19) LEFT WING ONLY
4 FORWARD UPPER ENGINE MOUNT LINK & BALL APPEARED WORN ON LEFT SIDE
5 TRAILING EDGE FLAP UP STOP FITTING (32-32506) LEFT SIDE ONLY

1750 HOURS

6 CRACKED WING ACCESS DOOR L/H (32-[illegible]030-101)
7 WING/FUSELAGE ATTACH FITTING CRACKED AT F.S. 308 "DOGBONE FITTING" (32-32086) LEFT SIDE ONLY
8 35% BEAM CENTER WING R/H SIDE CRACKS SUSPECTED AT FASTENER HOLES IN WEB FOR FASTENERS COMMON TO WEB & LOWER [illegible] HOLE LOC. 2822 & 2826 BETWEEN B.L. 15 & B.L. 24

4200 HOURS

9 OUTER WING LOWER SKIN GOUGE AT LRS 248 & 45% LINE (32-[illegible])
10 CRACK IN OUTBOARD SPEED BRAKE TUNNELS LEFT & RIGHT WING (32-[illegible]215-15 AND -16)
11 CRACK IN 53-[illegible]031 AFT HINGE SUPPORT ON LEFT SIDE (M.L.G. INBOARD DOOR HINGE ON WING)
12 LOWER SURFACE WING SKIN STREAKS (INNER WING)
13 FOLD R.H. AT B.L. 60 HOLE DAMAGE AND GOUGING CAUSED BY FAIRING REMOVAL (INNER WING 32-11040)

6000 HOURS

14 ENGINE FIRE WALL SHROUD CRACKED AT F.S. 332 (32-32081-765
15 CRACK IN OUTBOARD SPEED BRAKE TUNNELS, LEFT & RIGHT WING (32-11215-15 AND -16
16 CLIP AT F.S. 332 B.L. 47 STRINGER NO 6 ASSEMBLY, CENTER FUSELAGE (53-32154-21 & 22
17 SPEED BRAKE FORWARD CLOSURE WEB (LEFT SIDE ONLY) (53-11041)
18 7/8" BOLT CRACKED AT F.S. 249.65 BULKHEAD ASSY OUTBOARD BOLT ON RIGHT SIDE
19 CRACK IN 53-11031 AFT HINGE SUPPORT ON LEFT AND RIGHT SIDE (M.L.G. INBOARD DOOR HINGE ON WING)

8000 HOURS

20 CRACK IN ENGINE FIRE WALL SHROUD AT F.S. 332 CRACK EXTENDED 3/4" PAST HOLE STOP DRILLED AT 6000 HOURS (32-32081-765)
21 CRACK EXTENDING THROUGH VERTICAL ROW OF FIVE FASTENERS AT F.S. 318 IN FIRE WALL SHROUD (32-3208[illegible])
22 CRACK IN BOTH LEFT AND RIGHT INBOARD TRAILING EDGE FLAP UPSTOP FITTINGS (32-32506-301 & 302)
23 CRACK IN OUTBOARD SPEED BRAKE TUNNELS LEFT AND RIGHT WING (32-11215-15 & 16) CRACK FIRST FOUND AT 6000 HOURS DID NOT LENGTHEN
24 CLIP AT F.S. 332, BL 47, STRINGER NO 6 ASSEMBLY, LEFT AND RIGHT SIDE (53-32154-21 & 22) THESE CRACKS FOUND AT 6000 HOURS DID NOT LENGTH
25 AFT FASTENER MISSING TYING DOGBONE FITTING (32-32086) TO FIRE WALL SHROUD (32-3208[illegible])
26 AT F.S. 249 ONE FASTENER IS MISSING ON EACH SIDE MATING 32-11010 FRONT SPAR TO 53-31075 AND 53-31076 LEFT AND RIGHT KEEL WEB.
27 CRACK INDICATION IN 53-11031 AFT HINGE SUPPORT FOR MAIN LANDING GEAR INBOARD DOOR ON FUSELAGE NOT AS DEFINITIVE AS WHEN WITNESSED AT 8000 HOURS
28 CRACK IN DUCT SEAL AT F.S. 303 JUST ABOVE WING

[illegible] HOURS

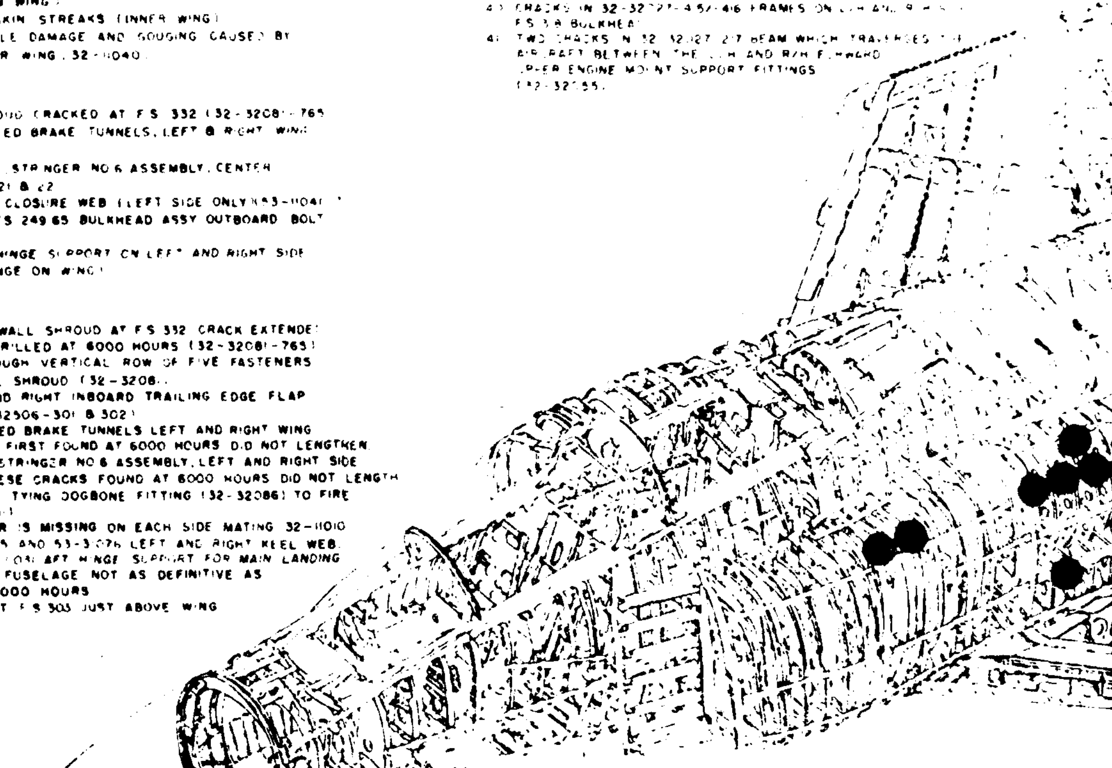
34 [illegible] (32-32[illegible]6 [illegible])
35 THREE [illegible] SHEAR RIVETS MISSING [illegible] HYDRAULIC ACCESS [illegible] FORWARD [illegible] SIDE OF AIRCRAFT
36 ONE HIGH SHEAR RIVET MISSING FROM 53-[illegible]84 [illegible] 94 5 TEE

[illegible] HOURS

37 FOUR FASTENERS HAVE FAILED WHICH [illegible] VERTICAL FLANGE OF THE 53-3250[illegible]16 R/H [illegible] ANGLE [illegible] THE HYDRAULIC ACCESS DOOR [illegible] FORWARD [illegible]

[illegible] HOURS

38 F.S. [illegible] BULKHEAD [illegible] ON BOTH R.H. AND [illegible] PLANE IN A [illegible] RUNNING A[illegible] 44 [illegible] AT THE POINT OF INTERSECTION BETWEEN [illegible] HORIZONTAL FLANGE [illegible]
39 32-32055 FORWARD ENGINE MOUNT SUPPORT [illegible] BOTH L/H AND R/H SIDES OF AIRCRAFT [illegible] AFT SIDE OF F.S. 318 B[illegible]
40 CRACKS IN 32-32[illegible]-457-416 FRAMES ON L/H AND R/H [illegible] F.S. 318 BULKHEAD
41 TWO CRACKS IN 32-32027-217 BEAM WHICH TRAVERSES THE AIRCRAFT BETWEEN THE L/H AND R/H FORWARD UPPER ENGINE MOUNT SUPPORT FITTINGS (32-32055)



42 CRACK PROPAGATING FROM FASTENER HOLE IN [illegible] MATING R/H SIDE PANEL AND FLOOR OF [illegible] FUE[illegible]
43 CRACK IN [illegible] FORMER [illegible] PART OF 32-3[illegible] ASSEMBLY
44 CRACK IN 32-32245-[illegible] FITTING TYING WING [illegible]
45 FASTENER MISSING TYING FRONT SPAR TO THE B.L. 34 SHEAR [illegible] F.S. 249.65 ON R/H SIDE OF AIRCRAFT
46 32-50028 [illegible] TYING [illegible]ATE TO BODY OF THE [illegible] ENGINE M[illegible]
47 ONE OF THE FOUR [illegible] NO 32-500[illegible] PAD TO THE R/H EN[illegible]
48 CRACKS IN 32-[illegible]219-187-[illegible] ANGLE AND IN 32-[illegible] 64 [illegible]
49 CRACK IN FIREWALL SHROUD [illegible] AT [illegible] PAST STOP DRILL HOLE
50 HYDRAULIC ACCESS [illegible] HALF [illegible] CRACKED
51 [illegible] BEL[illegible] CRACKED IN BOTH [illegible]
52 32-32127-[illegible]23 [illegible] LEFT SIDE [illegible] CRACKED
53 CRACKS IN 32-32[illegible] INTERCOSTAL ASSEMBLY [illegible] WARD OF F.S. 318

12000 HOURS

29 ENGINE FIRE W[illegible] EXTENDING [illegible] PAST PREVIOUS STOP DRILLED HOLE [illegible] F.S. 322.50 ([illegible] 32081-765)
30 SECOND CRACK IN DUCT SEAL AT F.S. 303 JUST ABOVE WING RESULTS IN DISCONNECTED [illegible]
31 TWELVE RIVETS MISSING FROM 53-32[illegible]-36 SPLICE ANGLE
32 CRACK IN FIRE WALL SHROUD (32-32081) EXTENDING HORIZONTALLY THRU FIVE FASTENERS [illegible] CRACK THRU VERTICAL ROW [illegible]
33 CHANNEL GUSSET (53-3[illegible]54-41) CRACKED [illegible]

SIGNIF[illegible]ANT FAILURES

[illegible] THE [illegible] AP UPSTOP FITTING

[illegible]SING TYING 53 32[illegible] 36 SPLICE ANGLE [illegible] FORWARD [illegible] F S 322 50 ON R/H

[illegible] FROM 53 32 154 19 [illegible] TYING [illegible] S TEE

[illegible] WHICH TIE THE VERTICAL [illegible] R/H SPLICE ANGLE [illegible] THE [illegible] FORWARD [illegible] F S 33[illegible]

[illegible] R/H [illegible] RUNNING [illegible] 44 [illegible] ARE [illegible] BETWEEN [illegible]

[illegible] AIRCRAFT [illegible]

[illegible] FRAMES ON L/H AND R/H SIDES OF

[illegible] BEAM WHICH TRAVERSES THE [illegible] AND R/H FORWARD [illegible] FITTINGS

[illegible] CRACKING IN [illegible] CENTER [illegible] ASSEMBLY [illegible] ON BOTH [illegible] THE AIRCRAFT BETWEEN F S [illegible] AND F S 326 95

55 CRACK NEAR [illegible] 53 32 [illegible] 34 [illegible] AIRCRAFT

[illegible] FASTENERS MISSING AND ONE [illegible] IN THE [illegible] PANEL BETWEEN F S 3[illegible] AND F S 322

57 [illegible] IN VERTICAL STIFFENERS LOCATED ON R/H AND [illegible] SIDES OF F S 249 [illegible] BULKHEAD [illegible] AT APPROXIMATE [illegible] AND [illegible]

58 [illegible] BETWEEN 53 32 [illegible] 6 [illegible] 62 BULKHEAD AND 3[illegible] WING TORQUE BOX ASSEMBLY

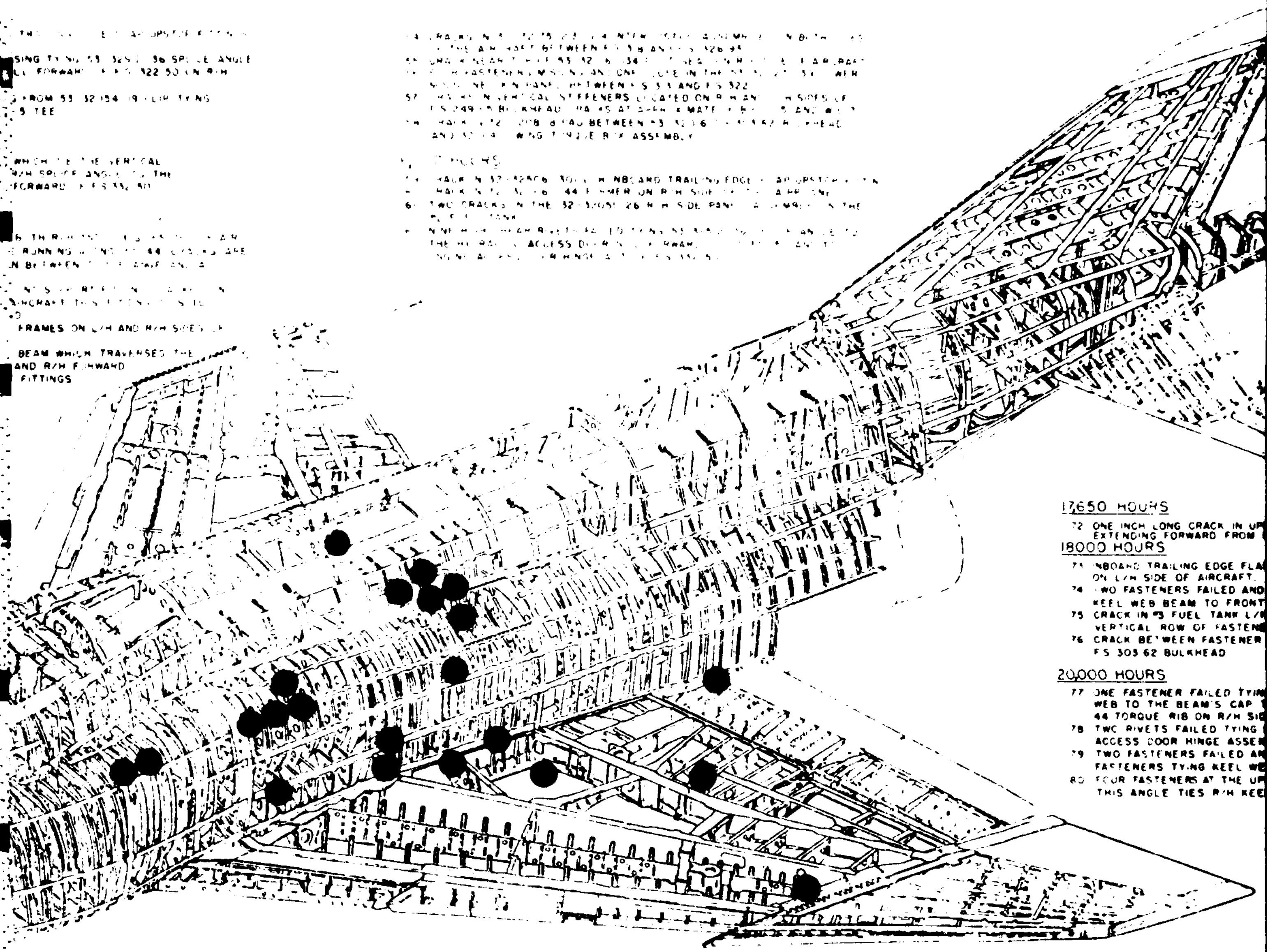
[illegible] HOURS

[illegible] CRACK IN 32 32506 30[illegible] INBOARD TRAILING EDGE [illegible] AP UPSTOP [illegible]

[illegible] CRACK IN 32 3[illegible] 6 44 FORMER ON R/H SIDE [illegible] AIRPLANE

6[illegible] TWO CRACKS IN THE 32 32051 26 R/H SIDE PANEL [illegible] IN THE [illegible] TANK

[illegible] NINE [illegible] RIVETS FAILED [illegible] THE HYDRAULIC ACCESS DOOR [illegible] FORWARD [illegible] HINGE [illegible]



17,650 HOURS

72 ONE INCH LONG CRACK IN UP[illegible] EXTENDING FORWARD FROM [illegible]

18000 HOURS

73 INBOARD TRAILING EDGE FLA[illegible] ON L/H SIDE OF AIRCRAFT.

74 TWO FASTENERS FAILED AND [illegible] KEEL WEB BEAM TO FRONT [illegible]

75 CRACK IN #3 FUEL TANK L/[illegible] VERTICAL ROW OF FASTEN[illegible]

76 CRACK BETWEEN FASTENER [illegible] F S 305 62 BULKHEAD

20,000 HOURS

77 ONE FASTENER FAILED TYIN[illegible] WEB TO THE BEAM'S CAP [illegible] 44 TORQUE RIB ON R/H SI[illegible]

78 TWO RIVETS FAILED TYING [illegible] ACCESS DOOR HINGE ASSE[illegible]

79 TWO FASTENERS FAILED A[illegible] FASTENERS TYING KEEL WE[illegible]

80 FOUR FASTENERS AT THE UP[illegible] THIS ANGLE TIES R/H KEE[illegible]

[illegible]TENER HOLE IN 32 32123 465 ANGLE MATING [illegible] #3 FUEL TANK

[illegible] PART OF 32 32016 137 L/H [illegible] NAME

[illegible] TYING WING PAD TO F S [illegible] NT SPAR TO THE B L 34 SHEAR WEB AT [illegible] AIRCRAFT

[illegible]TE TO BODY OF THE 32 5004[illegible] INBOARD

[illegible]NG 32-50014-301 FORWARD ENGINE MOUNT

[illegible] ANGLE [illegible] AND IN 32 [illegible] 84 [illegible]

[illegible] AT 10,000 H[illegible]

[illegible] HALF [illegible] CRACKED [illegible] ON BOTH SIDES OF AIRCRAFT [illegible] LEFT SIDE [illegible]

[illegible] INTERCOSTAL ASSEMBLY [illegible] FORWARD

63 CRACK IN OUTBOARD FIREWALL SHROUD, R/H SIDE, BETWEEN TWO FASTENERS IN VERTICAL ROW AT APPROXIMATELY F S 326

64 CRACK PROPAGATING FROM ONE FASTENER HOLE IN R/H FOLD RIB OF CENTER WING AND CRACKS PROPAGATING FROM FOUR HOLES IN L/H FOLD RIB

65 CRACK IN 32-11024-13 STIFFENER ON CENTERLINE RIB

66 THREE FASTENERS FAILED IN A VERTICAL COLUMN OF FASTENERS IN THE 35% BEAM NEAR THE JUNCTURE OF THE BEAM WITH THE TORQUE RIB

67 CRACK INBOARD FLANGE OF THE 32-32046-405 CHANNEL WHICH IS PART OF THE 32-32042-2339 KEEL WEB ASSEMBLY

68 TWO FASTENERS FAILED WHICH TIE THE 53-32512-84 WEB TO THE 53-32[illegible]4 HYDRAULIC ACCESS DOOR SILL ON EACH SIDE OF THE AIRCRAFT

