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Teardown Evaluation of a 1975 Piper Navajo Chieftain Model PA31-350 Airplane

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16. Abstract

To determine if potential continuing airworthiness problems exist for the small airplane fleet as a function of the aging process, the Federal Aviation Administration established a research program to conduct a destructive evaluation of four aged airplanes (two Cessna 402s, a Piper Navajo Chieftain, and a Beechcraft 1900D) used in commuter service. The intent of the program is to provide insight into the condition of a typical aged airplane. This program (1) determines if a correlation exists between the airplane's maintenance history and current condition from a safety of flight perspective and (2) generalizes recommendations on inspection programs and airplane condition to the small airplane fleet, as opposed to making specific recommendations on a particular model. This document provides findings in a summary report on the teardown evaluation of a 1975 Piper Navajo Chieftain model PA31-350. The results provide information for use in future investigations into the aged small airplane fleet and determine if additional research is required to address specific problems observed (if any). The destructive evaluation of the commuter-class airplane was separated into two main tasks: (1) inspection of the airframe, airplane systems, and wiring and (2) teardown evaluation of the airframe, airplane systems, and wiring. During the inspection phase, four subtasks were performed: a survey of the airplane maintenance records, visual inspection of the airframe, airplane systems and wiring, and the development and implementation of airworthiness airframe inspections. The teardown evaluation involved disassembly of the airframe and major airplane sections, inspection of airplane systems' components, a structural assessment using alternative nondestructive inspection techniques, and microscopic examination of critical structural areas. As part of the destructive evaluation, inspections and testing were also performed on the airplane wiring and other electrical components of the airplane to assess the condition and degradation of electrical wiring in small airplanes and to evaluate maintenance procedures. Specific observations are made regarding findings discovered during the teardown evaluation on the particular airplane selected.

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LIST OF ACRONYMS AND ABBREVIATIONS

AAIP	Approved Aircraft Inspection Program records
ACTT	Aircraft total time
AD	Airworthiness Directive
ASTM	American Society for Testing and Materials
ATA	Air Transport Association of America
BL	Body line
CAR	Civil Air Regulations
CFR	Code of Federal Regulations
DN	Down
ELT	Emergency locator transmitter
FAA	Federal Aviation Administration
FS	Fuselage station
Fwd	Forward
I/B	Inboard
JASC	Joint Aircraft System/Component
LBL	Left body line
LT	Left
LWS	Left wing station
MLG	Main landing gear
N/A	Not applicable
NDI	Nondestructive inspection
NIAR	National Institute for Aviation Research
NTSB	National Transportation Safety Board
O/B	Outboard
OC	Overcurrent
PSE	Principal structural element
RBL	Right body line
RT	Right
RWS	Right wing station
SAIB	Special Airworthiness Information Bulletin
SB	Service Bulletin
SDR	Service Difficulty Reports
WL	Water line
WS	Wing station

EXECUTIVE SUMMARY

To determine if potential continuing airworthiness problems exist for the small airplane fleet as a function of the aging process, the Federal Aviation Administration (FAA) established a research program to conduct a destructive evaluation of four aged airplanes (two Cessna 402s, a Piper Navajo Chieftain, and a Beechcraft 1900D) used in commuter service. The destructive evaluation of the commuter-class airplane was separated into two main tasks: (1) inspection of the airframe, airplane systems, and wiring and (2) teardown evaluation of the airframe, airplane systems, and wiring. During the inspection phase, three subtasks were performed: a survey of the airplane maintenance records; visual inspection of the airframe, airplane systems, and wiring; and the development and implementation of airworthiness airframe inspections. The teardown evaluation involved the disassembly of the airframe and major airplane sections, inspection of airplane systems' components, a structural assessment using alternative nondestructive inspection (NDI) techniques, and microscopic examination of critical structural areas. As part of the destructive evaluation, intrusive inspections were also performed on the airplane wiring and other electrical components of the airplane, and the functionality and performance of circuit breakers were tested. The intrusive wiring inspections and circuit breaker testing assess the condition and degradation of electrical wiring in small airplanes and an evaluation of maintenance procedures.

Of the 199 wiring defects documented during the routine visual inspections, 84 were wiring condition defects, 97 were installation defects, and 18 were termination defects. Of the 23 accessible principal structural elements inspected as part of the Piper Navajo Chieftain Service Manual airworthiness inspections, damage was identified in five areas: (1) the nose landing gear strut housing, (2) rudder torque tube, (3) left and (4) right wing rear spar attachment points, and (5) the wing spar main attachment points.

A structural assessment using alternative NDI techniques was performed on the spars of the wings, horizontal stabilizer, and vertical stabilizer in an effort to identify additional embedded defects in the primary structure prior to disassembly. This revealed 19 crack indications identified for further investigation during the teardown evaluation.

Of the 697 parts inspected during the teardown evaluation, 303 defects were identified and characterized. These defects included 161 cracks, 104 areas of corrosion, 28 areas of wear, and 10 areas of other damage. Of the 1022 findings documented during the intrusive visual inspections, 117 were categorized as critical findings. Eighteen percent of the wiring findings were rubbing/chafing of the outer insulation with an additional 18 percent classified as fluid contamination and another 17 percent identified as brittle and cracking insulation. Twenty-two percent of the findings were related to the heater-ventilating and deicing system, 15 percent were part of the airplane power system, and an additional 15 percent of the findings were related to the landing gear system. The data from the circuit breaker tests suggests the functional life of a circuit breaker can be extended if manual cycling is part of routine maintenance.

1. INTRODUCTION.

Economic and market conditions of present day aviation companies are requiring the use of airplanes far beyond their original design life objectives. The aging airplane concern exists for all types of airplanes including commercial, military, and general aviation. The concern is being amplified as more companies use aged airplanes and rely on standard inspection practices for a guarantee of airworthiness assurance. Standard practices to ensure continuing airworthiness include scheduled inspection and maintenance tasks contained in service manuals, Instructions for Continued Airworthiness, Airworthiness Directives (AD), and Service Bulletins (SB). These practices are not just limited to structural integrity, but also extend to wiring and systems integrity. These initiatives have provided timely preventative maintenance recommendations that permit continued safe operation of aging airplanes until retirement from service for economic reasons.

1.1 BACKGROUND.

Although the general public is primarily concerned with the airworthiness of commercial airplanes, which is where most research funding resources and efforts have been focused, a growing concern also exists in the small airplane fleet. While numerous investigations are being conducted on the aging aspects of large transport and military airplanes, limited resources are being expended to understand the aging aspects of the small airplane fleet. Investigations performed on large transport and military airplanes have focused on the structural integrity as well as wiring and systems-related aging concerns. The results of these investigations can benefit a similar research program that investigates the same issues on small airplanes.

The reliability and maintenance of electrical wiring and electrical components in aging airplanes have also become a major concern for the aviation industry. The Federal Aviation Administration (FAA) and Aging Transport Systems Rulemaking Advisory Committee have been working to assess the condition of electrical wiring and the effectiveness of wiring maintenance procedures. However, their efforts have been primarily focused on larger commercial transport airplanes thus far. Due to the large number of small airplanes in service, a need for examining the condition of wiring, electrical components, and maintenance procedures for these smaller airplanes exist.

Small airplanes are generally classified as general aviation airplanes. When one mentions general aviation, the traditional image usually involves a four passenger airplane like a Cessna 172; however, general aviation covers a wide range of airplanes. In the context of this program, a general aviation airplane will be defined as a Title 14 Code of Federal Regulations (CFR) Part 23 (or predecessor Civil Air Regulations (CAR) 3) airplane, which includes normal, utility, acrobatic, and commuter category airplanes. This classification includes airplanes operating in the commuter, cargo, or taxi service capacity under 14 CFR Part 135. The small airplane fleet includes approximately 210,000 fixed-wing airplanes with about 71% being single-engine piston airplanes; 10% being multiengine piston airplanes; 10% being experimental airplanes; and 9% being classified as turboprop, jet, glider, or lighter-than-air airplanes. Usage of the small airplane fleet is categorized as follows: 60% for personal use, 21% for business use, 6% for instruction, 4% for aerial application/observation, 3% for commuter service, 2% for public use, and 4% for other usage.

Due to the large number of small airplanes and their wide usage, the aging aspects of these airplanes must be addressed. In September 2002, the FAA Small Airplane Directorate and William J. Hughes Technical Center established a research program to begin addressing the aging concerns regarding small airplanes. The main purpose for this program is to provide insight into the condition of a typical aged small airplane and determine if a correlation exists between its maintenance history and the airplane's apparent condition. The research program is primarily conducted by the Aging Aircraft Research Laboratory at the National Institute for Aviation Research (NIAR), Wichita State University. This research program's major objective focuses on the integrity and aging aspects of small airplanes.

This research program was divided into two stages. The first stage, which was completed in September 2004, involved the comprehensive teardown of two Cessna 402 airplanes (A and C models). To achieve the level of inquiry desired, the airplanes were each assessed during two phases of research. The "Inspection Phase" included a survey of the airplane maintenance records; visual inspection of the airplane structure, systems, and wiring; and airworthiness nondestructive inspections (NDI) targeted at specific airframe structural components. The "Teardown Phase" consisted of detailed structural disassembly, visual inspection of the internal structure and critical systems' components, wiring tests to determine condition, a structural assessment using alternative NDI techniques, and microscopic/fractographic examinations of suspect and critical structural areas. The second stage of the research program expanded on and attempts to validate the results of the first stage teardown evaluation. The second stage involved the inspection and teardown evaluation of two additional airplanes, a Piper Navajo Chieftain and a Beechcraft 1900D, in an effort to generalize recommendations on inspection programs and airplane condition to the small airplane fleet, as opposed to making specific recommendations on the Cessna 402 fleet alone.

Like the first stage, the tasks for the second stage were divided into an inspection phase and a teardown evaluation phase. For the Piper Navajo Chieftain, the inspection phase consisted of a survey of the airplane maintenance records; visual inspection of the airframe, systems, and wiring; and the development and implementation of a set of airworthiness inspections using traditional NDI techniques to target specific airframe locations known as principal structural elements (PSE). The teardown evaluation phase consisted of the removal of major airplane sections, a structural assessment using NDI techniques, wiring inspections, circuit breaker tests, detailed disassembly, microscopic examination, and fractographic analysis of fracture faces to determine failure mode.

1.2 RESEARCH OBJECTIVES.

Much of the current concern related to aging effects on airplanes involves calendar age as opposed to flight hours. For instance, deterioration of wiring, aging effects on airplane systems (control systems, seals, cables, etc.), and corrosion are calendar related, and these effects may possibly be a continued safety of flight concern. For example, approximately 25,000 airplanes that are older than 50 years presently exist and still have the original electrical systems. Currently, no inspection criteria exist for evaluating the condition of aged wiring. In addition, major attachment fittings, such as wing attachment fittings, are typically never removed and inspected. These two areas of concern are the areas that this research program primarily focuses on to begin addressing and providing insight into the aging small airplane fleet.

The research program has a short-term objective to be achieved over 2 years during the first stage, and a long-term objective to be achieved over a 5-year period during the first and second stages. The short-term objective is to determine if potential continuing airworthiness problems exist for the small airplane fleet as a function of the aging process. The long-term objective is to establish guidance to ensure that current maintenance programs of small airplanes are providing acceptable levels of continued airworthiness. Achievement of the short-term objective should determine if generic degradation indicators exist in the small airplane fleet. These indicators will likely include structural (cracking/corrosion); electrical systems or wiring; airplane systems, such as fuel, hydraulic, pneumatic, mechanical, and flight control; and maintenance, service, and inspection quality. Determination of generic degradation indicators will assist in providing initial generic inspection guidance such as:

- Do maintenance inspection programs address all areas of concern appropriately?
- What was found in areas that normal maintenance would not see?
- Are additional inspection criteria required for aged airplanes?
- Should specialized one-time inspections at some age be required?
- Should inspections and maintenance programs become more extensive as the airplane ages?

To achieve the research objectives in the first stage, a destructive evaluation was conducted on two aged airplanes (both Cessna 402 models) used in commuter service. The intent of the program was not to provide statistical evidence for guideline development for inspection, but to provide insight into the condition of a typical aged airplane and to see if a correlation exists between its maintenance history and current condition from a safety of flight perspective. This program documented findings in a summary report for use in future investigations into the aged small airplane fleet and to determine if additional research is required to address specific problems observed (if any). Specific observations were made regarding the particular airplane selected, and generic recommendations were provided that are applicable to the small airplane fleet.

To achieve the long-term objectives of the research program, the second stage involves conducting an extensive teardown evaluation on a Piper Navajo Chieftain and a Beechcraft 1900D. The results from these teardown evaluations will allow small airplane fleet-wide conclusions to be drawn regarding the problems facing the small airplane fleet due to the aging process. Recommendations for guidance on maintenance programs to provide acceptable levels of continued airworthiness on small aging airplanes also will be provided.

1.3 TECHNICAL APPROACH.

The destructive evaluation of this Piper Navajo commuter-class airplane was divided into two main phases: (1) inspection of the airframe, systems, and wiring and (2) teardown evaluation of the airframe, airplane systems, and wiring. Each section below describes the airplanes selected for teardown evaluation and the subtasks conducted during the inspection phase and the teardown phase.

1.3.1 Airplane Selection.

The Piper Navajo Chieftain model PA31-350 (tail number N61510), which was used in commuter operations, was selected because it is representative of a large portion of the small airplane commuter fleet. It shares many design commonalities with other small twin-engine airplanes, such as the Cessna 402s. The design concepts of both systems (mechanical, electrical, and flight controls) and structures (layout and materials) are similar model-to-model and manufacturer-to-manufacturer. Therefore, findings from the destructive evaluation of the Piper Navajo Chieftain would be applicable to all small airplane models regardless of manufacturer.

The twin-engine airplane, shown in figure 1, had 17,269.9 total airframe hours with a current registration. Maintenance records, log books, the AD compliance list, and FAA 337 forms were included with the purchase of this airplane.

Figure 1. 1975 Piper Navajo Chieftain, Tail Number N61510

1.3.2 Inspection Phase Summary.

During the inspection phase, four subtasks were performed: a survey of the airplane maintenance records, routine visual inspections of the airframe and airplane systems as prescribed by the Piper Navajo Chieftain Service Manual, routine visual inspection of all accessible airplane wiring, and airworthiness inspections developed for accessible structural locations. The maintenance records survey provided information on the airplane maintenance history for correlation of maintenance practices to airplane condition, and the inspections determined the condition of the airplane based on normal maintenance activity.

1.3.3 Teardown Evaluation Phase Summary.

This research program focused on the destructive evaluation of a commuter-class airplane, yet a nondestructive evaluation was also conducted according to recommended practices prior to the destructive evaluation. The teardown evaluation involved disassembly of the airframe and major airplane sections, inspection of the airplane systems' components, a structural assessment using alternative NDI techniques, circuit breaker testing, microscopic examination of critical and suspect areas, and fractographic analysis of cracks to determine failure mode. All airplane systems' components and wiring were removed during disassembly allowing full access to all critical structural areas on the airplane. Inspection of the airplane systems' components assisted in determining if any signs of aging effects were apparent on the airplane systems. The structural assessment using alternative NDI techniques was conducted on the primary structure prior to disassembly. The microscopic examination of suspect and critical structural areas provided verification and detailed quantification of the extent of damage found during the airworthiness inspections, structural assessment using alternative NDI techniques, and disassembly of the entire airframe. Fractographic analysis was used to determine the failure mode of selected cracks.

2. INSPECTION PHASE.

The inspection phase of the Piper Navajo Chieftain teardown established the current condition of the airplane through a survey of the airplane maintenance records, routine visual inspections per the Piper Navajo Chieftain Service Manual, routine wiring inspections of all accessible wires and airworthiness inspections of primary airplane structure developed for selected airframe locations. These inspections allowed the NIAR team to determine which flaws could be found in the field during normal maintenance activity. The routine visual inspections were prescribed for specific airframe locations and systems' components. All accessible wires were also visually inspected for wiring condition defects, installation defects, and termination defects during the wiring routine visual inspections. The airworthiness inspections consisted of using typical NDI techniques targeted at specific primary structural locations on the airframe that were accessible and did not require destructive disassembly for the inspections. Inaccessible areas, such as the nacelle bulkheads, were not included in these airworthiness inspections. Since no supplemental inspection program has been formally established for this airplane, the NIAR team internally developed a set of airworthiness inspections with the assistance of the Piper Aircraft Company, the FAA Small Airplanes Directorate, and the FAA William J. Hughes Technical Center. The primary structural locations that were accessible and inspected during the airworthiness inspections were determined through a review of the applicable fleet-wide ADs, SBs, Service Difficulty Reports (SDR), and engineering knowledge of the airplane structure. Detailed inspection procedures were developed for each accessible airworthiness inspection location. Upon completion of the inspections, the NIAR team made an effort to correlate airplane condition with airplane usage and maintenance history through the survey of airplane maintenance records and a review of the Piper Navajo Chieftain SDRs.

2.1 SURVEY OF AIRPLANE MAINTENANCE RECORDS.

A survey of the airplane maintenance records was conducted in an attempt to correlate airplane condition with airplane usage and maintenance history. Maintenance log books, SBs, ADs, and

FAA 337 forms for major repairs and modifications were reviewed in this survey. The survey of the airplane records for the Piper Navajo Chieftain PA31-350 serial number 31-7552019 was completed using data obtained from the following resources:

- Airframe log books
- Approved Aircraft Inspection Program records (AAIP)
- FAA databases
- National Transportation Safety Board (NTSB) databases
- Piper Aircraft Maintenance and Service Publications
- Civil Aviation Registry records

Since leaving the production line, this airplane was registered under one tail number to 19 different owners/operators with 23 title changes, as shown in table 1. During its 29 years of operation, the airplane accumulated 17,269.9 total airframe hours. The acquisition dates and owner/operator information was obtained from the Civil Aviation Registry using applications for registration and airplane bill of sale records. As shown in table 1, the airplane spent approximately 3 years in Alaska, 6 years in the corrosive environment of the East Coast and approximately 12 years in Arizona and Nevada, likely performing Grand Canyon tours in an environment conducive to fatigue.

Table 1. Airplane History

Owner/Operator	Location	Acquisition Date
Aviation Development Co.	Oklahoma City, Oklahoma	01/13/1975
Catlin Aviation Co.	Oklahoma City, Oklahoma	01/21/1975
Polar Airways	Anchorage, Alaska	09/19/1975
Catlin Aviation Co.	Oklahoma City, Oklahoma	10/30/1978
Flight America Inc.	Lynchburg, Virginia	02/16/1979
J&F Leasing	Altavista, Virginia	07/02/1979
Catlin Aviation Co.	Oklahoma City, Oklahoma	11/05/1979
Reese R. Wagner John R. Hartman Maria K. Garcia-Rivenburgh John J. Kruse	Douglas, Arizona	01/16/1980
W.R. Eddins Michael G. Wystrach	Tuscon, Arizona	06/15/1982
Valley National Bank of Arizona	Tuscon, Arizona	06/20/1985
Nelson Clopine	Topeka, Kansas	06/26/1985
El Paso Aircraft Inc.	El Paso, Texas	Date not recorded
Nicholson Air Services Inc.	Cumberland, Maryland	07/24/1985
KPT Aviation Inc. Gaithersburg Aviation Inc.	Wheaton, Maryland	04/06/1990

Table 1. Airplane History (Continued)

Owner/Operator	Location	Acquisition Date
Gaithersburg Aviation Inc. KPT Aviation Inc. John. R Linderman Sr. T/A the Navajo Partnership	Wheaton, Maryland	04/09/1990
Navajo Partnership Ltd.	Wheaton, Maryland	07/31/1990
El Paso Aircraft Inc.	El Paso, Texas	03/13/1991
Lock Haven Aircraft Sales Inc.	Wilmington, Delaware	04/30/1992
Montage Aviation Corp Canyon Air Aviation Inc.	Keene, New Hampshire	07/24/1992
G&L Airline Properties Inc.	Las Vegas, Nevada	03/03/1995
Vegas Jet Financial and Leasing LLC	North Las Vegas, Nevada	04/14/1998
Clearwater Aircraft Inc.	Clearwater, Florida	11/21/2003
Corey Gerulis	Mount Juliet, Tennessee	02/23/2004

The ADs obtained from the FAA database that are applicable to this model of airplane are listed in table 2 This table includes the date of compliance for tail number N61510, the compliance aircraft total time (ACTT), the applicability to this particular serial number, and the compliance interval. Table 2 shows that AD 2003-24-07 was not complied with. This was due to NIAR's acquisition of the airplane in October 2004, which was still within the time of serve requirement for this AD. Ads relative to this research project are summarized as follows:

- AD 80-17-02 is used to ensure flutter integrity of the elevator system in accordance with Piper Aircraft Corporation SB No. 690, dated July 7, 1980, or in an equivalent manner.

- AD 85-08-05 precludes the loss of the main and nose landing gear by performing a detailed visual inspection to include Piper SB No. 779A (July 16, 1984) on each gear assembly. Upon finding a discrepancy, a maintenance action is necessary to restore airworthiness according to the appropriate maintenance manual.

- AD 96-10-14 is intended to prevent the main landing gear (MLG) from retracting because of a cracked MLG forward sidebrace, which, if not detected and corrected, could result in gear collapse and loss of control of the airplane during landing operations. Furthermore, when incorporating repetitive inspections on the left and right MLG, forward side braces maintenance action may be required if a failure is found.

- AD 96-10-15 requires that certain Piper models conduct repetitive inspections on the outboard flap tracks, wing rib flanges, and rear spar web at wing station (WS) 147.5 on both wings. The actions specified in this AD are intended to prevent structural failure under certain load conditions caused by cracked areas at WS 147.5, which, if not detected and corrected, could result in loss of control of the airplane.

- AD 96-12-12 requires that certain Piper models conduct repetitive inspections on the upper section of fuselage station (FS) 317.75 bulkheads for cracks and incorporating a certain reinforcement kit if any crack is found. This AD is intended to prevent structural failure of the vertical fin forward spar caused by cracks in the FS 317.75 bulkhead, which, if not detected and corrected, could result in loss of control of the airplane.

- AD 96-21-03 is directed at detecting cracks, using fluorescent liquid penetrant on certain models of Pipers that may be caused in the area of the inboard aileron hinge bracket, which, if not detected and corrected, could result in loss of control of the airplane.

- AD 96-21-11 requires that a reinforcement bracket on the MLG be replaced with a better designed bracket for certain models of Piper airplanes. The actions specified by this AD are intended to prevent the MLG from extending, when not selected and while the airplane is in flight, caused by actuator reinforcement bracket failure, which could result in substantial airplane damage or loss of control of the airplane.

- AD 98-09-25 applies to certain models of Piper airplanes and requires replacement and reworking of the lower spar splice plate and lower spar caps, respectively. The actions specified by this AD are intended to prevent failure of the lower spar splice plate caused by fretting and cracking, which could result in loss of control of the airplane.

- AD 99-06-01 applies to certain models of Piper airplanes, which requires repetitive inspections of the horizontal rear spar in the area of the outboard hinge attachment and the outboard hinge attachment bracket for cracks. When cracks are found at a certain accumulation of time-in-service, this AD also requires modifying the horizontal stabilizer spar by incorporating an improved stabilizer spar and hinge bracket assembly kit that will terminate the repetitive inspections. The actions specified by this AD are intended to prevent failure of the horizontal stabilizer rear spar caused by cracks at the elevator outboard hinge attachment, which could result in loss of control of the airplane.

- AD 99-12-05—The actions specified by this AD are intended to prevent failure of the elevator spar caused by fatigue cracking, which could result in reduced airplane controllability. As a result of reports of cracks, this AD is repetitive and requires installing inspection holes in the elevators and looking at the ice protection and structural systems. Replacement with an improved elevator design is required if a crack is found.

- AD 2000-25-01, per this repetitive AD, the owner/operator is required to inspect the MLG inboard door hinges and attachment angles for cracks on certain model-specific Piper airplanes. The actions specified by this AD are intended to detect and correct cracked MLG inboard door hinge assemblies. These cracked door hinge assemblies could result in the MLG becoming jammed, with consequent loss of control of the airplane during landing operations.

- AD 2001-06-01 supersedes three existing ADs and only applies to certain Piper model airplanes. This requires repetitive inspection and a modification of the elevator structure, which terminates the need for repetitive inspections. The actions specified by this AD are intended to continue to detect and correct damage to the elevator structure. A damaged elevator structure could lead to reduced or total loss of control of the airplane.

- AD 2003-24-07 requires that an inspection hole be installed in the rudder torque tube area on certain models of Piper airplanes in order to perform a detailed visual inspection of the rudder torque tube and its associated ribs as a result of reports of corrosion. The actions specified by this AD are intended to detect and correct corrosion in the rudder torque tube assembly and rudder rib, which could result in failure of the rudder torque tube.

Table 2. Airworthiness Directives

AD Number	Compliance Date	Compliance ACTT	Applicability	Interval
74-26-10			N/A by S/N	
75-06-04			N/A by S/N	
75-09-10			N/A by S/N	
75-26-18			Superseded	
76-04-10	09/05/1999	15,176.70	Applicable	Once
76-04-11	09/05/1999	15,176.70	Applicable	Once
76-10-06			Superseded	
76-15-07	09/05/1999	15,176.70	Applicable	Once
77-01-06			N/A by S/N	
77-07-03			N/A by S/N	
77-08-02R1	09/05/1999	15,176.70	Applicable	Once
77-09-10			N/A by S/N	
77-14-14			N/A by S/N	
77-19-06	09/05/1999	15,176.70	Applicable	Once
77-24-01R1			N/A by S/N	
77-26-02			N/A by S/N	
78-01-02			N/A by S/N	
78-05-05	09/05/1999	15,176.70	Applicable	Once
78-25-03	02/13/1979	3,819.33		
78-26-11	02/13/1979	3,819.33	Applicable	Once
79-01-04			Superseded	
79-12-02R1	09/05/1999	15,176.70	Applicable	Once
79-12-03R2			Superseded	
79-14-02	09/05/1999	15,176.70	Applicable	Once
79-20-07			N/A by S/N	
80-08-05	10/29/1980	4909.3		

Table 2. Airworthiness Directives (Continued)

AD Number	Compliance Date	Compliance ACTT	Applicability	Interval
80-09-10	10/29/1980	4909.3		
80-13-06	10/29/1980	4909.3	Applicable	Once
80-14-06			Superseded	
80-17-02	09/05/1999	15,176.70	Applicable	Once
80-17-06	10/29/1980	4909.3	Applicable	Once
80-18-10			N/A by Part	
80-20-04			Superseded	
80-22-04			Superseded	
80-26-05			Superseded	
81-11-03			Superseded	
81-11-04			Superseded	
81-15-04R1			Superseded	
82-08-06R1			Superseded	
82-16-05R1	03/11/1995	15176.7	Applicable	Recurring
82-27-13R2	09/01/2001	17232	Applicable	Recurring
85-08-05	09/10/1985	5968	Applicable	Once
86-17-07			Superseded	
87-21-01	02/04/1988	8189	Applicable	Once
88-05-05			Superseded	
92-26-02			Superseded	
92-27-05	03/09/1993	9272.1	Applicable	Once
93-02-13			Superseded	
93-23-13 C	09/05/1999	15176.7	Applicable	Recurring
93-24-02R1	03/26/1994	9483.3	Applicable	Once
93-25-08			Superseded	
96-10-14	05/18/1997	12546.8	Applicable	Recurring
96-10-15	09/05/1999	15,176.70	Applicable	Recurring
96-12-12 C	09/05/1999	15,176.70	Applicable	Once
96-21-03	11/20/1996	12026.9	Applicable	Recurring
96-21-04	09/05/1999	15,176.70	Applicable	Once
96-21-11	02/28/1995	9686.3	Applicable	Once
96-24-13	09/05/1999	15,176.70	Applicable	Once
97-07-03	09/05/1999	15,176.70	Applicable	Recurring
98-04-27			Superseded	
98-08-18	03/21/2001	16,654.40	Applicable	Recurring
98-09-25	09/05/1999	15,176.70	Applicable	Once
99-06-01	08/23/1999	15,123.50	Applicable	Recurring

Table 2. Airworthiness Directives (Continued)

AD Number	Compliance Date	Compliance ACTT	Applicability	Interval
99-12-05	03/31/2000	15,841.00	Applicable	Recurring
99-14-01	09/05/1999	15,176.70	Applicable	Once
2000-06-06	05/05/2000	15,922.10	Applicable	Once
2000-25-01	03/02/2001	16,619.30	Applicable	Recurring
2001-06-01	03/31/2000	15,841.00	Applicable	Recurring
2003-24-07	Not complied with	Not complied with	Applicable	Recurring

N/A = Not applicable
S/N = Serial number

Table 3 shows that SBs 1007 and 1105A were not performed on this airplane. These SBs and others of significance are summarized as follows:

- SB 636A describes modification kits that are available after an airplane has more than 2000 hours of operating time. These kits are to relieve the recurring inspection requirements and repair of any existing cracks in the attachment points for the vertical fin forward spar. If left unattended, this condition could eventually result in the loss of structural integrity of the vertical fin forward attachment.

- SB 647A, after several reports of cracks occurring in the outboard flap track attachment angles, wing rib flange, and the rear spar web at WS 147.5, this SB allows for a field modification, which allows for inspection and/or repairs. Left unattended, such cracks could eventually lead to flap operational interference.

- SB 1007 requires a final modification kit to be installed, due to continued reports of cracks being discovered in the horizontal stabilizer rear spar at the elevator outboard hinge attachment point even after compliance with a previous SB. Left uncorrected, such cracks could propagate and eventually result in impairment of elevator control.

- SB 1105A addresses corrosion in the rudder torque tube that may compromise the structural integrity of the torque tube and affect the rudder control.

Table 3 provides the following information regarding applicable SBs: requested compliance date, description, compliance method, actual compliance date, and ACTT.

Table 3. Service Bulletins

SB Number	Requested Compliance Date	Description	Compliance Method	Actual Compliance Date	ACTT
494B	07/17/1979	Inspection of flap control system	AD 82-27-13 R2	09/01/2001	17232
500	04/23/1976	Elevator balance weight modification	AD 76-15-07	09/05/1999	15176
604A	06/8/1979	Forward baggage door locking system modification	KIT 763-853	10/30/1980	4962
621	10/13/1978	Elevator cable inspection and cable guard installation	AD 78-26-11	02/13/1979	3819
626C	02/28/1997	Elevator down spring inspection/ replacement	AD 98-08-18	03/21/2001	16654
636A	08/26/1980	Bulkhead reinforcement at FS 317.75	KIT 964-028	10/30/1980	4962
647A	11/24/1980	Outboard flap track inspection and reinforcement	KIT 763-986	10/29/1980	4909
682	07/24/1980	Main landing gear inboard door hinge inspection/ replacement	AD 80-26-05	03/02/2001	16619
690	07/07/1980	Elevator balance weight inspection	AD 80-17-02	09/05/1999	15176
739	03/01/1982	Restriction of flap travel, inspection of flap flex shaft	AD 82-27-13 R2	08/05/1982	5701
779B	10/12/1987	Landing gear upper bearing retaining pin replacement	AD 85-08-05	09/10/1985	5968

Table 3. Service Bulletins (Continued)

SB Number	Requested Compliance Date	Description	Compliance Method	Actual Compliance Date	ACTT
845A	10/09/1987	Main landing gear forward side brace inspection replacement	AD 96-10-14	05/18/1997	12546
923	08/16/1989	Main landing gear actuator reinforcement bracket replacement	AD 96-21-11	02/28/1995	9686
974	10/19/1994	Aileron hinge inspection and replacement	AD 96-21-03	11/20/1996	12026
998A	08/04/1997	Elevator spar inspection	AD 99-12-05	04/05/2000	15841
1003	06/16/1997	Wing spar splice plate replacement	AD 98-09-25	06/10/1998	13823
1007	09/30/1997	Horizontal stabilizer spar outboard hinge inspection and modification	Not complied with		
1008	09/30/1997	Elevator butt rib and spar inspection	AD 99-12-05	04/05/2000	15841
1105A	09/22/2003	Rudder torque tube assembly inspection	Not complied with		

Table 4 lists all major repairs/alterations documented on FAA form 337 specific to the Piper Navajo Chieftain, tail number N61510, that were contained in the airplane records or in the ADs from the Civil Aviation Registry.

- 03/02/1982—The belly skin from FS 57.0 to FS 215.0 below water line (WL) -18.0 was replaced, as well as the left WS 120.0 to WS 138. All avionics antennas and doublers on the belly were replaced along with the inboard main landing gear doors. The nose landing gear doors and actuating linkages were replaced along with the pitot mast and tube. Repairs were made to the horizontal stabilizer in accordance with AD 81-15-04 by installing kit 764 055.

- 06/24/1997—The left engine compartment electrical wiring was modified to aid in quick engine changes, with the addition of a terminal strip and wire length/termination modification.

- 12/15/1998—The right engine compartment electrical wiring was modified to aid in quick engine changes, with the addition of a terminal strip and wire length/termination modification.