69 CRACK IN 32 32175-137 INTERCOSTAL WHICH IS PART OF 32-32175-821 STRINGER NO 3 ASSEMBLY

[illegible] HOURS

70 CENTERLINE SPLICE PLATE CRACKED THRU ELEVEN FASTENERS IN THE OUTBOARD ROW ON THE LEFT SIDE OF THE AIRPLANE JUST FORWARD OF THE MAIN SPAR

[illegible] HOURS

71 CRACK IN BEND RADIUS OF 32 [illegible]875-[illegible] 152 ANGLES TYING LEFT AND RIGHT KEEL WEBS [illegible] 249 55 BULKHEAD

81 32-3208[illegible]-1168 CHANNEL WH[illegible] 312 40 AND F S 318 FRAMES[illegible] F S 318 END

82 CRACK PROPAGATING FROM [illegible] FORMER AT F S 308 31 ON[illegible]

21,970 HOURS

83 CRACKS IN THE CENTERLI[illegible] THE 3RD, 7TH, AND 9 TH [illegible] 16,271 HOURS

22,225 HOURS

84 CRACKS PROPAGATING IN TH[illegible] FROM THE 4TH AND 5TH H[illegible]

23,140 HOURS

85 A CRACK IN THE CENTERLI[illegible] FROM THE 13TH HOLE FO[illegible]

[illegible]

A[illegible]URES

[illegible] CRACK IN 32-[illegible] INTERCOSTAL ASSEMBLY [illegible] THE AIRCRAFT [illegible] F S 326 93

5[illegible] CRACK [illegible] SEAL [illegible] AIRCRAFT

5[illegible] [illegible] MISSING AND [illegible] LOOSE IN THE 53-32127-135 LOWER [illegible] SKIN PANEL BETWEEN F S 3[illegible]3 AND F S 322

57 CRACK [illegible] VERTICAL STIFFENERS LOCATED ON R/H AND L/H SIDES OF F S 249 65 BULKHEAD [illegible] AT APPROXIMATELY B L [illegible]

58 CRACK IN [illegible] PAD BETWEEN F S 3[illegible] F S 303 62 BULKHEAD AND 32-[illegible] WING TORQUE BOX ASSEMBLY

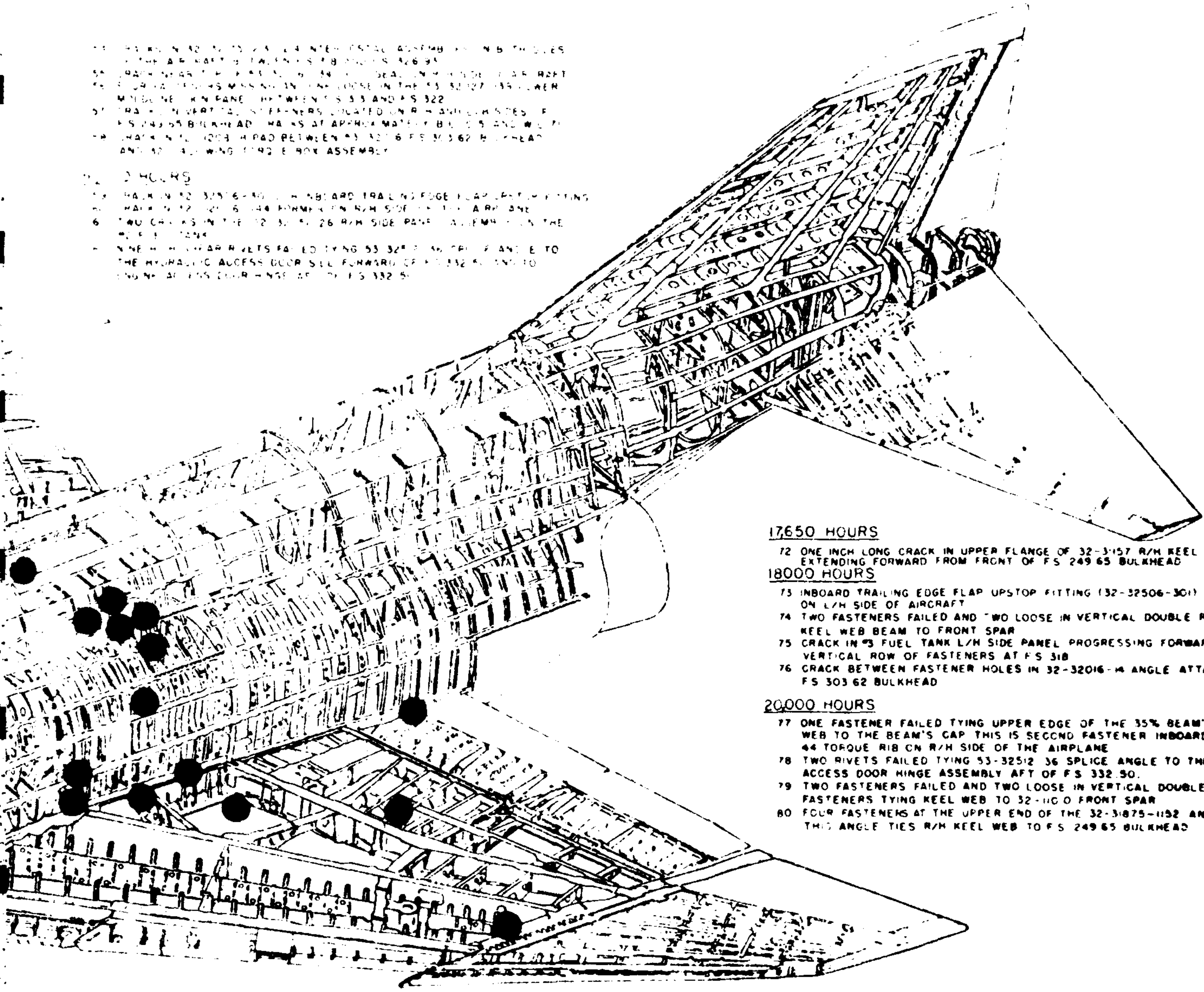
[illegible] HOURS

5[illegible] CRACK IN 32-32506-[illegible] INBOARD TRAILING EDGE FLAP UPSTOP FITTING

6[illegible] CRACK IN 32-[illegible] FORMER ON R/H SIDE OF THE AIRPLANE

6[illegible] TWO CRACKS IN THE 32-3[illegible] 26 R/H SIDE PANEL ASSEMBLY [illegible] THE [illegible] TANK

[illegible] NINE [illegible] RIVETS FAILED TYING 53-32[illegible] ANGLE TO THE HYDRAULIC ACCESS DOOR SILL FORWARD OF F S 332 [illegible] ENGINE ACCESS DOOR HINGE AFT OF F S 332 5[illegible]



17,650 HOURS

72 ONE INCH LONG CRACK IN UPPER FLANGE OF 32-3157 R/H KEEL WEB TRUSS EXTENDING FORWARD FROM FRONT OF F S 249 65 BULKHEAD

18000 HOURS

73 INBOARD TRAILING EDGE FLAP UPSTOP FITTING (32-32506-301) IS FAILED ON L/H SIDE OF AIRCRAFT

74 TWO FASTENERS FAILED AND TWO LOOSE IN VERTICAL DOUBLE ROW TYING KEEL WEB BEAM TO FRONT SPAR

75 CRACK IN #3 FUEL TANK L/H SIDE PANEL PROGRESSING FORWARD FROM VERTICAL ROW OF FASTENERS AT F S 318

76 CRACK BETWEEN FASTENER HOLES IN 32-32016-14 ANGLE ATTACHED TO F S 303 62 BULKHEAD

20,000 HOURS

77 ONE FASTENER FAILED TYING UPPER EDGE OF THE 35% BEAM'S VERTICAL WEB TO THE BEAM'S CAP THIS IS SECOND FASTENER INBOARD OF B L 44 TORQUE RIB ON R/H SIDE OF THE AIRPLANE

78 TWO RIVETS FAILED TYING 53-32512 36 SPLICE ANGLE TO THE ENGINE ACCESS DOOR HINGE ASSEMBLY AFT OF F S 332.50.

79 TWO FASTENERS FAILED AND TWO LOOSE IN VERTICAL DOUBLE ROW OF FASTENERS TYING KEEL WEB TO 32-1100 FRONT SPAR

80 FOUR FASTENERS AT THE UPPER END OF THE 32-31875-1152 ANGLE FAILED THIS ANGLE TIES R/H KEEL WEB TO F S 249 65 BULKHEAD

63 CRACK IN OUTBOARD FIREWALL SHROUD R/H SIDE BETWEEN TWO FASTENERS IN VERTICAL ROW AT APPROXIMATELY F S 324

64 CRACK PROPAGATING FROM ONE FASTENER HOLE IN R/H FOLD RIB OF CENTER WING AND CRACKS PROPAGATING FROM FOUR HOLES IN L/H FOLD RIB

65 CRACK IN 32-11024-15 STIFFENER ON CENTERLINE RIB

66 THREE FASTENERS FAILED IN A VERTICAL COLUMN OF FASTENERS IN THE 35% BEAM NEAR THE JUNCTURE OF THE BEAM WITH THE TORQUE RIB

67 CRACK INBOARD FLANGE OF THE 32-32046-405 CHANNEL WHICH IS PART OF THE 32-32042-2339 KEEL WEB ASSEMBLY

68 TWO FASTENERS FAILED WHICH TIE THE 53-32512-84 WEB TO THE 53-32514 HYDRAULIC ACCESS DOOR SILL ON EACH SIDE OF THE AIRCRAFT

69 CRACK IN 32-32175-137 INTERCOSTAL WHICH IS PART OF 32-32175-821 STRINGER NO 3 ASSEMBLY

[illegible] HOURS

70 CENTERLINE SPLICE PLATE CRACKED THRU ELEVEN FASTENERS IN THE OUTBOARD ROW ON THE LEFT SIDE OF THE AIRPLANE JUST FORWARD OF THE MAIN SPAR

[illegible] HOURS

71 CRACK IN BEND RADIUS OF 32-31875-1151 [illegible] ANGLES TYING LEFT AND RIGHT KEEL WEBS TO F S 249 65 BULKHEAD

81 32-32081-1168 CHANNEL WHICH ACTS AS AN INTERCOSTAL BETWEEN F S 312 40 AND F S 318 FRAMES FAILED BETWEEEN TWO FASTENERS NEAR F S 318 END

82 CRACK PROPAGATING FROM .56 DIA HOLE IN WEB OF 32-32006-144 FORMER AT F S 308 31 ON R/H SIDE OF AIRPLANE

21,970 HOURS

83 CRACKS IN THE CENTERLINE SPLICE PLATE PROPAGATING BOTH FORE AND A[illegible] THE 3RD, 7TH, AND 9 TH HOLES FORWARD OF THE DOUBLER ADDED AT 16,271 HOURS

2[illegible]22[illegible] HOURS

84 CRACKS PROPAGATING IN THE CENTERLINE SPLICE PLATE BOTH FORE AND AF[illegible] FROM THE [illegible]TH AND 5TH HOLES FORWARD OF THE DOUBLER ADDED AT 16,27[illegible]

23,140 HOURS

85 A CRACK IN THE CENTERLINE SPLICE PLATE PROPAGATING BOTH FORE AND [illegible] FROM THE 13TH HOLE FORWARD OF THE DOUBLER ADDED AT 16,271 HOURS

SECTION VI

MODIFICATIONS

The test aircraft was manufactured at McDonnell Douglas to represent an original F-4C/D. The modification on that would normally have been installed during mnufacturing was not incorporated.

As the test aircraft accumulated test hours, the modifications that were incorporated in the fleet were installed in the test aircraft (Table 4). These modifications were installed in the test aircraft at the average flight hours installation occurred in the fleet.

At 4000 hours the life extension modifications were installed.

The original planned modifications did not include the engine mount modification or the center splice doubler. These modifications were incorporated only after failure occurred in the fatigue test article. At 13,818 hours failure occurred in the upper engine mount back-up structure. The damage in the back-up structure was typical of that discovered in other test aircraft.

On the retrofit kit, developed by McDonnell Douglas for the Navy Aircraft at 16,271 test flight hours, the center line splice cracked in the left outer row just forward of the main spar. This was repaired by means of another Navy modification which incorporated a large doubler installed over the aft section of the center line splice.

TABLE 4

MODIFICATIONS

SPECTRUM HOURS	TCTO NO.	MODIFICATIONS
500	809	Taper Lok Mod on lower wing skin and main spar.
1875	915	Reinforcing doublers, outer wing assembly, lower skin.
	1014	Cold work of holes.
2750	1035	Kit
	1071	Fuel Cell Mod.
	1025	Modified AFT Locking Lugs.
	991	Aileron Closure Skin, Keel Webs, 303 Bl Head.
	SS359	Control Through.
4000	8166-3	Intermediate MLG rib.
	8167-1	Pylon hole.
	8167-2	Wing fold rib.
4000	8167-3	Cold Work TB Skin.
	-5	# 1 Stringer splice plate.
	-7	Cold work lower TB skin at torque rib.
	-9	Rework canopy sills.
13,818	598	Engine Mount Mod.
16,271		Navy Mod - Doubler installed over the mainspar, wing and splice plate at the aft section of the wing box.

SECTION VII

TEST PROGRAMS ADDITIONS

This program was generated in 1973 and completed in 1981. During this period several fleet failures occurred causing this test article to be used as a test bed for data gathering mechanism to define and analyze the various problem areas. The center line splice was instrumented because of a catastrophic failure during landing of the F-4 on a carrier deck. Instrumentation installed is described in Figures 24 and 25.

The outer wing just outboard of the wing fold area was also instrumented as described in Figure 26. This was also due to a failure that occurred in the fleet.

Add on or piggy back programs that were not a result of aircraft failures were the Crack Gage evaluation, Fatigue Sensor (annealed foil gage) and the installation of strain gages and strain survey conducted for use in survivability and vulnerability analyses.

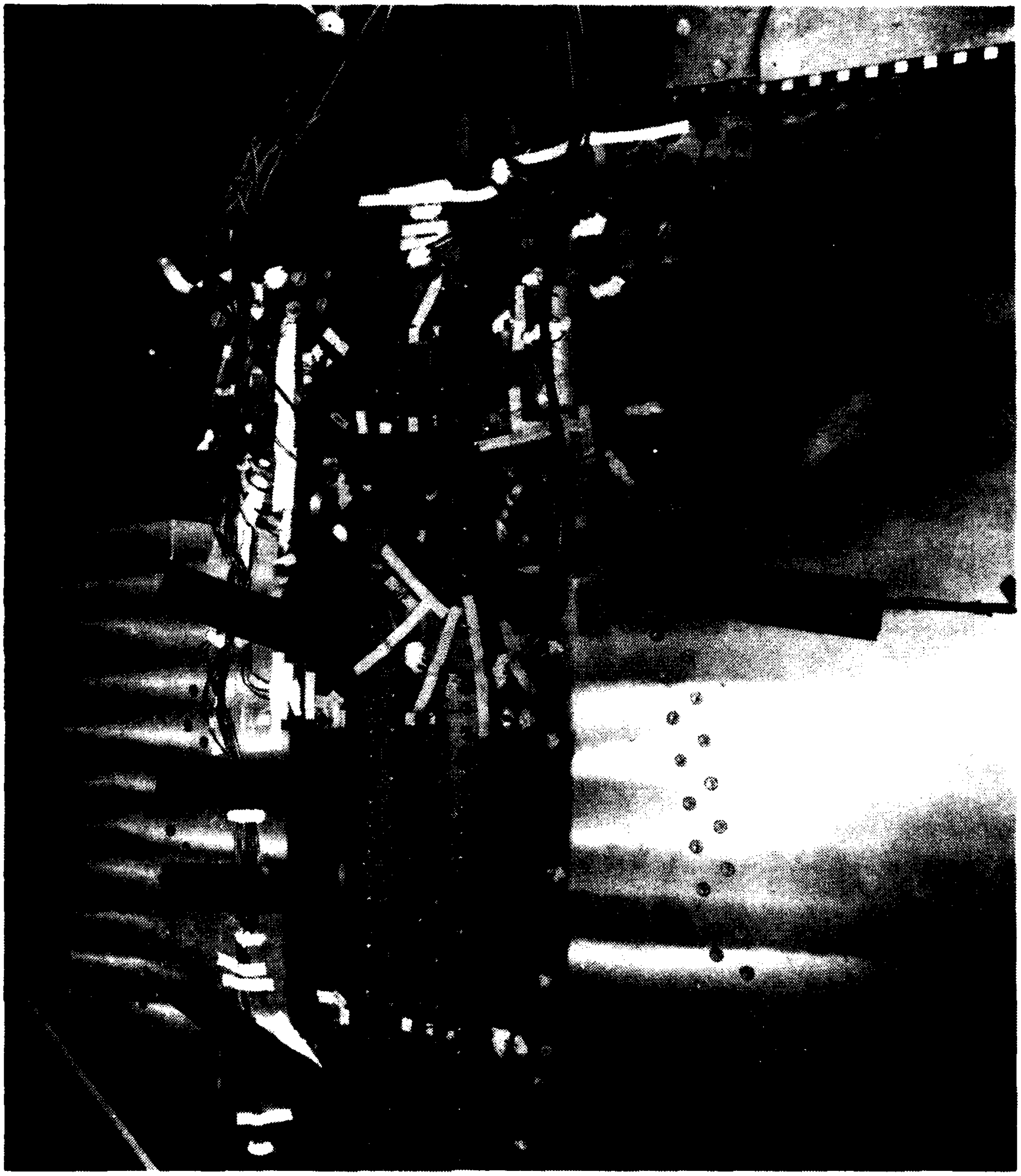


Figure 24. Instrumentation of Center Splice

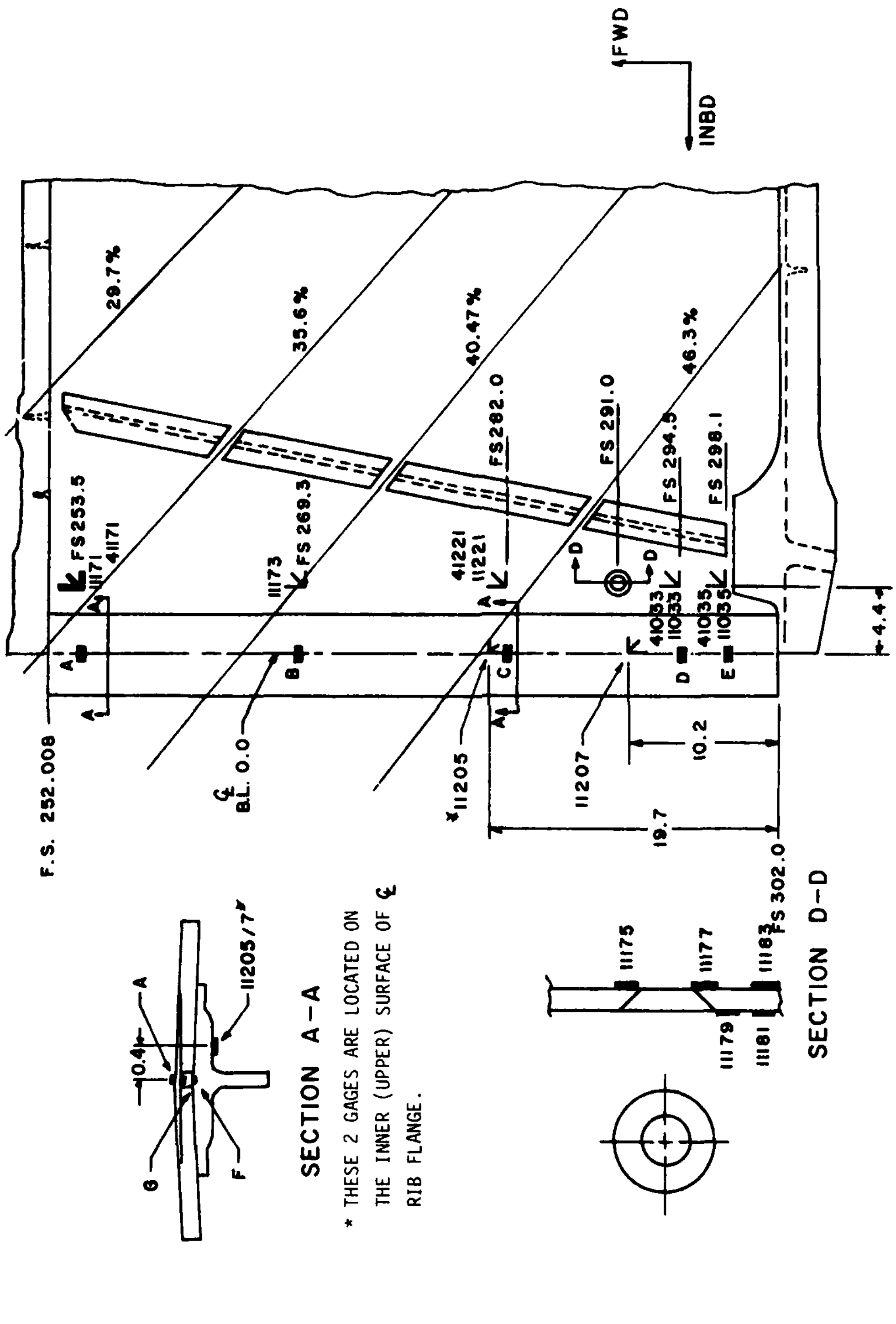


Figure 25. Drawing of Instrumentation of Center Splice

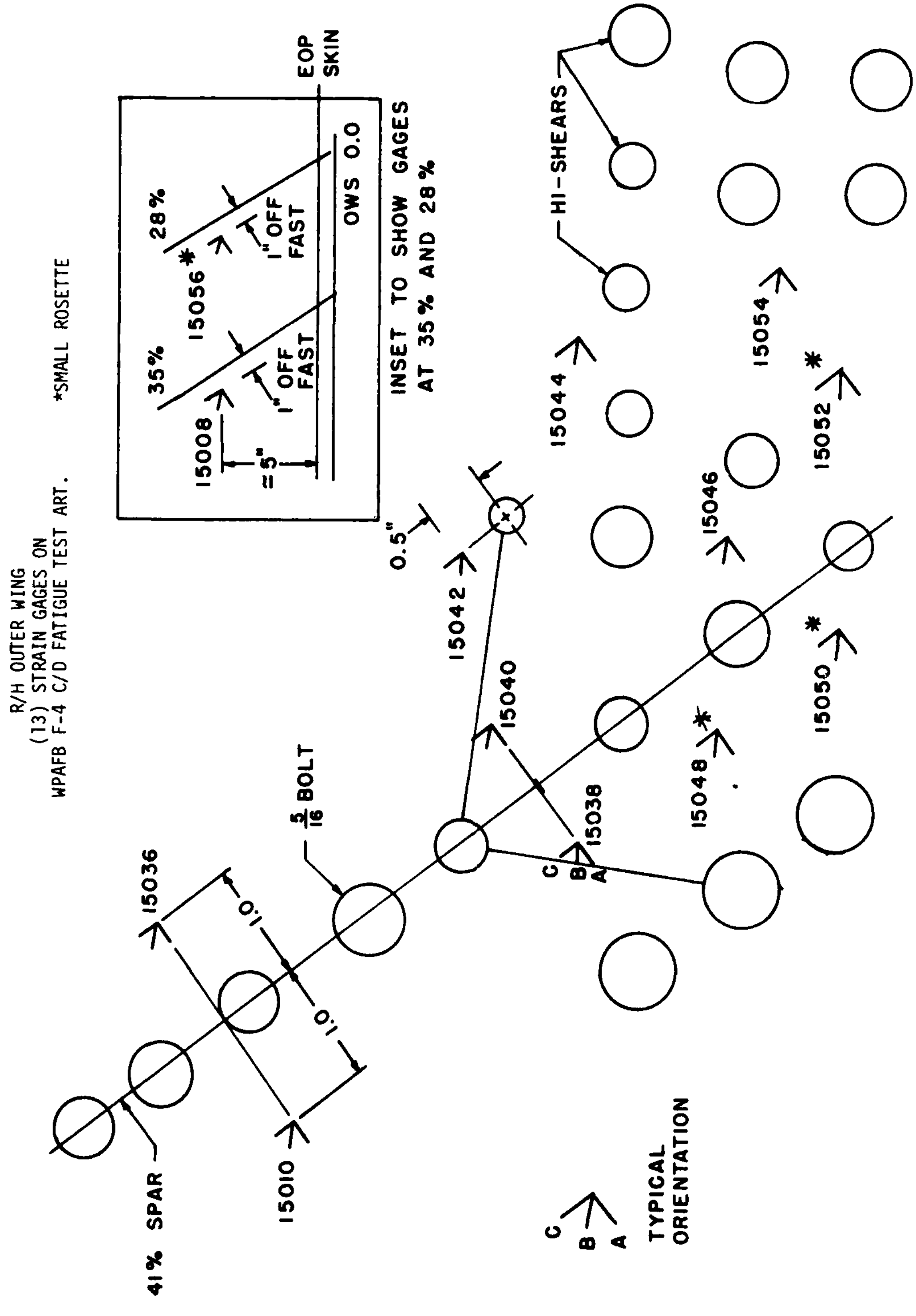


Figure 26. Outer Wing Instrumentation

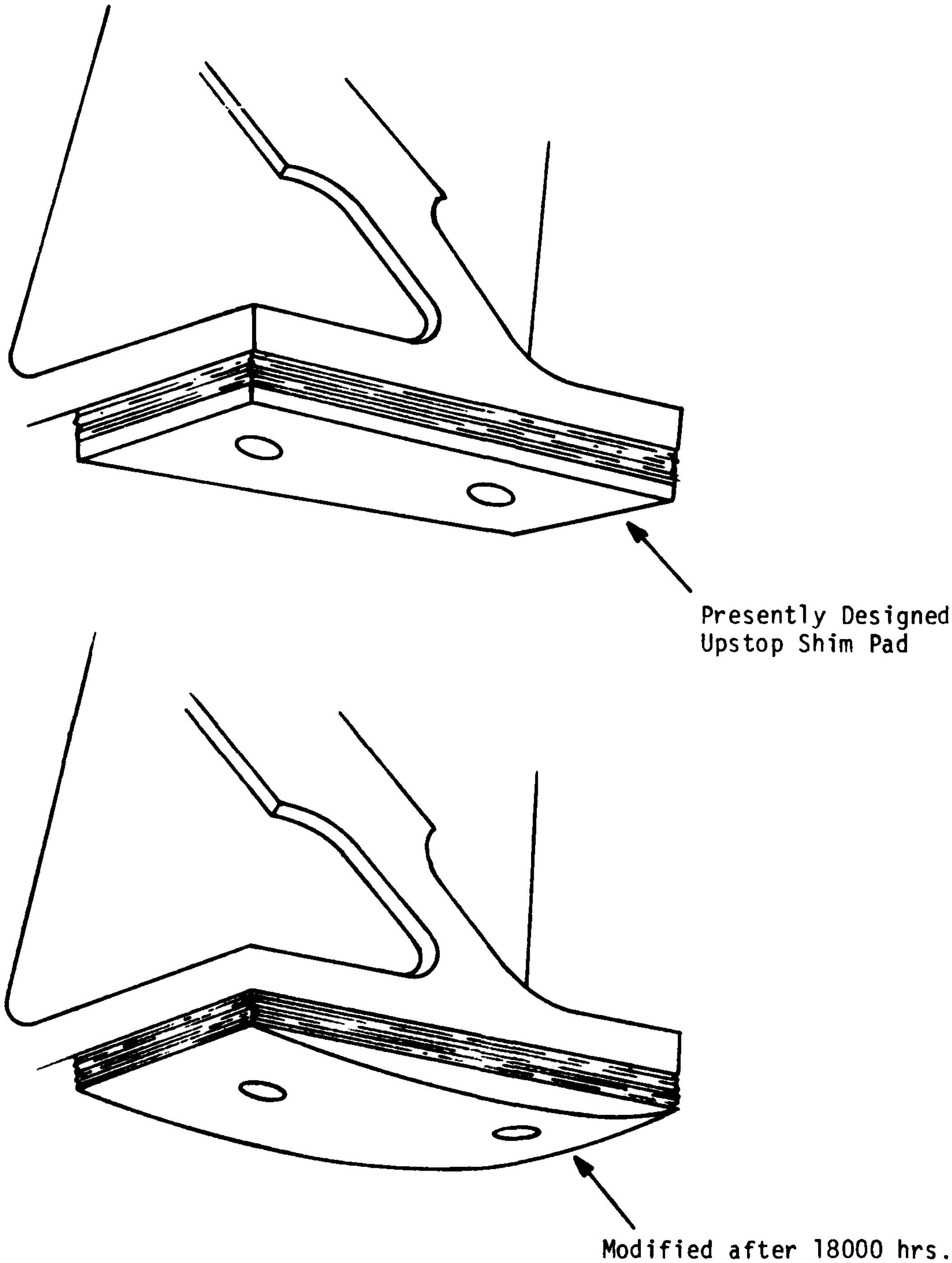
SECTION VIII

CONCLUSIONS

The test was a complete success and results were well documented. The tear down inspection that is being performed by McDonnell Douglas will determine the degree of effectiveness of these modifications and damage accumulated on the test structures.

The aircraft structure sustained its integrity through 24000 simulated flight hours of testing. In order to obtain this 24000 hours; two additional modifications (See Section VI) were installed to prevent catastrophic failure to the test aircraft. This is theoretically equivalent to three design lifetimes. All major discrepancies in this report were repaired as they occurred.

The flap upstop failed repeatedly and was replaced every 2000 hours. This failure constituted no major problem to the aircraft but remains a considerable nuisance in the maintenance of the aircraft. After 18,000 hours of testing, the flap upstop shims were modified and installed in the test aircraft (Figure 27). No further failures of the upstop occurred.



The rounded fore and aft edges assure that the upstop load is applied close to the middle of the pad and therefore reduces the bending associated with off center loads.

Figure 27. Upstop Modification

SECTION IX

RECOMMENDATIONS

Comparison of running time vs down time (Figure 1) indicated a disproportionate amount of overhead caused by having delays due to repairs, decisions and storage. It is recommended that future projects with the possibility of having long scheduled delays in testing be carefully considered when accepting bids of industry. Overhead cost generally continues during the down periods.

The tension pads used were the 2" X 2", RTV 60 type. It is recommended that a larger pad be used in the future and with a bonding adhesive that is not affected by the hydraulic oil, be used within the test facility.

The exceedance print-out (explained in Section III-7) proved its value and it is recommended for use on all future fatigue tests. This system compares original strains to that now produced on a given number of survey cycles. An example of this is where the exceedance print-out indicated a change in strain in the center wing lower skin above the set limit. Investigation of this area revealed a crack in the lower splice place some 8" in length. It is believed that, because of the massive wiring bundles in this area, this crack would not have been found and a catastrophic failure would have occurred if it had not been for the exceedance print-out.

It is also recommended that some consideration be given to the effects that the proposed inspections have on the fatigue life of the test aircraft. It is believed that inspections that involve the removal of fasteners, that are not normally removed in the fleet, effect the fatigue life considerably, depending on the fastener class fit and stress applied. A follow-up program to determine these effects, if any, has been initiated by AFWAL/FIBT.

APPENDIX A

SUMMARY OF F-4C/D FATIGUE TEST FAILURES

SUMMARY OF F-4C/D FATIGUE TEST FAILURES

	SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
	500	Gouge's in fastener holes at B.L. 100 and main spar lower surface was caused by fasterner removal.	Installed taper lok modification ahead of original schedule.
	1875	Shroud Cracks @ F.S. 312.90 (32-32081, Sht. 5, Sect. D-D, Z-H4)	Stop-Drilled Crack in Shroud. Replace Cut-Out Doubler.
	1875	B.L. 34 Forward Fuselage Splice AD Rivets Missing on R/H Side	Rivets Replaced.
*	1875	Speed Brake Aft Closure Angles (53-11040-19) Left Wing Only (Figure 28).	Replaced Aluminum Angles with Titanium Angles (ECP 1014).
	1875	Forward Upper Engine Mount, Link & Ball Appeared Worn on Left Side	Replaced Ball and Link on L/H Side Only.
*	1875	Trailing Edge Flap Up-Stop Fitting (32-32506) Left Side Only (Figure 29).	Replaced with Beefed-Up Part (ECP 1014).
	2750	Cracked Wing Accessor Door, L/H (32-11030-101)	Replaced Door.
*	2750	Wing/Fuselage Attach. Fitting Cracked @ F.S. 308 "Dogbone Fitting" (32-32086) Left Side Only.	Replaced with Beefed-Up Part (ECP 1014).
	2750	35% Beam Center Wing R/H Side Cracks Suspected @ Fastener Holes in Web for Fasteners Common to Web & Lower Cap. Fast. Hole Loc.: 2822 & 2826 Between B.L. 15 & B.L. 24	No Action Taken.
	4000	Outer Wing Lower Skin Gouge at LRS 248, & 45% Line (32-15531).	
	4000	Crack in Outboard Speed Brake Tunnels, Left & Right Wing (32-11215-15 and 16)	No Action Taken.
	4000	Crack in 53-11031 Aft Hinge Support on Left Side (M.L.G. Inboard Door Hinge on Wing)	No Action Taken.

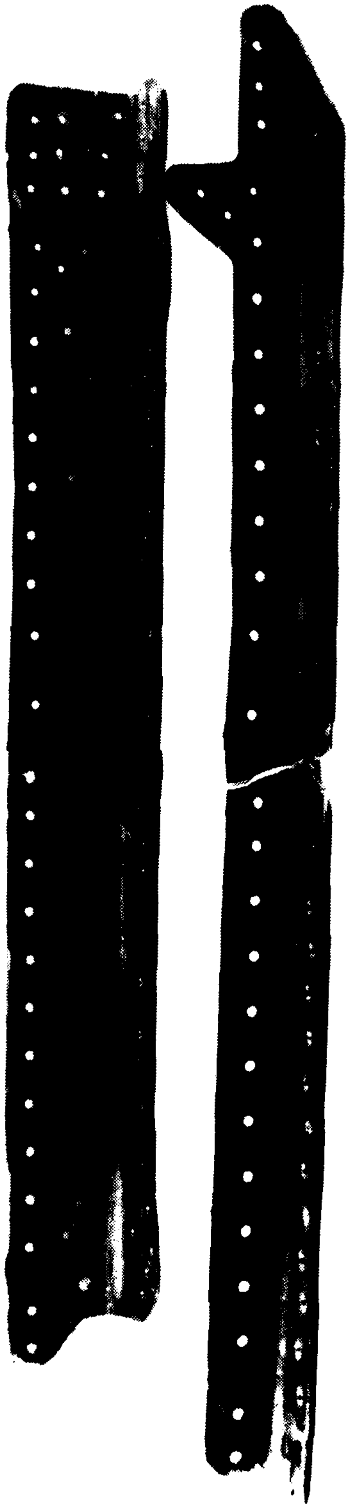


Figure 28. Speed Brake Failure

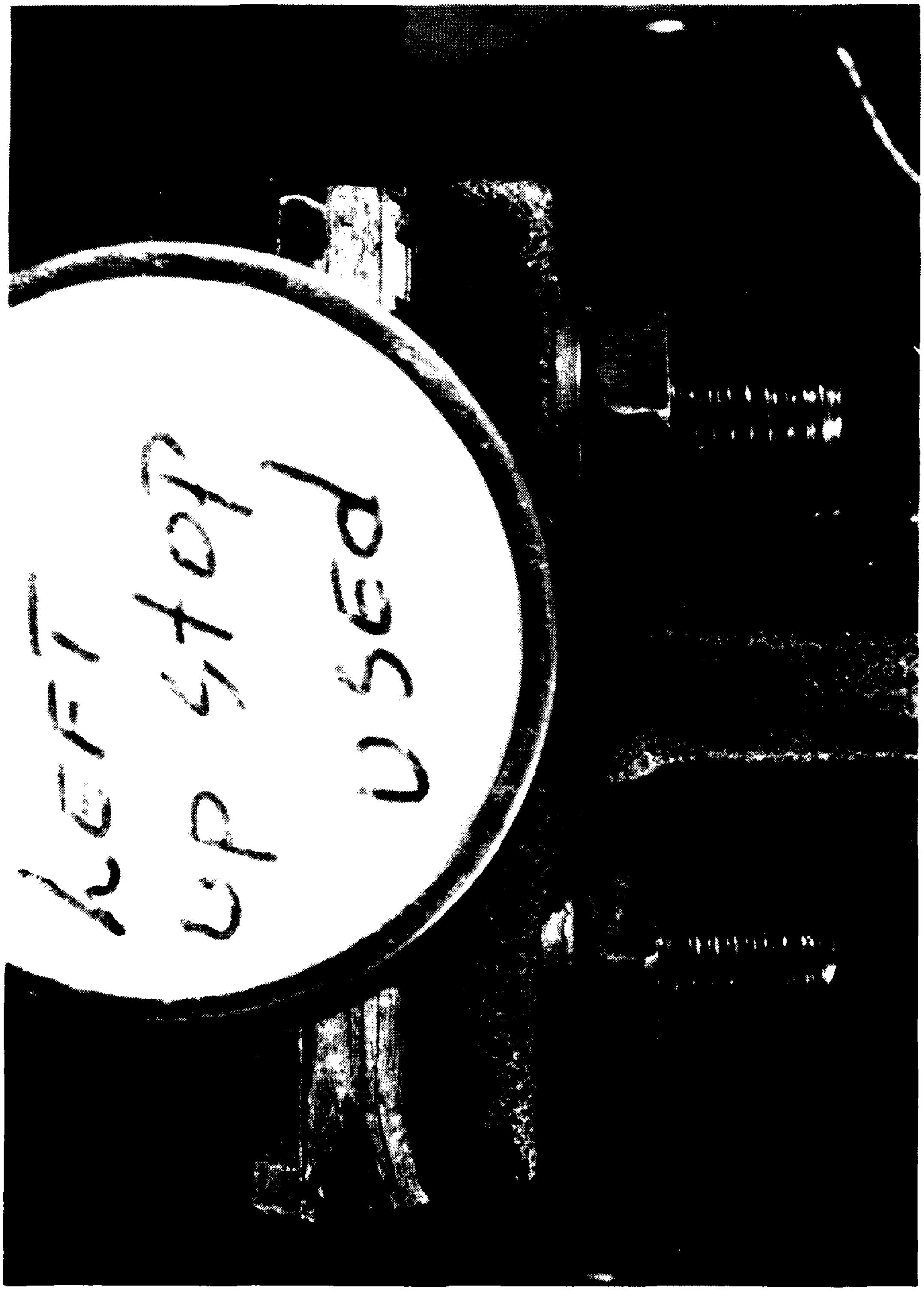


Figure 29. Upstop Failure

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
4000	Lower Surface Wing Skin Streaks (Inner Wing)	No Action Taken.
4000	Fold Rib at B.L. 160, Hole Damage and Gouging Caused by Fairing Removal (Inner Wing, 32-11040)	Blended Out Gouges.
6000	Engine Fire Wall Shroud Cracked @ F.S. 332 (32-32081-765) (Figure 30).	Stop Drilled
6000	Crack in Outboard Speed Brake Tunnels, Left & Right Wing (32-11215-15 & 16) (Figure 31).	No Action Taken.
6000	Clip at F.S.332, B.L. 47; Stringer No. 6 Assembly, Center Fuselage (53-321154-21 & 22)	No Action Taken.
* 6000	Speed Brake Forward Closure Web (Left Side Only) 53-11041-7 (Figure 32).	Aluminum Web Replaced with Titanium Web (Left Side Only).
6000	7/8" Bolt Cracked @ F.S. 249.65 Bulkhead Assy. Outboard Bolt on Right Side	Replaced Bolt.
6000	Crack in 53-11031 Aft Hinge Support on Left and Right Side (M.L.G. Inboard Door Hinge on Wing)	No Action Taken.
8000	Crack in Engine Fire Wall Shroud F.S. 332. Crack extended 3/4" Past Hole Stopped Drilled at 6000 Hours (32-32081-765)(Figure 33).	Stop Drilled with 3/8"
8000	Crack extending through vertical row of five fasteners at F.S. 318 in Fire Wall Shroud (32-32081)(Figure 34).	No Action Taken.
* 8000	Crack in Both Left and Right-Hand Inboard Trailing Edge Flap Up-Stop Fittings (32-32506-301 & - 302)	Replaced Fittings.
8000	Crack in Outboard Speed Brake Tunnels Left and Right Wing (32-11215-15 & 16). Crack first found at 6000 hours did not lengthen (Figure 35).	No Action Taken.