Table 4. Major Repairs and Alterations

Date	Description
12/10/1976	Woodward Prop governors installed
04/06/1978	Engine overhauled S/N: L-295-68A
04/27/1978	Engine overhauled S/N: L-2834-61A
03/02/1982	Extensive airplane repair following gear up landing
08/25/1987	King distance measuring equipment installed
03/25/1993	Superseded-passenger tour system installed
02/18/1995	Bracket air filter assemblies installed
07/07/1995	Tape players and associated equipment installed
11/21/1996	TCB Composites left spinner and bulkhead installed
05/01/1997	TCB Composites right spinner and bulkhead installed
06/24/1997	Left engine compartment electrical wiring modified
07/15/1998	Vortex generators and nacelle strakes installed
07/21/1998	Airspeed indicator modified
12/15/1998	Right engine compartment electrical wiring modified
02/26/2001	Janitrol heater P/N FR65D79-3EL installed

S/N = Serial number
P/N = Part number

Table 5 lists the significant minor repairs/alterations not listed as major repairs/alterations on FAA form 337 for the Piper Navajo Chieftain, tail number N61510. These were discovered in the maintenance records and consist mainly of replaced or repaired items.

- 12/01/1992—Landing gear strut housing, part number 40273-00, replaced.

- 03/25/1998—Cracked right main landing gear side brace attachment, part number 40294-00, replaced.

- 06/15/1998—Spar splice reinforcement plate and kit 766-641 installed in accordance with AD 98-09-25.

- 12/07/1998—Elevator stop bolts, part number 71533-02, replaced.

- 01/10/1999—New bulkheads, part numbers 41050-08 and 41112-08, at FS 220.2 and FS 236.7.

- 04/05/2000 New elevator spars, part numbers 40075-20 and 40075-21, and elevator butt rib kit 766-642 were installed in accordance with AD 99-12-05, SB 998A, and SB 1008.

Table 5. Minor Repairs and Alterations

Date	Description
06/09/1989	Left wing flap trailing edge patched
11/17/1992	Left wing tip trailing edge repaired after being bent
12/01/1992	Nose gear strut replaced due to damaged steering stop
09/30/1993	Nose gear trunnion replaced
03/25/1998	Cracked right main landing gear side brace attachment replaced
06/15/1998	Spar splice reinforced per AD 98-09-25
12/07/1998	Damaged elevator stop bolts replaced
01/10/1999	Cabin bulkheads at FS 220.2 and FS 236.7 replaced
04/05/2000	Elevator spars and butt ribs replaced per AD 99-12-05
10/05/2000	Loose rivets in the tail replaced
07/24/2001	Loose nacelle rivets near spars of both wings replaced

Table 6 lists the Special Airworthiness Information Bulletins (SAIB) that were retrieved from the FAA database. An SAIB is an information tool that alerts, educates, and makes recommendations to the aviation community. SAIBs contain nonregulatory information and guidance that does not meet the criteria for an AD.

Table 6. Special Airworthiness Information Bulletins

SAIB Number	SAIB Date	Description
ACE-96-16	12/24/1996	Advisement of SB compliance
ACE-98-01	10/17/1997	Guidance for AD 82-27-13R2
ACE-98-44	08/26/1998	Drawing change for AD 98-09-25
CE-04-86	09/13/2004	Service publication for heater

Table 7 lists the General Aviation Airworthiness Alerts also known as AC 43-16A by Air Transport Association of America (ATA) code. Items related to wiring, structure, flight controls, fuel, and landing gear are listed. Listed items are airplane make and model specific, not serial number specific.

Table 7. General Aviation Airworthiness Alerts

ATA Code	Description
2400	Landing gear safety solenoid circuit breaker defective
2460	Battery relay burned and broken
2897	Left fuel boost pump wire chafing fuel line, caused brief fire in left wing root
3230	Broken wire in the left main landing gear downlock switch
3230	Flexible hose assembly (P/N 17766-04) used on the nose gear actuator ruptured
3230	Nose landing gear upper right drag brace (P/N 40336-00) broken
3230	Left main landing gear retraction arm (P/N 42042-00) cracked
3230	Nose landing gear right upper drag brace link assembly (P/N 40336-00) cracked
5312	Bulkhead doubler (P/N 40682-10) cracked
5320	Upper support bracket (P/N 40987-02) for the nose landing gear actuator cracked
5510	Elevator outboard hinge assembly (P/N 71700-2) cracked
5540	Rib (P/N 40440-00) used to attach rudder balance weight broken
7332	Fuel pressure hose (P/N 80048-04) leaking at the fuel pressure gauge

P/N = Part number

A search of the NTSB database of accidents did not indicate that any accidents/incident were ever recorded for the Piper Navajo Chieftain, tail number N61510. A review of the Type Certificate Data Sheet A20SO and the Supplemental Type Certificates indicates the airplane was in compliance with requirements for airworthiness at the time of inspection.

2.2 SURFACE DIFFICULTY REPORTS DATABASE REVIEW.

The database review of the Piper Navajo Chieftain SDRs involved over 6200 reports from 1974 to 2004. Table 8 lists some of the SDRs obtained from the FAA database by the Joint Aircraft System/Component (JASC) code, which is used to categorize the reports. It is important to note that the SDR database is make and model specific, not serial number specific. Table 9 lists the SDR categories of interest to this research program and the number of occurrences per category. The breakdown of SDRs by JASC code is shown graphically in figure 2, which shows that 17% are related to the landing gear systems and an additional 15% are related to the reciprocating engines. SDRs are not always generated for each issue that arises; therefore, common problems may exist that are not appropriately represented by the SDR database.

Table 8. Service Difficulty Reports

JASC Code	Description
2497	Smoke in cabin emanating from instrument panel caused by a broken wire
2497	Smoke in cabin emanating from circuit breaker panel caused by a shorted wire
2710	Discovered crack in right aileron bellcrank assembly during 100-hour inspection
2720	Corrosion found on rudder torque tube and attaching ribs due to water entrapment

Table 8. Service Difficulty Reports (Continued)

JASC Code	Description
3210	Landing gear retract arms cracked, found during cleaning, difficult to detect when painted
3210	Shear failure of rivets attaching uplock brackets to attaching rib
3211	Nose wheel well reinforcement channel cracked at cutout, caused by loose bolts
3213	Right landing gear trunnion cracked at upper center web
3213	Right main landing gear cracked under brake line clamp
3213	Main landing gear cracked on forging seam line, found during NDI
3220	Failure of nose landing gear assembly housing due to improper towing
5210	Fuselage upper cabin door hinge support structure cracked
5280	Main landing gear inboard door hinge cracked, post SB/AD repair
5312	Left and right fuselage mounted forward wing support bulkheads were found to be cracked on their lower mounting flanges
5510	Horizontal stabilizer outboard hinge brackets cracked
5520	Elevator horn found cracked at bolthole
5520	Elevator channel structure found cracked at nutplate
5521	Left and right elevator spars cracked at tips on both sides of the top and bottom flanges
5531	Vertical stabilizer rib cracked, rivets loose and drag angle cracked
5531	Front spar reinforcing angle of the vertical fin cracked at bushing
5540	Rudder torque tube has small holes due to corrosion
5541	Rib below rudder balance weight broken out
5710	Left and right wing frames have numerous cracks due to contact with wheel brakes
5711	Left and right forward spar attachment structures cracked
5720	Stiffener on rib in wheel well cracked
5740	Left and right upper main spar attachment fittings corroded and rivets sheared
5740	Web P/N 4029400 cracked
5751	Left and right aileron spars have 1.0-inch cracks
5751	Inboard rib and spar of right aileron cracked
5753	Flap leading edge behind engine separated due to corrosion and rivet failure
7120	Bracket broken at engine attachment points
7120	Engine bracket broken at weld

P/N = Part number

Table 9. Number of SDR Occurances by Category

JASC Code	Number of Reports
32xx-Landing Gear	1045
85xx-Reciprocating Engine	953
81xx-Turbocharging	734
74xx-Ignition	444
61xx-Propellers/Propulsors	318
27xx-Flight Controls	297
28xx-Fuel	262
24xx-Electrical Power	250
73xx-Engine and Fuel Control	195
21xx-Air Conditioning	182
52xx-Doors	171
55xx-Stabilizers	167
57xx-Wings	139
29xx-Hydraulic Power	132
37xx-Vacuum	132
78xx-Engine Exhaust	121
34xx-Navigation	107
71xx-Power Plant	107
79xx-Engine Oil	102
53xx-Fuselage	67
80xx-Starting	57
22xx-Autoflight	53
77xx-Engine Indicating	40
25xx-Equipment/Furnishings	39
30xx-Ice and Rain Protection	36
72xx-Turbine/Turboprop Engine	29
33xx-Lights	28
76xx-Engine Controls	28
23xx-Communications	26
56xx-Windows	20
14xx-Hardware	5
54xx-Nacelles/Pylons	5
31xx-Instruments	2
35xx-Oxygen	2
36xx-Pneumatic	2

Table 9. Number of SDR Occurances by Category (Continued)

JASC Code	Number of Reports
26xx-Fire Protection	1
83xx-Accessory Gearboxes	1
Total	6299

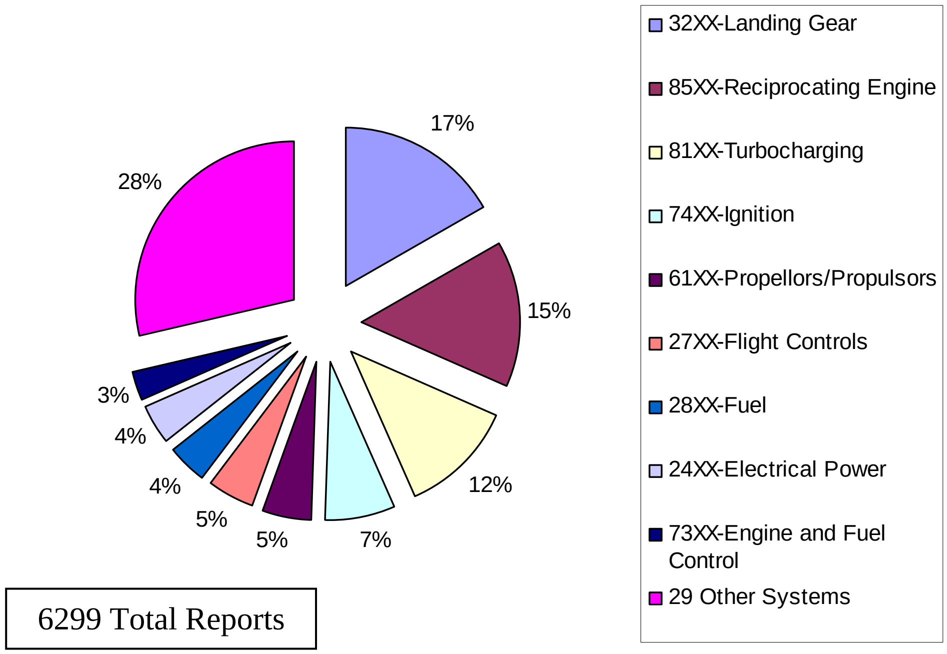


Figure 2. Breakdown of SDRs by Category

2.3 ROUTINE VISUAL INSPECTION OF AIRFRAME, SYSTEMS, AND WIRING.

The general condition of the airframe and systems' components was evaluated through routine visual inspections prescribed in the Piper Navajo Chieftain Service Manual. These visual inspections are typically required on an annual basis. All accessible wiring was also visually inspected for wiring condition defects, installation defects, and termination defects. The intent of these inspections was to discover all possible visual defects and document each defect's severity and location. Each inspection was duplicated by independent inspectors. The results of these inspections were reviewed by a licensed airframe mechanic to determine which findings were noteworthy. Noteworthy findings are defined as defects that would require further maintenance action.

2.3.1 Operational Checks.

Operational checks, listed below and obtained from the Piper Navajo Chieftain Service Manual, were performed to determine the functionality of airplane systems prior to teardown. The operational checks were performed by inspectors under the supervision of a licensed airframe mechanic. All systems were operational during these checks.

- Trim System
- Rudder Pedals
- Control Wheels, Column, Pulleys, and Cables
- Landing Lights
- Navigation Lights
- Cabin Lights
- Elevator Trim Mechanism
- Electronic Installations
- Instrument Lights
- Instruments
- Parking Brake
- Switches to Indicators Registering Fuel Tank Quantity
- Ailerons, Trim Cables, Pulleys, and Bellcranks
- Warning Horn and Lights

All fuel was removed from the airplane prior to the operational checks, resulting in the un-inspectability of the power plant systems. The following systems were not checked for operational functionality due to the system being disabled:

- Pressure Operated Instruments
- Electric Turn and Bank Indicator
- Fuel Selector Valves
- Fuel Crossfeed Valves
- Emergency Shutoff Valves
- Heater Fuel Pump
- Fuel Selectors
- Fuel Lines, Valves, and Gauges
- Cabin Heater
- Pneumatic Deicers

2.3.2 Airframe and Systems Routine Visual Inspections.

The Piper airframe and systems' components inspections, listed in the Piper Navajo Chieftain Service Manual, were divided into two main groups: the cabin group and the fuselage, wing, and empennage group. All routine visual inspections were performed by inspectors under the supervision of a licensed airframe mechanic. One hundred and one different visual inspection locations, listed below, were performed on the Piper Navajo Chieftain with the intent of finding every detectable flaw on the airplane. This inspection methodology led to a large number of documented flaws. Twenty-four inspection areas were listed in the cabin group with the remaining 76 listed in the fuselage, wing, and empennage group. Only 11 findings were determined to be noteworthy, and the remaining findings were deemed minor by airframe mechanics. With aged airplanes, flaws such as minor cracks, scratches, dents, paint chips, and slight corrosion are to be expected and pose no immediate threat to the safety of the airplane. In addition to dents and areas of missing paint, slight to moderate corrosion was noted on a majority of the screws on the airplane. It was also noted that some disposable parts, such as seals and hoses, were due to be replaced. All defects were recorded by location relative to the airplane coordinate system, by right wing station (RWS), left wing station (LWS), FS, WL, right body line (RBL), or left body line (LBL), as established in the Piper Navajo Service Manual.

- Cabin Group
  - Cabin Doors
    - Entrance
    - Pilot
    - Cargo
    - Baggage
  - Windows
  - Emergency Exit
  - Upholstery
  - Seats/Seat Belts
  - Trim System
  - Rudder Pedals
  - Control
    - Wheel
    - Column
    - Pulleys
    - Cables
  - Aileron
    - Sprocket
    - Chain
  - Lights
    - Landing
    - Navigation
    - Cabin
    - Instrument
  - Instrument and Lines
  - Parking Brake
  - Pitot Tubes and Lines
  - Static Vents
  - Fuel
    - Crossfeed Line
    - Selector Valve
    - Crossfeed Valve
    - Emergency Shutoff Valve
    - Quantity Indicating Switches
  - Heater Fuel Pump
  - Heat Ducts
  - Emergency Hydraulic Hand Pump
  - Oxygen
    - Outlets
    - System Components
- Fuselage Wing and Empennage Group
  - Battery
    - Box
    - Cables
  - Electronic Installations
  - Antenna Mounts
  - Anticollision Lights
  - Emergency Locator Transmitter (ELT)
    - Battery
    - Transmitter
  - Heater
  - Brake Reservoir
  - Receiver-Dehydrator
  - Air Conditioning System
  - Hydraulic Power Pack
    - Fluid Level
    - Lines
  - Pneumatic Deicers
  - Fuel Valves and Gauges
  - All Lines
  - Fuel Cells
    - Nylon Support Cords
    - Baffles
    - Flapper Valve
    - Vents
  - Engine Exhaust Shields
  - Engine Mount Structure
  - Bulkhead and Stringers
  - Vertical Fin
    - Surface
    - Attachment
  - Rudder
    - Tab Hinges
    - Horns
    - Attachments
    - Tab Hinge Bolt
    - Balance Weights
    - Trim
    - Cables
  - Elevator
    - Surfaces
    - Tab Hinges
    - Horns
    - Attachments
    - Stop Screws and Nuts
    - Balance Weights

        - Trim
        - Balance Spring
        - Cables
    - Horizontal Stabilizer
        - Surfaces
        - Attachments
    - Aileron
        - Cables
        - Tab Hinges
        - Balance Weight and Arm
        - Trim
        - Bellcranks
        - Hinge Bolts
    - Flaps
        - Attachments
        - Transmission
        - Actuator Cable
        - Actuator Motor
        - Hinge Bolts
    - Wing Attachments and Bolts
    - Engine Mount Attaching Structure
    - Landing Gear
        - Oleo Struts
        - Nose Gear Steering Control
        - Wheels
        - Tires
        - Bearings
        - Brakes
        - Shimmy Damper
        - Gear Fork
        - Struts
        - Attachments
    - Surfaces, Skins and Tips
    - Landing Gear
        - Torque Links
        - Bearing Blocks
        - Down Locks
        - Torque Link Bolts
        - Gear Doors
        - Warning Horn and Lights
        - Retraction Handle
        - Retraction Solenoid
        - Actuating Cylinders
        - Position Switches
        - Lock Rods

Eleven of 59 findings in the fuselage, wing, and empennage group would require corrective action to maintain airworthiness. These findings are listed below.

- Corrosion and cracking of the battery box and tray
- ELT antenna wire severed
- Wear of fuel cell transmitter float arms
- Chafing of fuel line by engine control cable
- Damaged repairs to the ribs in the left and right gear wheel wells
- Cracked stringers in left and right main gear wheel wells
- Damaged repair to floor plate gusset assembly in nose baggage compartment
- Cabin floor bulkhead assembly cracked/torn
- Corrosion of the rudder torque tube
- Chafing of coaxial cable between trim control cables
- Nose gear oleo strut housing damaged

Figure 3 shows the location of the battery box assembly, and figure 4 shows corrosion of the battery box assembly as well as a 1-inch crack in the angle that retains the battery box, part number 53960-00. This defect is located at FS 2, WL 13, and body line (BL) 0. The location of the ELT antenna, part number 597 257, is shown in figure 5, while the ELT antenna, located at FS 261, WL 40, and RBL 6, is shown in figure 6. The wire connecting the antenna to the transmitter was severed, rendering the ELT system ineffective. Figure 7 shows the locations of the fuel gauge assemblies, part numbers 40648-02 and 40648-03. Wear was observed on the float arms, part number 54751-02, with a depth of 0.1 to 0.3 times the arm diameter, as shown in figure 8. The damage on the float arm shown in figure 9 is representative of all fuel gauge assemblies observed on this airplane. The wear locations coincide with the arm travel stops of the assemblies, implying damage due to repetitive contact. The location of an engine control cable, part number 24894-18, observed chafing of a fuel line, part number 23745-15, as shown in figure 10. Although this cable only moves inside a stationary housing, the contact between the housing and fuel line is inappropriate. The engine control cable assembly contacting the fuel line in the left nacelle at FS 110, WL 15, and LWS 64 shown in figure 11, as found on the airplane.

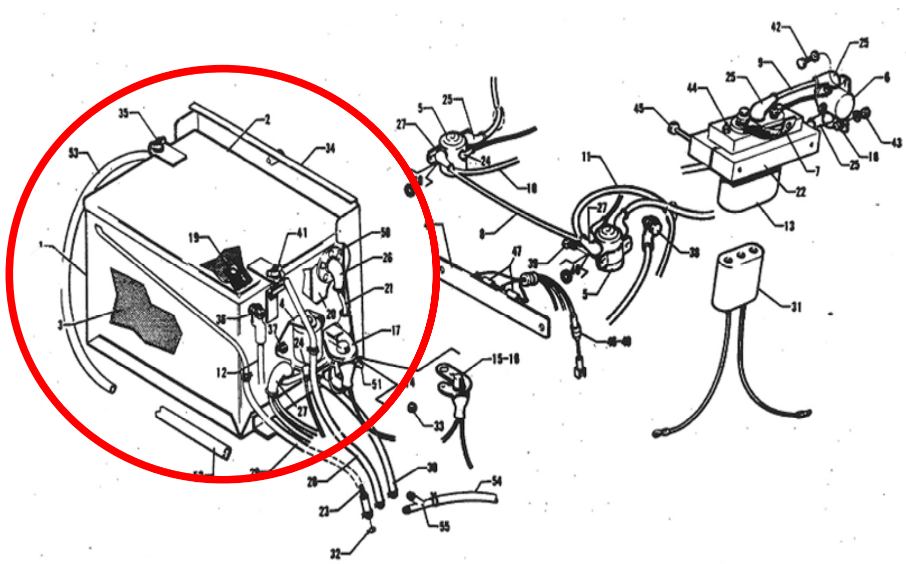
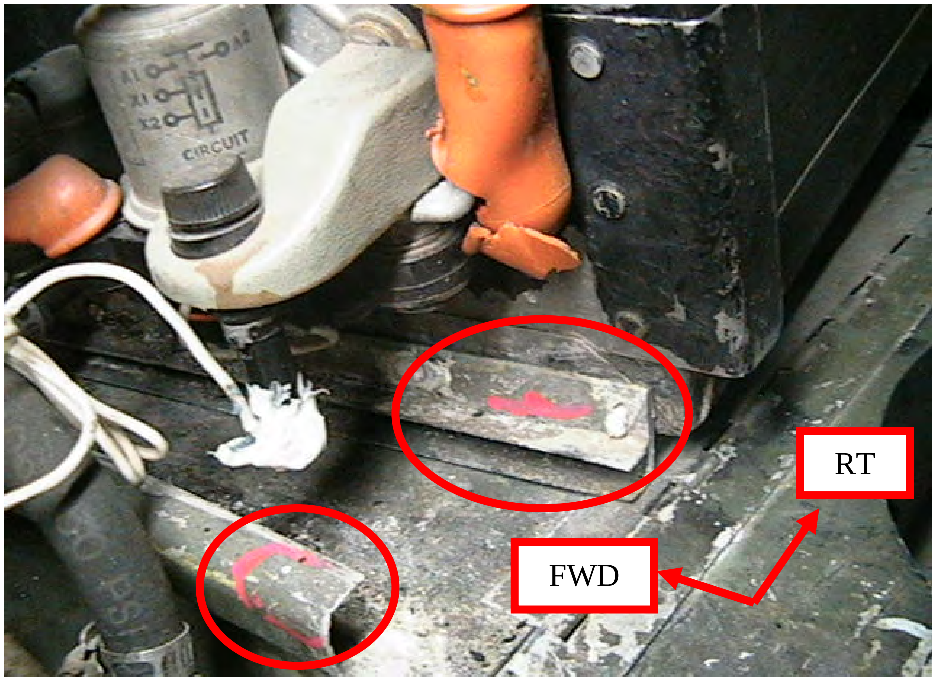


Figure 3. Location of the Battery Box Assembly



Forward (FWD)
Right (RT)

Figure 4. Corrosion and Crack on the Battery Box FS 2, WL 13, and BL 0

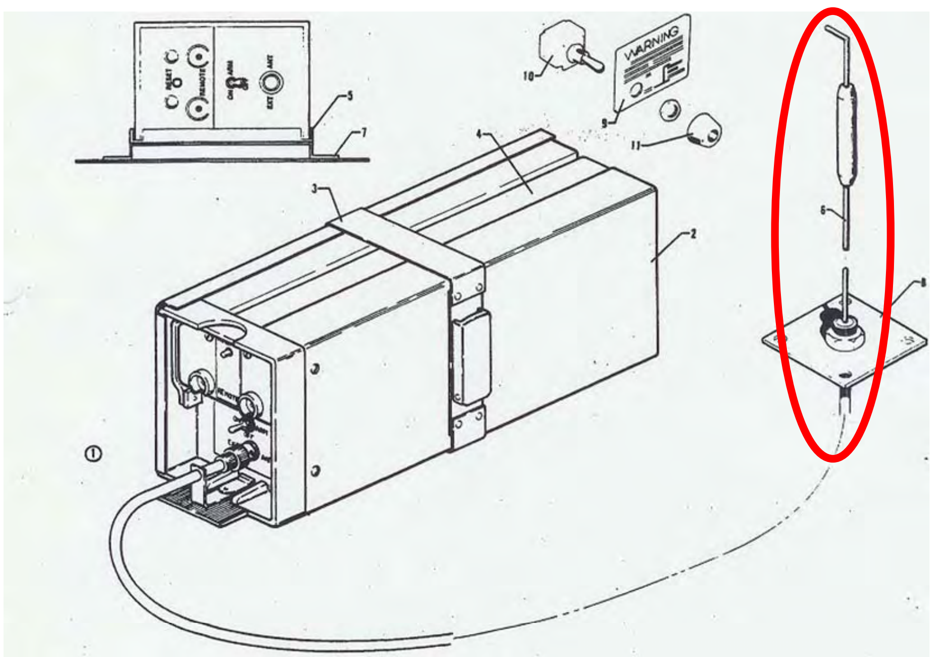


Figure 5. Location of the ELT

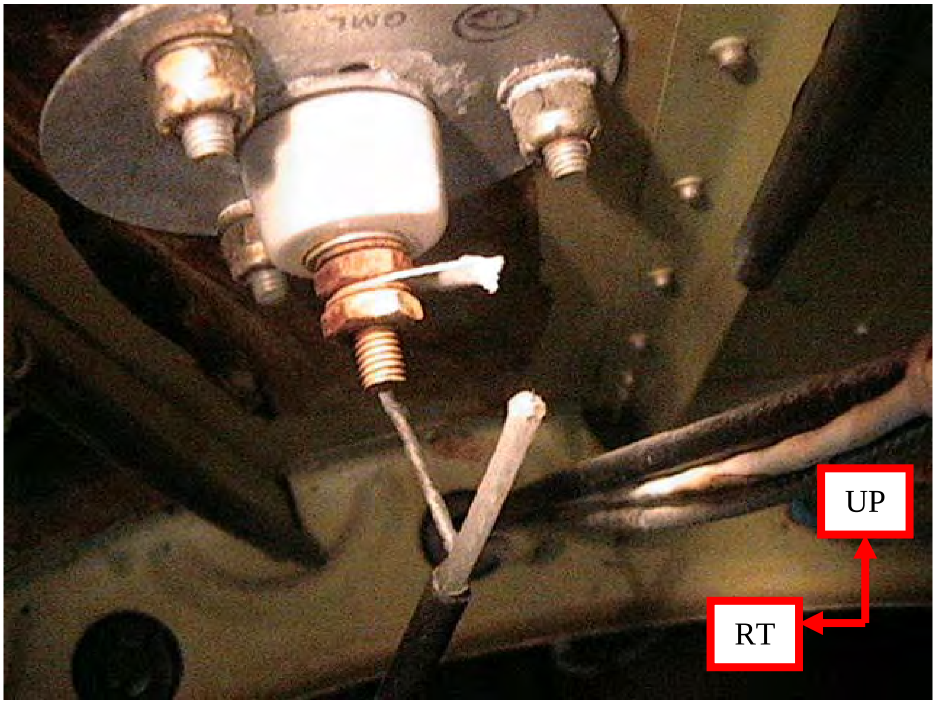


Figure 6. Severed ELT Antenna Wire FS 261, WL 40, and RBL 6

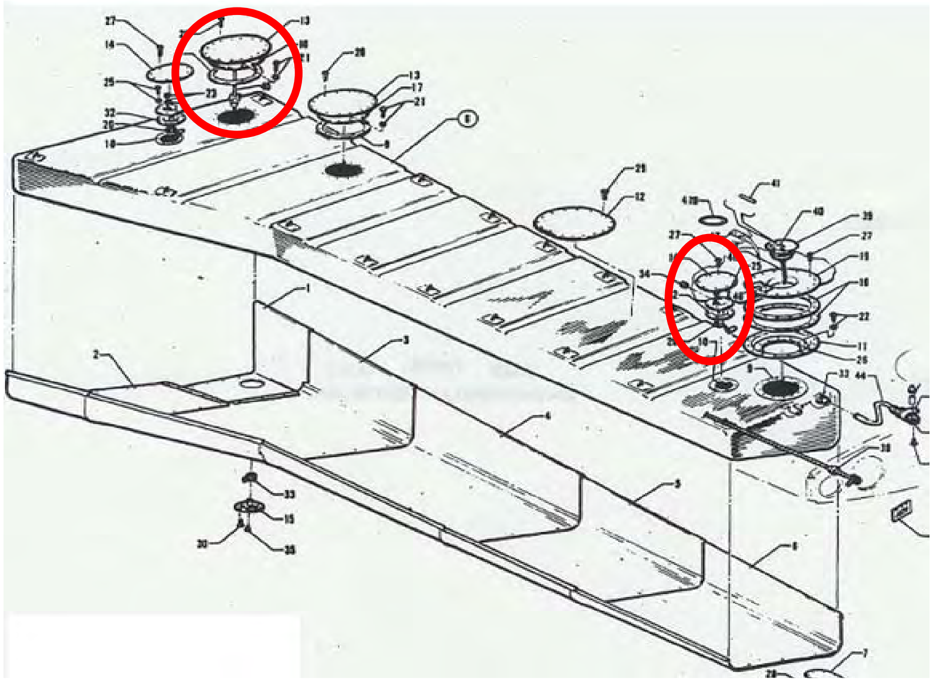


Figure 7. Location of the Fuel Gauge Assemblies

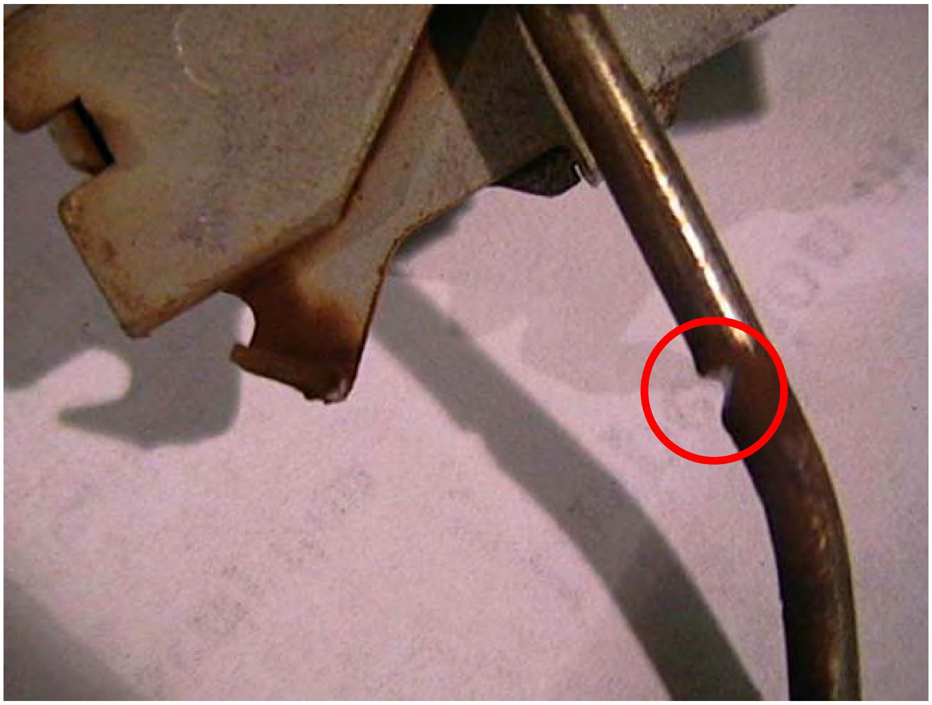


Figure 8. Worn Fuel Gauge Assembly Float Arms All Wing Fuel Cell Gauge Locations

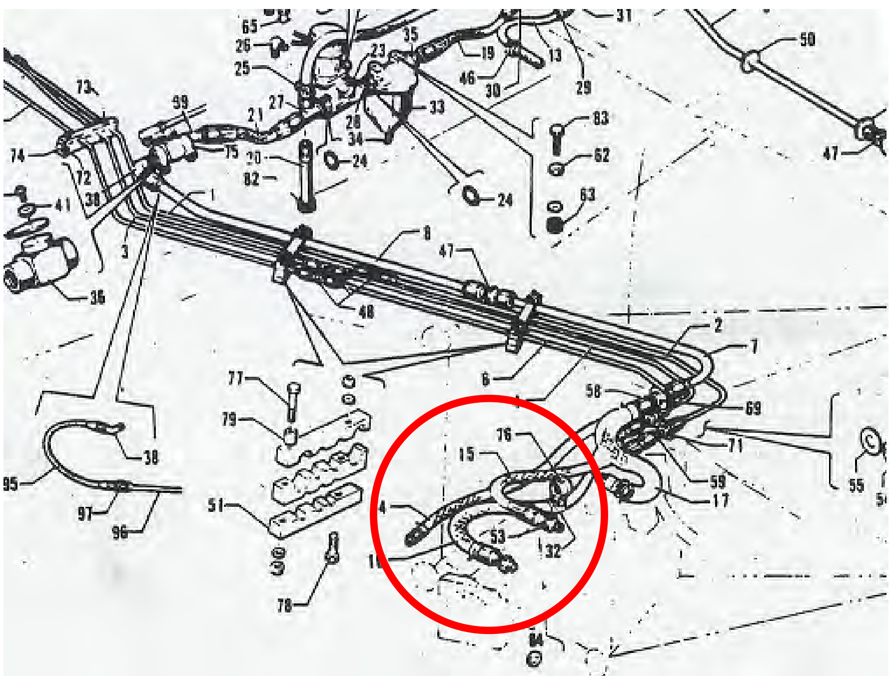
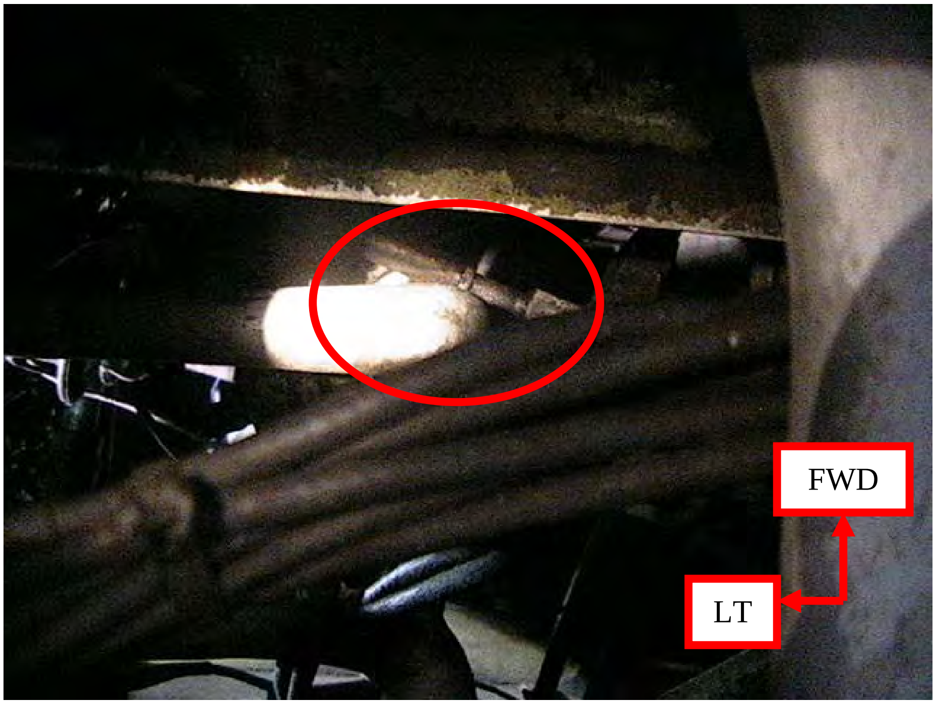


Figure 9. Location of the Engine Control Cable Contacting Fuel Line



Left (LT)

Figure 10. Contact Between Fuel Line and Engine Control Cable Assembly FS 110, WL 15, and LWS 64

The locations of the right and left main landing gear wheel well ribs and stringers are shown in figures 11 and 12, respectively, and the damage to these areas at FS 150, WL 16, and RWS 39.5 to RWS 49, and LWS 39.5 to LWS 49 is shown in figures 13 and 14. Information obtained during the survey of the airplane maintenance records revealed numerous instances of the wheel brake assemblies contacting the damaged areas. Repairs have been performed in both the left and right main landing gear wheel wells to correct this issue; however, these repairs have also been damaged in the same manner. Figure 15 shows the location of the floor gusset plate, part number 41278-00, located at FS 28, WL 13, and RBL 12, and figure 16 shows cracking and damage at this location. Figure 17 illustrates the location of the lower center bulkhead assembly, part number 41233-03. Tearing past the stop drilled cracks on the left and right sides of the lower center bulkhead assembly at FS 162.6, WL 16, RBL 6, and LBL 6 was observed and is shown in figure 18.