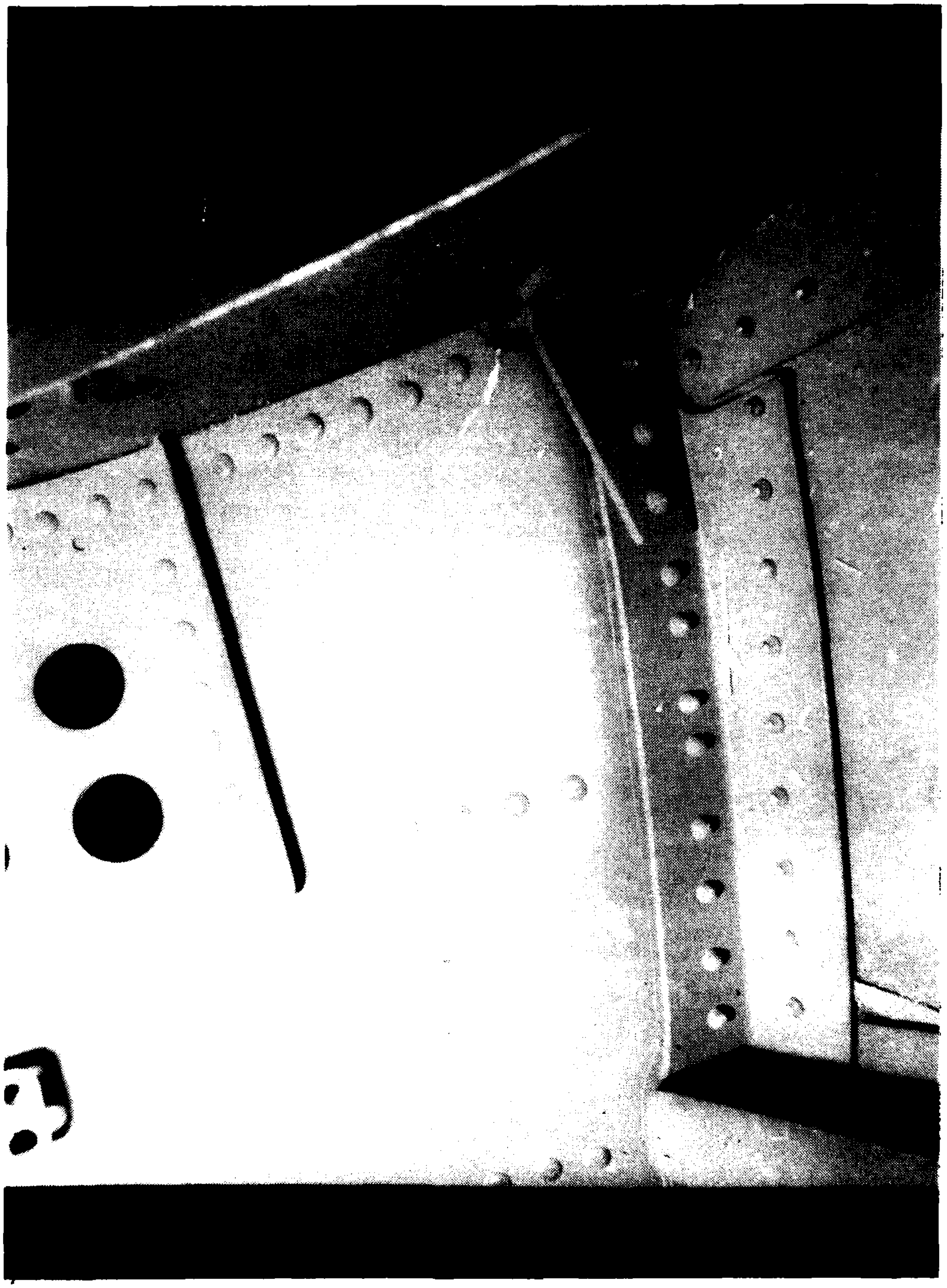


Figure 30. Crack in Engine Shroud

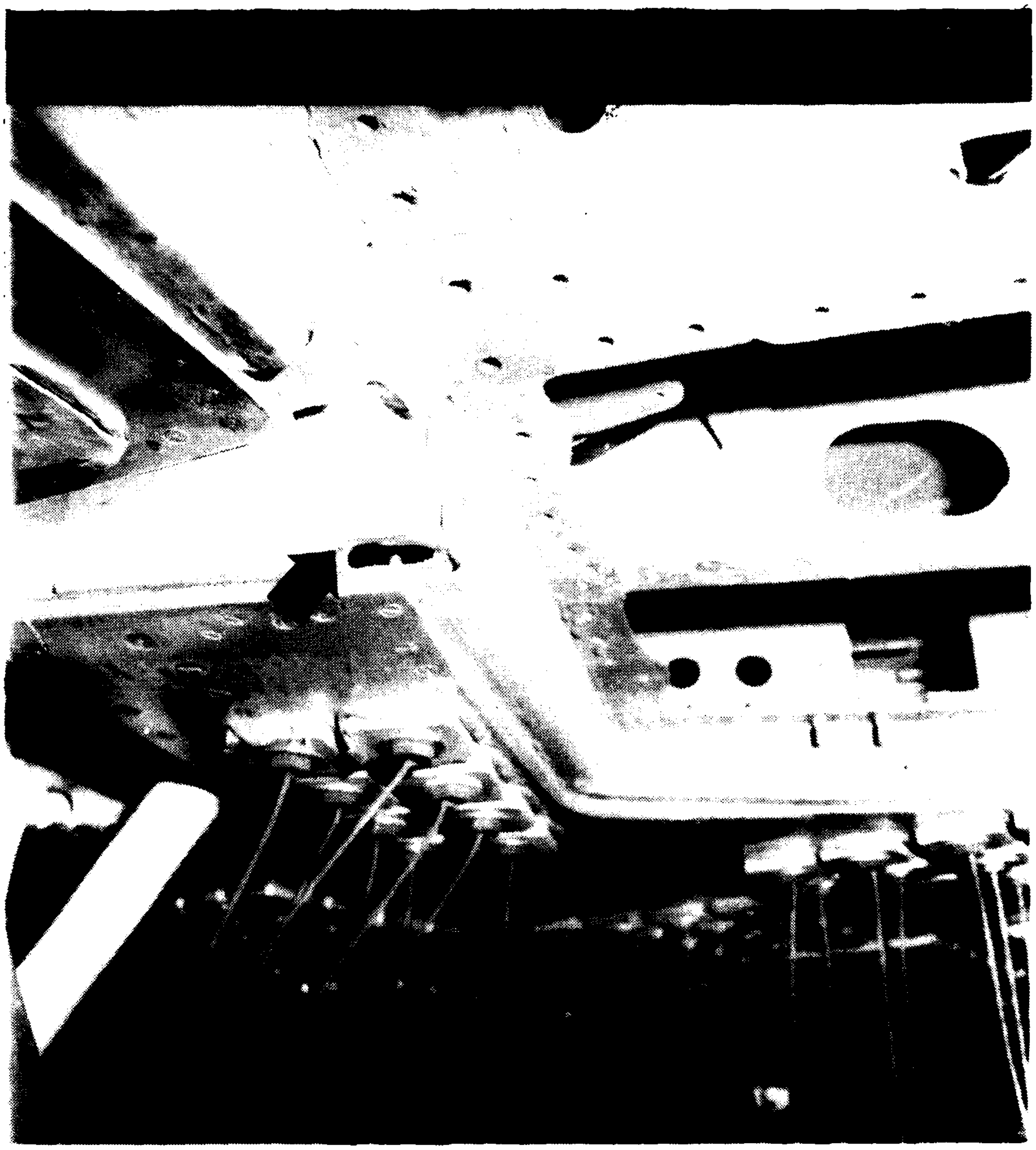


Figure 31. Cracks in Speed Brake Tunnels

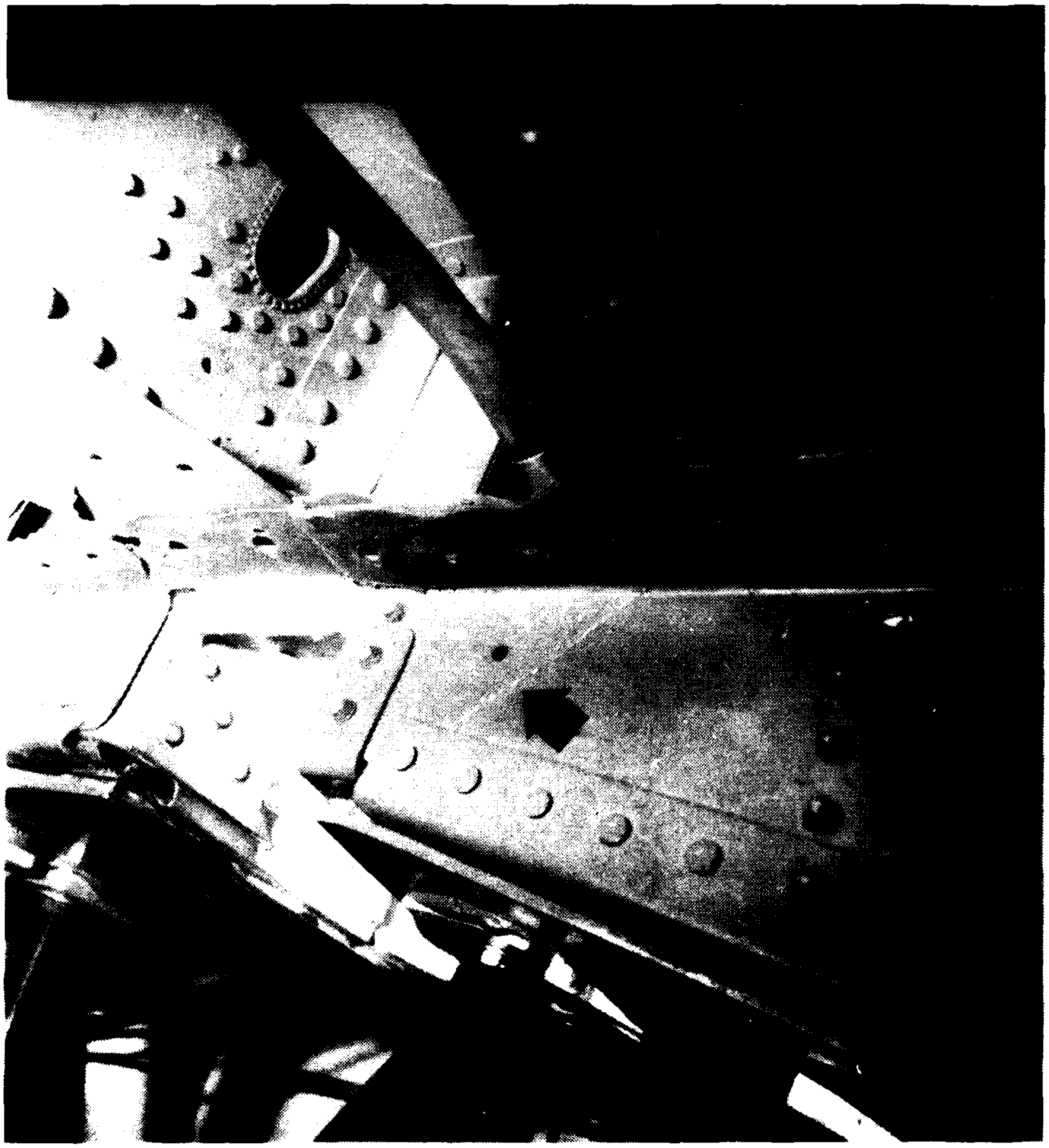


Figure 32. Failure of Speed Brake Enclosure

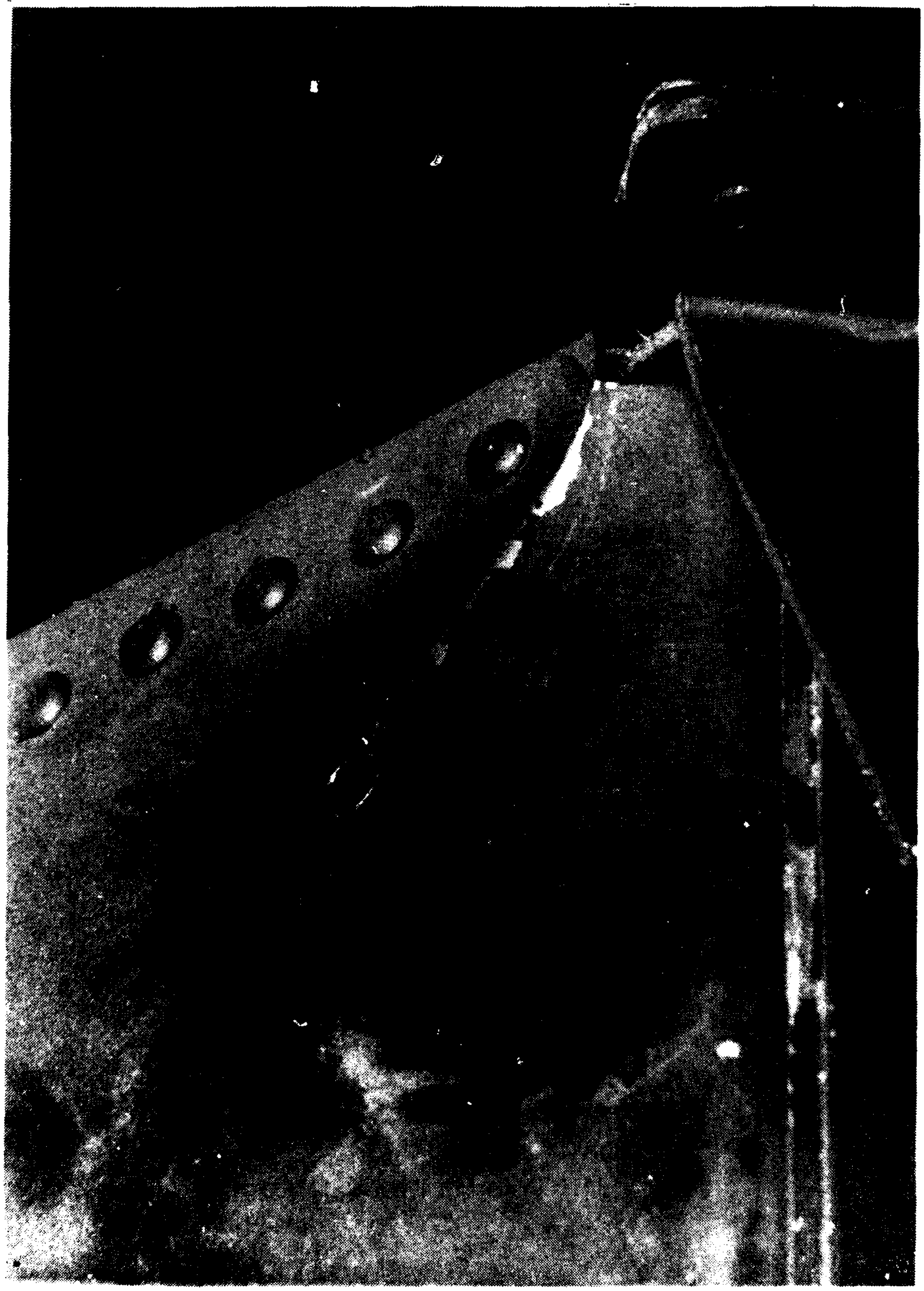


Figure 33. Crack in Shroud FS-332

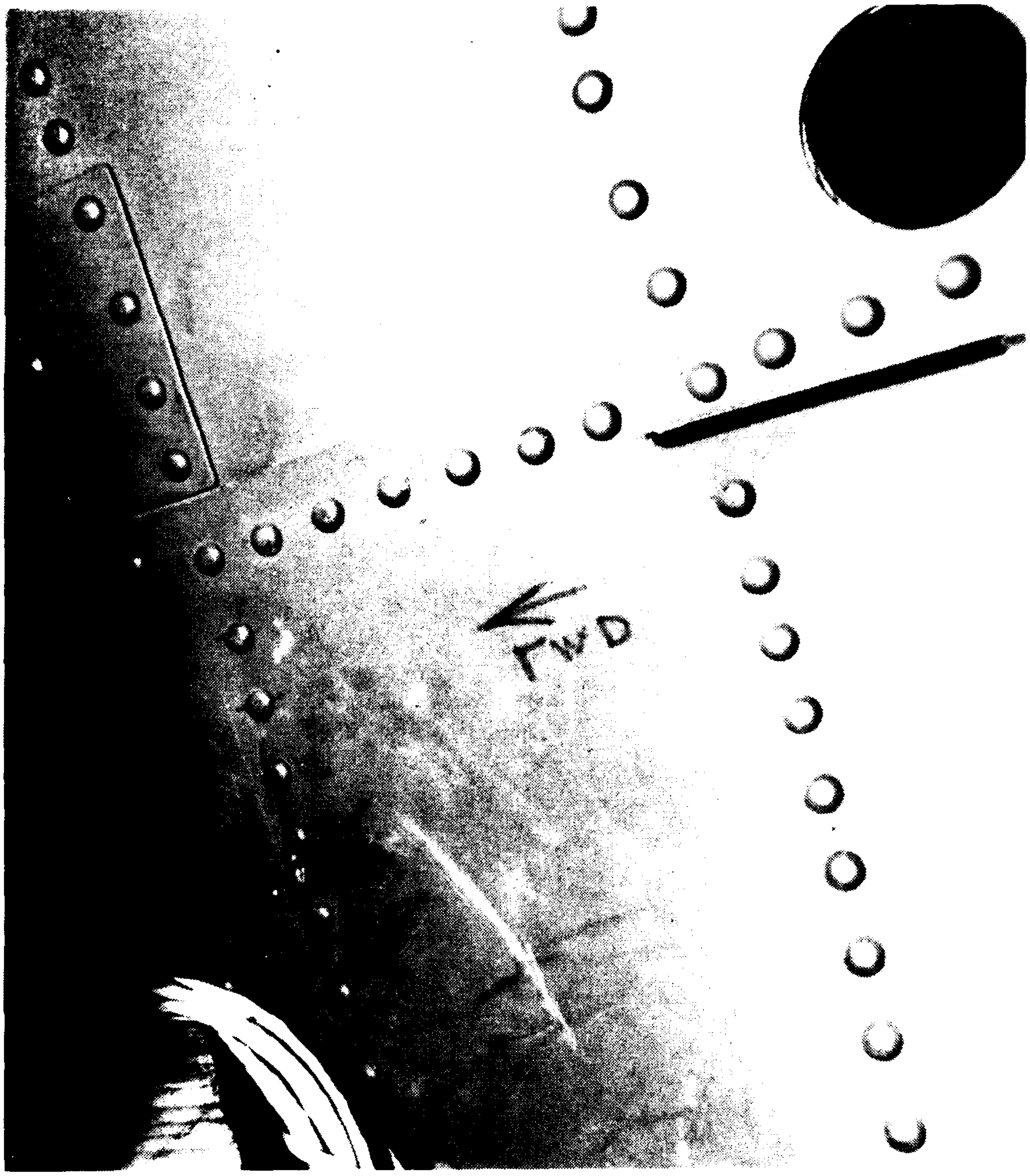


Figure 34. Crack in Fire Wall FS-318

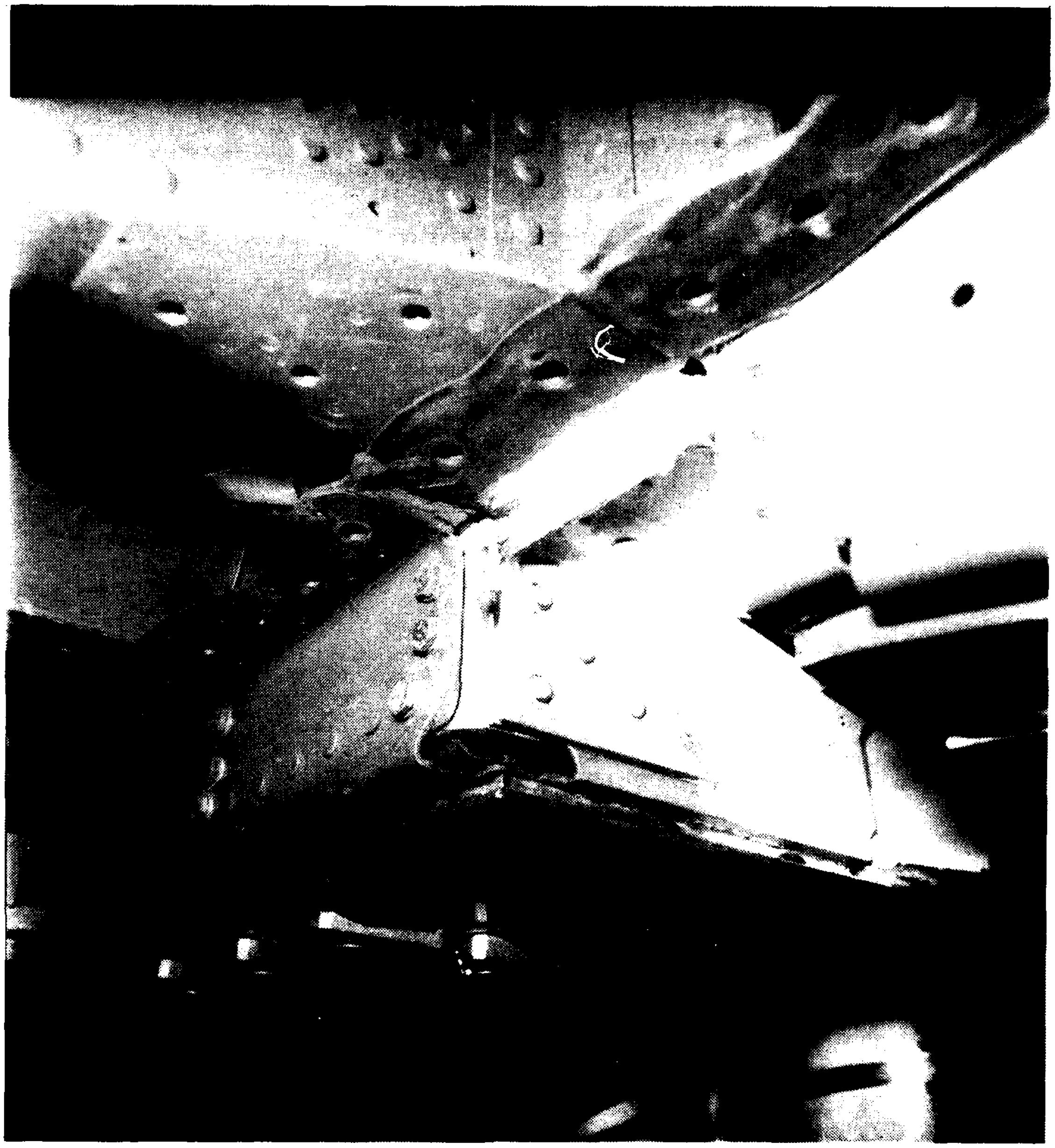


Figure 35. Crack in Speed Brake Tunnels

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
8000	Clip at F.S. 332, B.L. 47; Stringer No. 6 Assembly, Left and Right Side (53-321154-21 & 22). These cracks found at 6000 hours did not lengthen.	No Action Taken.
* 8000	Aft Fastener Missing Tying Dogbone Fitting (32-32086) to Fire Wall Shroud (32-32081) (Figure 36).	Replaced Fastener.
8000	At F.S. 249, One Fastener was Missing on Each Side Mating 32-11010 Front Spar to 53-31075 and 53-31076 Left and Right Keel Webs. (Figure 37).	No Action Taken.
8000	Crack indication in 53-11031 Aft Hinge Support for Main Landing Gear Inboard Door on Fuselage. Not as definitive as When Witnessed at 6,000 Hours.	No Action Taken.
8000	Crack in Duct Seal at F.S. 303 Just Above Wing.	No Action Taken.
9000	Right Speed Brake Inclosure Cracked (Figure 38).	Replaced
10,000	Engine Fire Wall Shroud had Crack Extending 3/8" Past Previously Stop Drilled Hole at F.S. 332.50 (32-32081-766).	Stop Drilled With 1/2" Hole.
10,000	Second Crack in Duct Seal at F.S. 303 Just Above Wing Results in Disconnected Piece of Material (53-32016).	No Action Taken.
*10,000	Twelve Rivets Missing From 53-32512-36 Splice Angle (Figure 39).	Replaced with Bolts.
10,000	Crack in Fire Wall Shroud (32-321081) Extending Horizontally Through Five Fasteners. This Crack Connected with Previously Noted Crack Through Vertical Row of Five Fasteners at F.S. 318.	Stop Drilled Cracks and Add. 063 Aluminum Doubler.
10,000	Channel Gusset (53-321154-14) Cracked in Bend Radius.	No Action Taken.

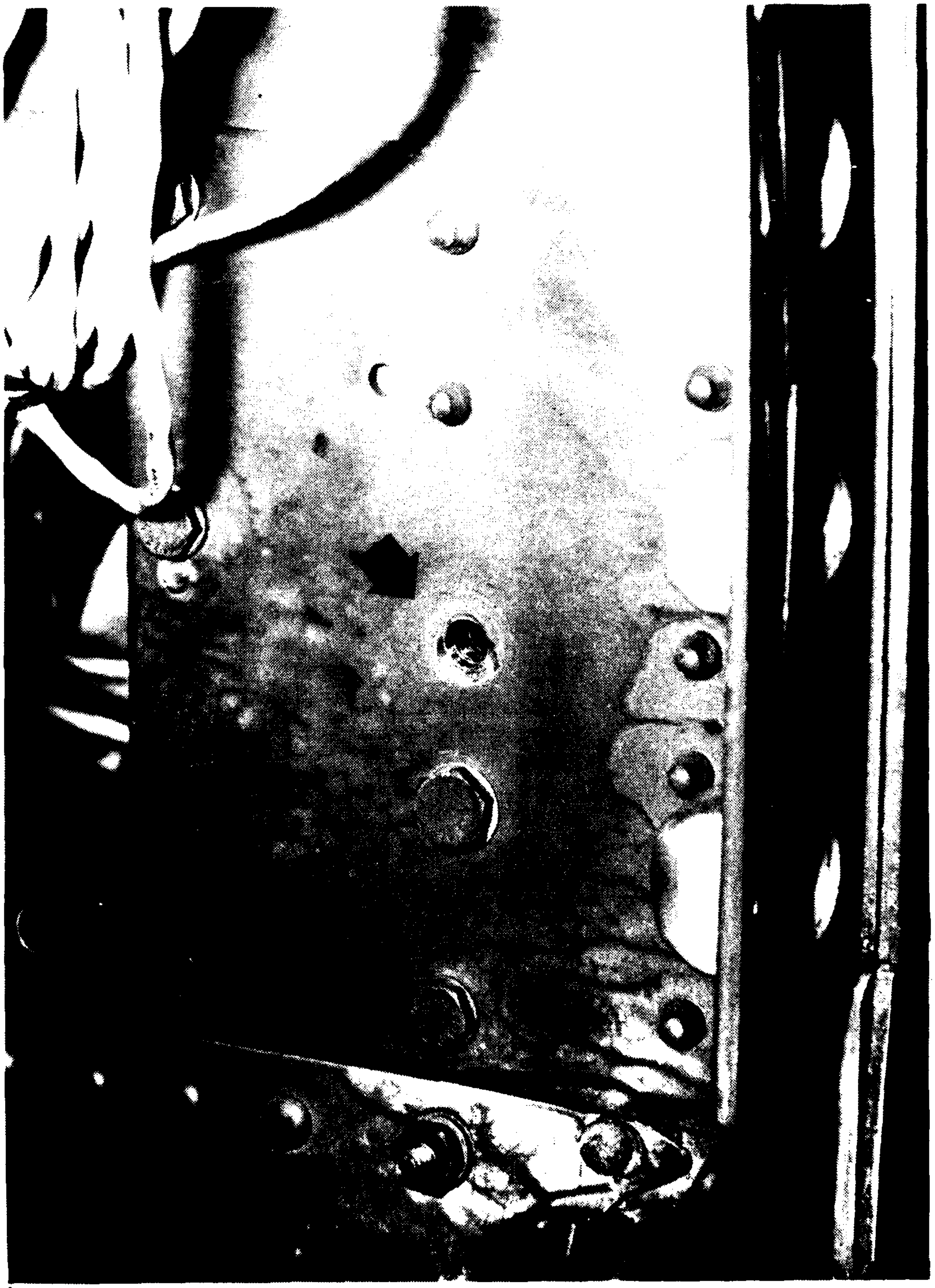


Figure 36. Failure of Fastener Tying Dogbone to Fuselage Vertical Wall

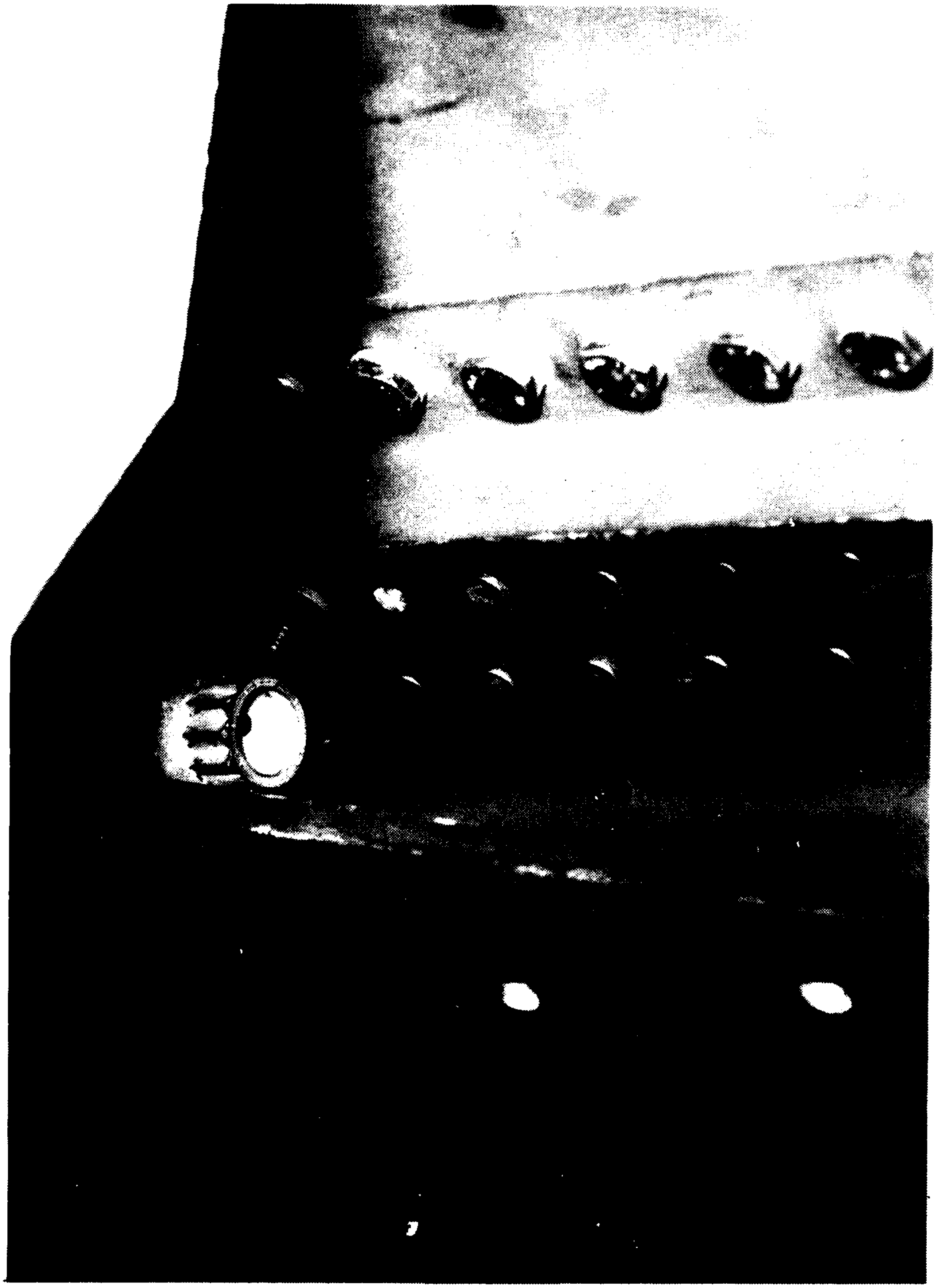


Figure 37. FS-249 Fastener Missing

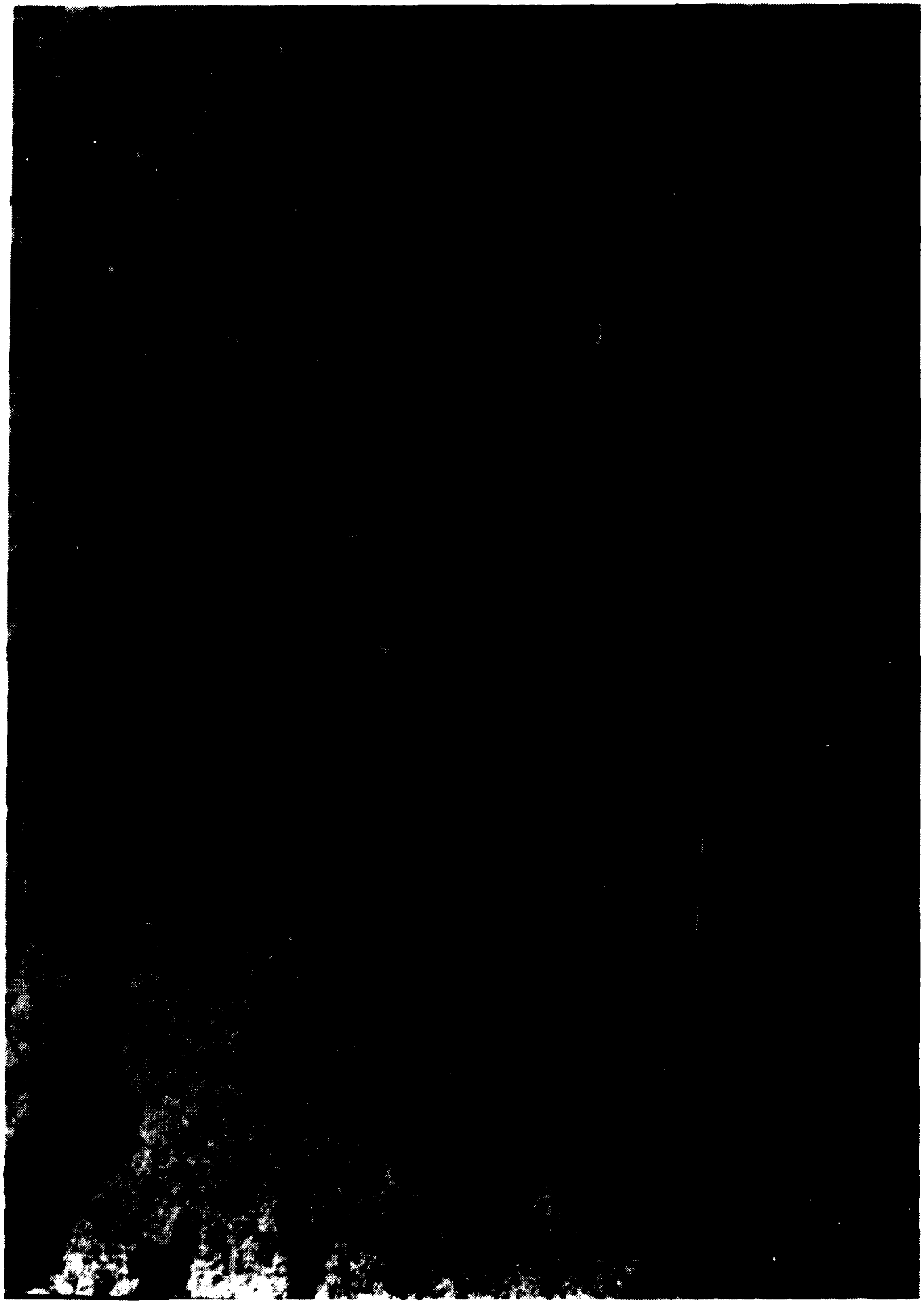


Figure 38. Right Speed Brake Enclosure Cracked

Figure 39. Rivets Missing from Splice Angle

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
* 12,000	Cracks in L/H & R/H Inboard Trailing Edge Flap Up-Stop Fittings (32-32506-301-302).	Replaced.
* 12,000	Three High Shear Rivets Missing Tying 53-32512-36 Splice Angle to Hydraulic Access Door Sill Forward of F.S. 332.50 on R/H Side of Aircraft.	Replaced with Bolts.
* 12,000	One High Shear Rivet Missing from 53-321154-19 Clip Tying 53-32512 Hinge to 53-321154-5 Tee.	Replaced
* 13,570	Four Fasteners Failed Which Tied the Vertical Flange of the 53-32512-36 R/H Splice Angle to the Hydraulic Access Door Sill Forward of F.S. 332.50.	Replaced with Bolts.
* 13,818	F.S. 303 Bulkhead Cracked on Both R/H and L/H Sides of the Airplane in a .080 Thick Flange Running Along B.L. 44. Cracks were at the Point of Intersection Between this Flange and a Horizontal Flange at W.L. 22.	Cracks Stop Drilled.
* 13,818	32-32055 Forward Engine Mount Support Fitting Cracked on Both L/H and R/H Sides of Aircraft. This Fitting Ties to Aft Side of F.S. 318 Bulkhead (Figure 40).	Replaced with Titanium Fitting as Used on Navy Aircraft.
* 13,818	Cracks in 32-32027-415/-416 Frames on L/H and R/H Sides of F.S. 318 Bulkhead.	Repaired per Repair Drawing Furnished by MCAIR.
* 13,818	Two Cracks in 32-320270217 Beam which Traverses the Aircraft Between the L/H and R-H Forward Upper Engine Mount Support Fittings (32-32055).	Replaced
* 13,818	Crack Propagating from Fastener Hole in 32-32123-465 Angle Mating R/H Side Panel and Floor of # 3 Fuel Tank. (Figure 41).	Replaced Angle.
13,818	Crack in -143 Former that is Part of 32-32006-137 L/H Frame Assembly.	No Action Taken.