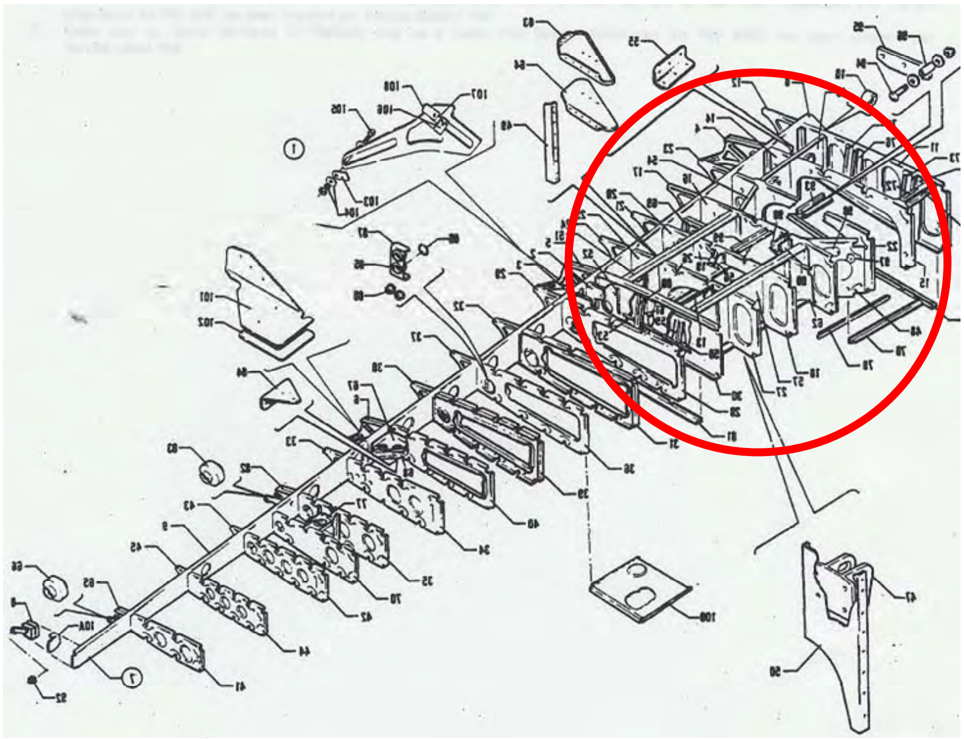


Figure 11. Location of Damage Observed on the Right Main Landing Gear Wheel Well Structure

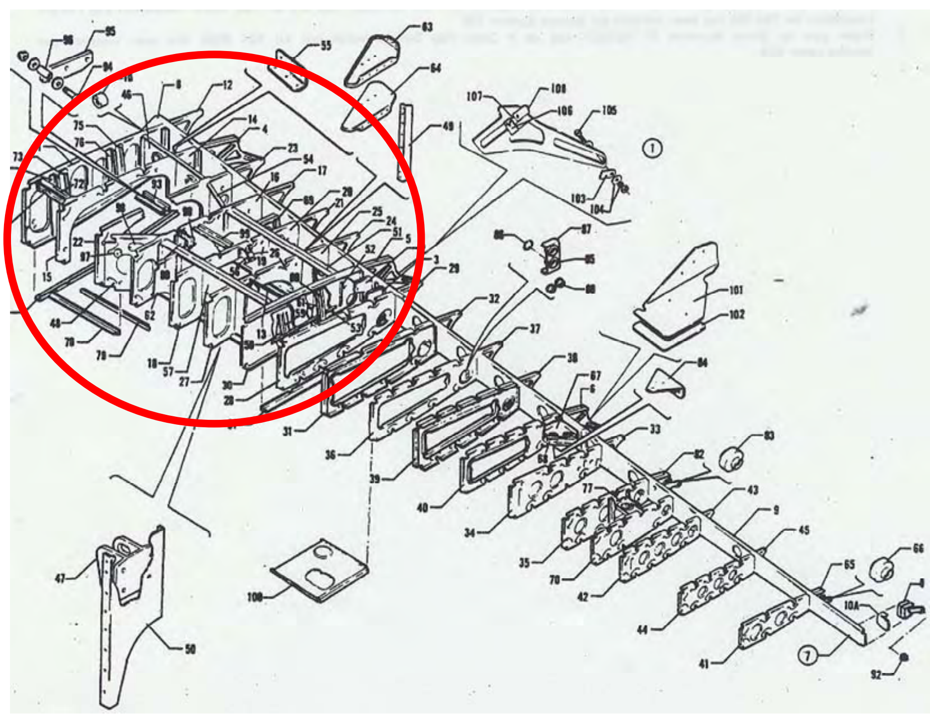


Figure 12. Location of Damage Observed on the Left Main Landing Gear Wheel Well Structure

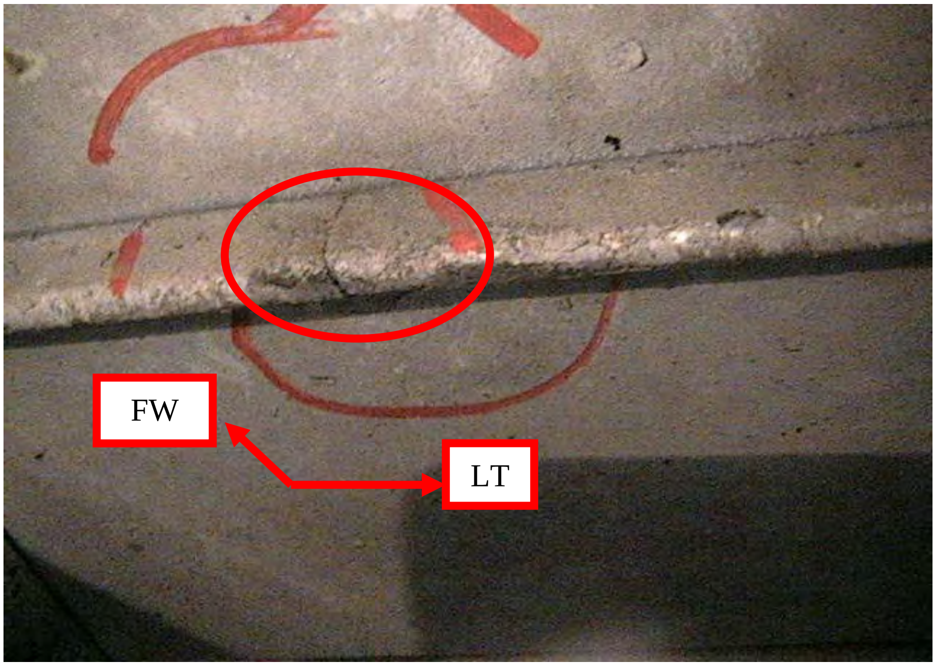


Figure 13. Damaged and Cracked Right Wing Stringers in the Main Gear Wheel Well FS 50, WL 16, and RWS 39.5 to RWS 49

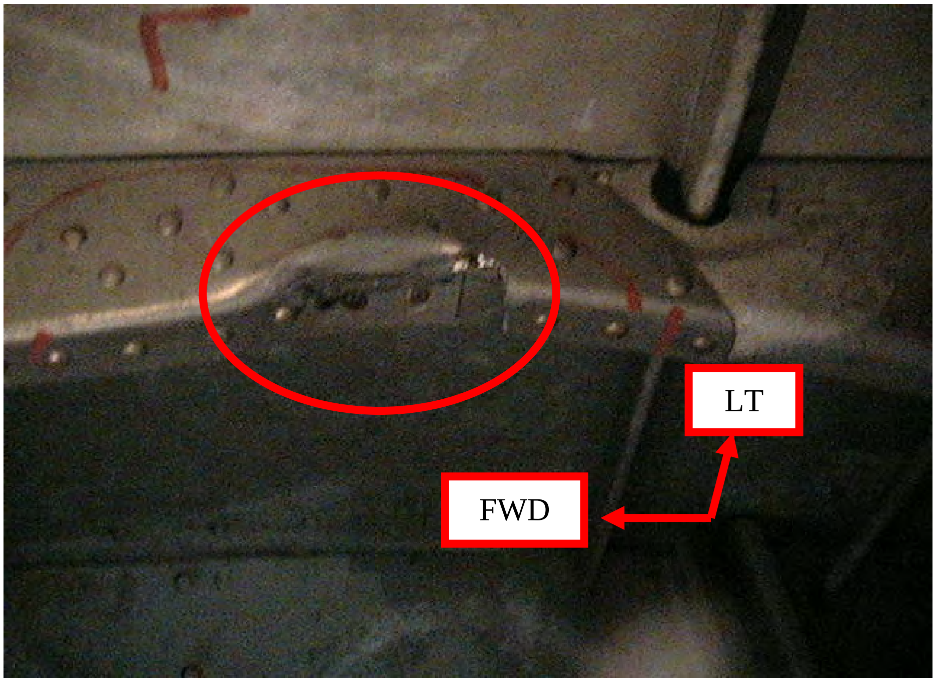


Figure 14. Damaged Left Wing Ribs in the Main Gear Wheel Well FS 150, WL 16, and LWS 39.5 to LWS 49

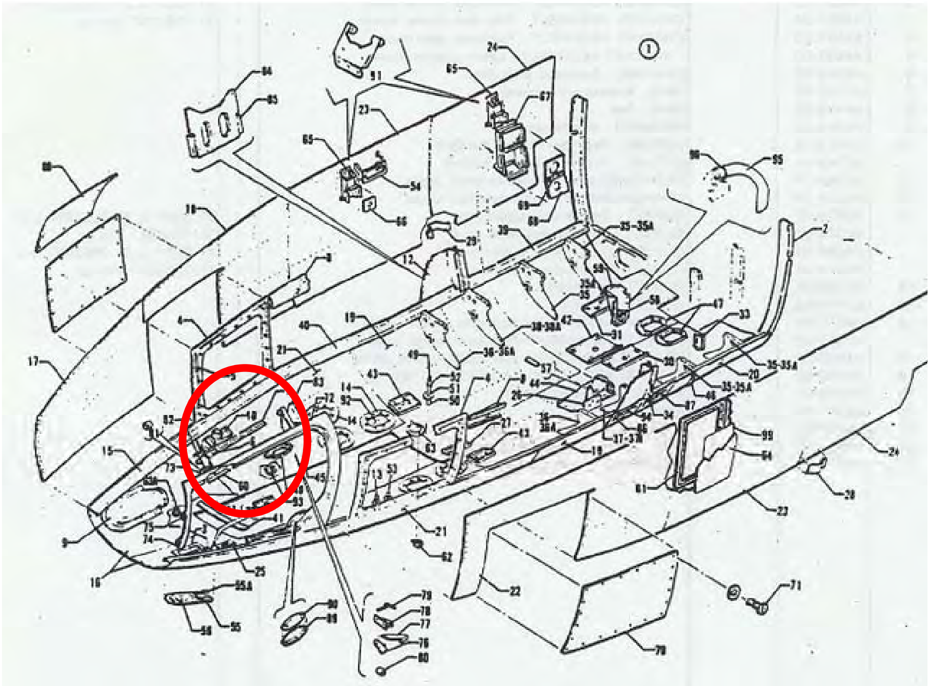
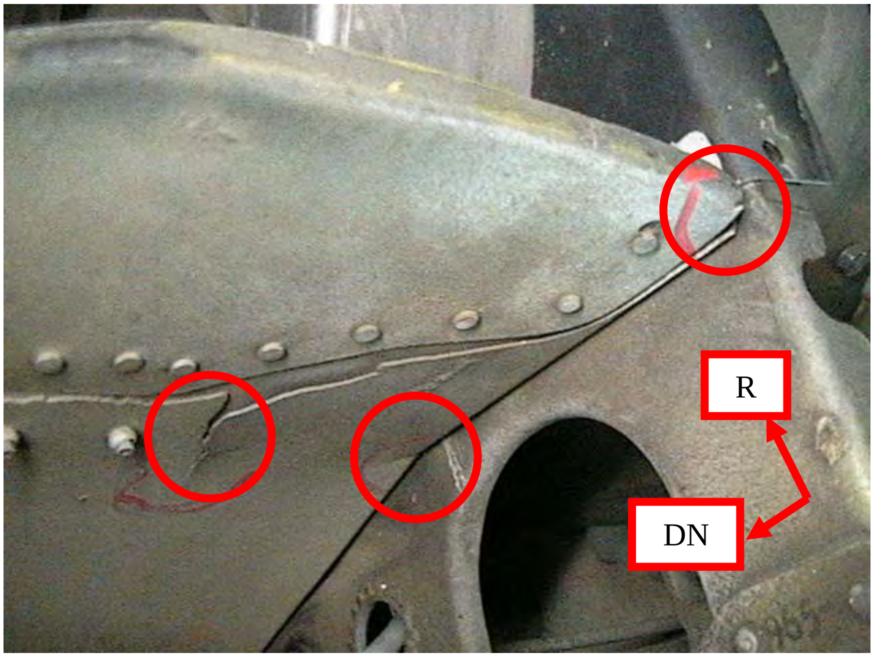


Figure 15. Location of Floor Gusset Plate FS 28, WL 13, and RBL 12



DN = Down

Figure 16. Cracks in the Floor Gusset Plate Assembly FS 28, WL 13, and RBL 12

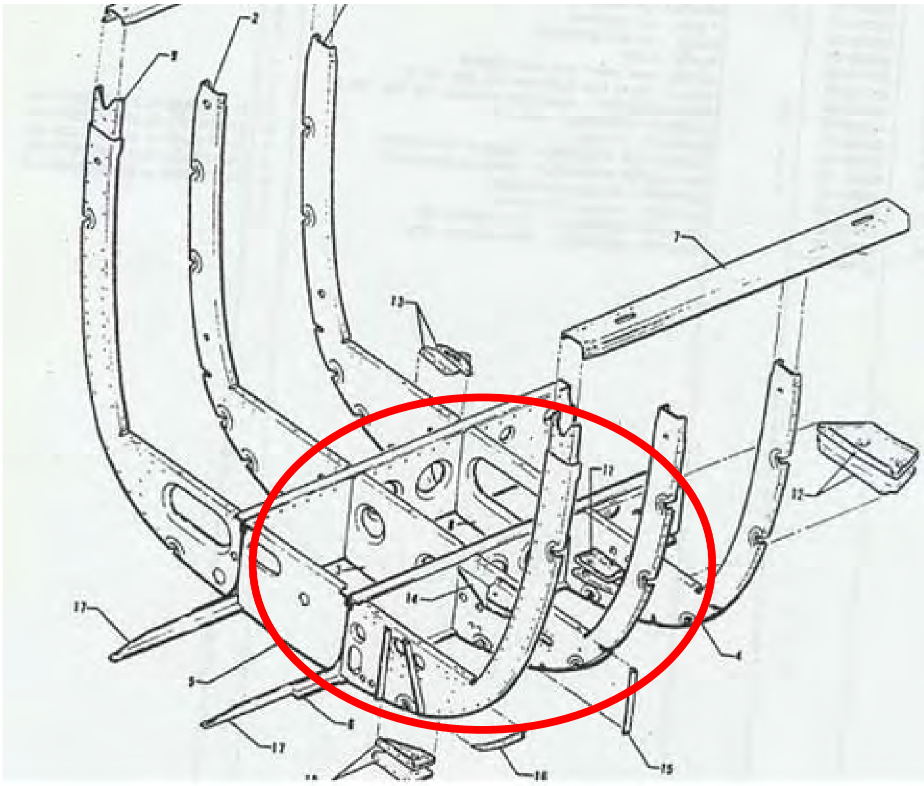


Figure 17. Location of Damage on the Lower Center Bulkhead

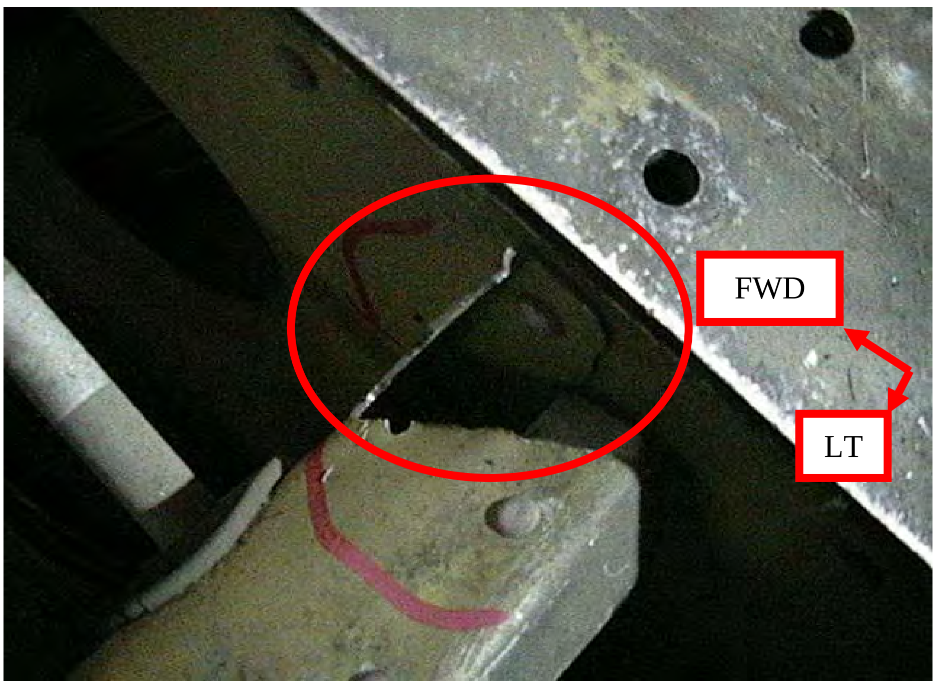


Figure 18. Torn Lower Center Bulkhead Assembly FS 162.6, WL 16, RBL 6, and LBL 6

The location of the rudder torque tube, part number 40040-09, is shown in figure 19. Figure 20 shows corrosion of the rudder torque tube at FS 342.25, WL 20, and BL 0. Figure 21 shows the location of a chafed coaxial cable, while figure 22 shows chafing of the coaxial cable by the aileron trim tab control cables at FS 84, WL 17, and BL 0. The location of the nose landing gear oleo strut housing, part number 40273-00, is shown in figure 23. A 1-inch crack, located at FS 25, WL 32, and RBL 2, began near the steering stop, as shown in figure 24. The survey of the airplane maintenance records revealed that numerous reports have been made of similar damage fleet-wide due to improper towing procedures and exceeding the turn limitations of the airplane.

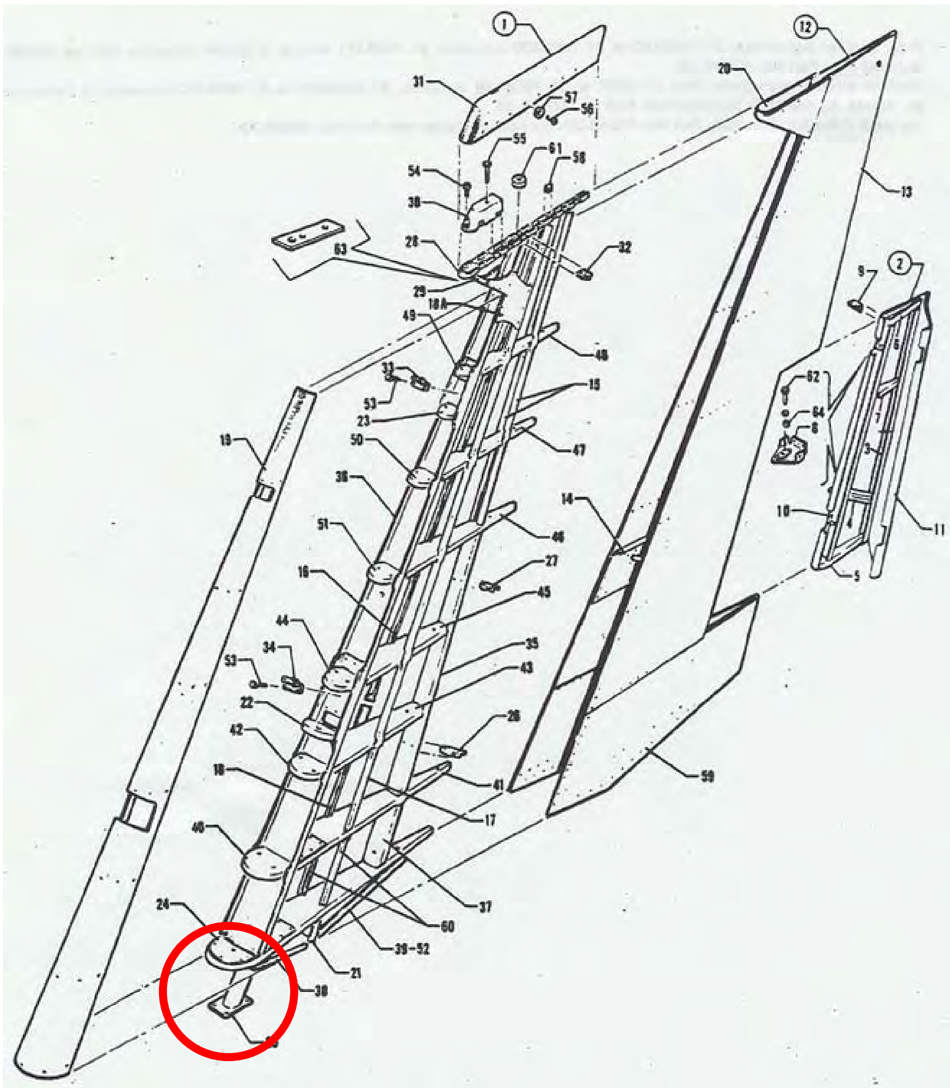


Figure 19. Location of the Rudder Torque Tube

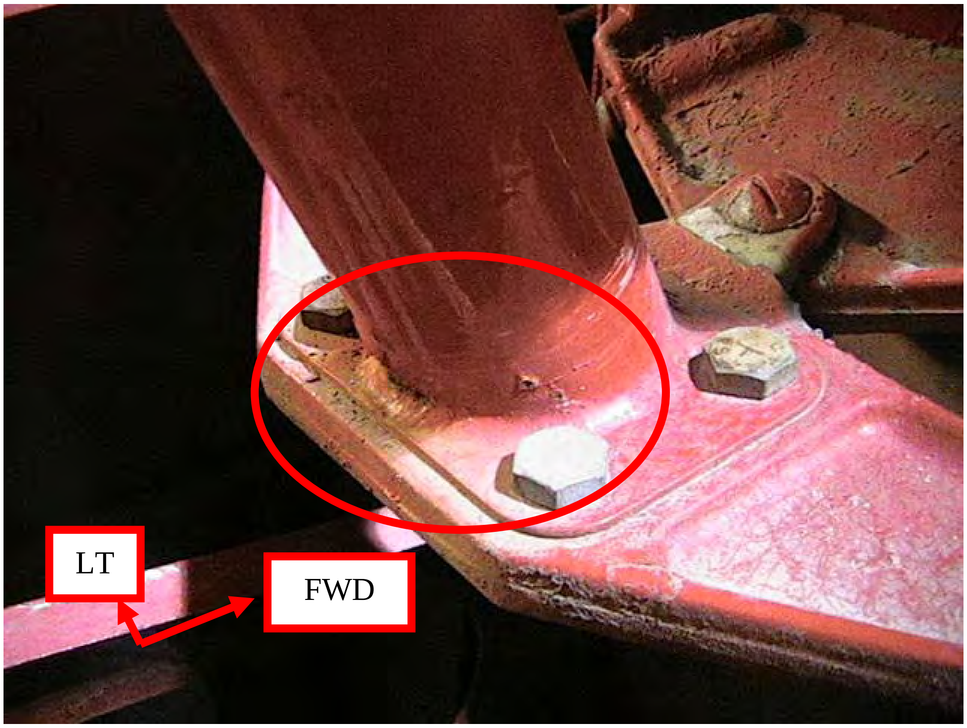


Figure 20. Corrosion on the Rudder Torque Tube FS 342.25, WL 20, and BL 0

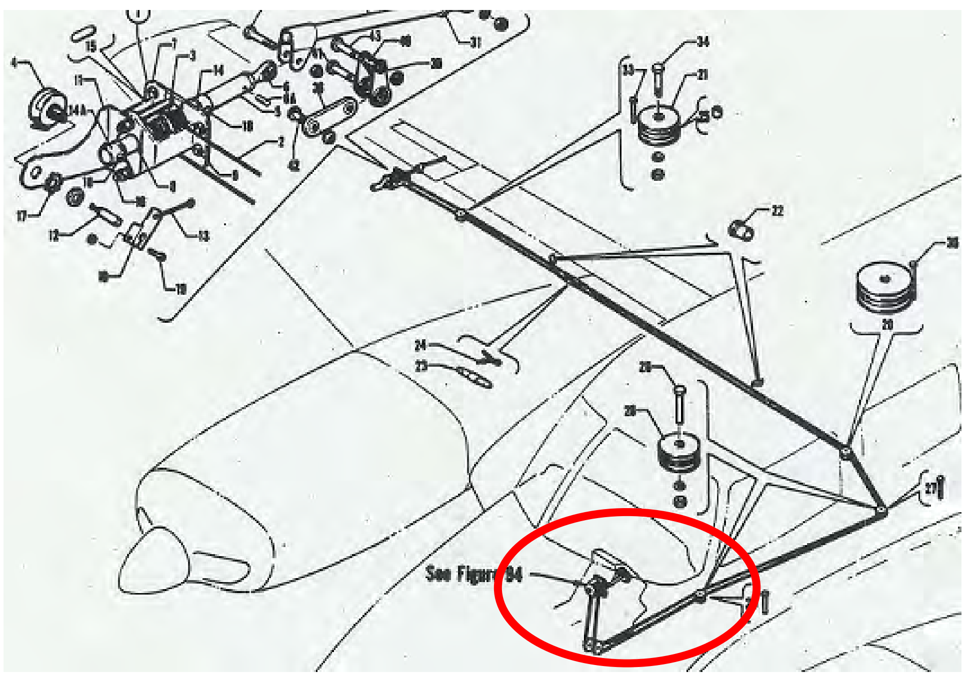


Figure 21. Location of Coaxial Cable Chafed by the Aileron Trim Tab Cables

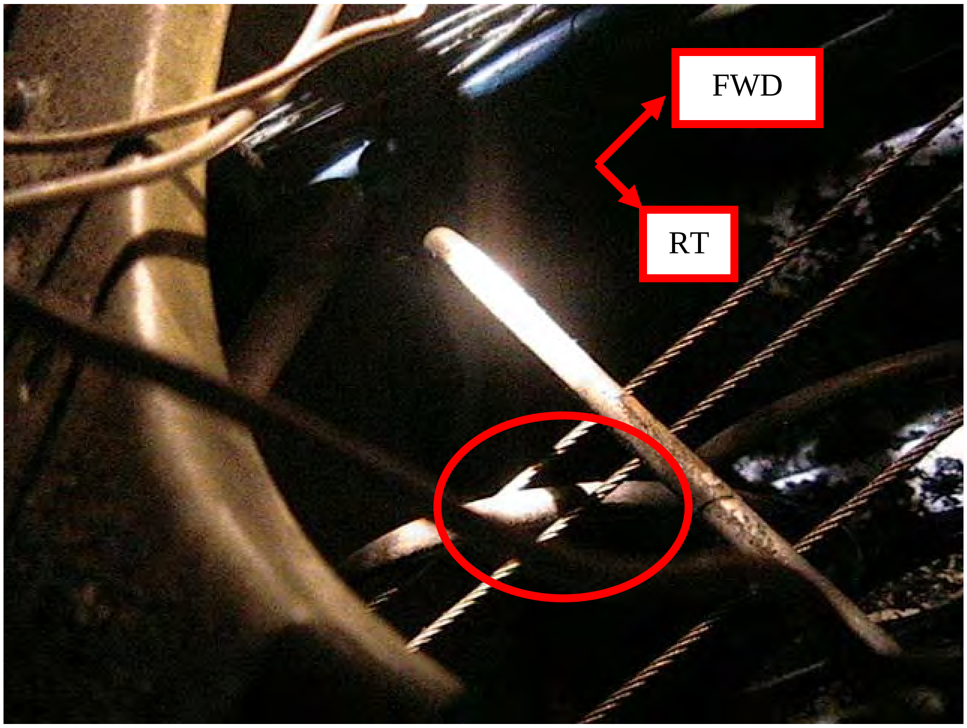


Figure 22. Chafing of a Coaxial Cable by Trim Tab Cables FS 84, WL 17, and BL 0

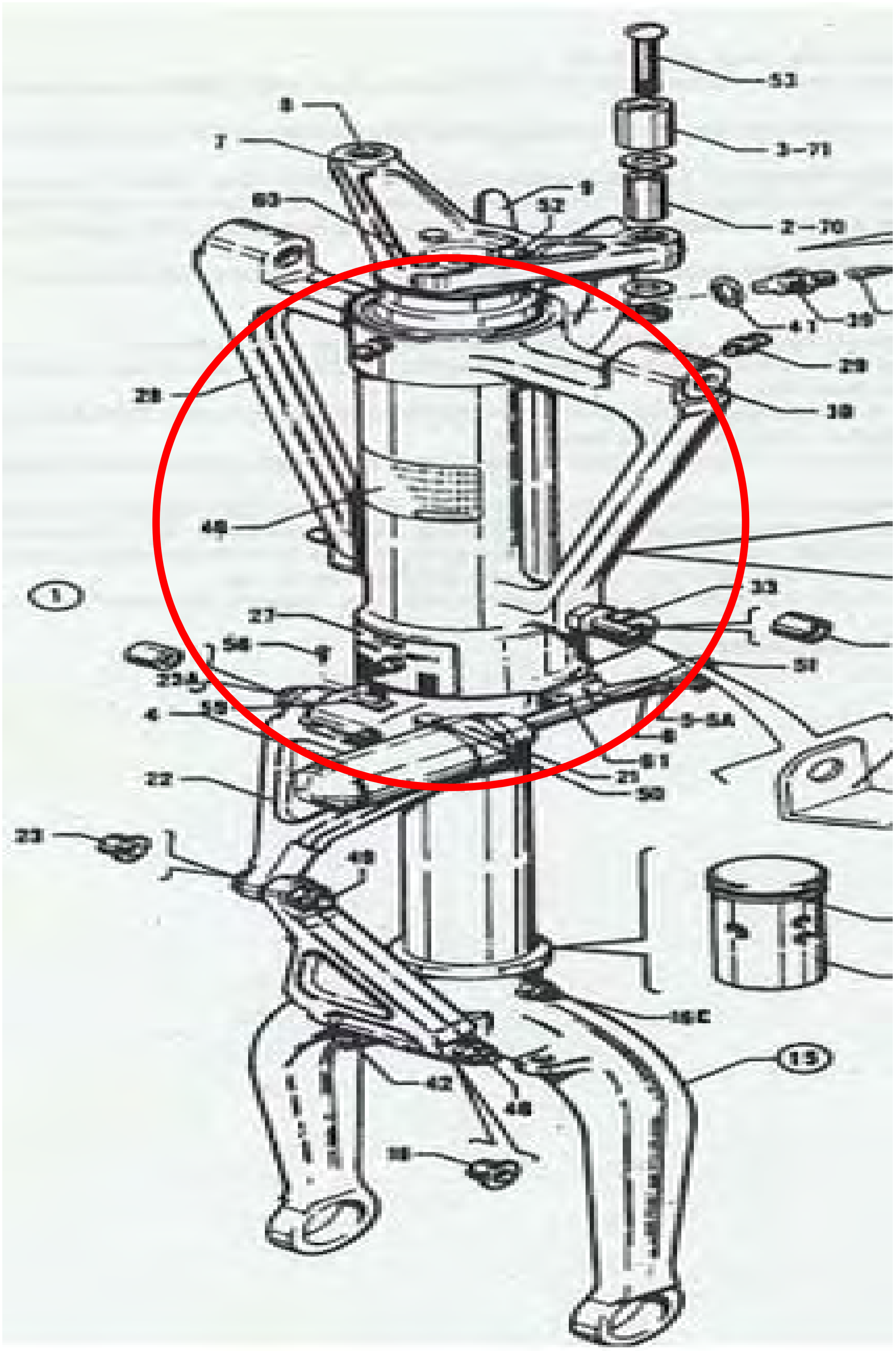


Figure 23. Location of the Nose Landing Gear Strut Housing

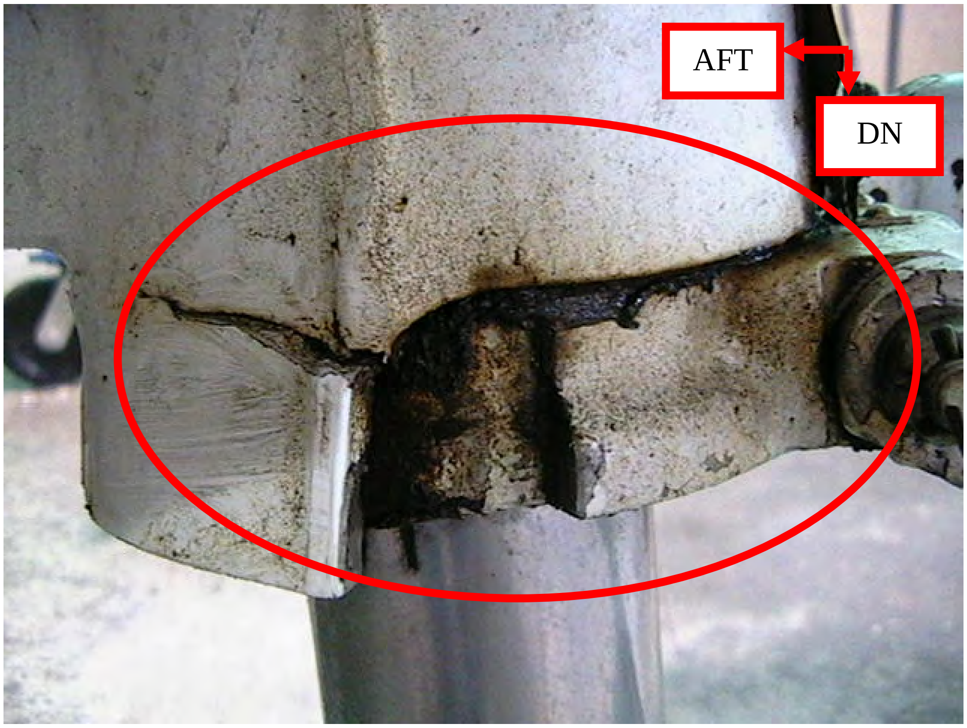


Figure 24. Nose Landing Gear Strut Housing Cracked FS 25, WL 32, and RBL 2

2.3.3 Airplane Wiring Routine Visual Inspections.

The purpose of performing the routine visual inspections on the wiring systems of this airplane was to check the conditions of the wires for any defects or flaws, which could be hazardous to the airplane in flight. This is the first step performed prior to any other wiring inspections that are performed to assess the maintenance activities performed on the airplane.

The routine visual inspections were performed without disturbing the condition of the wires in the airplane. The inspections were performed to assess the general wiring condition (chafing, rubbing, burning, or tearing of the wires), installation defects, terminations, connectors, groundings, and circuit breakers. The condition of the wires was documented and photographed during the inspection process. The most frequent issues found in the routine visual inspections were broken terminals, chafing, rubbing, cutting, exposed shield, exposed inner conductor, cutting though the outer insulation, improper terminations of end terminals, improper termination of wires (no cap), unused wires improperly stowed, and corroded terminals.

2.3.3.1 Inspection Areas.

The airplane was divided into eight zones to perform the routine visual inspections on the wiring components. A location code was assigned to each zone for documentation convenience, as shown in table 10. Each zone represented a wiring system or part of a wiring system of the airplane. The initial conditions of each zone were observed and recorded.

Table 10. Definition of Airplane Zones for Routine Wiring Inspections

Zone	Location
1	Nose compartment
2	Crew compartment
3	Passenger compartment
4	Empennage
5	Left wing
6	Right wing
7	Left nacelle
8	Right nacelle

2.3.3.2 Inspection Codes

The inspection codes were designed in accordance with the wiring inspection/practices training presentation of the FAA academy. The coding was adopted so that it would be less complicated while tagging various defective locations on a particular wire. The defect codes and their descriptions are given in table 11.

Table 11. Wiring Defect Codes

Type	Defect Description	Defect Code
Wiring conditions	Rubbing/chafing of outer insulation	01A
	Cutting through outer insulation	01B
	Exposed shield	01C
	Damaged shield	01D
	Chafing/cutting of inner insulation	01E
	Exposed inner conductor	01F
	Damaged inner conductor	01G
	Heat damage	01H
	Fluid/chemical/dust contamination	01J
	Corroded shield/conductors	01K
	Label illegible	01L
	Others	01X
Installations	Inadequate clearance to structure	02A
	Improper wire riding on other wire bundle	02B
	Improper bend radius (10 by wire/bundle diameter)	02C
	Missing/deteriorated ties	02D
	Missing/deteriorated grommets	02E
	Improper clamp condition/size	02F
	Excessive slack/sag between clamps	02G
	Excessive strain on wires	02H
	Improper T or Y breakout	02J
	Repaired wires	02K

Table 11. Wiring Defect Codes (Continued)

Type	Defect Description	Defect Code
	Improperly labeled	02L
	Unused wires improperly stowed	02M
	Other	02X
Terminations	Loose/broken terminals	03A
	Corroded terminals	03B
	Improper grounding condition	03C
	High bonding resistance	03D
	Others	03X
Connectors	Insert damage/deterioration	04A
	Contact arcing/fretting	04B
	Missing/damaged/loose back shells	04C
	Missing hardware	04D
	Other	04X

2.3.3.3 Recording Wiring Conditions.

In each zone, all defective wires and wire bundles were tagged separately for ease of identification. Every defect or flaw observed on a wire was labeled and tagged. Each defect was identified with a number and a letter. The number categorizes each defect by zone, while the letter reflects the specific defect. The first defect in a zone is documented as finding “a” in that specific zone; the second is labeled “b” and so on. The tags placed on the defective area of the wire contain the defect identification number (zone number and finding letter), the location of the defect relative to the airplane coordinate system, and the type of defect from table 11. Photographs were taken for each defect during the inspections that clearly identify the defect and location/orientation to the airplane.

2.3.3.4 Routine Wiring Visual Inspection Results.

The routine visual inspections were performed on all accessible wires without disturbing them in the airplane. The purpose of this inspection was to determine the general condition of the wires in the nose, crew, and passenger compartments, the empennage, the right and left wings, and the right and left nacelles. Wiring defects such as rubbing/chafing of outer insulation, exposed inner conductor, damaged shield, repaired wires, contamination, cracked wires, corroded terminals, improper termination, and heat damage were documented during this inspection.

2.3.3.4.1 Nose Compartment.

The defects observed in the nose compartment included improper termination, missing/damaged grommets, which resulted in chafed insulation, and overheating and cracking of the voltage clamper. Figure 25 shows an improper termination using electrical tape, while figure 26 shows a wire with the insulation cut near the termination point. A chafed wire due to a damaged grommet is shown in figure 27. A cracked and overheated voltage clamper is shown in figure 28.

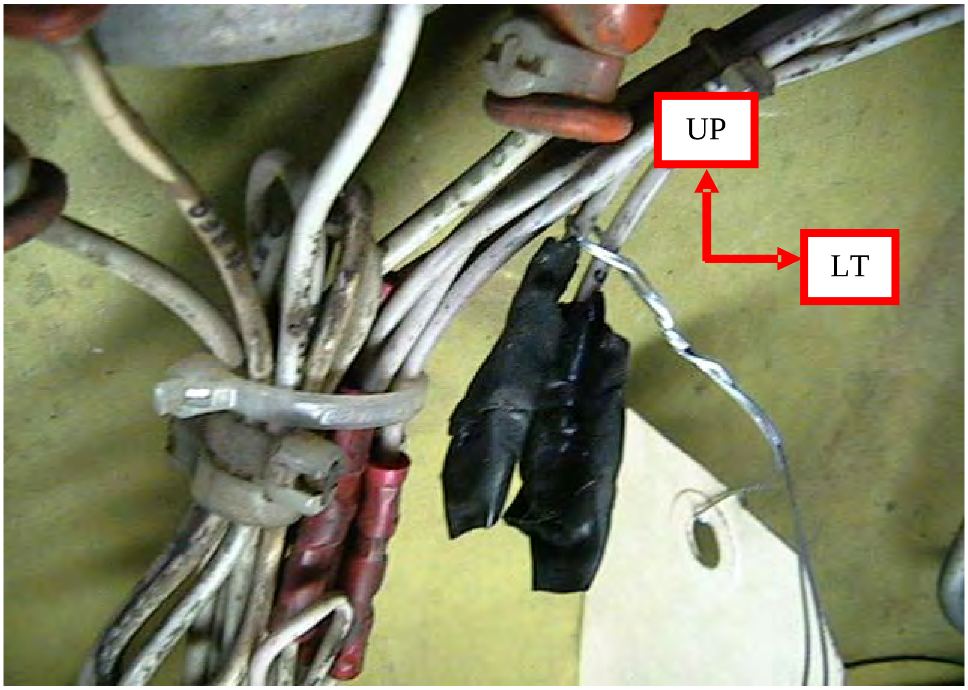


Figure 25. Improper Terminations in Nose Compartment FS 57, WL 8, and RBL 18

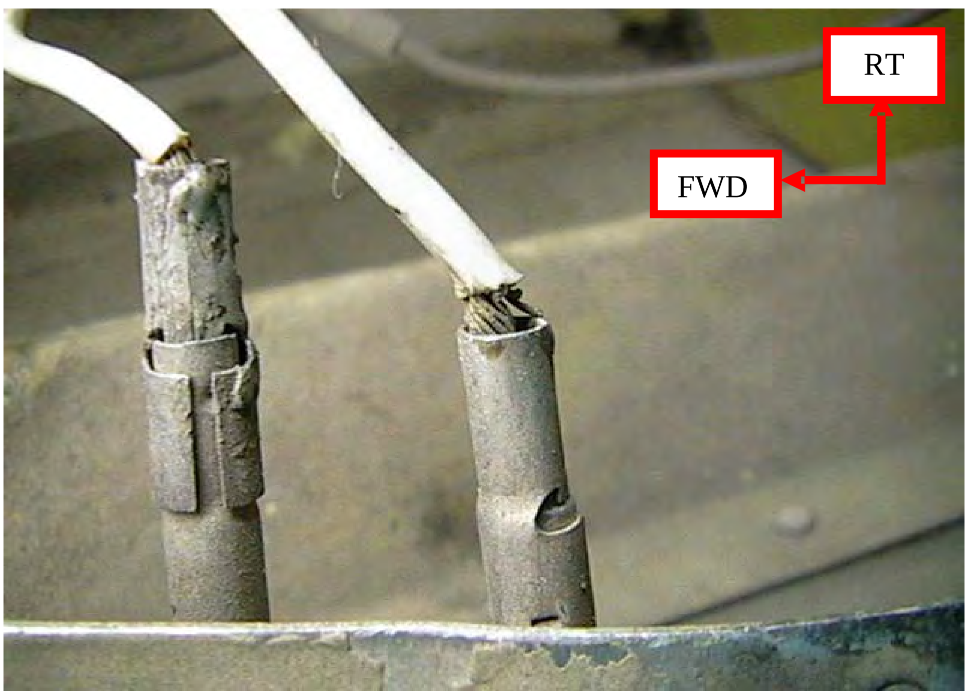


Figure 26. Cut Insulation at Terminations FS 24, WL 15, and LBL 18

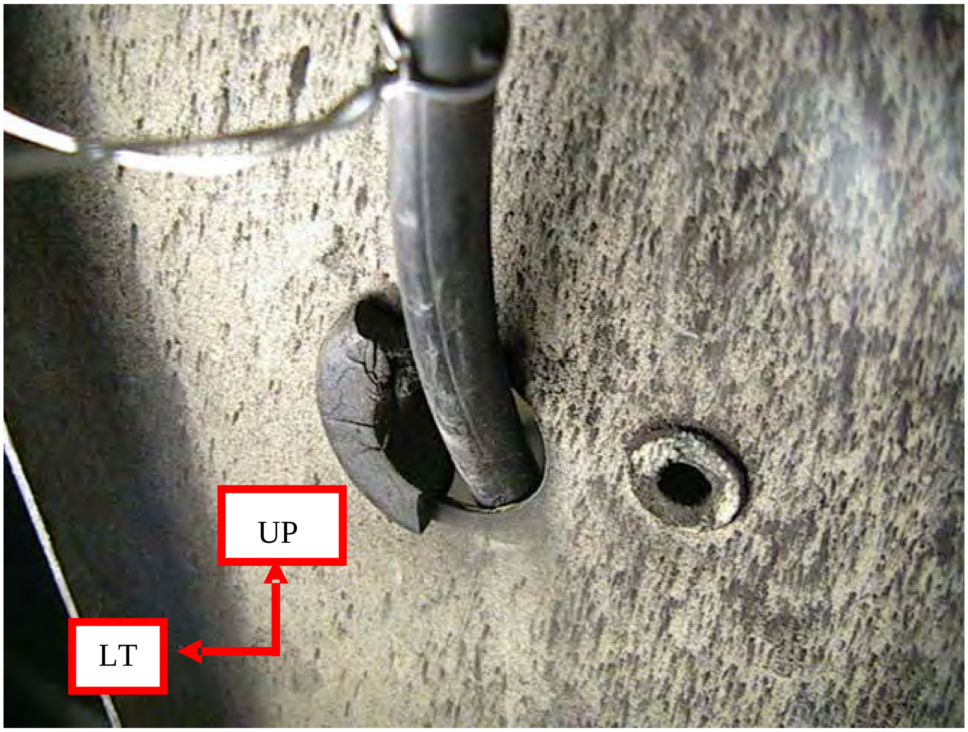


Figure 27. Damaged Grommet Resulting in Wire Chafing FS 35, WL 11, and RBL 18

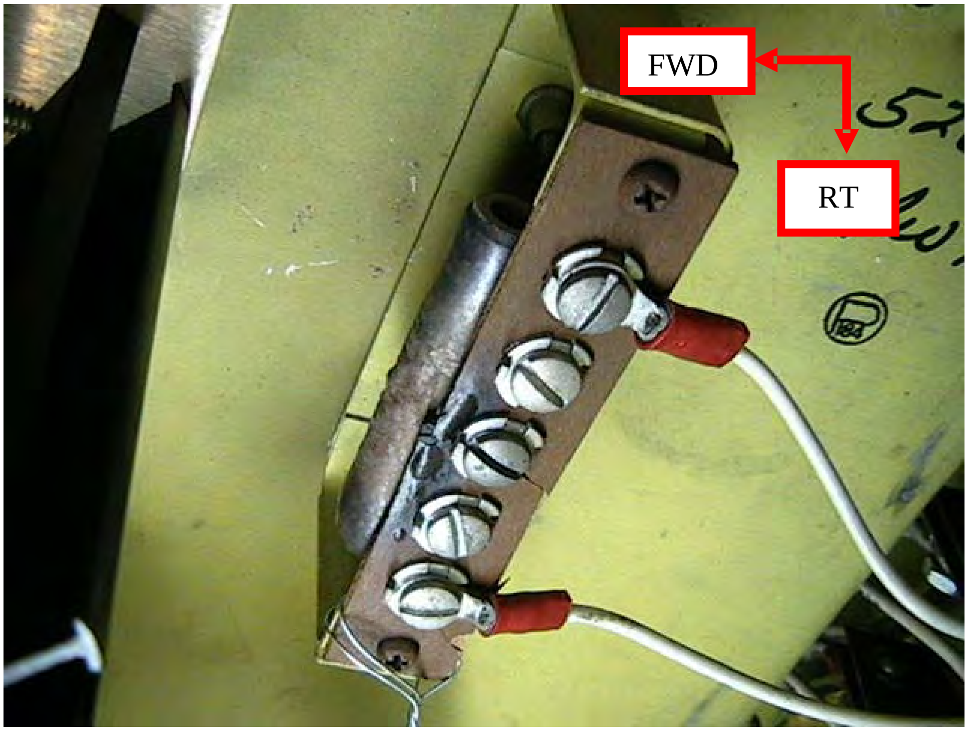


Figure 28. Cracked and Burnt Voltage Clamper FS 52, WL 0, and LBL 12

2.3.3.4.2 Crew Compartment.

Most of the defects in the crew compartment were located forward of the instrument panel and consisted of improperly terminated wires and broken wires as shown in figures 29 and 30, respectively. Other defects observed included chafing of wires, resulting in exposed conductors and damaged insulation, as shown in figure 31.

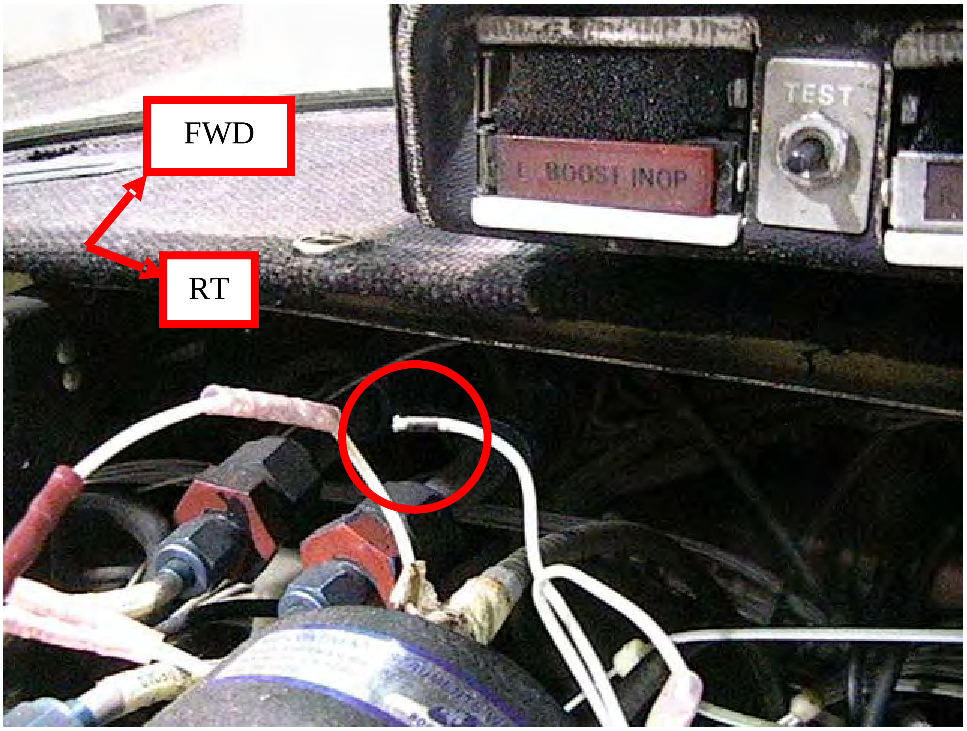


Figure 29. Improper Wire Terminations FS 76, WL 24, and BL 0

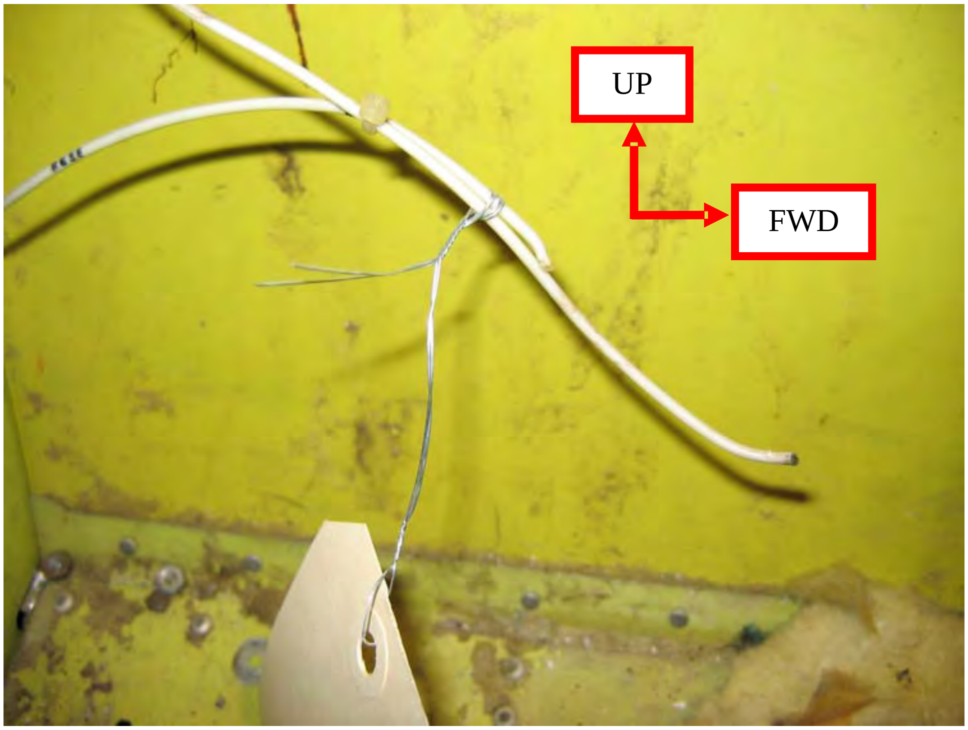


Figure 30. Broken Wire FS 116, WL 10, and LBL 20

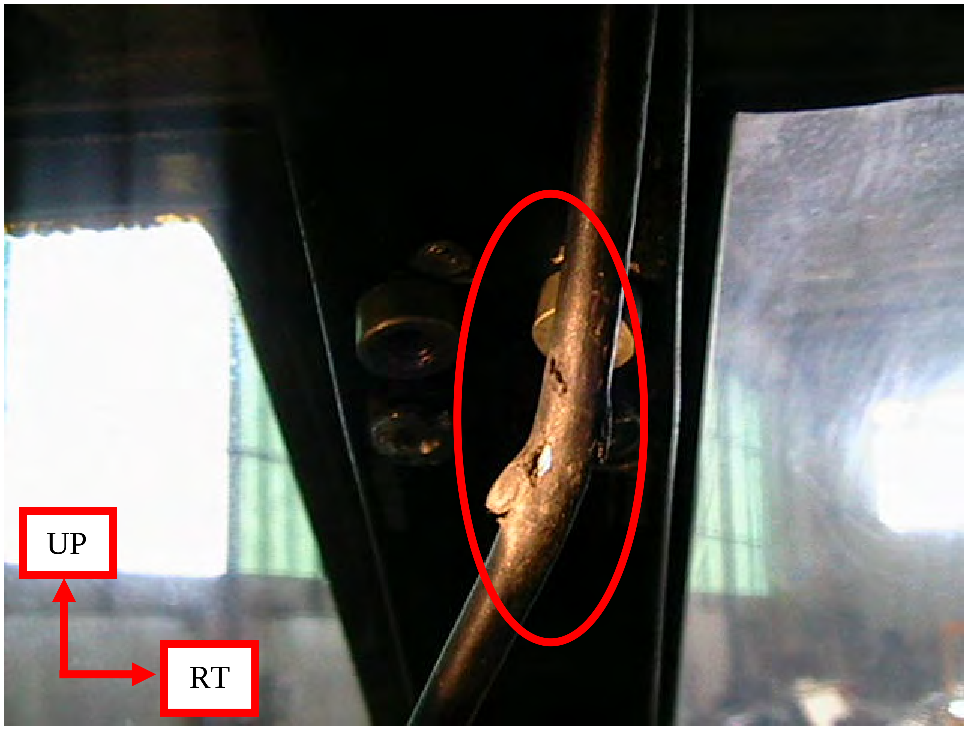


Figure 31. Damaged Wire and Insulation FS 70, WL 30, and BL 0

2.3.3.4.3 Passenger Compartment.

Wiring defects in the passenger compartment consisted of chafing and damage due to missing or deteriorated grommets in the structure as shown in figures 32 and 33, respectively. Figure 33 shows a missing grommet in a cabin frame. Eighteen instances of missing grommets in frames were noted on this airplane. A few additional defects including exceeding the wire bend radius and improper terminations were also noted.

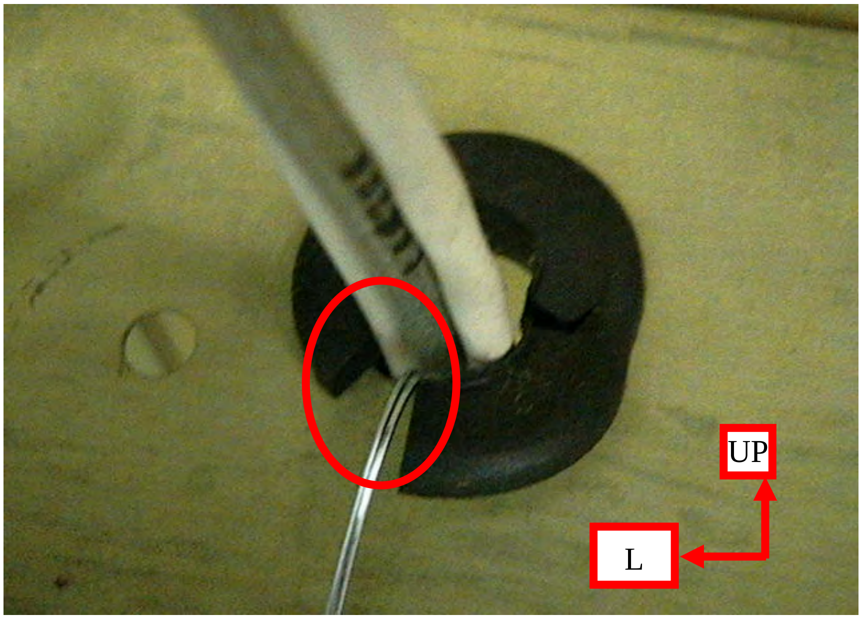


Figure 32. Damaged Grommet in Cabin Frame FS 261, WL 21, and BL 0

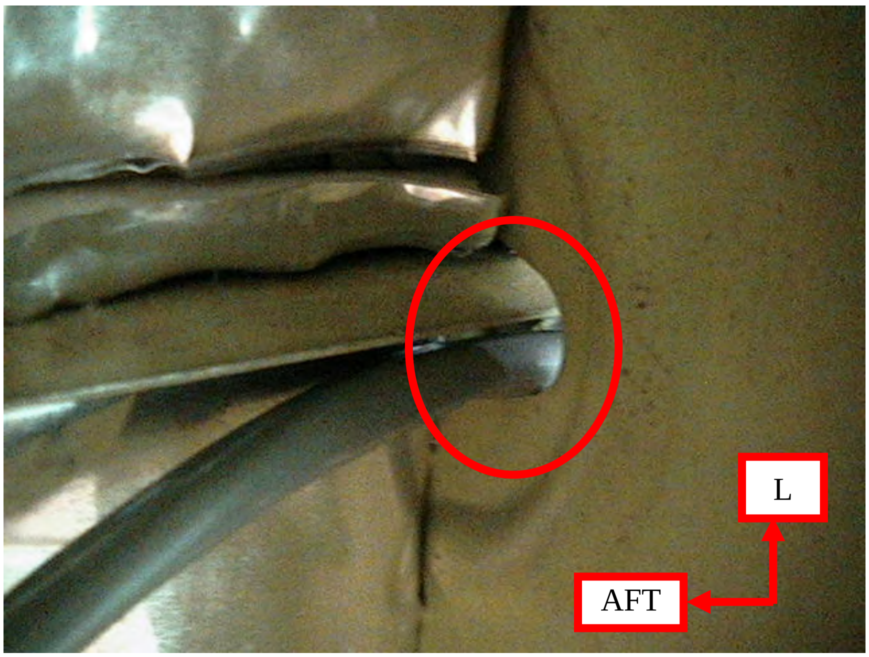


Figure 33. Missing Grommet in Cabin Frame

2.3.3.4.4 Empennage.

Findings in the empennage were limited to lack of support for wires and wire bundles, as shown in figure 34, and missing grommets that resulted in wire chafing, as shown in figure 35.

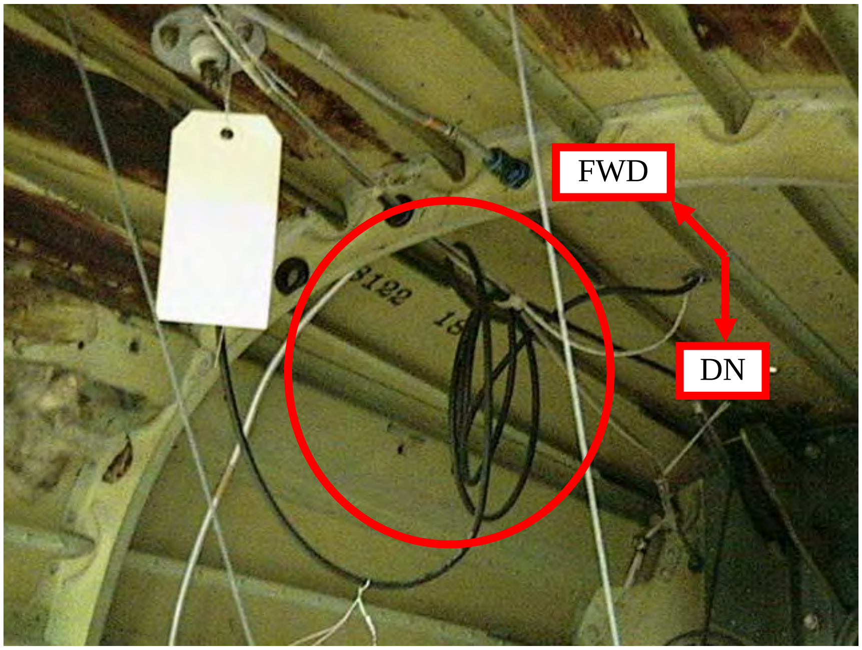


Figure 34. Improperly Supported Wires FS 298, WL 36, and BL 0

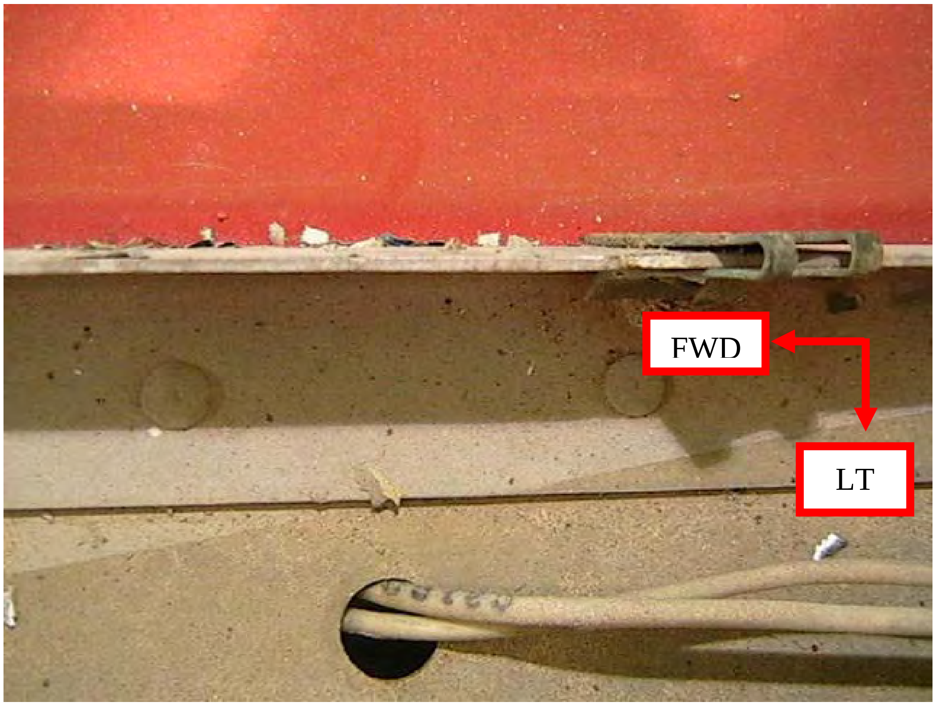


Figure 35. Chafing of Wires Due to Missing Grommet FS 311, WL 30, and BL 0

2.3.3.4.5 Left Wing.

Installation defects were numerous in the left wing zone. Wiring defects observed in the left wing included chafing, overheating, exposed inner conductor, improper clamping of wire bundles, and missing grommets that led to the chafing of wires. Figure 36 shows a wire component chafing flammable fluid lines, while figure 37 shows an overheated wire. A wire with an exposed inner conductor is shown in figure 38. Examples of improper clamping and support of wires and wire bundles observed in the left wing are shown in figure 39. The line shown in figure 39 is a deactivated de-ice system line. Missing grommets in the fuel cells caused chafing of the fuel transmitter wire insulation.