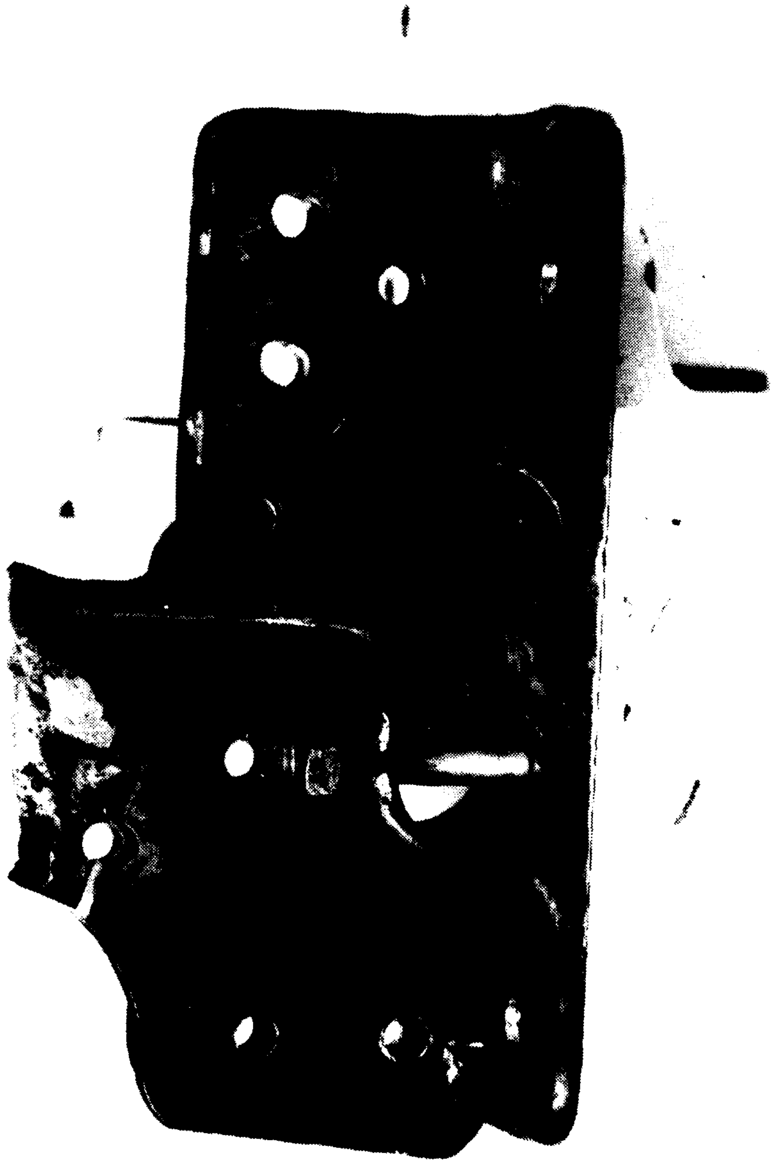


Figure 40. Forward Engine Mount Cracked

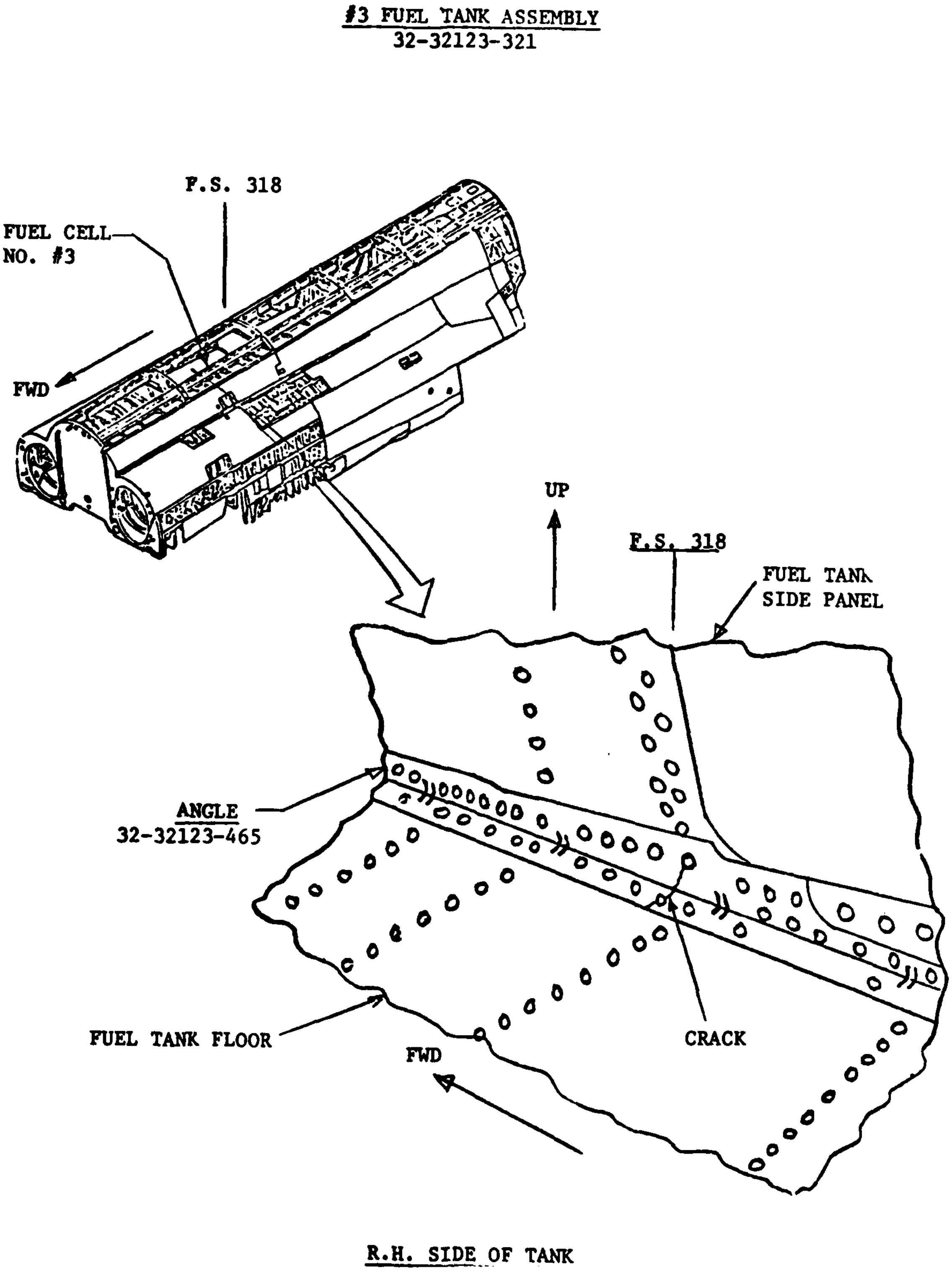
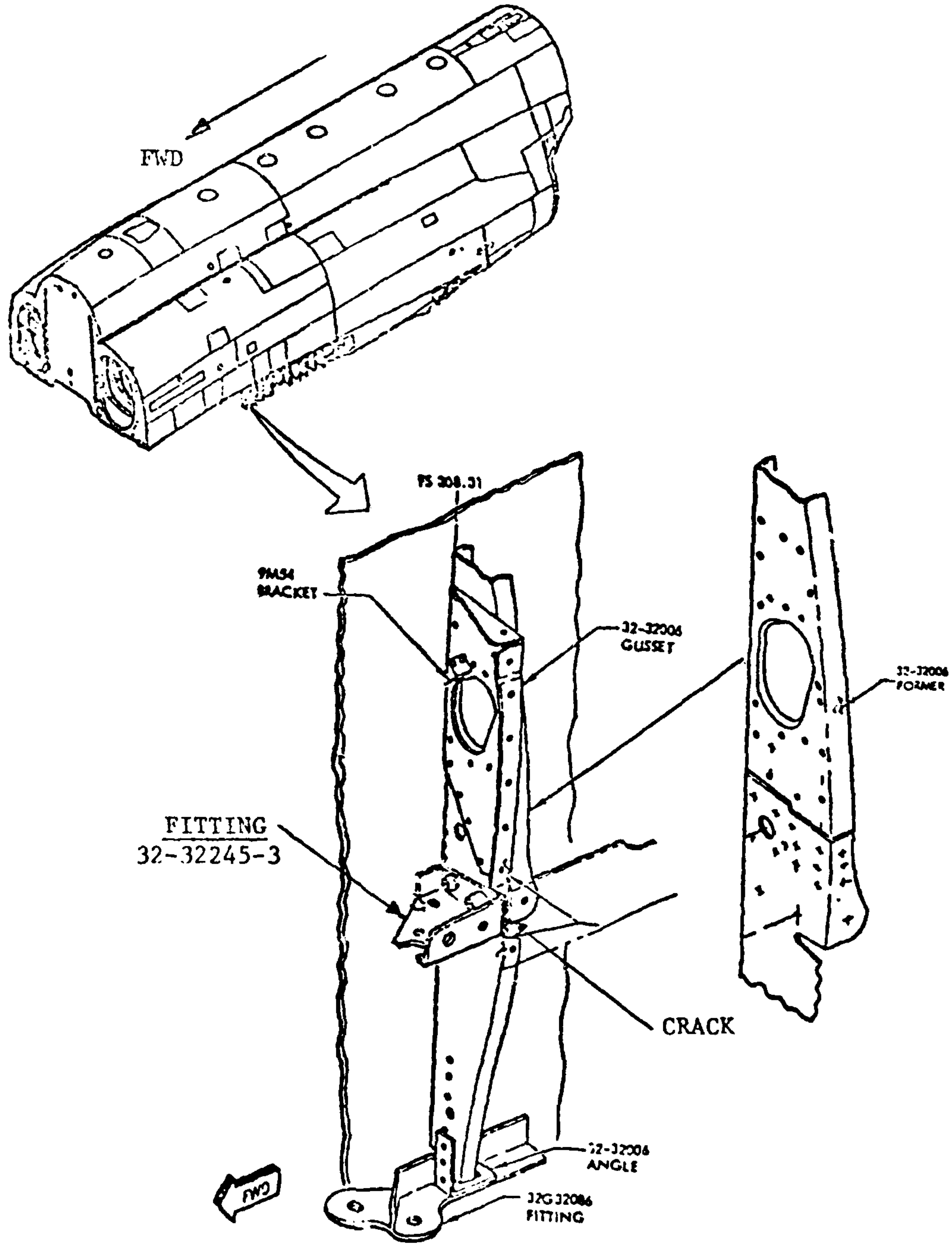


Figure 41. #3 Fuel Tank Cracked

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
13,818	Crack in 32-32245-5 Fitting Tying Wing Pad to F.S. 308.31 Frame. (Figure 42).	Replaced
13,818	Fastener Missing Tying Front Spar to the B.L. 34 Shear Web at F.S. 249.65 on R/H Side of Aircraft.	Fastener Replaced.
13,818	32-50025 Eyebolt Tying Gate to Body of the 32-50043 Aft Inboard Engine Mount. (Figure 43).	Replaced
13,818	One of the Four Bolts Tying 32-50014-301 Forward Engine Mount Pad to the R/H Engine.	Replaced
*13,818	Cracks in 32-11215-16/-18 Tunnel Angles and in 32-11212-84 Aft Closure Angle. (Figure 44).	Replaced 32-11212-84 Angle. No Action Taken on 32-11215-16/-18 Angles.
13,818	Crack in Fire Wall Shroud Found at 10,000 Hours had Propagation Past Stop Drill Hole.	Stop Drilled with 3/8" Hole.
*13,818	Hydraulic Access Door Hinge Half Tied to Fuselage Cracked.	Replaced
*13,818	Fixed Bellmouth Rings Cracked on Both Sides of Aircraft.	Replaced
*13,818	32-32027-123 Strap Installed on Left Side of F.S. 318 Frame Cracked.	Replaced as Part of Engine Mount Back-Up Structure Retrofit Kit Installation.
*13,818	Cracks in 32-32175-449/-450 Intercostal Assemblies Forward of F.S. 318 (Figure 45).	Replaced as Part of Engine Mount Back-Up Structure Retrofit Kit Installation.
*13,818	Cracks in 32-32175-213/-214 Intercostal Assemblies on Both Sides of the Aircraft Between F.S. 318 and F.S. 326.93 (Figure 46).	Replaced
13,818	Crack Near Top of 53-32016-134 Duct Seal on R/H Side of Aircraft. (Figure 47).	No Action Taken.

FITTING - WING PAD TO F.S. 308.31 ATTACH
32-32245-5

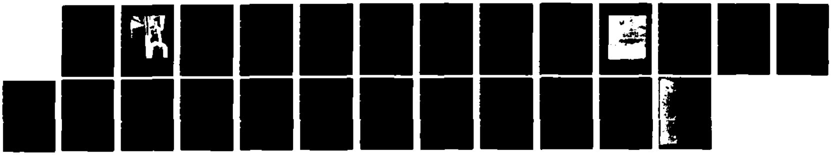


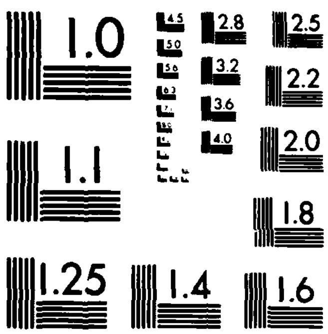
MATERIAL: 7075-T73 ALUMINUM PRESSING
THICKNESS: .10 INCH

Figure 42. Wing Pad to FS-308 Frame Cracked

AD-A127 495   F-4C/D LIFE EXTENSION PROGRAM(U) AIR FORCE WRIGHT   2/2
AERONAUTICAL LABS WRIGHT-PATTERSON AFB OH
R L SCHNEIDER OCT 82 AFWAL-TR-82-3047

UNCLASSIFIED   F/G 1/3   NL

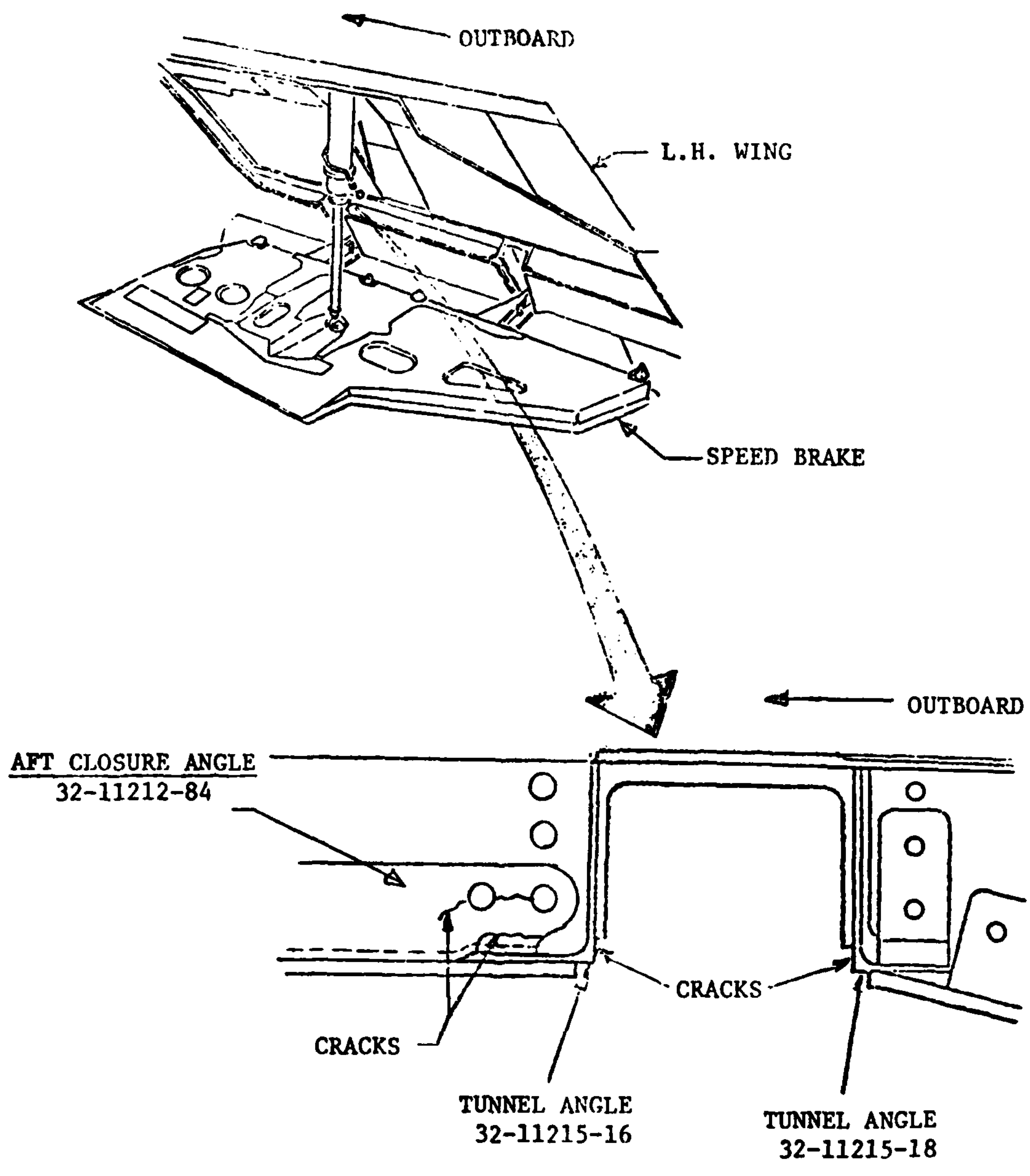




MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

Figure 43. Eye Bolt Failed Aft Engine

OUTBOARD SPEED BRAKE TUNNEL ASSEMBLY
32-11215-52



NOTE: LEFT HAND ASSEMBLY IS PICTURED.
HOWEVER, DASH NUMBERS AS INDICATED
CORRESPOND TO THE RIGHT HAND ASSEMBLY
ON WHICH THE FAILURES OCCURRED