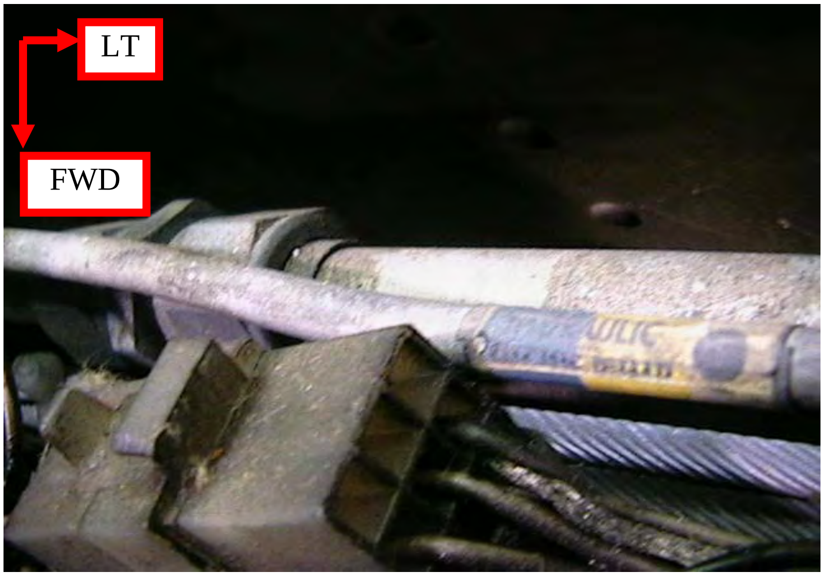


Figure 36. Chafing of Fluid Tube by Wire Connector FS 104.5, WL 12, and LWS 40

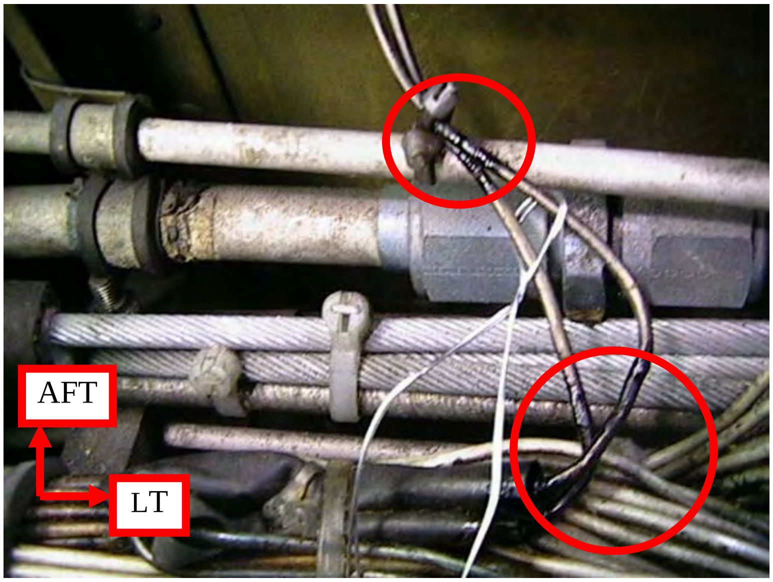


Figure 37. Overheated Wires FS 104.5, WL 12, and LWS 39

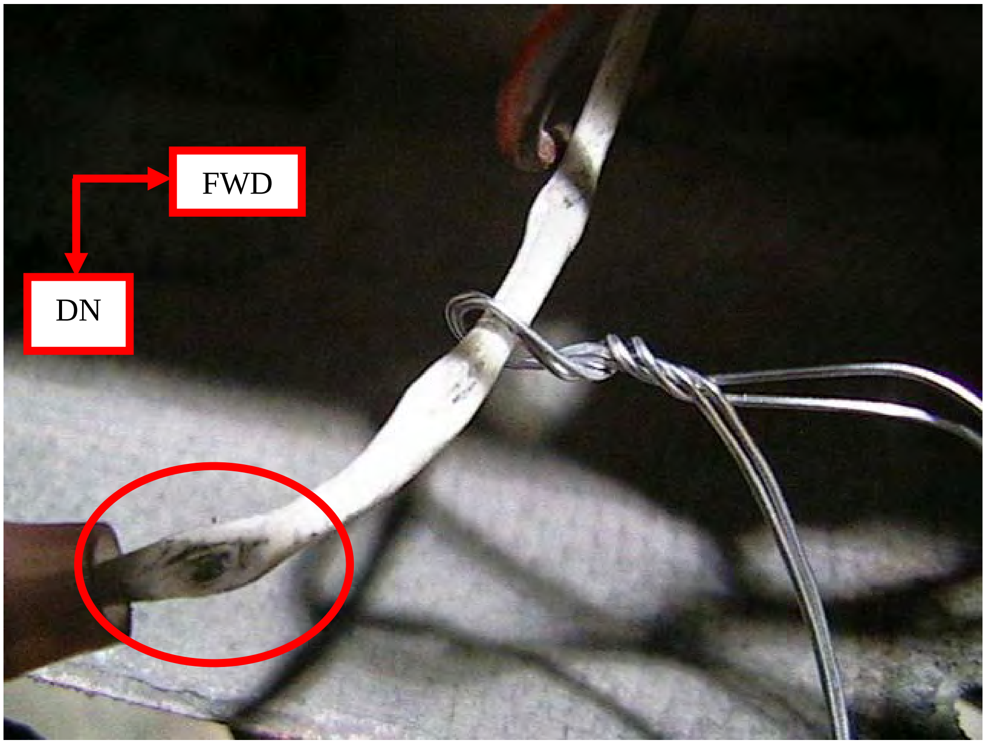


Figure 38. Exposed and Damaged Wire Inner Conductor FS 135, WL 0, and LWS 242

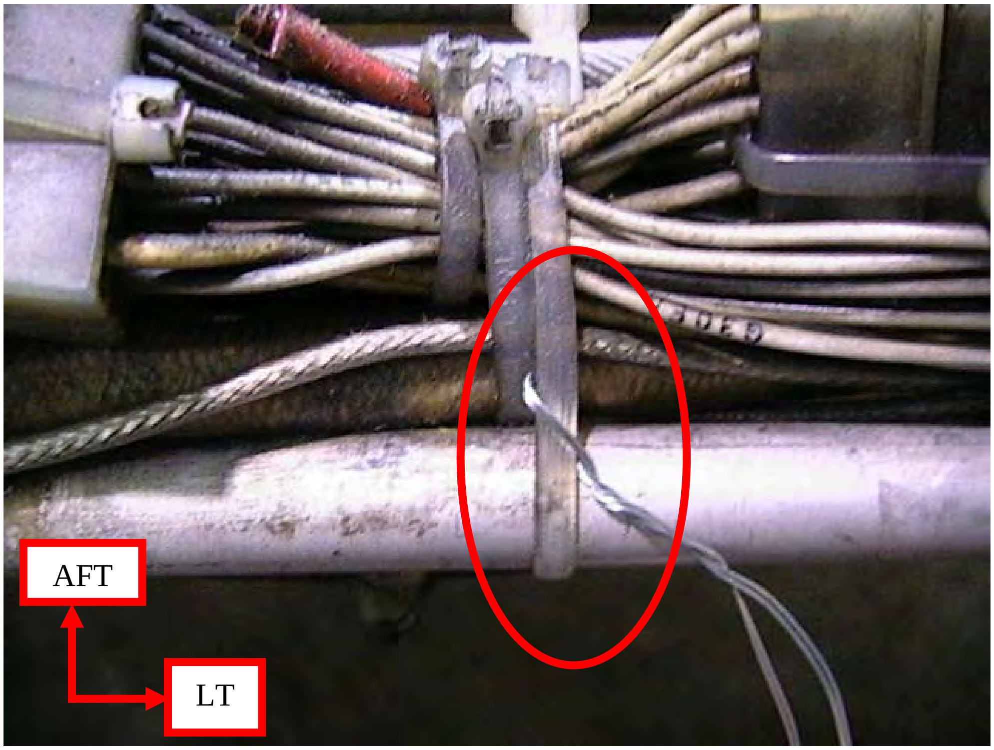


Figure 39. Improper Support of Wire Bundle to Fluid Tubing FS 106, WL 12, and LWS 45

2.3.3.4.6 Right Wing.

The inspection of the right wing revealed similar discrepancies noted during the inspection of the left wing. Wiring defects, such as improper termination, improper clamping, and missing grommets, were observed in the right wing. Figure 40 shows the improper termination of a wire, while figure 41 illustrates improper clamping of a wire. The right wing also was missing grommets in the fuel cells, which resulted in chafing of the fuel transmitter wire insulation.

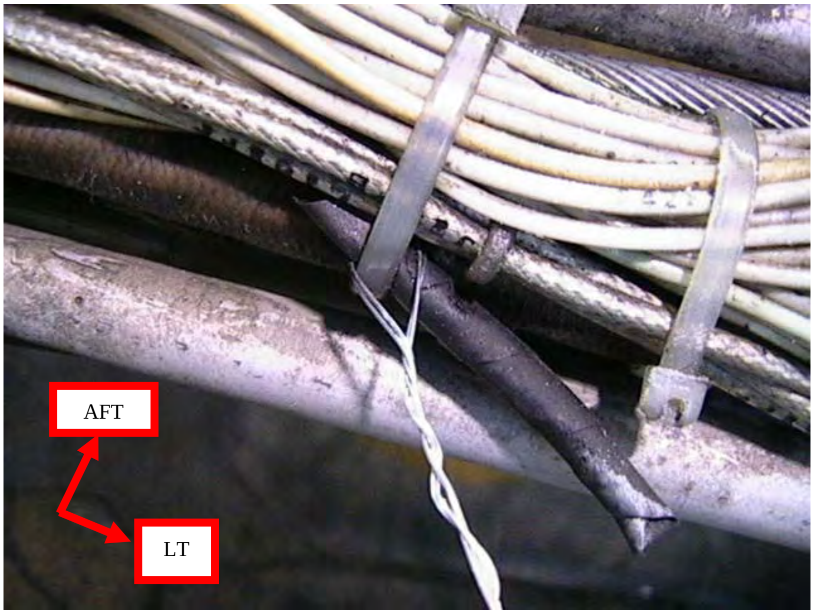


Figure 40. Improper Wire Termination FS 105, WL 13, and RWS 51

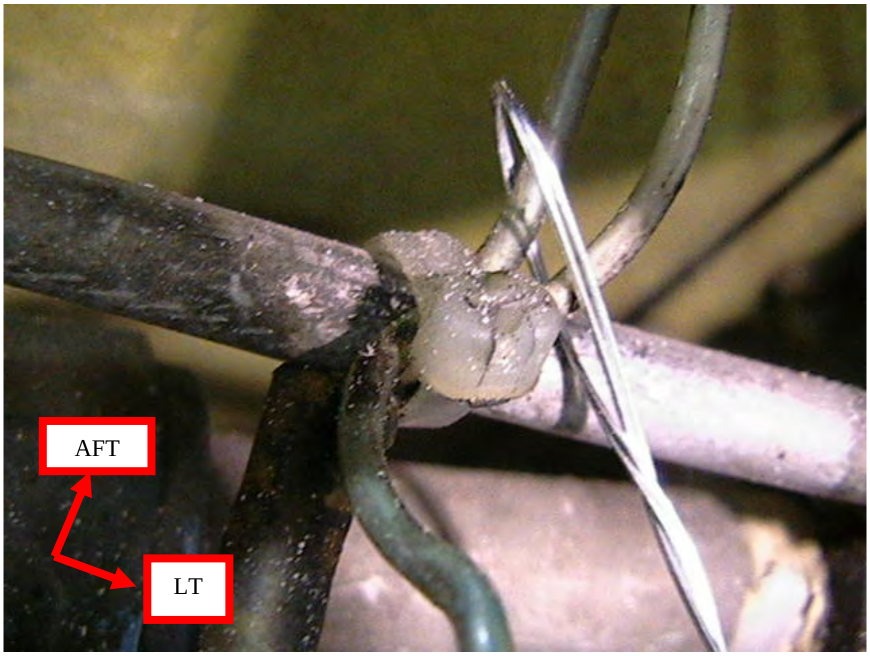


Figure 41. Improper Clamping/Support of Wire FS 105, WL 13, and RWS 36

2.3.3.4.7 Left Nacelle.

Defects in the left nacelle mainly consisted of damaged insulation due to abrasion and chemical damage, as shown in figures 42 and 43. Wires improperly stowed that led to chafing are shown in figure 44.

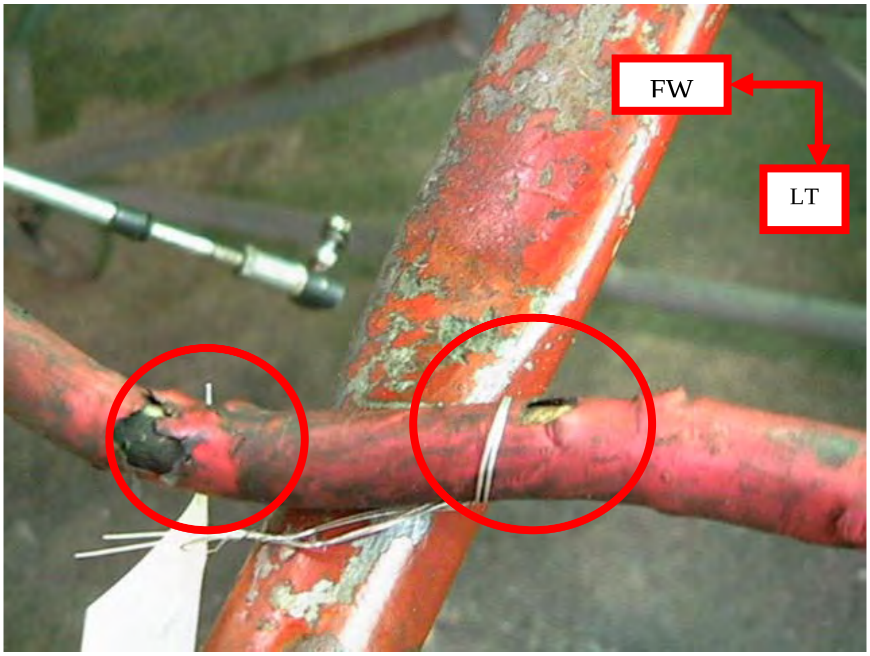


Figure 42. Damaged Insulation on the Starter Wire FS 87, WL 15, and LWS 77

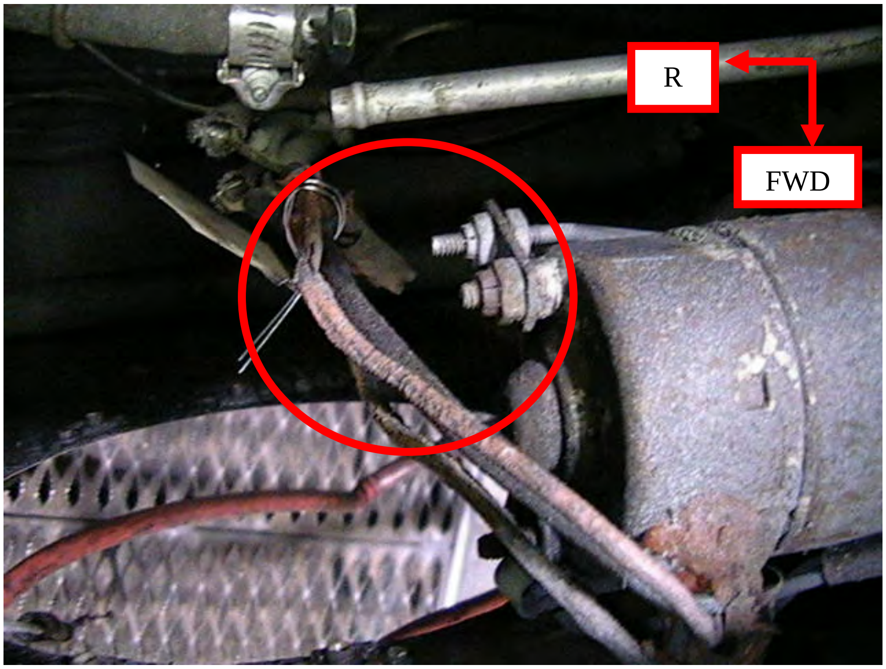


Figure 43. Chemically Damaged Cowl Flap Motor Wires FS 110, WL 0, and LWS 77

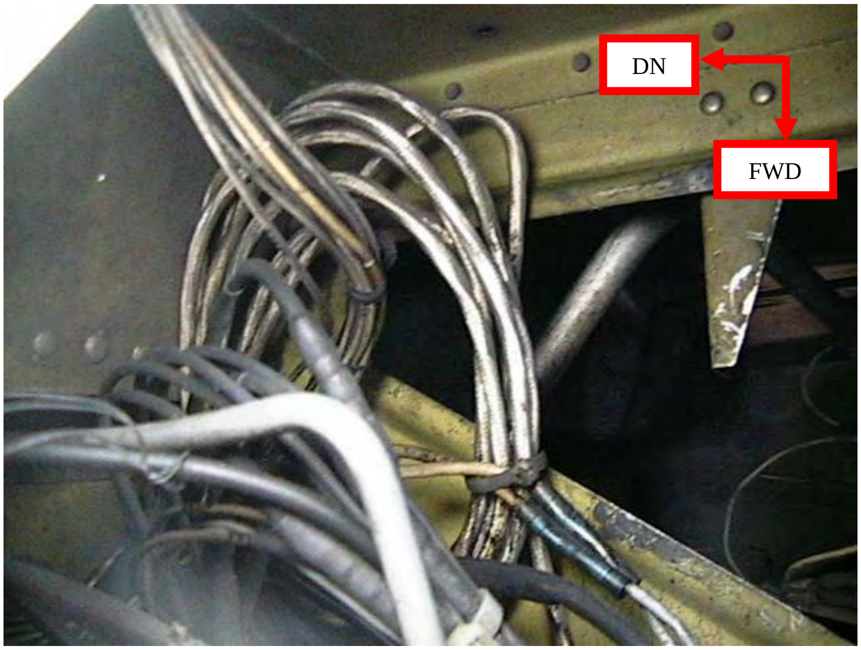


Figure 44. Insufficiently Stowed Wires FS 110, WL 6, and LWS 62

2.3.3.4.8 Right Nacelle

Similar to the findings in the left nacelle, the right nacelle defects included dirty/oily wires, as shown in figure 45, and insufficient support of wires and wire bundles, shown in figure 46. Figure 46 shows wires improperly supported on an air conditioner condenser line.

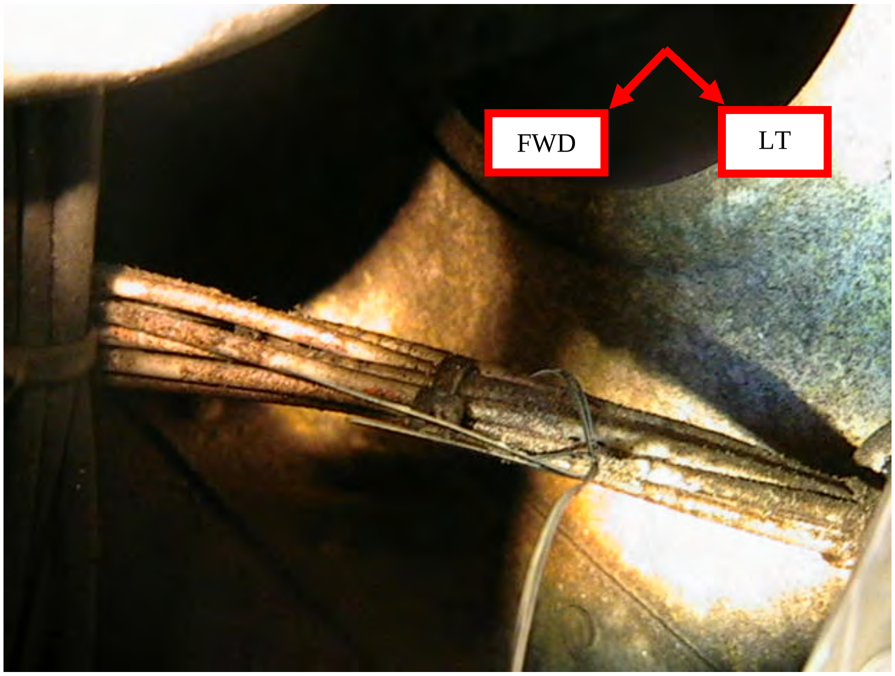


Figure 45. Dirty/Oily Wires FS 110, WL 0, and RWS 77

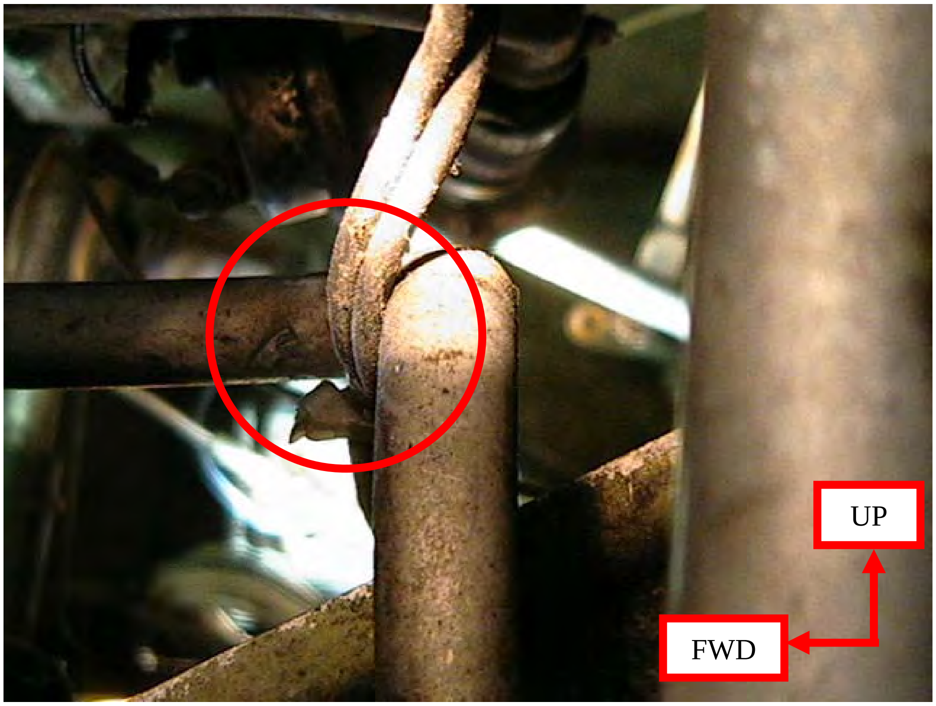


Figure 46. Poor/Improper Support of Wires FS 110, WL 0, and RWS 77

2.3.3.5 Analysis of Routine Wiring Visual Inspection Results.

Following the routine wiring visual inspections, the wiring condition defects, installation defects, termination defects, and zone-wise location of defects were analyzed to determine which flaws occurred most frequently and what zone locations experienced the most defects. Table 12 shows the results of the routine wiring visual inspections. Of the 199 wiring defects documented during the routine visual inspections, 84 were wiring condition defects. Of the wiring condition defects, 51% consisted of rubbing or chafing of the outer insulation, and 30% of the wires had exposed inner conductors. Of the 97 installation defects noted, 28% were missing/deteriorated grommets, with an additional 35% with inadequate clearance to structure. Eighteen termination defects were noted, with seven being loose/broken terminals. No connector defects were observed on this airplane. Defects in the passenger compartment accounted for 35% of the total wiring defects observed. Another 22% of the defects were located in the crew compartment with an additional 16% located in the left wing. The number of occurrences of the different types of wiring condition defects is shown in figure 47, while the occurrences of installation defects are illustrated in figure 48. Termination defects are shown in figure 49, and the zone-wise distribution of all wiring defects is shown in figure 50.

Table 12. Routine Wiring Visual Inspection Results

Type	Defect Description	Number of Defects
Wiring Conditions	Rubbing/chafing of outer insulation	43
	Cutting through outer insulation	4
	Exposed shield	0

Table 12. Routine Wiring Visual Inspection Results (Continued)

Type	Defect Description	Number of Defects
	Damaged shield	1
	Chafing/cutting of inner insulation	0
	Exposed inner conductor	25
	Damaged inner conductor	6
	Heat damage	1
	Fluid/chemical/dust contamination	3
	Corroded shield/conductors	0
	Label illegible	0
	Others	1
	Total Wiring Condition Defects	84
Installations	Inadequate clearance to structure	34
	Improper wire riding on other wire bundle	0
	Improper bend radius (10 by wire/bundle diameter)	2
	Missing/deteriorated ties	6
	Missing/deteriorated grommets	27
	Improper clamp condition/size	5
	Excessive slack/sag between clamps	1
	Excessive strain on wires	0
	Improper T or Y breakout	0
	Repaired wires	20
	Improperly labeled	2
	Unused wires improperly stowed	0
	Other	0
	Total Installation Defects	97
Terminations	Loose/broken terminals	7
	Corroded terminals	0
	Improper grounding condition	0
	High bonding resistance	0
	Others	11
	Total Termination Defects	18
Connectors	Insert damage/deterioration	0
	Contact arcing/fretting	0
	Missing/damaged/loose back shells	0
	Missing hardware	0
	Other	0
	Total Connector Defects	0
Total Defects		199

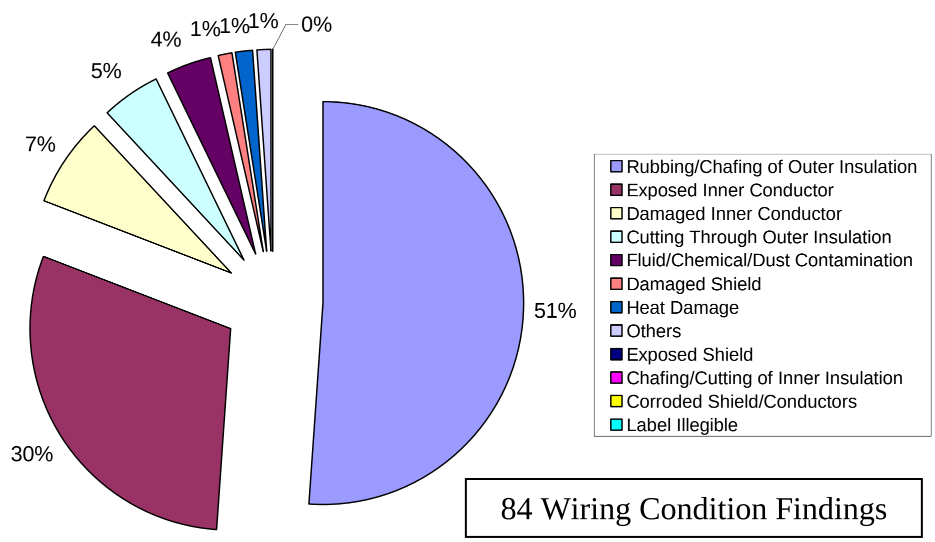


Figure 47. Routine Wiring Visual Inspection Results Wiring Condition Defects

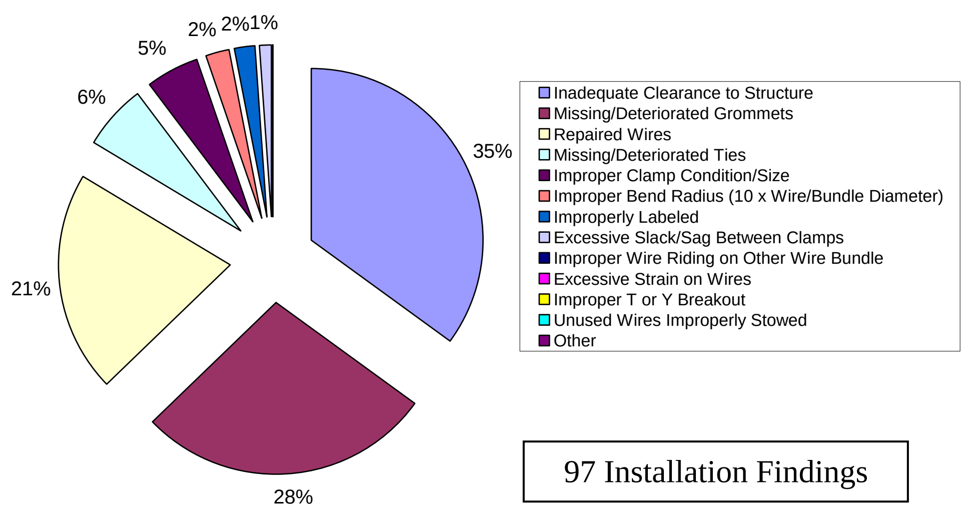


Figure 48. Routine Wiring Visual Inspection Results Installation Defects

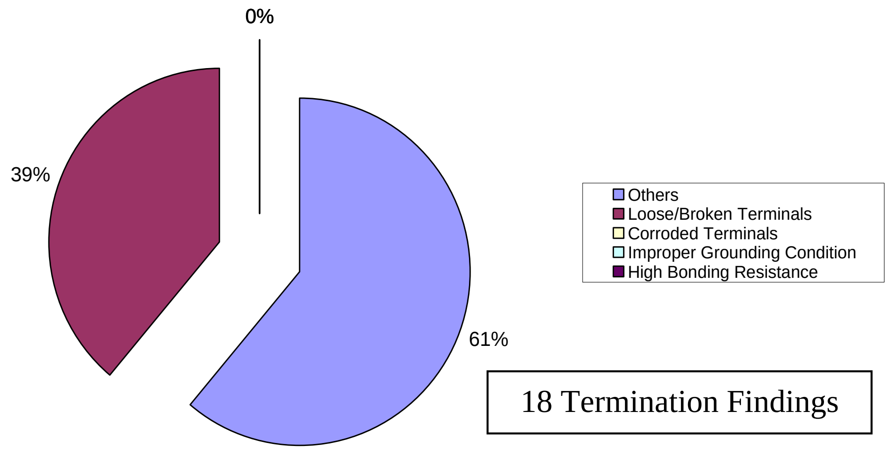


Figure 49. Routine Wiring Visual Inspection Results Termination Defects

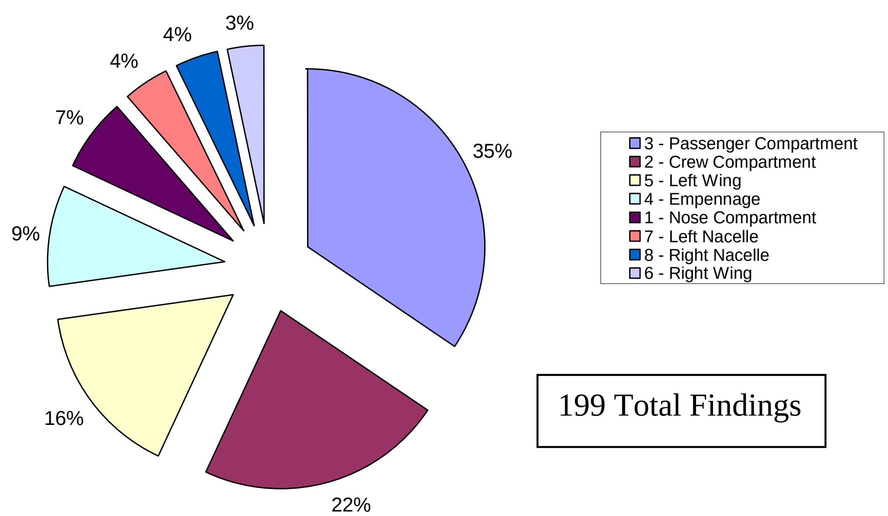


Figure 50. Routine Wiring Visual Inspection Results Zone-Wise Distribution of Defects

2.4 AIRWORTHINESS INSPECTIONS FOR ACCESSIBLE STRUCTURAL LOCATIONS.

As small airplanes approach and surpass their design life objectives, in-service failures due to corrosion and fatigue become more probable. To address this issue for the Piper Navajo Chieftain model PA31-350, the NIAR team developed airworthiness inspections for accessible structural locations based on engineering judgment and airplane maintenance history. These airworthiness inspections target PSE, using standard NDI techniques for structural areas that are not inspected on a routine basis. These inspections are more intrusive, and they are typically not performed until the number of flight hours approaches or exceeds the design life of the airplane. Detailed NDI procedures were developed for each accessible PSE identified in this report based on the type of defect to be found, access to the inspection area, and the type of material in the inspection area. The NDI methods used for these airworthiness inspections included visual inspection, fluorescent liquid penetrant inspection, magnetic particle inspection, surface scan eddy-current inspection, and bolthole eddy-current inspection.