Figure 44. Speed Brake Tunnel Angles Cracked

INTERCOSTAL ASSEMBLY, F.S. 312.90 TO F.S. 318
32-32175-449, -450

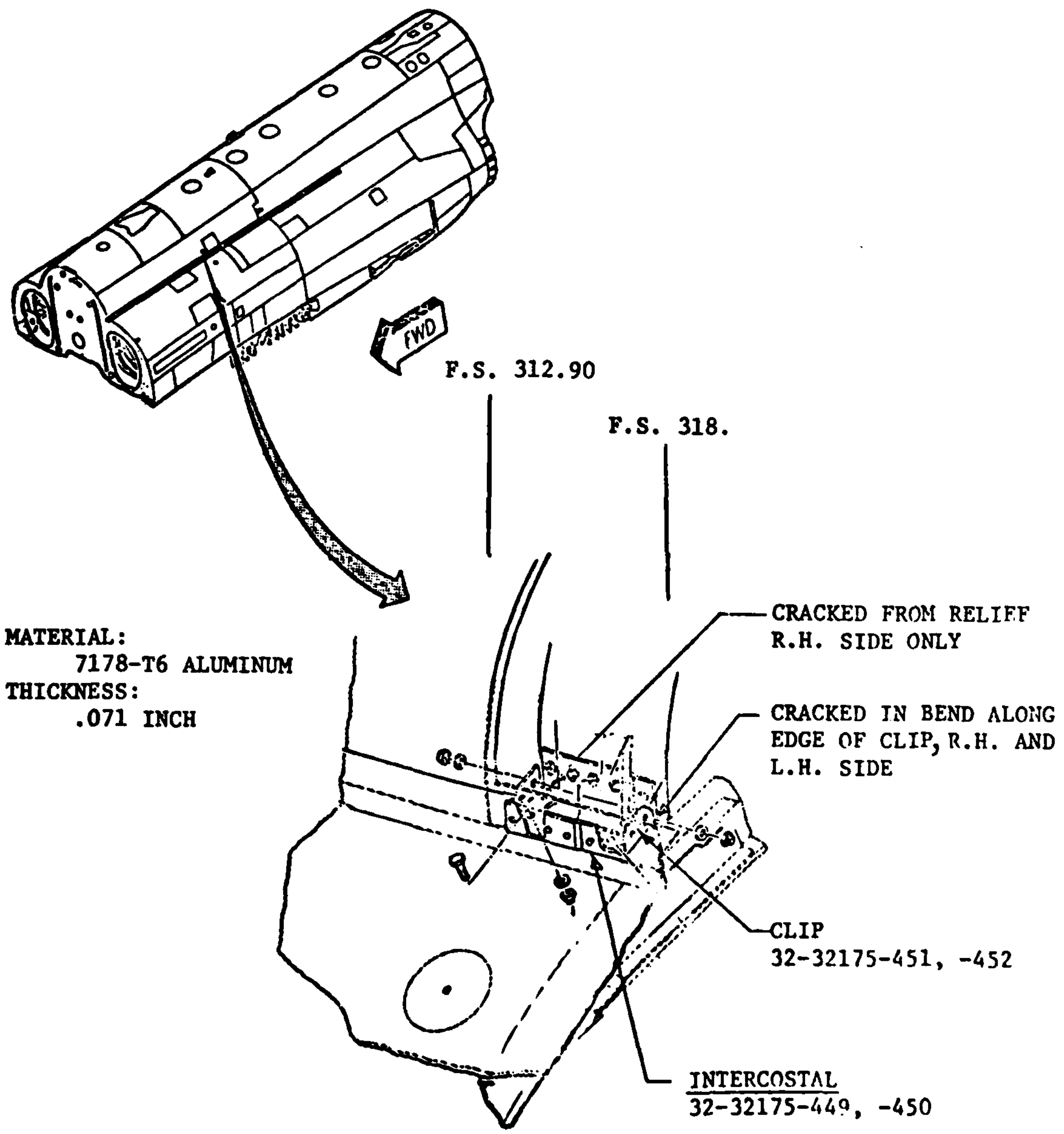


Figure 45. Intercoastal FS-318 Cracked

INTERCOSTAL ASSEMBLY, F.S. 318 TO F.S. 326.93
32-32175-213, -214

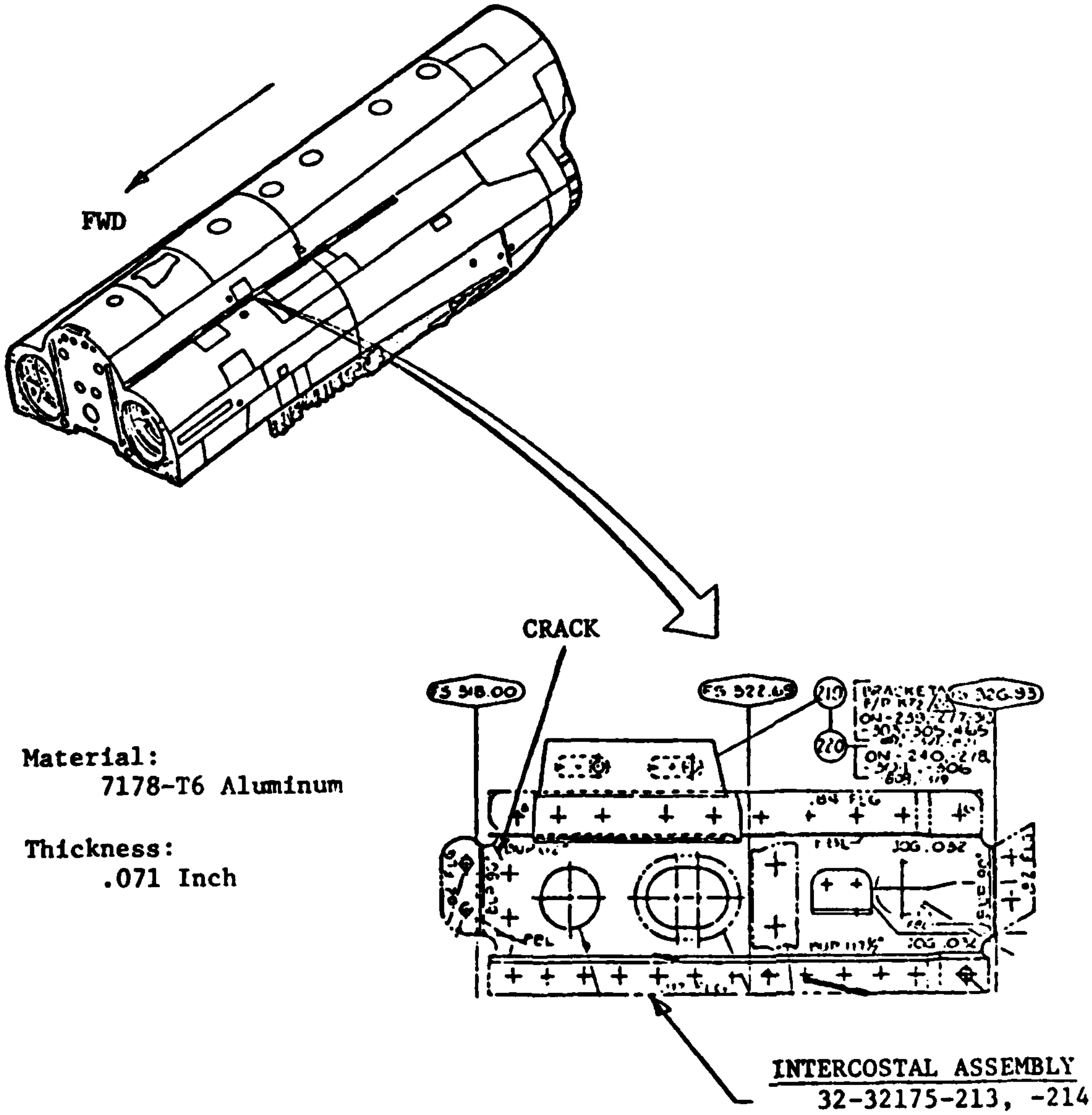
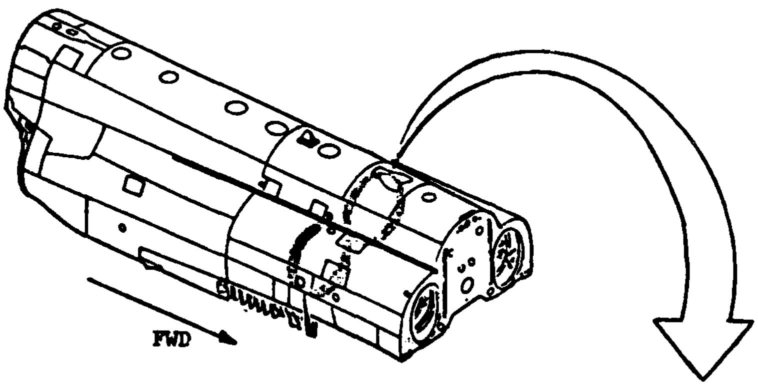
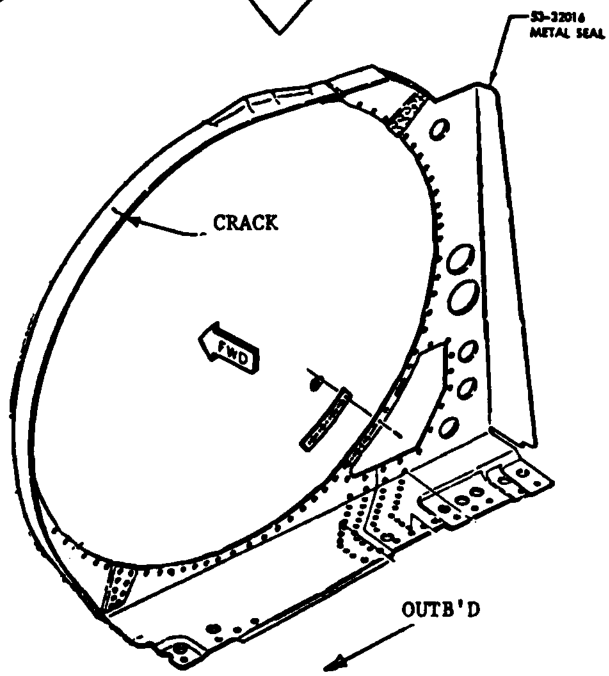


Figure 46. Intercoastal between FS-318 & 326.9 Cracked

DUCT SEAL AT F.S. 303
53-32016-134



NOTE: The seal as shown in this figure is on the left side of the airplane. The seal found to be cracked on the T.D. 89 aircraft is on the right side.

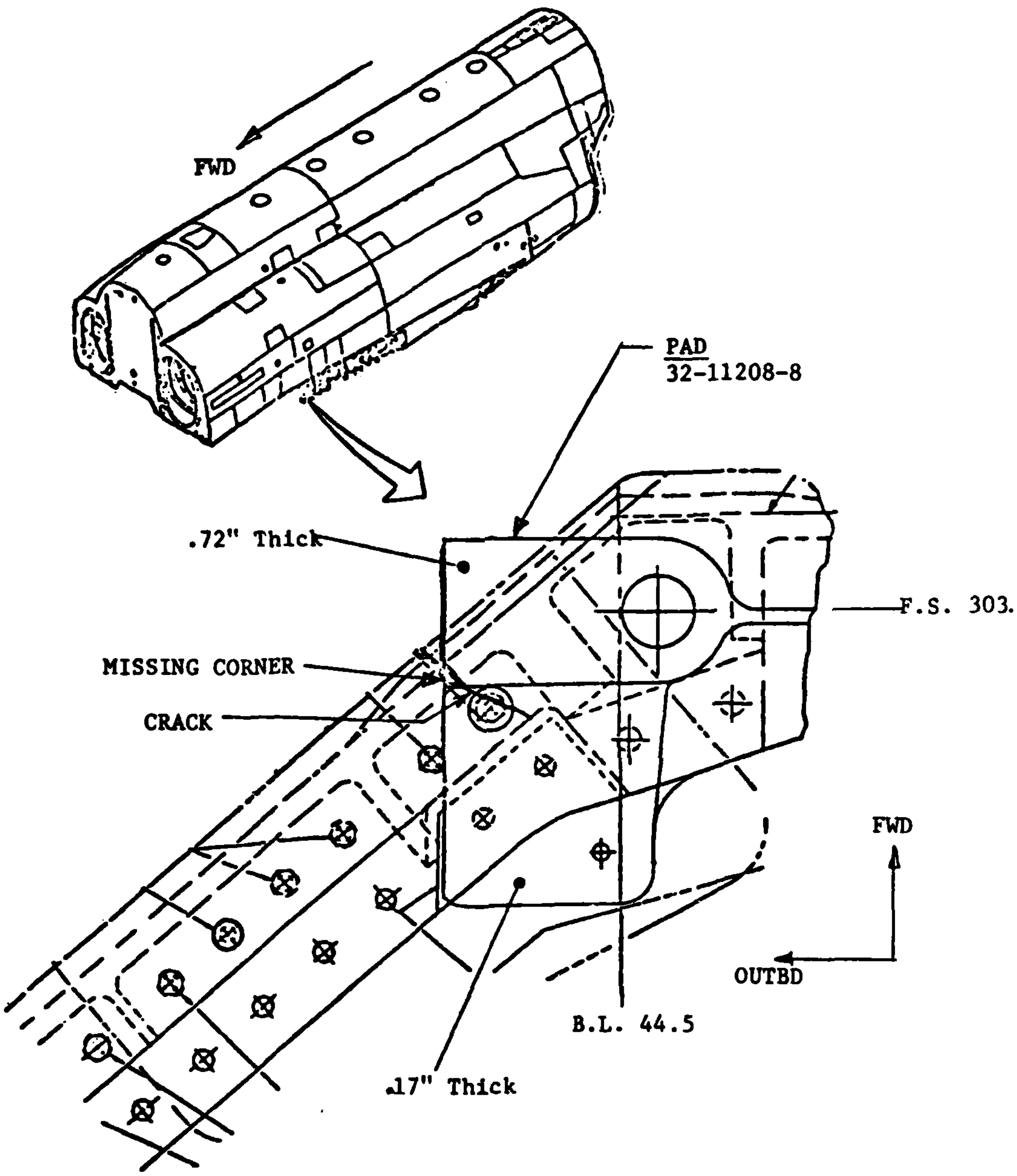


MATERIAL

2024-T3
Thickness = .040 In.

Figure 47. Duct Seal Cracked

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
13,818	Four Fasteners Missing and One Loose in the 53-32027-139 Lower Moldline Skin Panel Between F.S. 313 and F.S. 322.	Fasteners Replaced.
* 13,818	Cracks in Vertical Stiffeners Located on R/H and L/H Sides of F.S. 249.65 Bulkhead. Cracks at Approximately B.L. 10.5 and W.L. 71.	Repair Performed per 53-07997 Repair Drawings Furnished by MCAIR.
13,818	Crack in 32-11208-8 Pad Between 53-32016 F.S. 303.62 Bulkhead and 32-11401 Wing Torque Box Assembly (Figure 48).	No Action Taken
* 16,000	Crack in 32-32506-301 L/H Inboard Trailing Edge Flap Up-Stop Fitting.	Replaced
16,000	Crack in 32-32006-144 Former on R/H Side of Airplane.	Stop Drilled with 1/4" Diam. Hole.
16,000	Two Cracks in the 32-32051-26 R/H Side Panel Assembly of the # 2 Fuel Tank.	Stop Drilled with 3/8" Diam. Holes.
* 16,000	Nine High Shear Rivets Failed Tying 53-32512-36 Splice Angle to the Hydraulic Access Door Sill Forward of F.S. 332.50 and to Engine Access Door Hinge Aft of F.S. 332.50.	Replaced with Bolts.
16,000	Crack in Outboard Firewall Shroud, R/H Side, Between Two Fasteners in Vertical Row at Approximately F.S. 326.	No Action Taken.
16,000	Crack Propagated from One Fastener Hole in R/H Fold Rib of Center Wing and Cracks Propagated from Four Holes in L/H Fold Rib.	No Action Taken.
16,000	Crack in 32-11024-13 Stiffener on Centerline Rib.	Stop Drilled with 3/8" Diam. Hole.
16,000	Three Fasteners Failed in a Vertical Column of Fasteners in the 35% Beam Near the Juncture of the Beam with the Torque Rib.	Replaced with Jo-Bolts.



NOTE:
Assembly shown is on the left side of aircraft. However, failed pad is on right side.

Figure 48. Crack in Pad between Torque Box and 303 Bulkhead

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
16,000	Crack Inboard Flange of the 32-32046-405 Channel which is Part of the 32-32042-2339 Keel Web Assembly.	No Action Taken.
16,000	Two Fasteners Failed which Tie the 53-32512-87 Web to the 53-32514 Hydraulic Access Door Sill on Each Side of the Aircraft.	Fasteners Replaced.
16,000	Crack in 32-32175-137 Intercostal which is Part of 32-32175-821 Stringer No. 3 Assembly.	Stop Drilled with 3/8" Diam. Hole.
* 16,271	Centerline Splice Plate Cracked thru Eleven Fasteners in the Outboard Row on the left side of the Airplane just Forward of the Main Spar Figure 49.	Repair Doubler (53-010156) was Installed.
16,700	Crack in Bend Radius of 32-31875-1151/-1152 Angles Tying Left and Right Keel Webs to F.S. 249.65 Bulkhead.	No Action Taken.
17,650	One Inch Long Crack in Upper Flange of 32-31157 R/H Keel Web Truss Extended Forward from Front of F.S. 249.65 Bulkhead.	No Action Taken.
* 18,000	Inboard Trailing Edge Flap Up-stop Fitting (32-32506-301) Failed on L/H Side of Aircraft.	Replaced
18,000	Two Fasteners Failed and Two Loose in Vertical Double Row Tying Keel Web Beam to Front Spar.	Replaced
* 18,000	Crack in # 3 Fuel Tank L/H Side Panel Progressing Forward from Vertical Row of Fasteners at F. 318.	Doubler Added.
18,000	Crack Between Fastener Holes in 32-32016-14 Angle Attached to F.S. 30.62 Bulkhead.	No Action Taken.
* 20,000	One Fastener Failed Tying Upper Edge of the 35% Beam's Vertical Wb to the Beam's Cap. This was Second Fastener Inboard of B.L. 44 Torque Rib on R/H Side of the Airplane.	Fastener Replaced.

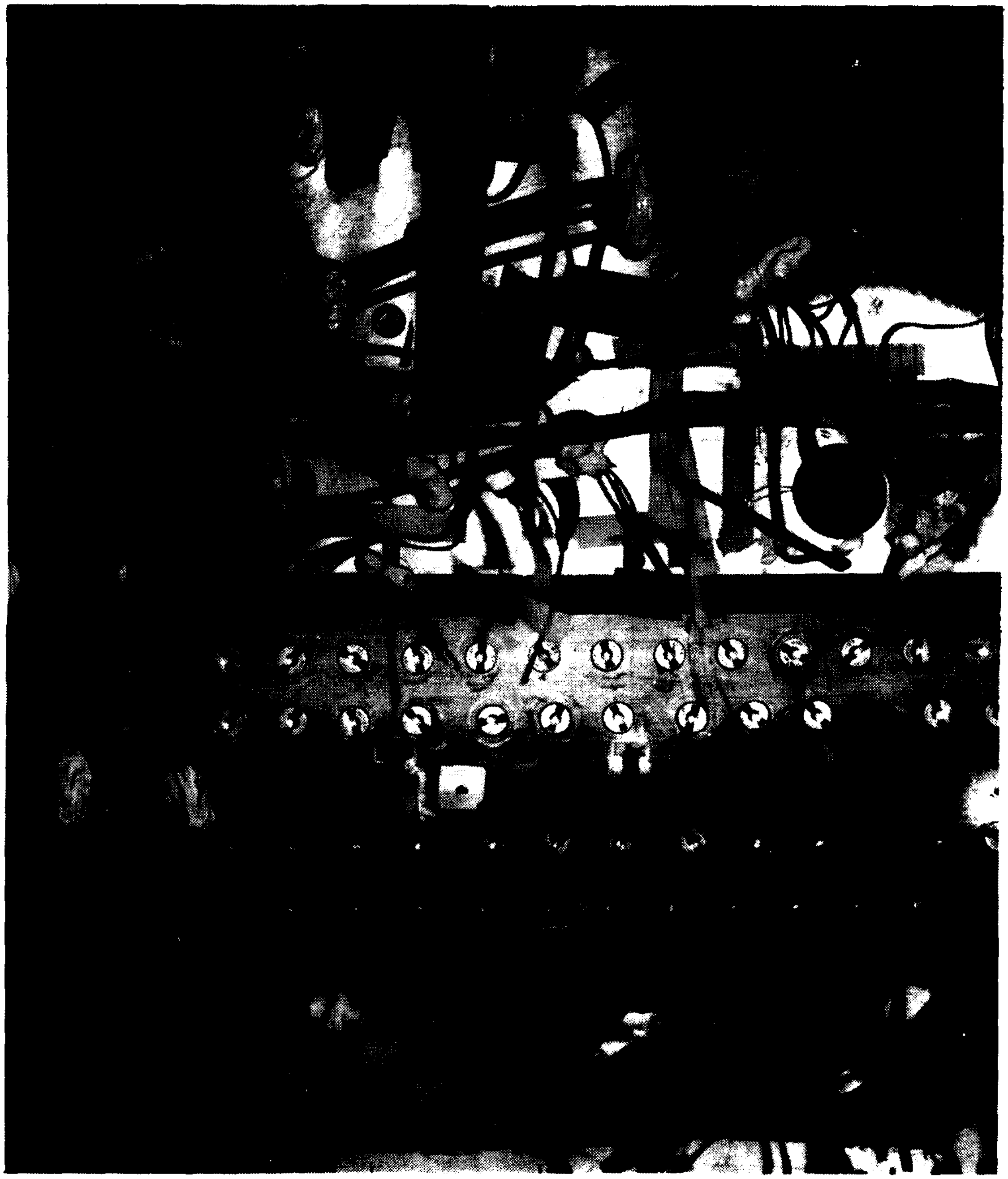


Figure 49. Center Splice Plate Cracked

SPECTRUM HOURS	DESCRIPTION OF FAILURES	DISPOSITION
* 20,000	Two Rivets Failed Tying 53-32512-36 Splice Angle to the Engine Access Door Hinge Assembly Aft of F.S. 332.50.	Replaced with Bolts.
* 20,000	Two Fasteners Failed and Two Loose in Vertical Double Row of Fasteners Tying Keel Web to 32-11010 Front Spar.	Replaced with Bolts.
* 20,000	Four Fasteners at the Upper End of the 32-31875-1152 Angle Failed. This Angle Ties R/H Keel Web to F.S. 249.65 Bulkhead.	Fasteners Replaced.
20,000	32-32081-1168 Channel which Acts as an Intercostal Between F.S. 312.40 and F.S. 318 Frames Failed Between Two Fasteners Near F.S. 318 End.	No Action Taken.
20,000	Crack Propagated from .56 Diam. Hole in Web of 32-32006-144 Former at F.S. 308.31 on R/H Side of Airplane.	Stop Drilled with 1/4" Diam. Hole.
21,970	Cracks in the Centerline Splice Plate Propagated Both Fore and Aft from the 3rd, 7th, and 9th Holes Forward of the Doubler Added at 16,271 Hours,	No Action Taken.
22,225	Cracks Propagated in the Centerline Splice Plate Both Fore and Aft from the 4th and 5th Holes Forward of the Doubler Added at 16,271 Hours.	No Action Taken.
23,140	A Crack in the Centerline Splice Plate Propagating both Fore and Aft from the 13th Hole Forward of the Doubler Added at 16,271 Hours.	No Action Taken.

APPENDIX B

INDEX OF DATA RECORDED

INDEX OF DATA RECORDED

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
574/0	237	4.0 Strain Survey (S.S.)1	Sampled While Holding (SWH)	1000
881/0	143	+5.5 (S.S.1)	SWH	1001
574/0	237	+4.0 (S.S.2)	SWH	1002
881/0	143	+5.5 (S.S.2)	SWH	1002
2210/0	335	+4.0 (S.S.2)	SWH	1003
7880/0	237	-1.5 (S.S.2)	SWH	1004
12370/0	335	-2.0 (S.S.2)	SWH	1005
33/0	299	Landing (S.S.2)	SWH	1005
7880/0	237	-1.5 (S.S.1)	SWH	1006
12370/0	335	-2.0 (S.S.1)	SWH	1006
33/0	299	Landing (S.S.1)	SWH	1006
2210/0	335	+4.0 (S.S.1)	SWH	1006
238/1	265	+6.5	SWH After Dump	1008
3158/1	-	--	SWH After Dump	1010
5652/1	299	Landing	SWH After Cycle	1012
8866/1	133	4.5	SWH	1014
20605/1	557	+5.0 (S.S.2)	SWH	1022
20605/1	557	Linkage Failure (S.S.2)	Continuous Record Thru Peak (CRTP)	1022
20605/2	557	+7.5 (S.S.2)	Baseline at Peak	1030
2814/3	557	+6.5	100 Point File	1044
4721/3	647	+4.0 (S.S.2)	SWH	1045
4721/3	647	+7.0 (S.S.2)	CRTP	1045
13141/3	566	+7.0 (Random)	100 Point File	1052
14897/3	665	+7.0 (Random)	100 Point File	1053
18087/3	557	+7.0 (Random)	100 Point File	1054

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER.NO.
18822/3	143	+5.0	CRTP	1055
10333/4	237	+4.0	Continuous Record Thru Cycle (CRTC)	1060
24970/4	656	+5.5 (Random)	100 Point File	1062
45/5	236	+4	CRTC	4000
134/5	237	+5	CRTP	4000
163/5	237	+6.5/0	CRTC	4000
1361/5	143	+5/+1	CRTC	1063
2210/5	335	+4/+1 (S.S.)	CRTC	1064
7880/5	237	+3/-1.5 (S.S.)	CRTC	1066
8027/5	557	+6/+1 (Random)	CRTC	1066
12344/5	665	+5.5/+1 (Random)	CRTC	1067
12370/5	335	+2.5/-2 (S.S.)	CRTC	1067
12587/5	299	Landing (S.S.)	CRTC	1068
14659/5	655	+6.0/+1 (Random)	CRTP	1069
19209/5	143	+5/+1	SWH	1070
20097/5	237	+3/0	SWH	1070
20104/5	237	+4.5/-0.5	SWH	1070
20265/5	576	+6.5 (Random)	CRTC	1070
20407/5	237	+5/0	SWH	1070
20603/5	566	+6.5 (Random)	CRTC	1070
20605/5	557	+7.5 (S.S.)(Random)	CRTP	1070
21374/5	143	+5.5/0	CRTP	1071
21590/5	143	+5.5/+1	CkTP	1071
23238/5	-	Ship Floating	SWH	1071

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
23239/5	-	Ship Floating	SWH	1072
23239/5	-	Ship Floating (No Gear Load)	SWH	1072
23557/5	143	+5.5/+1 (S.S.)	SWH 1G Steps	1072
574/6	237	+4/+1 (S.S.)	SWH 0.5G Steps	1073
881/6	143	+5.5/+1 (S.S.)	SWH	1073
1483/6	-	Holding 1G	SWH	1073
5058/6	-	-	F/S	-
6642/6	-	Holding 1G	SWH	1074
6786/6	143	+5.5/+1	SWH	1074
9045/6	-	Holding 1G	SWH	1074
19901/6	345	+6.5/0	100 Point File	1075
20603/6	566	+6.5 G (Random)	CRTC	1076
20605/6	557	+7.5 G (S.S.)(Random)	CRTC	1076
21374/6	143	+5.5/0	CRTP	1077
21590/6	143	+5.5/+1	CRTP	1077
23557/6	143	+5.5/+1	CRTC	1077
574/7	237	+4/+1 (S.S.)	CRTC	1078
2210/7	335	+4/+1 (S.S.)	SWH	1079
3854/7	143	+5.5/+1	CRTP	1079
7992/7	226	+5/+1	CRTC	1080
8016/7	256	+3/+1	CRTC	1080
8027/7	557	+6 (Random)	CRTP	1080
8358/7	143	+5/+1	CRTC	1080
15297/7	256	+7/+1	CRTC	1081(Data N/A)
15328/7	247	+7/0	100 Point File	1081
20097/7	237	+3/0	SWH	1082

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
20104/7	237	+4.5/-0.5	SWH	1082
20407/7	237	+5/0	SWH	1082
20603/7	566	+6.5 (Random)	CRTC	1082
20605/7	557	+7.5 (Random)	CRTC	1082
321/8	566	+6.5 (Random)	CRTP	1083
330/8	575	+6.5 (Random)	CRTP	1083
574/8	237	+4/+1 (S.S.)	CRTC	1083
881/8	143	+5.5/+1 (S.S.)	CRTC	1083
2210/8	335	+4/+1 (S.S.)	CRTC	1084
2814/8	557	+6.5 (Random)	CRTP	1084
4721/8	647	+7 (Random)	CRTC	1084
4791/8	335	+3.5/+1	CRTC	1084
4792/8	655	+6.5 (Random)	CRTC	1084
18087/8	574	+7 (Random)	CRTP	1085
20097/8	237	+3/0	SWH	1085
20104/8	237	+4.5/-0.5	SWH	1085
20407/8	237	+5/0	SWH	1085
20502/8	566	+7 (Random)	CRTC	1085
20504/8	557	+7.5 (Random)	CRTP	1085
326/9	557	+6.5 (Random)	CRTP	1086
574/9	237	+4/+1 (S.S.)	CRTC	1086
881/9	143	+5.5/+1 (S.S.)	CRTC	1086
1096/9	465	+5.5 (Random)	CRTP	1086
1146/9	463	+7 (Random)	CRTP	1086
1704/9	473	+7.5 (Random)	CRTP	1086

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
2210/9	335	+4/+1 (S.S.)	CRTC	1086
2350/9	664	+6.5 (Random)	CRTP	1086
3506/9	473	+6.5 (Random)	CRTP	1086
3618/9	463	+7.5 (Random)	CRTP	1086
4667/9	655	+7 (Random)	CRTP	1087
4714/9	647	+5.5 (Random)	CRTP	1087
5002/9	647	+6.5 (Random)	CRTP	1087
5621/9	573	+7.5 (Random)	CRTP	1087
6565/9	474	+6 (Random)	CRTP	1087
7262/9	657	+5 (Random)	CRTP	1087
7319/9	664	+7.5 (Random)	CRTP	1087
7436/9	656	+6 (Random)	CRTP	1087
7545/9	663	+6.5 (Random)	CRTP	1087
8361/9	447	+5 (Random)	CRTP	1087
10044/9	655	+7 (Random)	CRTP	1087
10659/9	237	+4	SWH	1088
11584/9	143	+5.5	CRTP	1088
13064/9	573	+8 (Random)	CRTP	1088
14968/9	664	+7 (Random)	CRTP	1088
14989/9	647	+6 (Random)	CRTP	1088
15075/9	575	+8 (Random)	From Peak Downward	1088
15075/9	575	+8 (Random)	100 Point File	1088
16434/9	473	+6 (Random)	CRTP	1088
17648/9	573	+7 (Random)	CRTP	1089
18089/9	585	+7 (Random)	CRTP	1089
18666/9	465	+6 (Random)	CRTP	1089

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
18713/9	447	+5 (Random)	CRTP	1089
20495/9	575	+8 (Random)	CRTP	1089
21428/9	463	+7 (Random)	CRTP	1089
22183/9	656	+7 (Random)	CRTP	1089
22381/9	663	+7.5 (Random)	CRTP	1089
22478/9	684	+6 (Random)	CRTP	1089
22672/9	567	+6 (Random)	CRTP	1089
22738/9	574	+8 (Random)	CRTP	1089
23165/9	573	+8 (Random)	CRTP	1089
23352/9	456	+5 (Random)	CRTP	1089
24118/9	464	+6.5 (Random)	CRTP	1089
24876/9	657	+6 (Random)	CRTP	1089
24959/9	665	+6.5 (Random)	CRTP	1089
25031/9	663	+7 (Random)	CRTP	1089
574/10	237	+4/+1 (S.S.)	CRTC	1090 (Data N/A)
881/10	143	+5.5/+1 (S.S.)	CRTC	1090 (Data N/A)
2210/10	335	+4/+1 (S.S.)	CRTC	1090 (Data N/A)
20603/10	566	+6.5 (Random)	CRTC	1091
20605/10	557	+7.5 (Random)	CRTC	1091
168/11	255	+6.5/+1	CRTP	1092
574/11	237	+4/+1 (S.S.)	SWH	1093
881/11	143	+5.5/+1 (S.S.)	SWH to 4G; CRTP above 4G	1094
2210/11	335	+4/+1 (S.S.)	SWH	1094
20609/11	246	+6.5	CRTP	1095
574/12	237	+4/+1 (S.S.)	SWH	1096 (EPM)

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
881/12	143	+5.5/+1 (S.S.)	SWH to 4G; CRTP above 4G	1096 (EPM)
2210/12	335	+4/+1 (S.S.)	CRTC	1097 (EPM)
20603/12	566	+6.5 (Random)	CRTC	1098
20605/12	557	+7.5 (Random)	CRTC	1098
321/13	566	+6.5 (Random)	CRTP	1099
330/13	575	+6.5 (Random)	CRTP	1099
574/13	237	+4/+1 (S.S.)	CRTC	1099
881/13	143	+5.5/+1 (S.S.)	CRTC	1099
2210/13	335	+4/+1 (S.S.)	CRTC	1099
4721/13	647	+7 (Random)(S.S.)	CRTC	1100
4792/13	655	+6.5 (Random)	CRTP	1100
14897/13	665	+7 (Random)	CRTP	1101
17646/13	575	+7 (Random)	CRTP	1101
18087/13	574	+7 (Random)	CRTP	1102
20502/13	566	+7 (Random)	CRTP	1102
20504/13	557	+7.5 (Random)	CRTP	1102
22670/13	567	+6 (Random)	CRTP	1102
24970/13	656	+5.5 (Random)	CRTP	1102
321/14	566	+6.5 (Random)	CRTP	1103 (Data N/A)
330/14	575	+6.5 (Random)	CRTP	1103 "
574/14	237	+4/+1 (S.S.)	CRTC	1103 "
881/14	143	+5.5/+1 (S.S.)	CRTC	1103 "
4721/14	647	+7 (Random)(S.S.)	CRTC	1103 "
4792/14	655	+6.5 (Random)	CRTP	1103 "
8363/14	464	+6 (Random)	CRTP	1104
18087/14	574	+7 (Random)	CRTP	1105

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
20502/14	566	+7 (Random)	CRTP	1106
20504/14	557	+7.5 (Random)	CRTP	1106
22670/14	567	+6 (Random)	CRTP	1108 (Data N/A)
574/15	237	_4/+1 (S.S.)	CRTC	1109
881/15	143	+5.5/+1 (S.S.)	CRTC	1109
2210/15	335	+4/+1 (S.S.)	CRTC	1110 (Data N/A)
8027/15	557	+6.0 (Random)	CRTP	1111
20603/15	566	+6.5 (Random)	CRTC	1112
20605/15	557	+7.5 (Random)	CRTC	1112
12835/16	237	+6/0	CRTC	1113
20603/16	566	+6.5 (Random)	CRTC	1114
20605/16	557	+7.5 (Random)	CRTC	1114
574/17	237	+4/+1 (S.S.)	CRTC	1115
986/17	143	+6.5/+1	CRTC	1115
2210/17	335	+4/+1 (S.S.)	CRTC	1116
6856/17	143	+4.5	CRTP	1117
20603/17	566	+6.5 (Random)	CRTC	1118
20605/17	557	+7.5 (Random)	CRTC	1118
574/18	237	+4/+1	CRTC	1119
2330/18	335	+5	CRTP	1120
3381/18	143	+5.5	CRTC	1121
4721/18	647	+7 (Random)	CRTC	1121
4792/18	566	+6.5 (Random)	CRTP	1121
13141/18	566	+7 (Random)	CRTP	1122
13177/18	585	+6.5 (Random)	CRTP	1122
14897/18	655	+7 (Random)	CRTP	1123

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
17249/18	647	+5 (Random)	CRTP	1124
17646/18	575	+7 (Random)	CRTP	1124
18087/18	574	+7 (Random)	CRTP	1125
20502/18	566	+7 (Random)	CRTP	1125
20504/18	557	+7.5 (Random)	CRTP	1125
22670/18	567	+6 (Random)	CRTP	1126
24970/18	656	+5.5 (Random)	CRTP	1127
326/19	557	+6.5 (Random)	CRTP	1128
574/19	237	+4/+1 (S.S.)	CRTC	1128
881/19	143	+5.5/+1 (S.S.)	CRTC	1128
1703/19	143	+5.0	CRTC	1129
1704/19	473	+5.5 (Random)	CRTP	1129
3506/19	473	+6.5 (Random)	CRTP	1130
3618/19	463	+7.5 (Random)	CRTP	1130
4714/19	647	+5.5 (Random)	CRTP	1131
4828/19	674	+6 (Random)	CRTP	1131
5002/19	647	+6.5 (Random)	CRTP	1131
6565/19	474	+6 (Random)	CRTP	1131
7262/19	657	+5 (Random)	CRTP	1132
7319/19	664	+7.5 (Random)	CRTP	1132
7545/19	663	+6.5 (Random)	CRTP	1132
8025/19	585	+7 (Random)	CRTP	1132
8361/19	447	+5 (Random)	CRTP	1132
10044/19	655	+7 (Random)	CRTP	1133
12367/19	665	+7.5 (Random)	CRTP	1134
13064/19	573	+8 (Random)	CRTP	1134

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
15075/19	575	+8 (Random)	CRTP	1135
16434/19	473	+6 (Random)	CRTP	1135
17648/19	573	+7 (Random)	CRTP	1136
18089/19	585	+7 (Random)	CRTP	1136
18666/19	465	+6 (Random)	CRTP	1136
18713/19	447	+5 (Random)	CRTP	1136
20495/19	575	+8 (Random)	CRTP	1137
21428/19	463	+7 (Random)	CRTP	1137
22183/19	656	+7 (Random)	CRTP	1138
22381/19	663	+7.5 (Random)	CRTP	1138
22478/19	684	+6 (Random)	CRTP	1138
22672/19	567	+6 (Random)	CRTP	1138
22738/19	574	+8 (Random)	CRTP	1138
23165/19	573	+8 (Random)	CRTP	1138
23352/19	456	+5 (Random)	CRTP	1138
24876/19	657	+6 (Random)	CRTP	1139
24959/19	665	+6.5 (Random)	CRTP	1139
25031/19	663	+7 (Random)	CRTP	1139
574/20	237	+4/+1 (S.S.)	CRTC	1139
881/20	143	+5.5/+1 (S.S.)	CRTC	1139
20603/20	566	+6.5 (Random)	CRTC	1140
20605/20	557	+7.5 (Random)	CRTC	1140
21122/21	143	+5.5/+1	CRTC	1141
692/22	237	+5.0	CRTC	1142
881/22	143	+5.5/+1 (S.S.)	CRTC	1142
2210/22	335	+4/+1 (S.S.)	CRTC	1142

INDEX OF TEST DATA RECORDED (Continued)

CYCLE/BLOCK	MISSION	G LEVEL	DATA TYPE	SER. NO.
20603/22	566	+6.5 (Random)	CRTC	1143
20605/22	557	+7.5 (Random)	CRTC	1143
574/23	237	+4/+1 (S.S.)	CRTC	1144
926/23	132	+6.5	CRTC	1144
1411/23	143	+6.0	CRTC	1145
2134/23	335	+6.0	CRTC	1145
4721/23	647	+7 (Random)(S.S.)	CRTC	1146
8363/23	464	+6 (Random)	CRTP	1147
14897/23	665	+7 (Random)	CRTP	1148
17646/23	575	+7 (Random)	CRTP	1149
18087/23	574	+7 (Random)	CRTP	1150
20502/23	566	+7 (Random)	CRTC	1151
20504/23	557	+7.5 (Random)	CRTP	1151
22670/23	567	+6 (Random)	CRTP	1151
881/24	143	+5.5/+1 (S.S.)	CRTC	1153
2210/24	335	+4/+1 (S.S.)	CRTC	1153
4721/24	647	+7 (Random)(S.S.)	CRTC	1154
4792/24	655	+6.5 (Random)	CRTP	1154
8363/24	464	+6 (Random)	CRTP	1155
14897/24	665	+7 (Random)	CRTP	1156
18087/24	574	+7 (Random)	CRTP	1157
20502/24	566	+7 (Random)	CRTP	1157
20504/24	557	+7.5 (Random)	CRTP	1157
22670/24	567	+6 (Random)	CRTP	1158
24970/24	656	+5.5 (Random)	CRTP	1159

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