PSEs were identified for the Piper Navajo Chieftain based on its similarities to other models with supplemental inspections, specifically the Cessna 402 airplanes. It is important to note that only a service history/engineering judgment approach along with accessibility to the structural areas was taken to develop this set of airworthiness inspections, and no damage tolerance assessment was performed. Therefore, no initial inspection time or inspection intervals were established. This set of airworthiness inspections provides an owner/operator with information on where to inspect a high-time Piper Navajo Chieftain and provides inspection procedures developed for specific structural stackup found on the accessible PSEs of this airplane.

2.4.1 Development of Airworthiness Inspections for Accessible Structural Locations.

An airplane component was classified as a PSE if the component contributes significantly to carrying flight and ground loads and if failure of the component could result in catastrophic failure of the airframe. The monitoring of the PSEs is the main focus of these airworthiness inspections. Using engineering judgment, PSEs were determined to include all accessible major attachment fittings of the airplane and all accessible structural members reacting high stresses, such as the wing root and engine beam rib. Structural areas that were not accessible, such as nacelle bulkheads, were not inspected since destructive disassembly would have been required to access these areas for inspection. Other significant areas were determined based on maintenance history. The only four areas selected due to maintenance history were the rudder torque tube, elevator torque tube, landing gear strut housing attachment points, and the horizontal stabilizer butt rib. The Piper Navajo Chieftain has a history of corrosion related issues on the rudder torque tube. AD 2003-24-07 was prescribed to inspect the torque tube for corrosion; therefore, it was essential to list this part as a PSE for developing these airworthiness inspections. Like the rudder torque tube, the elevator torque tube also has a history of corrosion, as does the horizontal stabilizer butt rib. The landing gear strut housing attachment points are an area identified from the SDRs as a common fleet-wide issue.

The following is a complete listing of all the accessible PSE components identified for the airworthiness inspections. Note that since only the accessible PSEs were identified, this list does not constitute a complete list of structural PSEs for this model:

- Aileron Hinges and Attachment Fittings
- Rudder Torque Tube
- Elevator Torque Tube
- Landing Gear Strut Housing Attachment Points
- Nose Landing Gear Attachment Points Common to Fuselage
- Horizontal Stabilizer Butt Rib
- Elevator Hinges and Attachment Fittings
- Vertical Stabilizer Front Spar Attachment Points
- Vertical Stabilizer Rear Spar Attachment Points
- Rudder Hinges and Attachment Fittings
- Horizontal Stabilizer Front Spar Attachment Points
- Horizontal Stabilizer Rear Spar Attachment Points
- Wing Front Spar Attachment Points
- Engine Rib Lower Cap to Main Spar Attachment
- Wing Aft Auxiliary Spar Landing Gear Attachment Points
- Wing Forward Auxiliary Spar Landing Gear Attachment Points
- Wing Main Spar Lower Cap
- Wing Rear Spar Attachment Points
- Wing Main Spar Splice Attachment Points
- Wing Main Spar to Fuselage Attachment Points
- Wing Main Spar Landing Gear Attachment Points
- Flap Attachment Fittings
- Engine Structure

Following the teardown evaluation phase, section 3.4, components that were inspected as part of these airworthiness inspections and that exhibited defects in other areas were investigated to determine if a revised inspection method and/or locations should be included in the airworthiness inspections. These revised inspections would strengthen the airworthiness inspections and further validate the safety and/or structural integrity of the Piper Navajo airplane. Table 13 displays the inspection methods that were revised in the airworthiness inspections and areas of each component that required additional inspections based on results from the teardown evaluation phase.

Table 13. Revised Airworthiness Inspections

Part Nomenclature	Revised Inspection	Location
LT wing main gear aft fitting	Remove paint and perform BHEC and LPI	Figure 179
LT wing trunnion landing gear aft	Remove paint and perform BHEC and LPI	Figure 198
RT Wing trunnion landing gear aft	Remove paint and perform BHEC and LPI	Figure 332

Table 13. Revised Airworthiness Inspections (Continued)

Part Nomenclature	Revised Inspection	Location
RT Front spar attachment	Eddy-current surface scan if accessible from floor	Figure 420
LT front spar attachment	Eddy-current surface scan if accessible from floor	Figure 420
Vertical stabilizer front spar	Remove stabilizer and perform eddy-current surface scan around mounting holes and at 45 degree angle	Figure 586
Vertical stabilizer front spar right angle	Remove stabilizer and perform eddy-current surface scan around mounting holes	Figure 603
Vertical stabilizer front spar left angle	Remove stabilizer and perform eddy-current surface scan around mounting holes	Figure 598

BHEC = Bolthole eddy-current
LPI = Liquid penetrant inspection

Visual inspection constituted an important part of practical quality control and quality assurance. This inspection method is the most extensively used NDI method. It is easy to apply, very quick, and often requires no special equipment other than good eyesight and relatively inexpensive tools when necessary. All accessible PSEs received a visual inspection prior to additional NDI inspections; however, some components required an additional comprehensive visual evaluation. The engine structure, rudder torque tube, elevator torque tube, horizontal stabilizer butt rib, and landing gear strut housings were inspected using this comprehensive visual inspection.

Fluorescent liquid penetrant inspection is another inexpensive and reliable NDI method for detecting discontinuities open to the surface of a material under evaluation. It can be used on metals and other nonporous materials not harmed by the penetrant. The standard American Society for Testing and Materials (ASTM) E 1417-99 practices were used during this evaluation. This method was performed on the aileron hinges and attachment fittings, flap attachment fittings, wing front spar attachment fittings, elevator hinges and attachment fittings, and rudder hinges and attachment fittings. Type 1, level 3 penetrant materials were selected for this application with a minimum dwell time of 10 minutes.

Magnetic particle inspection is used to reveal surface and near subsurface discontinuities in magnetic materials. This inspection can only be used on materials that can be magnetized (ferrous). The standard ASTM E 1444-01 practices were used during this evaluation. On some material, magnetic particle was used instead of liquid penetrant because it is faster and requires less surface preparation; however, liquid penetrant can be used on most inspections that require magnetic particle inspection, if time is not a constraint. Magnetic particle inspections were performed on the landing gear strut housing attachment points and on all welds on the engine structure. Longitudinal and circular magnetism were selected for this application to cover all possible defect orientations.

Eddy-current inspection is used to detect discontinuities in parts that are conductors of electricity. In general, the eddy-current method of NDI is used to inspect relatively small areas; however, the probe design and test parameters must be established with a good understanding of the flaw to be detected. Since eddy currents tend to concentrate at the surface of a material, they only can be used to detect surface and near surface defects. Optimum sensitivity to cracks or other flaws generally occurs in specific frequency ranges for each combination of metal, flaw size, and flaw depth. Eddy-current inspections were performed on the wing forward and aft auxiliary spar landing gear attachment points, the wing main spar to fuselage attachment points, the wing main spar landing gear attachment points, wing main spar lower cap, engine rib lower cap to main spar attachment, wing main spar splice attachment points, wing rear spar attachment points, horizontal stabilizer front and rear spar attachment points, and the vertical stabilizer front and rear spar attachment points. Due to the material thickness of the PSEs and the anticipated flaws to be detected for the necessary materials, a frequency of 200 kHz was selected for surface scan eddy current and 500 kHz was selected for bolthole eddy current.

Appendix A to this report provides the documentation for performing the airworthiness inspections of accessible structural locations identified in this teardown evaluation. The document includes detailed information on the accessible PSEs and the detailed inspection procedures developed for this airplane. Often, multiple inspection techniques implementing various inspection methods were developed for the same PSE due to limited inspection access.

2.4.2 Results From Airworthiness Inspections of Accessible Structural Locations.

The airworthiness inspections of accessible structural locations employed several NDI techniques including visual inspection, eddy-current surface and bolthole inspection, fluorescent liquid penetrant inspection, and magnetic particle inspection. These inspections were performed on specified areas of the wings, fuselage, horizontal stabilizer, vertical stabilizer, and landing gear. Qualified NIAR level I and II inspectors independently performed all of the airworthiness inspections. During the development of the airworthiness inspections, accessible PSEs were selected based on two methods: (1) engineering evaluation, and (2) maintenance history. All inspections listed in tables 14 to 18 are superscripted according to their reason for selection. Table 13 includes the name of the airworthiness inspection and the method performed on each wing. Table 15 lists the inspections performed on the fuselage, and tables 16 and 17 detail the inspections performed on the horizontal and vertical stabilizers and their respective flight control surfaces. Table 18 contains the other airworthiness inspection areas.

Table 14. Airworthiness Wing Inspections

Title	Method of Inspection
Aileron hinges and attachment fittings	Liquid penetrant
Flap attachment fittings	Liquid penetrant
Wing aft auxiliary spar landing gear attachment points	Eddy current
Wing forward auxiliary spar landing gear attachment points	Eddy current
Wing front spar attachment points	Eddy current
Wing main spar to fuselage attachment points	Eddy current
Wing main spar landing gear attachment points	Eddy current

Table 14. Airworthiness Wing Inspections (Continued)

Title	Method of Inspection
Wing main spar lower cap	Eddy current
Engine rib lower cap to main spar attachment	Eddy current
Wing main spar splice attachment points	Eddy current
Wing rear spar attachment points	Liquid penetrant

Table 15. Airworthiness Fuselage Inspections

Title	Method of Inspection
Wing front spar attachment points	Liquid penetrant
Wing main spar to fuselage attachment points	Eddy current
Wing rear spar attachment points	Liquid penetrant
Horizontal stabilizer front spar attachment points	Eddy current
Horizontal stabilizer rear spar attachment points	Eddy current
Vertical stabilizer front spar attachment points	Eddy current
Vertical stabilizer rear spar attachment points	Eddy current

Table 16. Airworthiness Horizontal Stabilizer and Elevator Inspections

Title	Method of Inspection
Elevator hinges and attachment fittings	Liquid penetrant
Elevator torque tube	Visual
Horizontal stabilizer front spar attachment points	Eddy current
Horizontal stabilizer rear spar attachment points	Eddy current
Horizontal stabilizer butt rib	Visual

Table 17. Airworthiness Vertical Stabilizer and Rudder Inspections

Title	Method of Inspection
Rudder hinges and attachment fittings	Liquid penetrant
Rudder torque tube	Visual
Vertical stabilizer front spar attachment points	Eddy current
Vertical stabilizer rear spar attachment points	Eddy current

Table 18. Other Airworthiness Inspections

Title	Method of Inspection
Landing gear strut housing attachment points	Visual and magnetic particle
Engine structure	Visual and magnetic particle

2.4.2.1 Visual Airworthiness Inspection Results.

Visual testing is the original and most commonly used NDI method. It is the simplest to perform since it does not require additional equipment. The inspector simply looks at the inspection areas for surface damage. A ten power hand-held magnifying glass under a minimum 100-lumen light is often used to detect smaller flaws. Since paint is rarely removed from the inspection area, the inspector must decide whether a crack is in the paint only or if it extends into the underlying structure. The results of the visual airworthiness inspections are listed in table 19. In addition, these visual findings are shown in figures 51 through 55. Figure 51 is the global view of the nose landing gear strut assembly. Figure 52 shows a 1-inch crack on the nose landing gear strut housing. Figure 53 shows the location of the rudder torque tube. Corrosion, approximately 2 square inches, on the rudder torque tube can be seen in figures 54 and 55.

Table 19. Visual Airworthiness Inspection Results

Name of Inspection	Discrepancy Reported
Landing gear strut housing attachment points	Nose landing gear housing is cracked
Rudder torque tube	Corrosion is completely through the tube

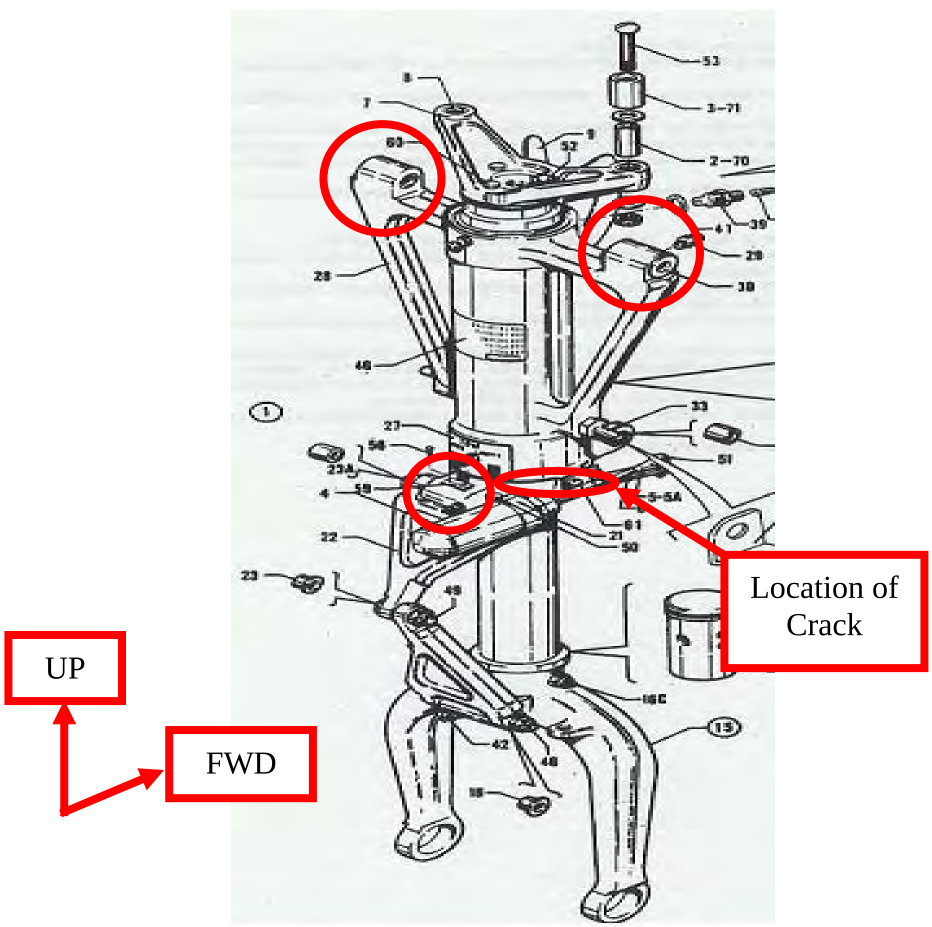


Figure 51. Visual Inspection Areas on the Nose Landing Gear Strut Housing

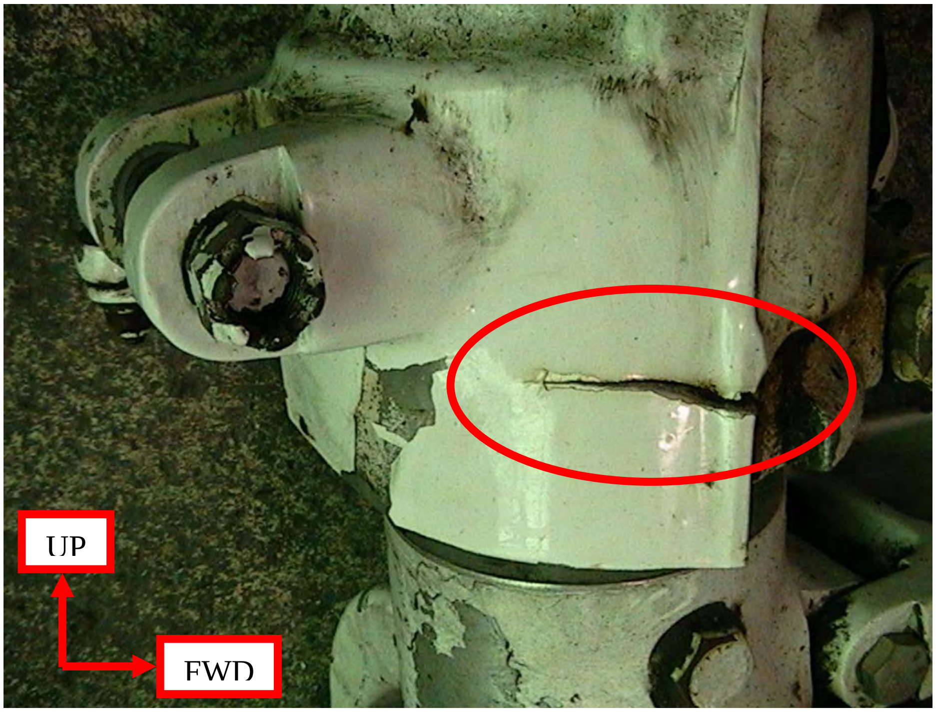


Figure 52. Crack in the Nose Landing Gear Strut Housing FS 25, WL 32, and RBL 2

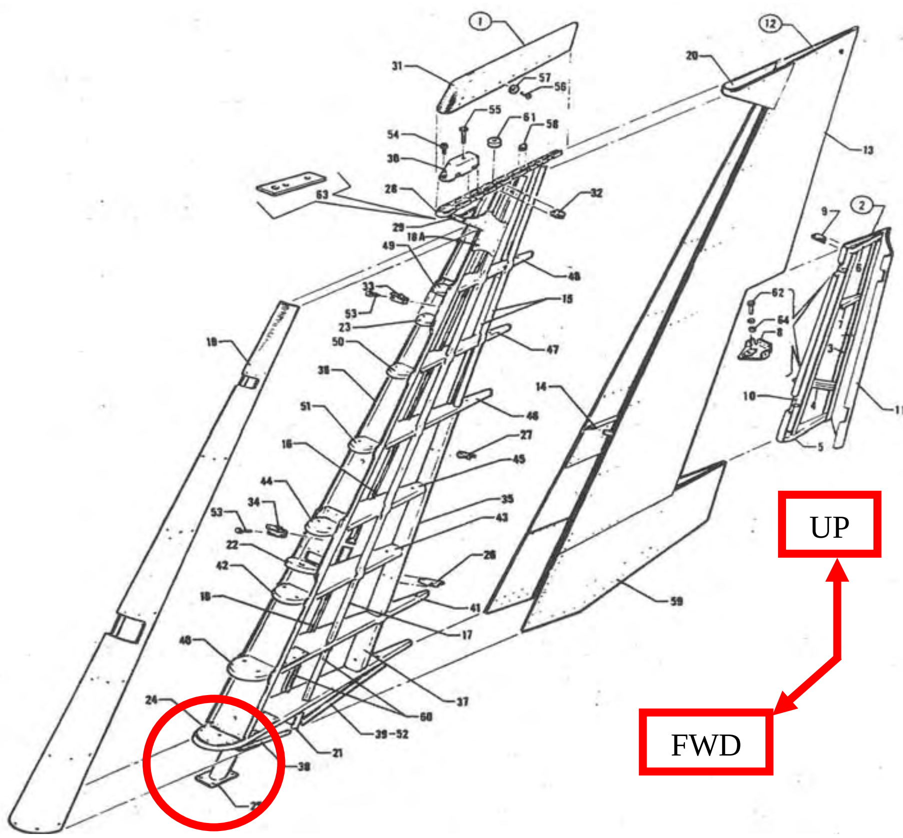


Figure 53. Location of the Rudder Torque Tube

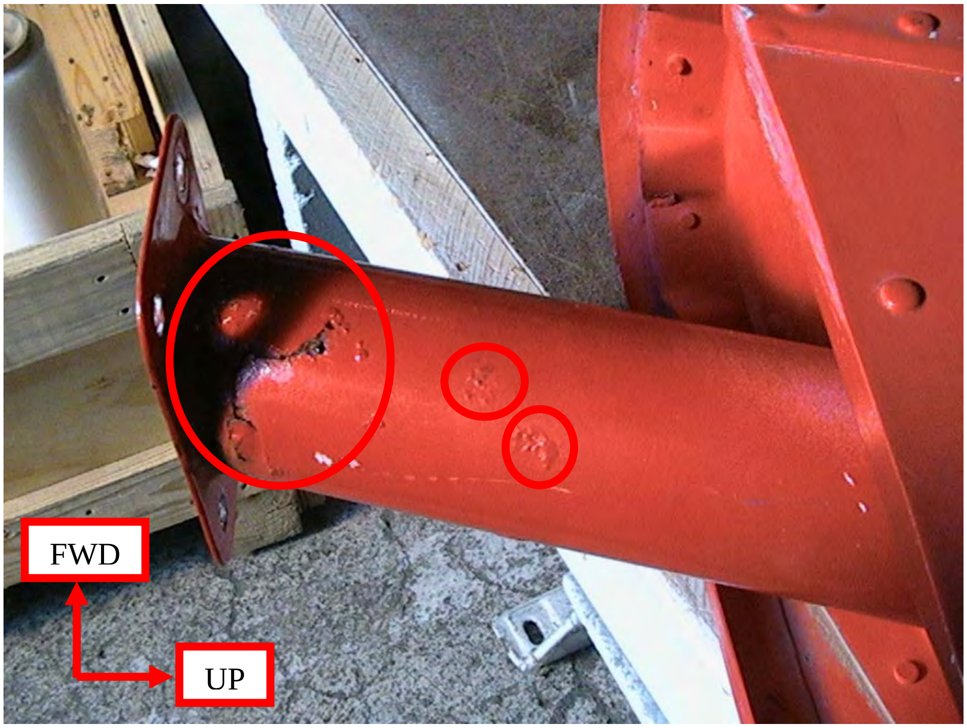


Figure 54. Corrosion on the Rudder Torque Tube FS 342.25, WL 20, and BL 0

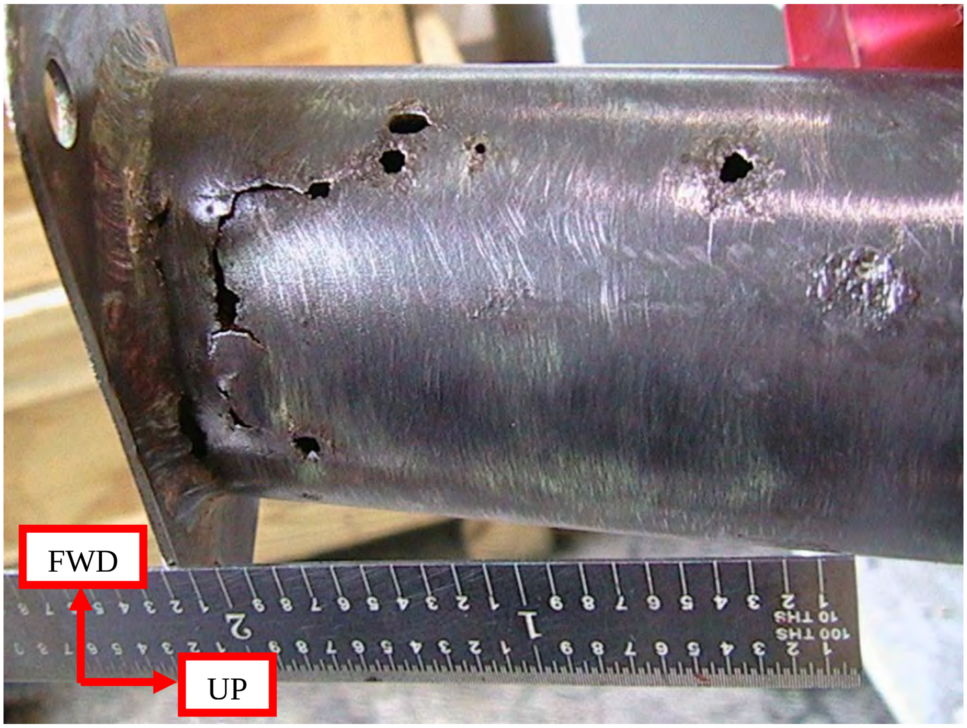


Figure 55. Corrosion on the Rudder Torque Tube After Paint Removal FS 342.25, WL 20, and BL 0

2.4.2.2 Fluorescent Liquid Penetrant Airworthiness Inspection Results.

The fluorescent liquid penetrant inspection is illustrated in figure 56.

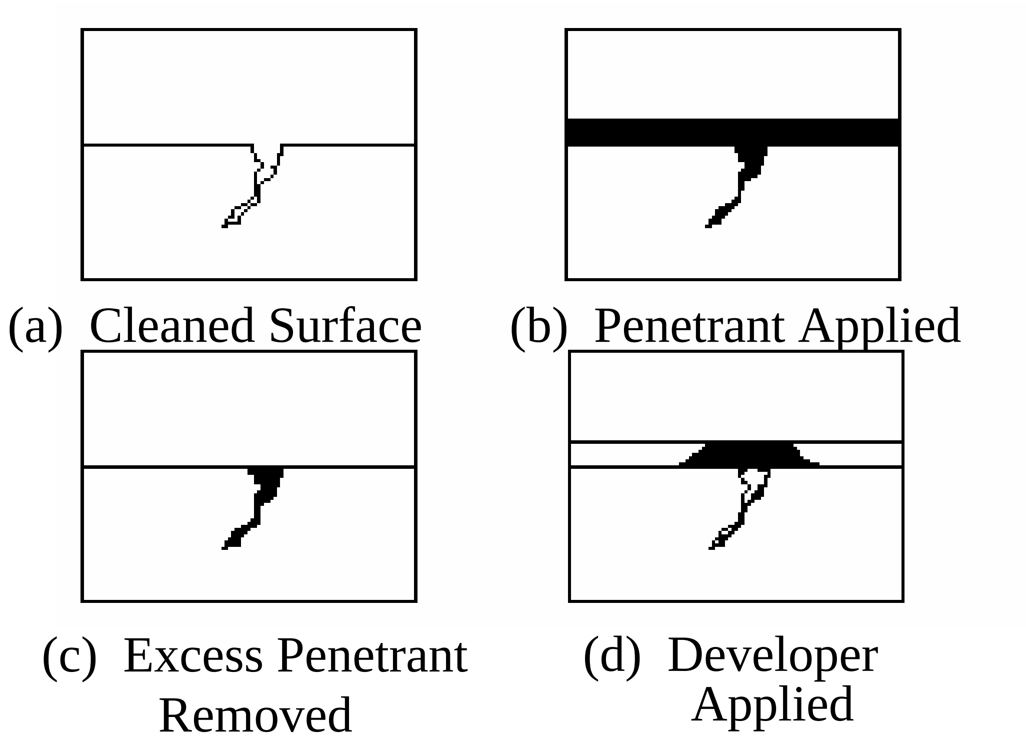


Figure 56. Penetrant Inspection Process

Penetrant inspections tend to take longer than visual or eddy current to perform since surfaces must be stripped of paint before this inspection can be conducted. A penetrating liquid is placed on the surface of the part, and this liquid penetrates into the cavity formed by a discontinuity, such as a crack, in the material. The surface liquid is then removed gently to avoid disturbing the penetrant that settled in the discontinuity. The liquid left in the cavity is brought back to the surface with the application of a developer. A black light is used to view the flaw if a fluorescent penetrant is used. Fluorescent penetrant is limited to surface flaws and requires about 20 minutes of inspection preparation. Table 20 shows the discrepancies of the liquid penetrant airworthiness inspections, which included the inspection of elevator hinges and attachment fittings. The corrosion findings from liquid penetrant are shown in figures 57 through 61. The global view of the liquid penetrant areas on the wing front and rear spar attachment points is shown in figure 57. The left wing fuselage attachment points with corrosion are shown in figures 58 and 59. Corrosion on the right wing fuselage attachment points is shown in figures 60 and 61.

Table 20. Fluorescent Liquid Penetrant Airworthiness Inspection Results

Name of Inspection	Discrepancy Reported
Front spar attachment point	Corrosion was found on the left wing
Rear spar attachment point	Corrosion was found on the left and right wing

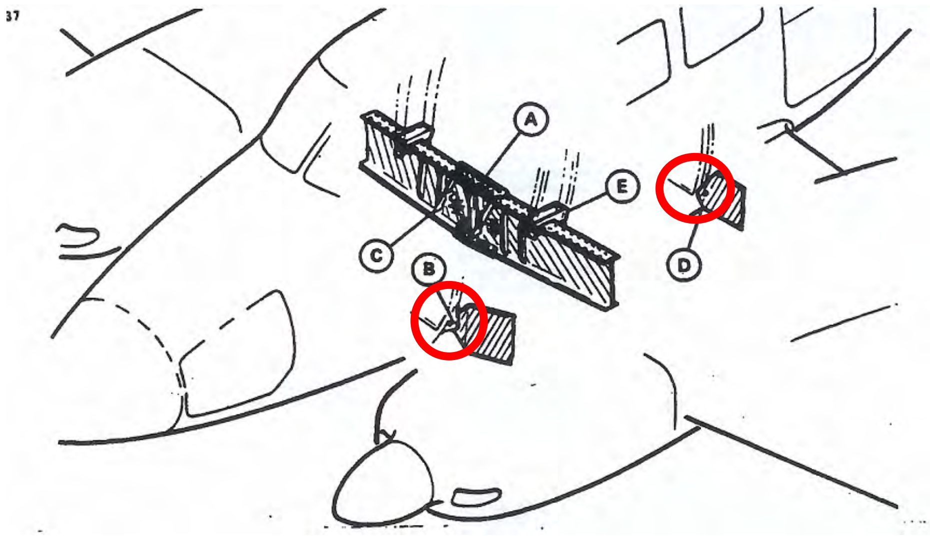


Figure 57. Location of the Front and Rear Spar Attachment Points

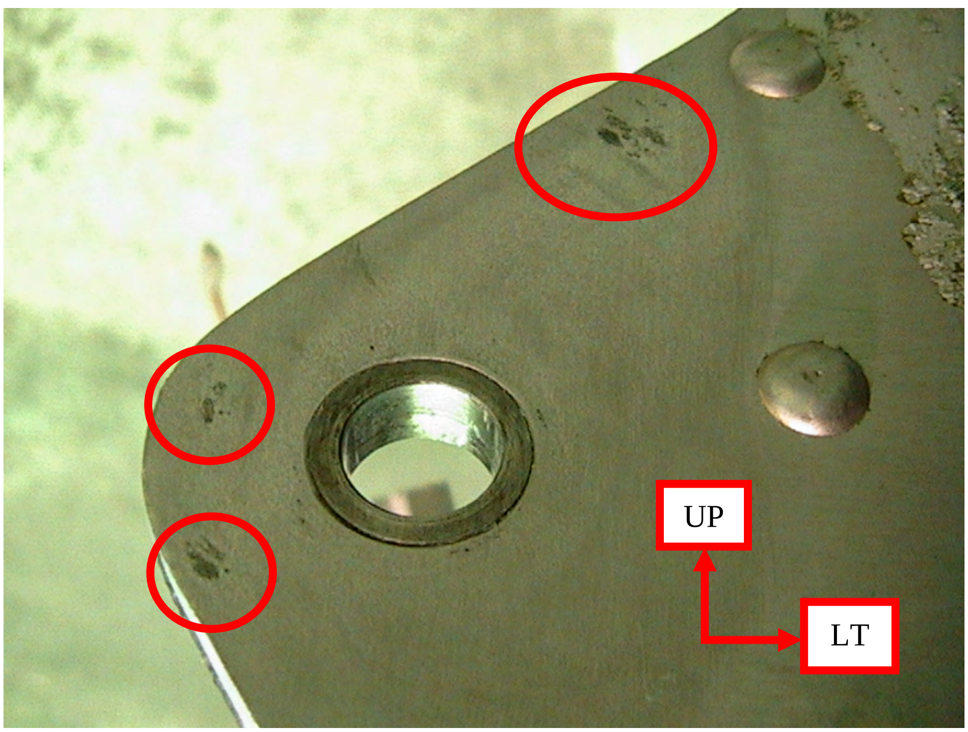


Figure 58. Corrosion on the Left Wing Front Spar Wing Side Attachment Point WS 30

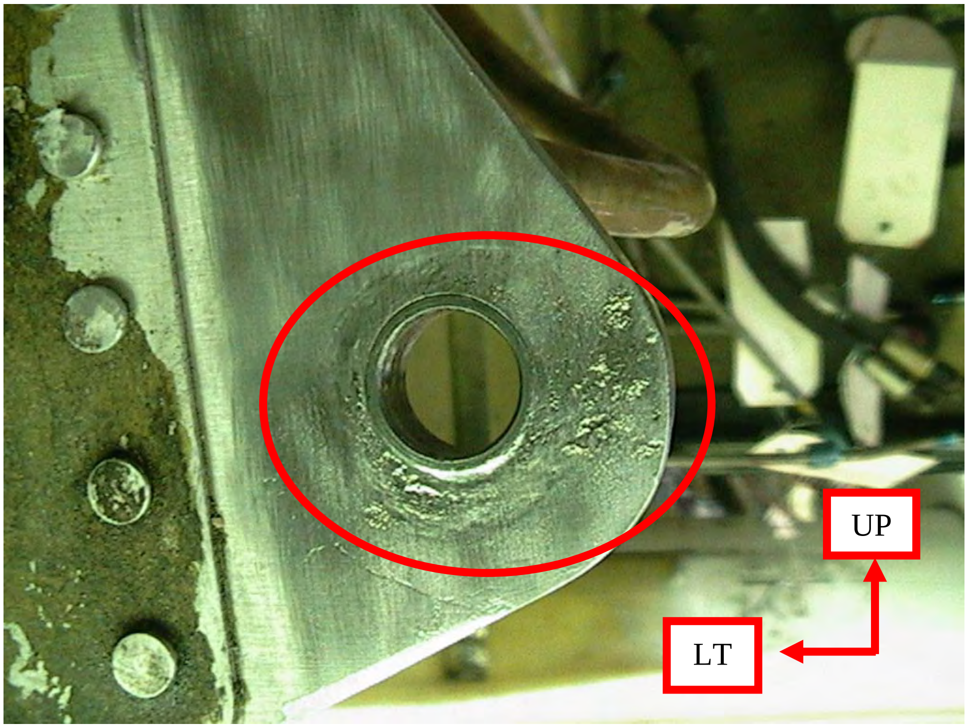


Figure 59. Corrosion on the Left Wing Rear Spar Wing Side Attachment Point WS 30

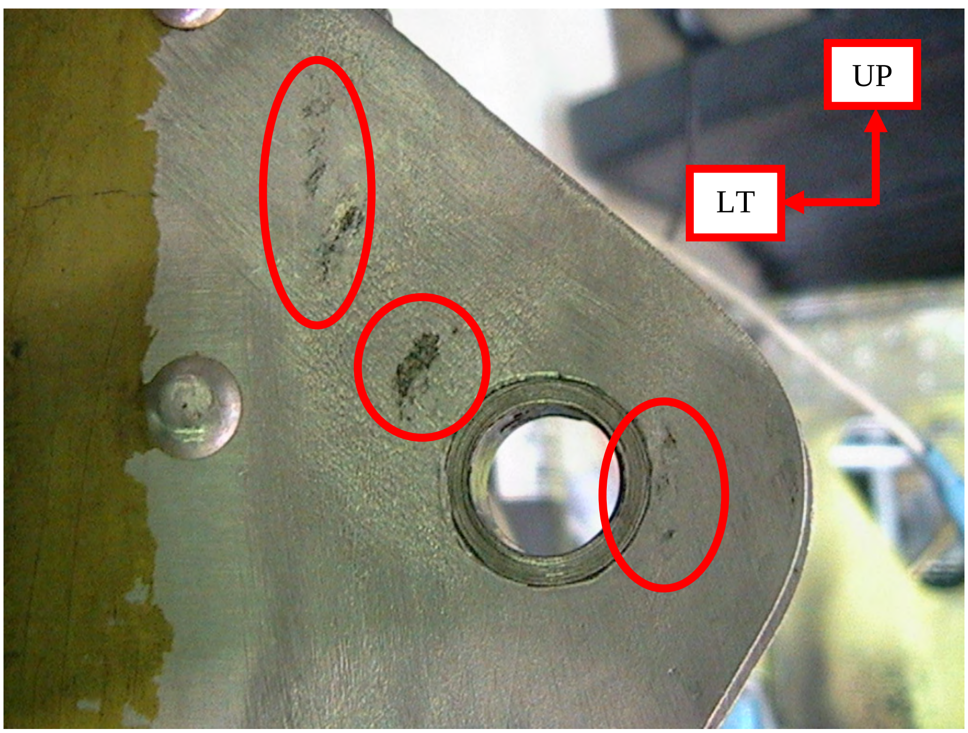


Figure 60. Corrosion on the Right Wing Front Spar Wing Side Attachment Point WS 30

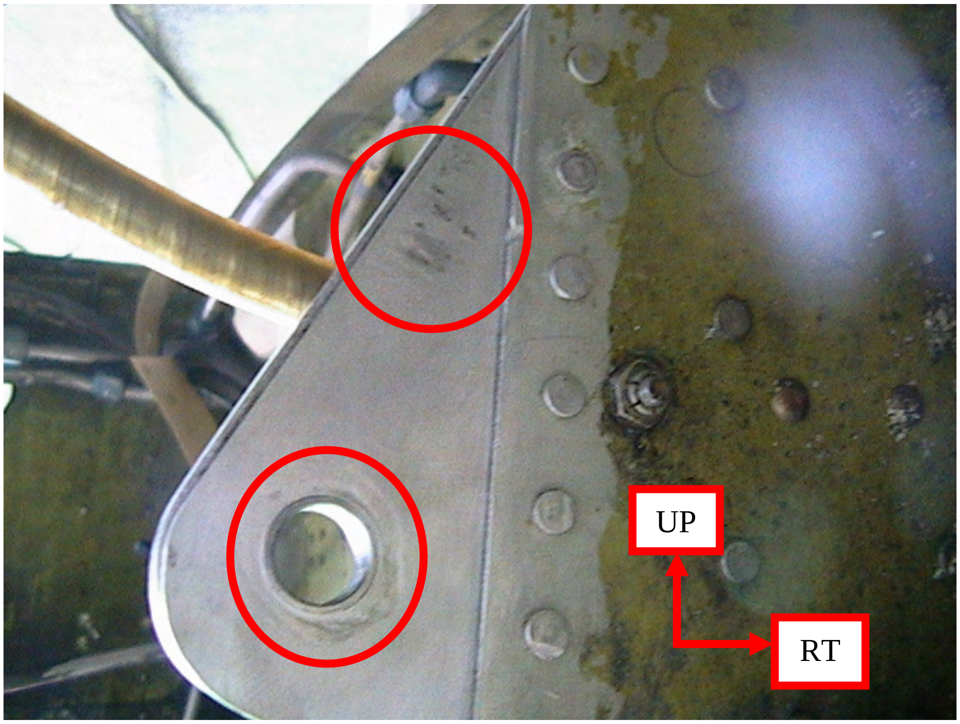


Figure 61. Corrosion on the Right Wing Rear Spar Wing Side Attachment Point WS 30

2.4.2.3 Magnetic Particle Airworthiness Inspection Results.

Magnetic particle inspections consist of several basic steps. The first step is to magnetize the part. In the second step, magnetic particles, such as iron filings, are then sprayed onto the part. These particles are the media through which the flaws become visible, and indications are interpreted by a qualified inspector. Fluorescent magnetic particles are most commonly used for airplane inspections. Figure 62 shows a typical indication of a surface crack. Magnetic particle is easy to use, requires minimal training, and results can be achieved quickly. However, it is limited to relatively small inspection areas composed of a magnetic material. Only the landing gear attachment points and the engine structure welds were inspected during the magnetic particle airworthiness inspections and no relevant indications were detected.

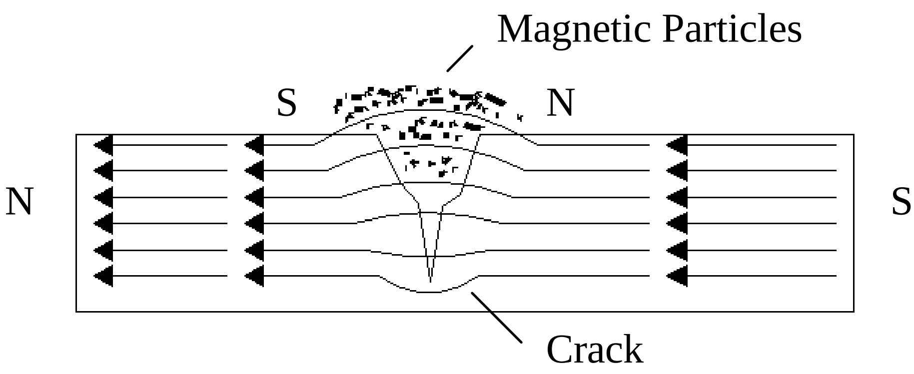


Figure 62. Surface Crack Indications

2.4.2.4 Eddy-Current Airworthiness Inspection Results.

The basic setup for an eddy-current probe is shown in figure 63. A primary magnetic field created by alternating current will cause the current to flow in a circular direction, known as eddy currents. For the current to flow, the test part must be made of conductive materials. A secondary magnetic field forms in the part and opposes changes to the primary magnetic field. Changes in material properties alter the eddy currents, which in turn changes the opposing secondary magnetic field. This change alters the electrical characteristics of the primary magnetic field, which can be detected by the eddy-current instrument. Interpretation of the electrical signals requires experienced inspectors.

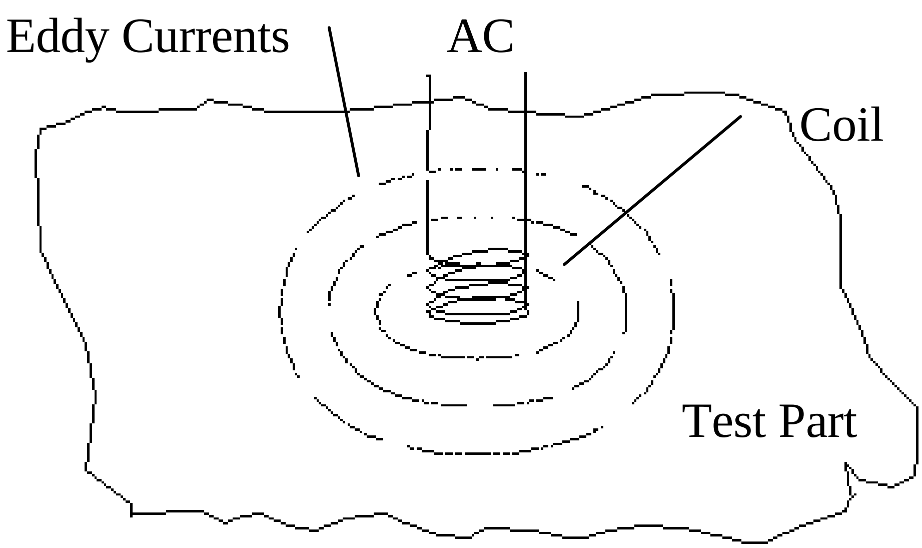


Figure 63. Eddy-Current Probe

A majority of the airworthiness inspections were performed using the eddy-current method. Table 21 shows the results from the eddy-current airworthiness inspections. These eddy-current findings are shown in figures 64 through 69. The global view of the inspected area on the wing main spar splice attachment points is shown in figure 64. The left wing main spar splice attachment points with the gouge indications are shown in figures 65 and 66. Figure 67 shows the average gouges found on the left wing main spar splice attachment points. The right wing main spar splice attachment points with gouge indications are shown in figure 68. Figure 69 shows the average gouges found on the right wing main spar splice attachment points.

Table 21. Eddy-Current Airworthiness Inspection Results

Name of Inspection	Discrepancy Reported
Wing main spar splice attachment points	10 indications from gouges

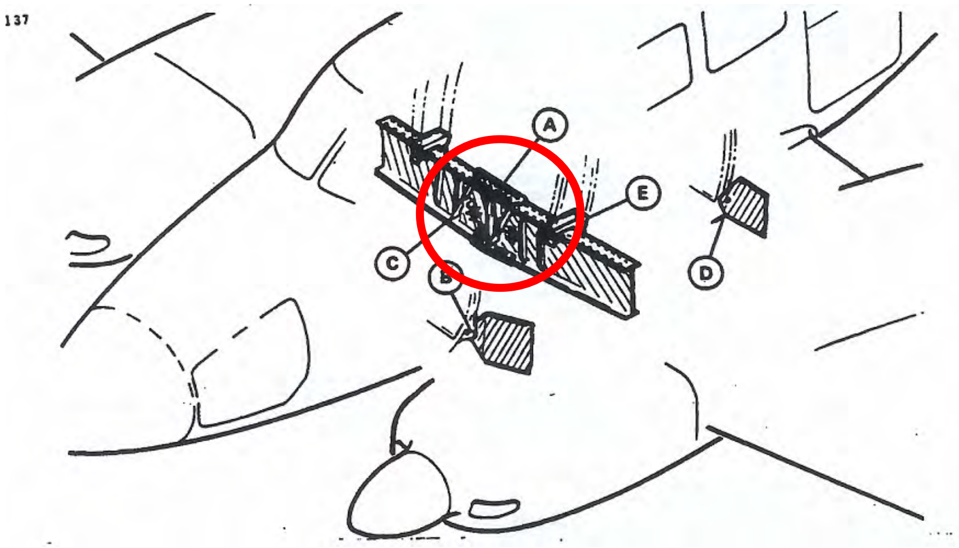


Figure 64. Location of the Wing Main Spar Splice

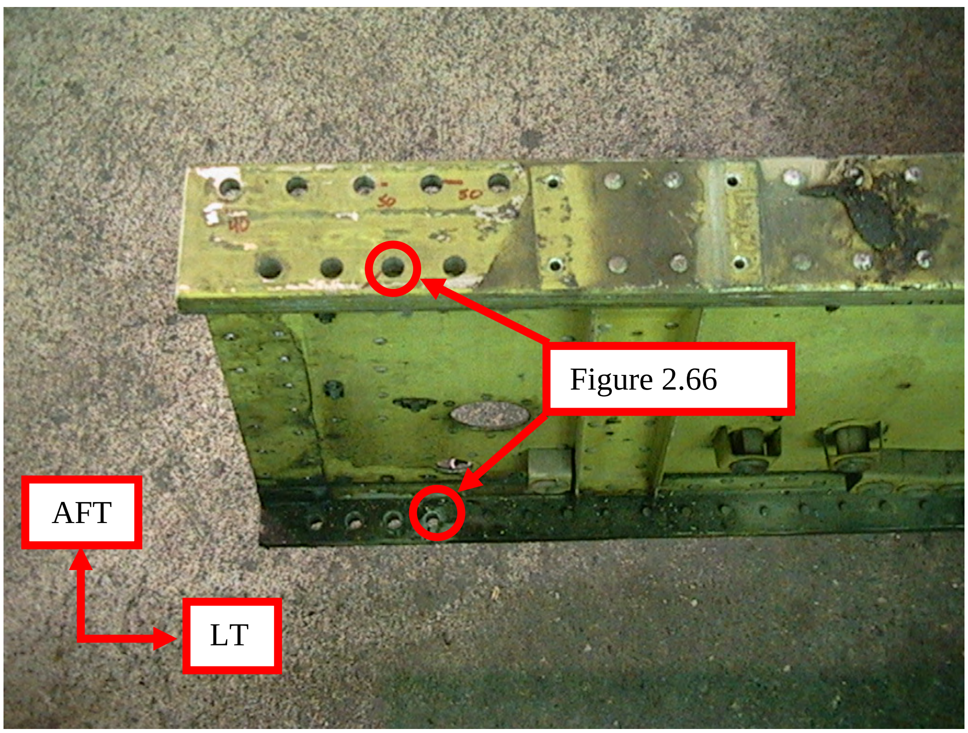


Figure 65. Left Wing Main Spar Splice Lower Surface Attachment Points BL 0 Through BL 6, View 1

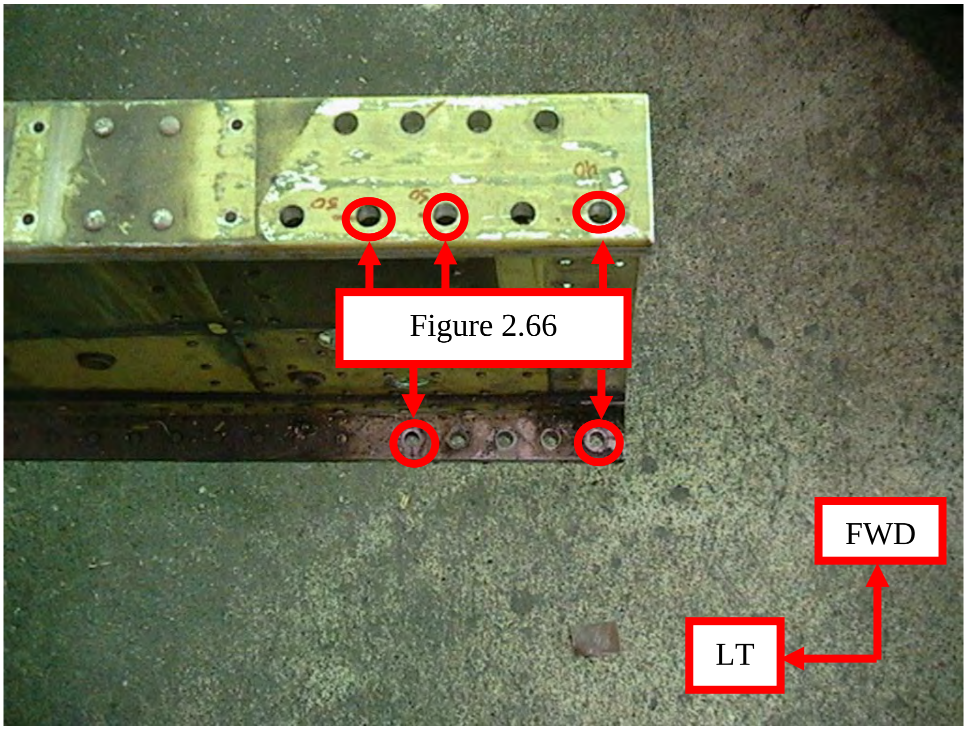


Figure 66. Left Wing Main Spar Splice Lower Surface Attachment Points BL 0 Through BL 6, View 2

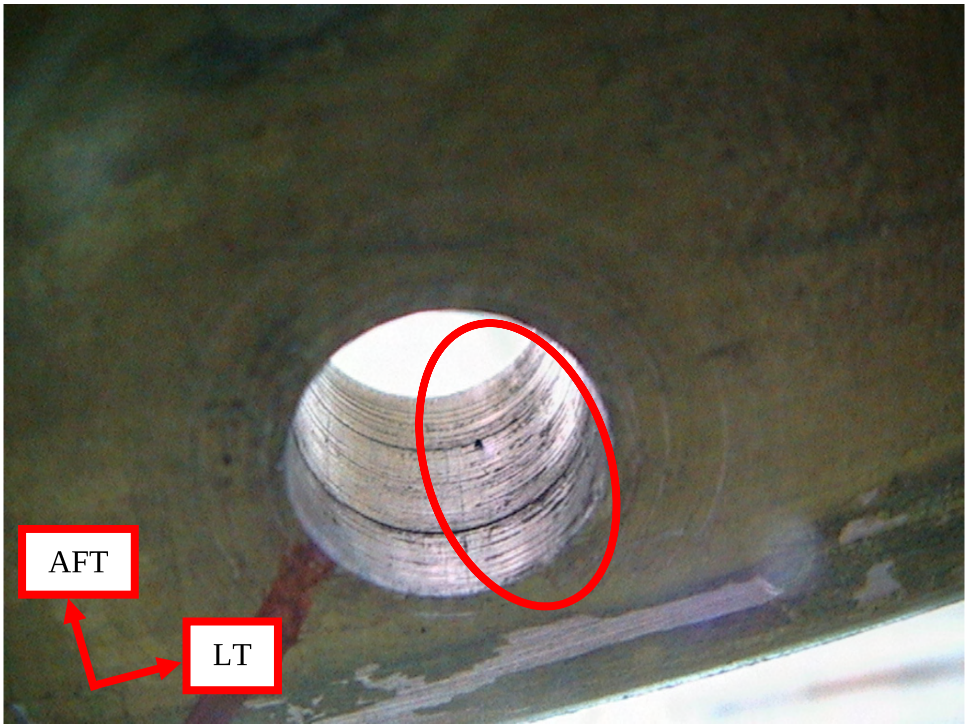


Figure 67. Gouges on the Left Wing Main Spar Splice Attachment Points BL 0 Through BL 6

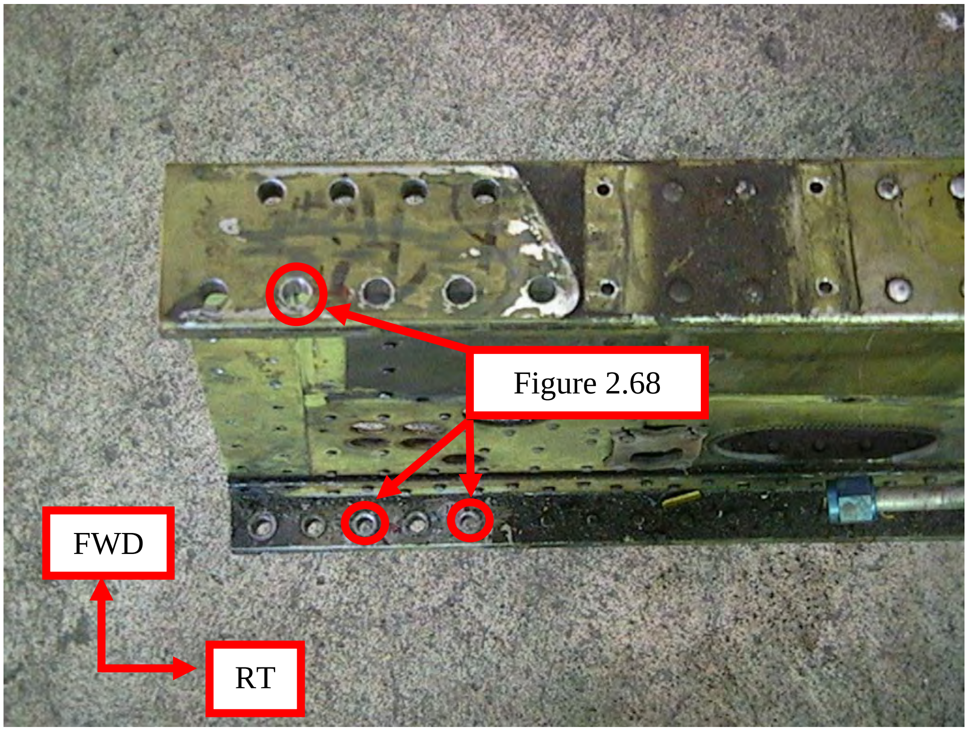


Figure 68. Right Wing Main Spar Splice Lower Surface Attachment Points BL 0 Through BL 6

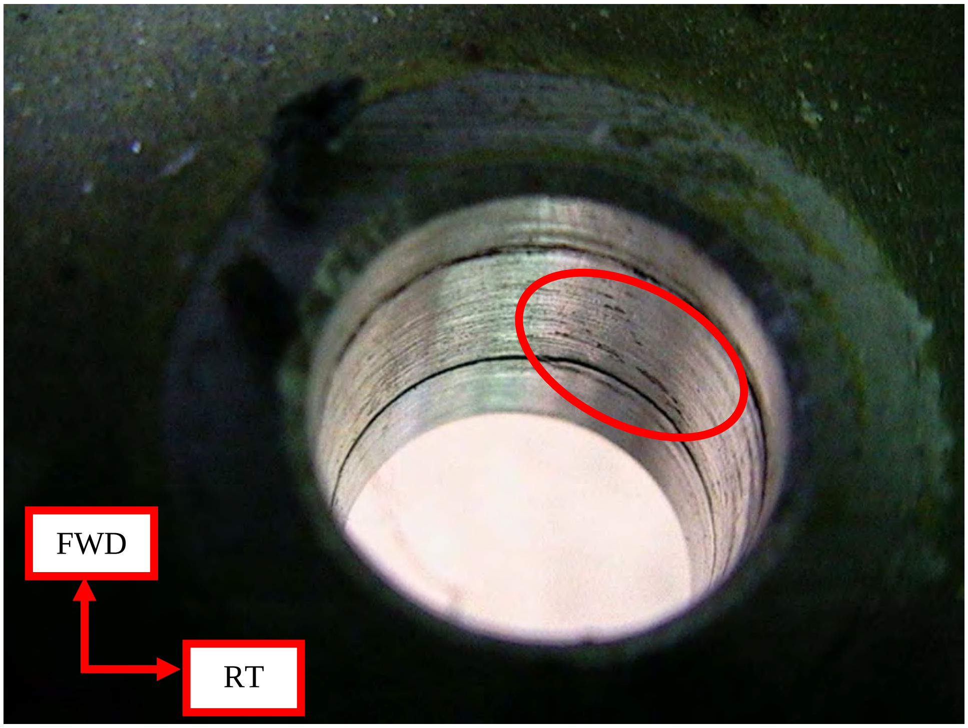


Figure 69. Gouges on the Right Wing Main Spar Splice Lower Attachment Points BL 0 Through BL 6

3. TEARDOWN EVALUATION PHASE.

Following the completion of the inspection phase, the teardown evaluation phase was performed to find defects not found by typical maintenance or airworthiness inspections. Every defect found in the inspection phase and the teardown evaluation phase was completely characterized during the microscopic examination portion of this project. The airplane was first disassembled into the major airplane sections to include: wing, forward fuselage, aft fuselage, cabin, landing gear, horizontal stabilizer, and vertical stabilizer. The systems' components, which include the right pitot probe, air conditioning unit, system wiring, tubes and hoses, were then inspected both on the airplane and, if necessary, after removal. A structural assessment of primary airplane structure using alternative NDI techniques was used to examine the wing spars and the horizontal and vertical stabilizer spars prior to disassembly in an effort to identify additional defects. Following the assessment using alternative NDI techniques, the major airplane sections were disassembled to remove the critical structural details and suspect locations identified during inspections. These details were then examined microscopically to find and characterize all defects in the structure. Certain cracks were sectioned and mounted for fractographic analysis to determine the extent of the flaw and the mode of failure for cracks (i.e., fatigue, stress corrosion cracking, etc.).

3.1 DISASSEMBLY OF AIRPLANE INTO MAJOR SECTIONS.

To facilitate the alternative NDI and detailed disassembly, the airplane was disassembled per the Piper Navajo Chieftain Maintenance and Service Manuals and Parts Catalog into its major airframe sections: wings, horizontal stabilizer, vertical stabilizer, landing gear, forward fuselage, aft fuselage, and cabin. Prior to any disassembly of the airplane sections, the engines were removed and were not investigated as part of this program. All systems' components, including wiring, were also removed and inspected using Advisory Circular 43.13-1, "Acceptable Methods, Techniques, and Practices." All disassembly was conducted using techniques to minimize damage to wiring, system components, and the underlying structure of the airplane.

3.2 INSPECTION OF SYSTEMS' COMPONENTS.

As airplanes age, degradation of mechanical systems may occur to some degree, and this degradation must be considered when designing airplane components. System components' inspections were accomplished to identify where mechanical wear, cracks, or corrosion affected critical areas. All mechanical system parts on the Piper Navajo Chieftain airplane were visually inspected for signs of degradation during the teardown process to determine if these parts were in airworthiness status. These mechanical systems included flight control cables and linkages, landing gear actuators, combustion heater, hydraulic actuator, hydraulic reservoir, airplane battery box, right engine pump, nose steering tube assembly, and pressurized lines.

As airplanes age, pressurized lines, systems' components, and wiring encounter conditions that may initiate corrosion, cracks, leaks, and wear. As these conditions occur, severe consequences may ensue with regard to the safety of flight. To determine if these conditions have occurred and their degradation severity, a teardown inspection of system components tubes, lines, and wiring was performed. A leak test was developed by constructing a system for pressurizing the lines in a consistent and reproducible manner. The inspection criteria and authorization used to evaluate selected components include the following:

- Advisory Circular 43.13.1B, Acceptable Methods, Techniques, and Practices
- Advisory Circular 43.13.2A, Acceptable Methods, Techniques, and Practices
- Piper Illustrated Parts Catalog
- Piper Navajo Chieftain Service Manual
- 14 CFR Parts 43 and 65
- ASTM E1417-99 Standard Practice for Liquid Penetrant Examination

3.2.1 Inspection Process for Systems' Components.

All components removed from the Piper Navajo Chieftain received a general visual inspection per available technical data. Ten components were selected for an extensive teardown to include a more in-depth visual inspection. Fluorescent liquid penetrant was used on selected components to help identify damage and defects. Lines and tubing were checked for leaks through pressurization, while wiring was visually inspected for defects and unsafe conditions.

3.2.1.1 Component Selection.

Each component was removed from the airplane during the detailed disassembly. The components listed in table 22 were selected for teardown and inspection based on component location in the airplane, component criticality, and functionality.

Table 22. Inspection Results of Systems' Components

Component	Part Number	Discrepancy	Inspection Type
Hydraulic actuator	451 826	No discrepancies found	Visual inspection
Pitot probe	42530-03	Plating worn Attach holes corroded	Visual inspection
Nose steering tube assembly	41437-00	No discrepancies found	Fluorescent liquid penetrant inspection
Hydraulic reservoir	757 457	No discrepancies found	Visual inspection
Right starter solenoid	487 152	Minor surface corrosion	Visual inspection
Torsion link assembly	40280-01	No discrepancies found	Paint stripped Visual inspection
Heat duct switch assembly	757 726	No discrepancies found	Visual inspection
Air conditioning unit	52670-02	Minor surface corrosion Crack on frame	Visual inspection

Table 22. Inspection Results of Systems' Components (Continued)

Component	Part Number	Discrepancy	Inspection Type
Battery box	53960-00	Crack Minor surface corrosion	Visual inspection
Right engine hydraulic pump	26802-08	Mount bracket corroded Minor corrosion on bolts	Visual inspection

3.2.1.2 Disassembly of Components.

Selected components identified in table 22 were disassembled followed by a close visual inspection from a qualified licensed airframe and power plant mechanic with an inspection authorization. Each component was disassembled and inspected for defects not conforming to the general guidelines of Advisory Circular 43.13.1B and 43.13.2A. The Piper Navajo Chieftain Service Manual was also used for inspection support criteria.

3.2.1.3 Systems' Component Inspection Findings.

Fifty percent of the ten components inspected revealed no notable discrepancies; however, the remaining 50 percent revealed cracks, excessive wear, and mild to severe corrosion. Aside from the corrosion identified on the right pitot probe, no other findings of note were identified throughout the component teardown phase.

The teardown inspection of the air conditioning unit, shown in figure 70 before disassembly and figure 71 after disassembly, revealed a 0.5-inch crack in the frame mount hole and some minor corrosion. Several heat exchanger cooling fins had minor damage on the top and bottom end. No other defects beside normal wear were found on this unit.

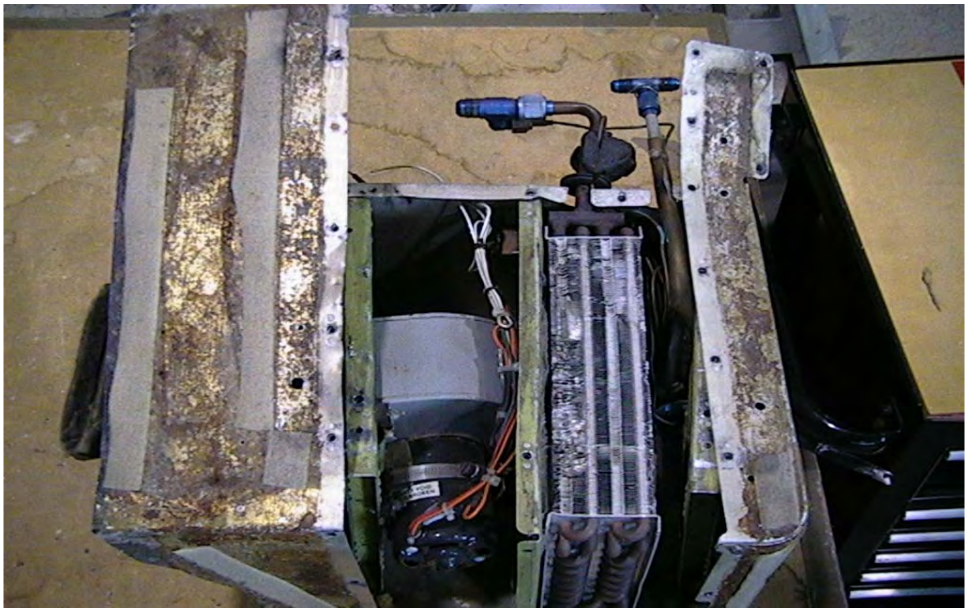


Figure 70. Air Conditioning Unit Before Disassembly

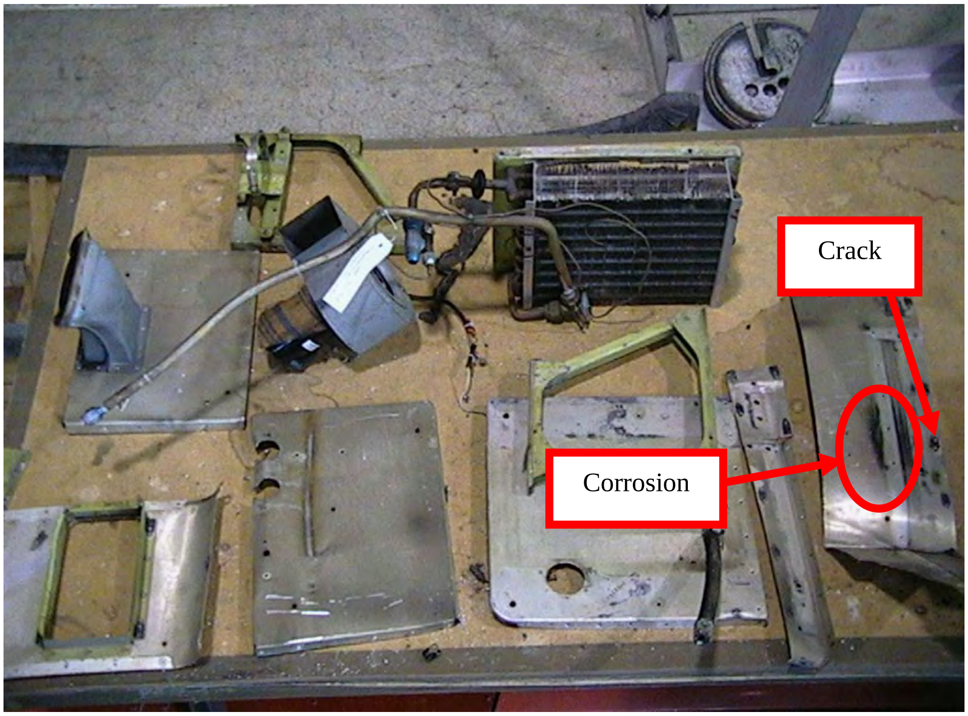


Figure 71. Air Conditioning Unit Cracked and Corroded

The inspections of the right pitot probe, shown in figure 72, revealed extensive galvanic corrosion around the element heating probe pins. This condition has the potential to inhibit the probe from heating, causing the probe to freeze in flight, which may result in false airspeed indications. There was also a small amount of corrosion found around the bolt attachment holes. Approximately 20% of the surface plating on the top side of the probe had deteriorated. This condition was attributed to possible high wind-speed environment for which the probe operates in.

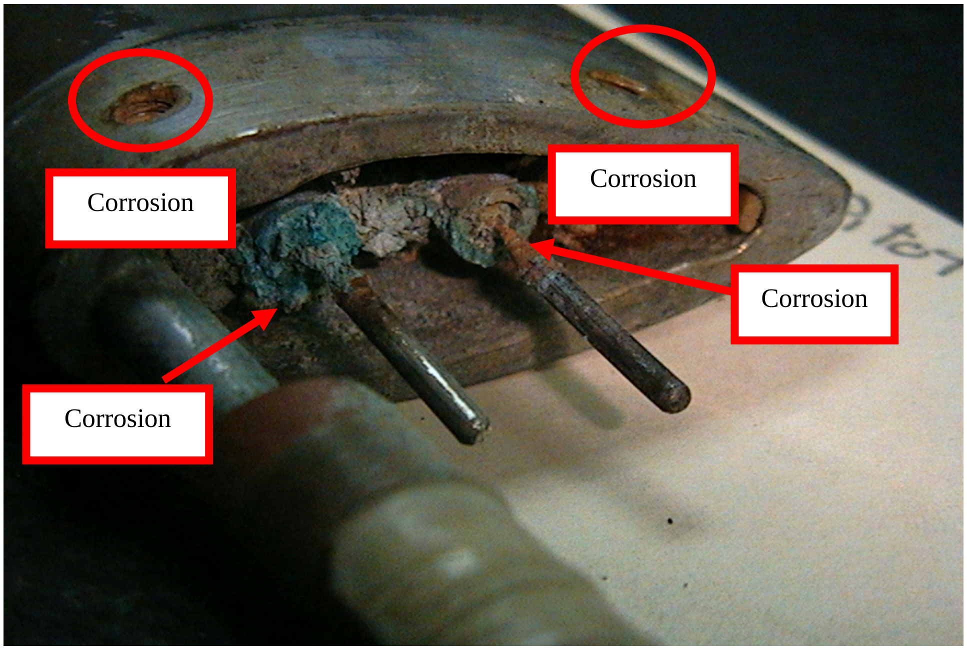


Figure 72. Corrosion on Pitot Probe

Figures 73 and 74 show the nose wheel steering rod before and after NDI inspection. The component was initially visually inspected by an airframe inspector. A fluorescent liquid penetrant inspection was performed by a qualified NDI inspector, revealing no defects. Figure 75 shows the landing gear torsion links after disassembly. A visual inspection was also performed with no defects found. Figure 76 shows the right engine hydraulic pump. Minor surface corrosion was found on both valve attachment bolt heads. The bolts on the valve attachments were disassembled for further inspection, and no corrosion was found beyond the bolt heads.

Figure 73. Nose Wheel Steering Rod Under Visible Light

Figure 74. Nose Wheel Steering Rod Under Ultraviolet Light

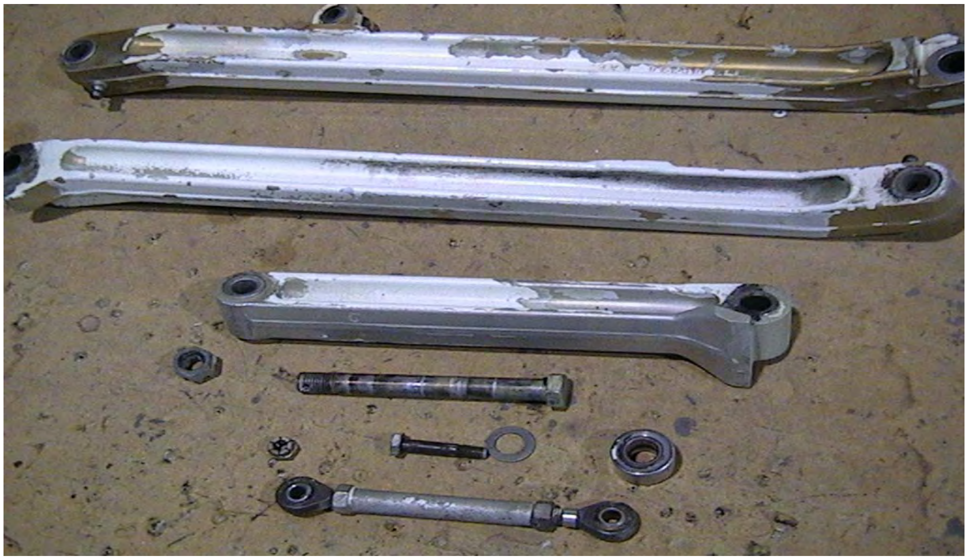


Figure 75. Torsion Links Disassembled

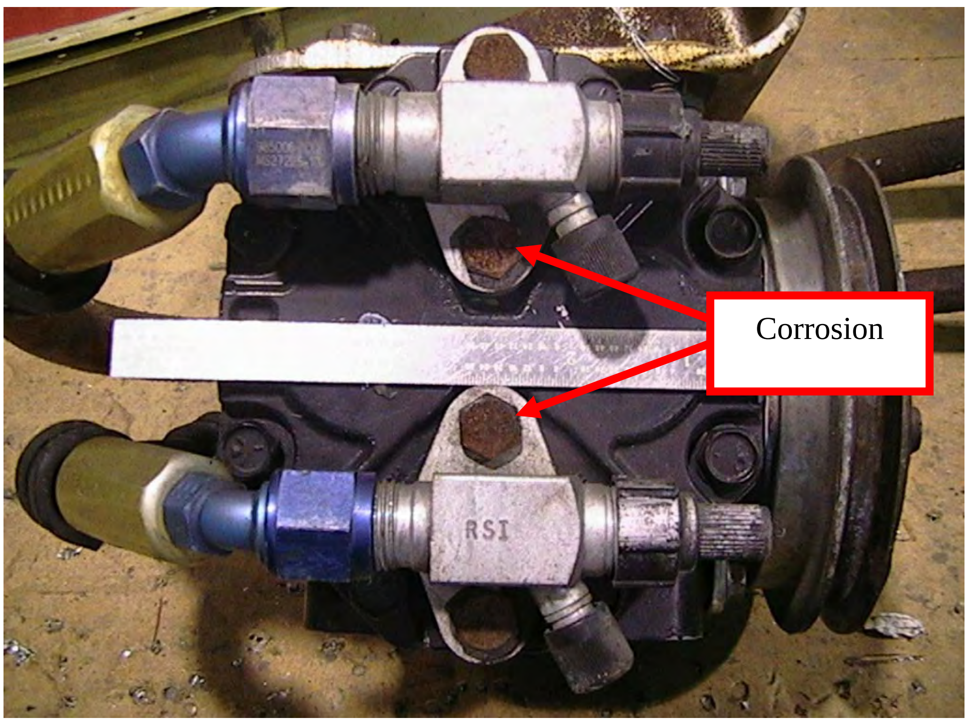


Figure 76. Right Engine Hydraulic Pump

The hydraulic actuator, shown in figure 77 before disassembly and figure 78 after disassembly, revealed no defects during an extensive visual inspection. Figure 79 shows a hydraulic reservoir, and figure 80 shows a heat duct switch assembly. Both were visually inspected, resulting in no defects found. The inspection of the battery box, shown in figure 81, revealed a 0.5-inch crack and minor uniform surface corrosion.

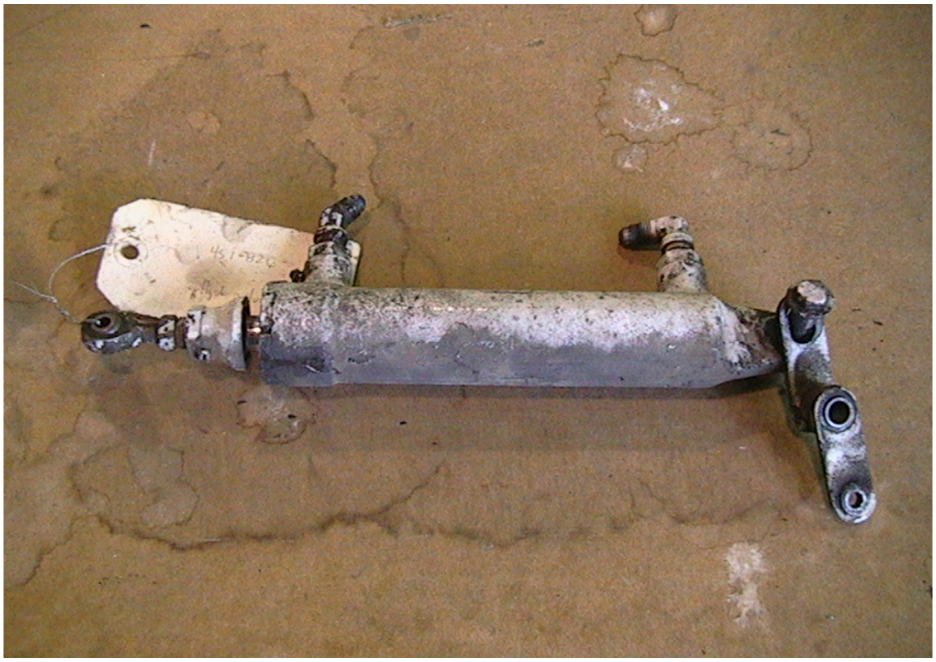


Figure 77. Hydraulic Actuator Before Disassembly

Figure 78. Hydraulic Actuator After Disassembly

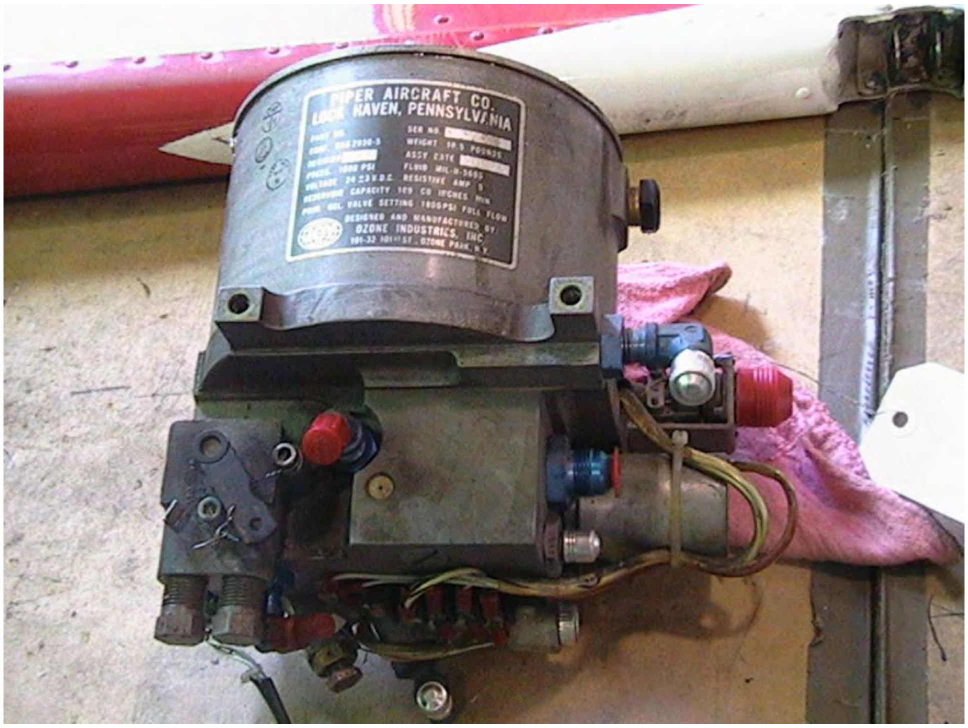


Figure 79. Hydraulic Reservior

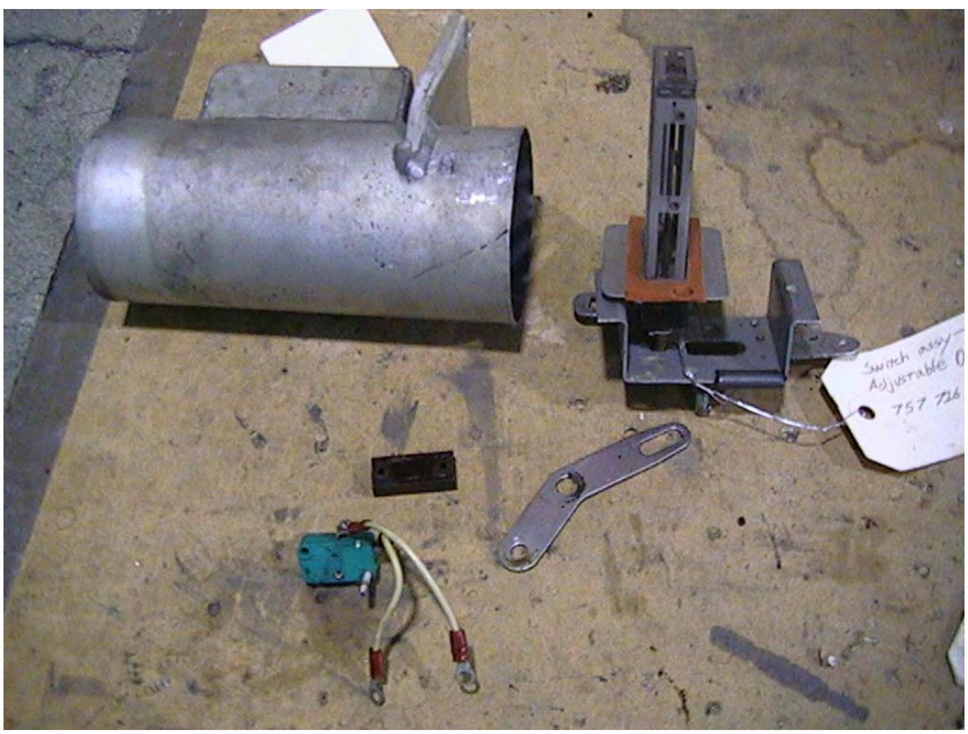


Figure 80. Heat Duct Switch Assembly

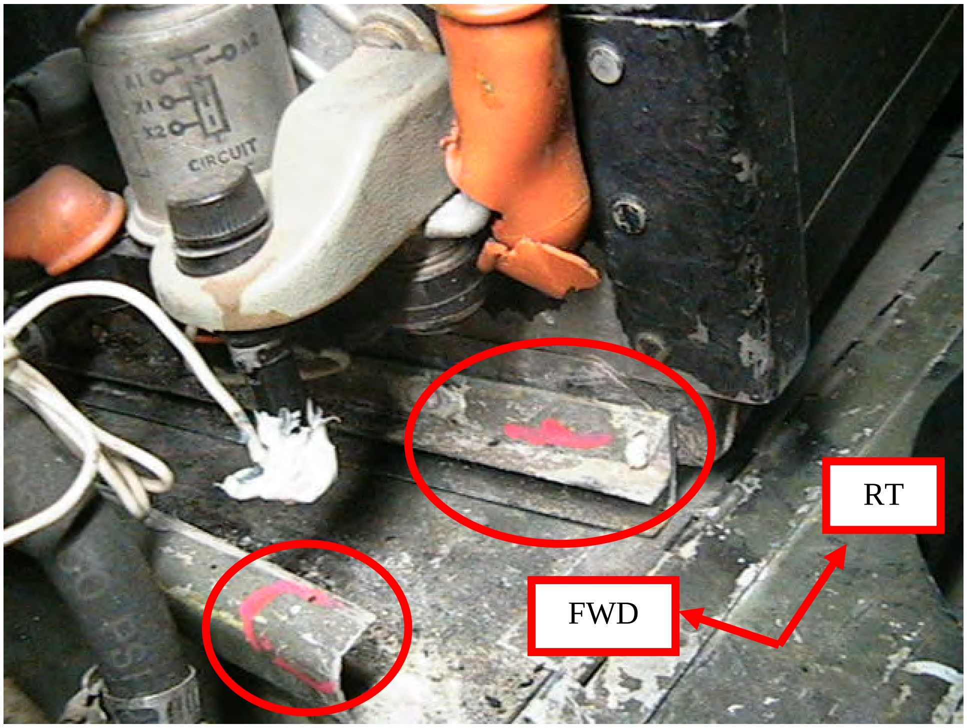


Figure 81. Battery Box Cracked and Corroded

3.2.1.4 Visual/NDI Inspections of Components Summary.

The Piper Navajo Chieftain systems' components shown in table 22 were visually inspected individually by a qualified licensed airframe and power plant mechanic for cracks, corrosion, and any signs of visible degradation that could possibly affect system performance. Initial inspections revealed no visible signs of severe degradation, which could possibly lead to a system failure. Fluorescent liquid penetrant was also used to evaluate component conditions of selected components; however, this inspection method resulted in no additional defects found.

The right pitot probe, shown in figure 72, the battery box, shown in figure 81, and the air conditioner unit, shown in figure 70, along with other selected components identified in table 22, were identified for anomalies not conforming to inspection criteria. One small crack and mild-to-extensive corrosion were identified on these components through visual inspection. The battery box showed minor corrosion around the weld seams. This is common due to its corrosive contents. The right pitot probe's inspection revealed minor corrosion around both attachment holes, while moderate to severe corrosion was discovered around both sensor pins, which could possibly affect pitot probe heating capabilities. An estimated 20% of surface silver plating was missing from the upper surface of the probe, which could possibly be attributed to wind and heat conditions that occur during normal operations. The air conditioner unit inspection revealed a 0.5-inch crack on the edge of the side frame. An extensive visual and NDI inspection revealed no defects on the nose steering tube assembly.

3.2.2 Leak Testing of Hydraulic, Oil, Fuel, and Oxygen Lines.

As airplanes are operated, the pressurized lines encounter conditions that may initiate corrosion or cracking. This degradation of the lines, when allowed to mature, may create a hole in the line from which the pressurized contents may escape. If a leak occurs, severe consequences may ensue with regard to the safety of the airplane. Pressurized lines are designed to withstand their predicted operating environment over the lifetime of the line. To determine the severity of the degradation in pressurized lines, a test was developed to look for leaks in the tubing by constructing a system for pressurizing the lines in a consistent and reproducible manner. Figure 82 illustrates the pressurization system.

Figure 82. Apparatus Used to Pressurize Lines

Two hundred and three lines, tubes, and hoses were visually inspected and leak checked during the teardown phase. The majority of findings were minor scratches, nicks, gouges, flared tube ends, and minor corrosion. One major finding, shown in figures 83 and 84, was a manifold pressure line with a small 1/16-inch-diameter hole causing it to fail the leak check. This pressure line had a small gouge surrounding the hole, measuring approximately 1 inch. This is indicative of a chafing condition.

Figures 85 and 86 show two separate line fittings with flared ends, which could cause leaking. The line shown in figure 86 also had a small gouge on the b-nut, and figure 87 shows the pneumatic de-ice valve lower fitting with mild corrosion on the fitting threads. The overall condition of all tubes, fittings, and hoses inspected were classified as satisfactory. The majority of findings may be contributed to airplane normal wear and aging effects.

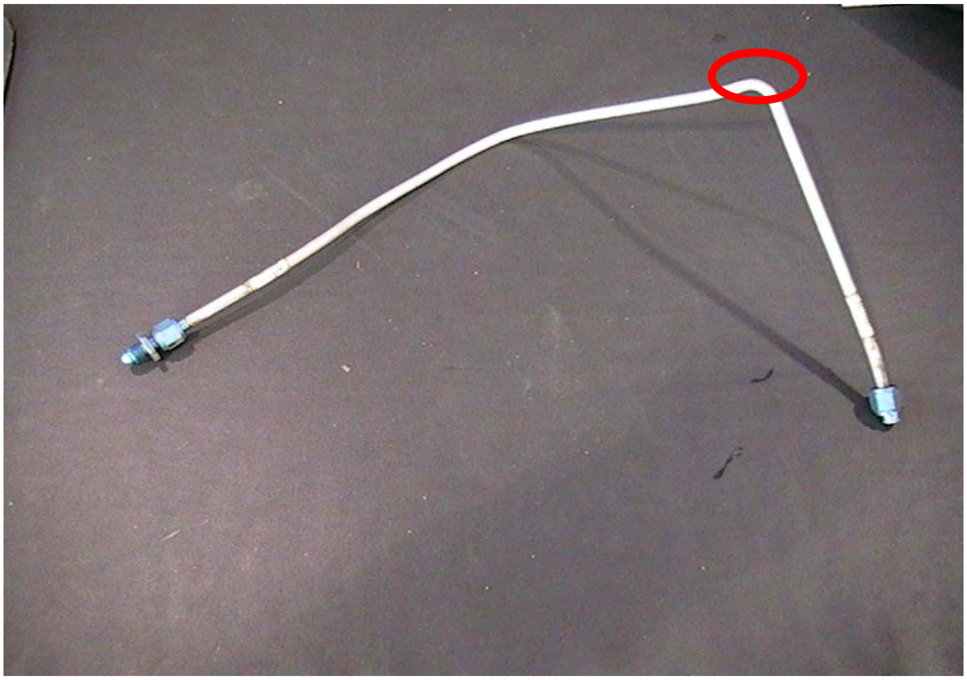


Figure 83. Location of Hole in the Left Wing Manifold Pressure Line

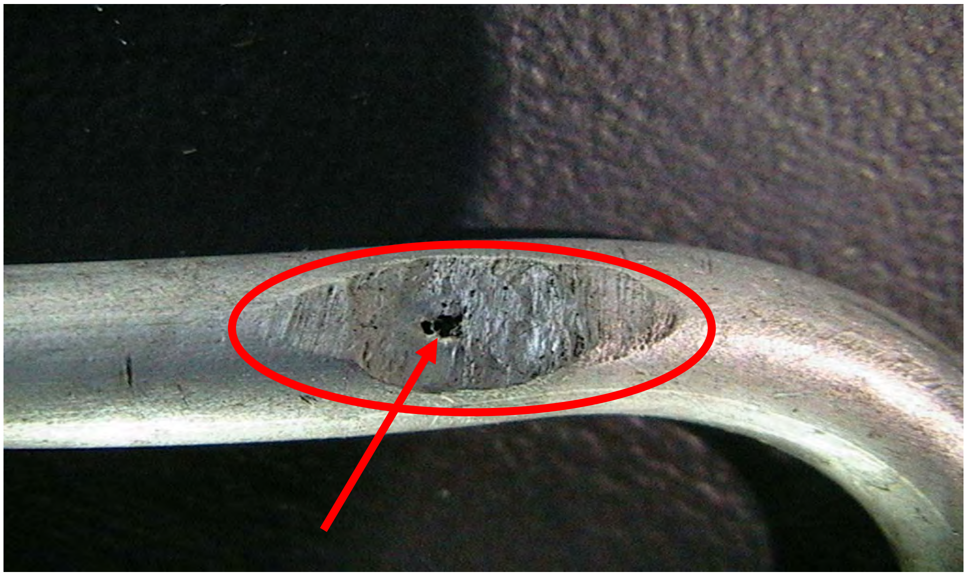


Figure 84. Close-Up View of Hole in the Left Wing Manifold Pressure Line

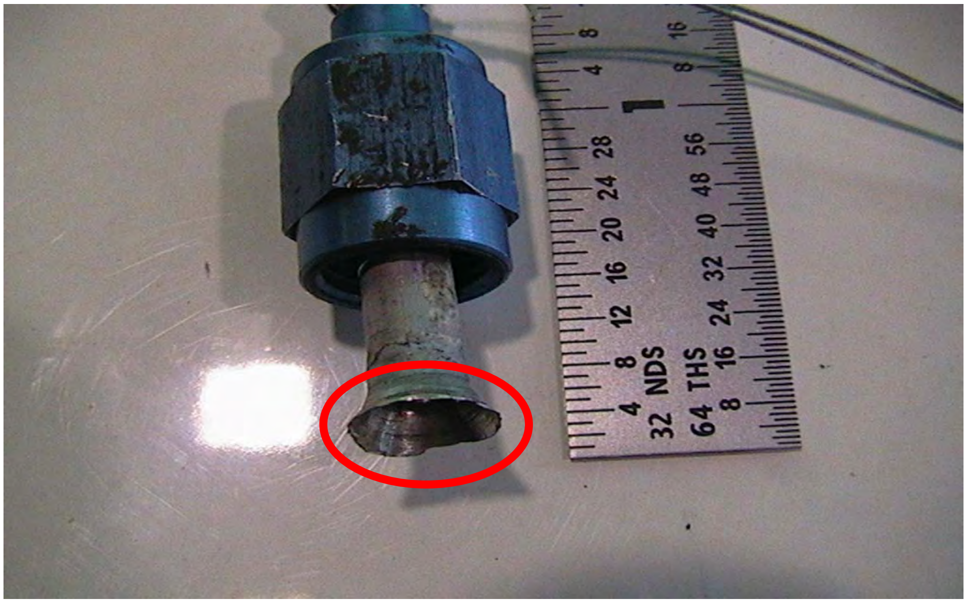


Figure 85. Upper Oil Pressure Line Fitting With Flared End

Figure 86. Lower Oil Pressure Line Fitting With Gouge and Flared End

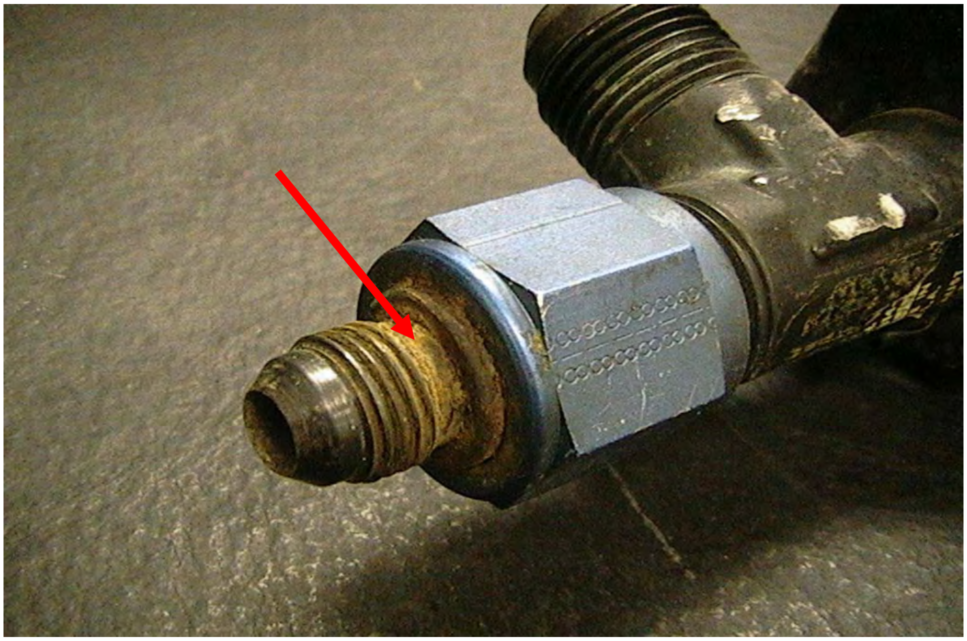


Figure 87. Corrosion on Pneumatic De-Ice Valve

3.3 STRUCTURAL ASSESSMENT USING ALTERNATIVE NDI TECHNIQUES.

During this research program, alternative NDI techniques were implemented to assess the primary airplane structure beyond the airworthiness inspections prior to teardown. The wing spars and horizontal and vertical stabilizer spars were inspected using two eddy-current-based inspection techniques capable of detecting surface and subsurface cracks and areas of corrosion. The sliding probe, shown in figures 88 and 89, detects cracks in up to three layers of metal, and the spot probe, shown in figures 90 and 91, can detect corrosion in multiple layers.

The primary purpose of these inspections was to find additional defects in the airframe using alternative NDI techniques prior to disassembly. These techniques are not called out in the Piper Navajo Chieftain maintenance manuals, and they are not typically used for inspection on general aviation airplanes. Therefore, no procedures had been established or validated for using these techniques on the Piper Navajo Chieftain airplane. Using existing structure, inspectors attempted to identify target areas for further microscopic examination. No effort was made to evaluate the capabilities of the alternative NDI techniques, and conclusions about the capabilities of the sliding probe or spot probe should not be made from the results presented in this report.

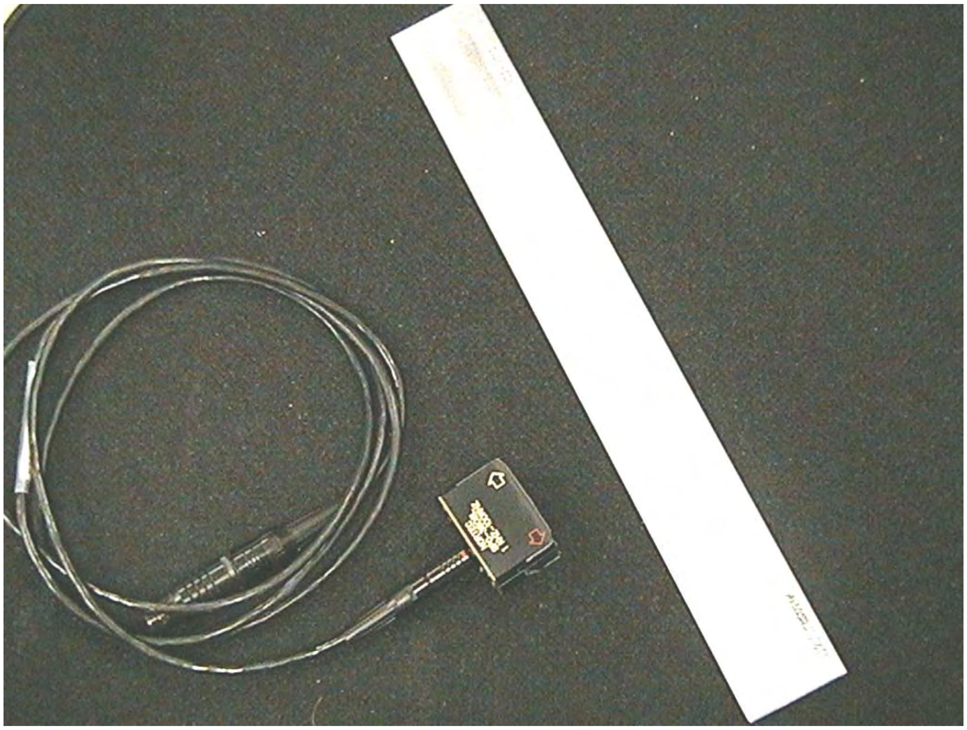


Figure 88. Sliding Probe and Standard

Figure 89. Demonstration of the 1- to 100-kHz Sliding Probe

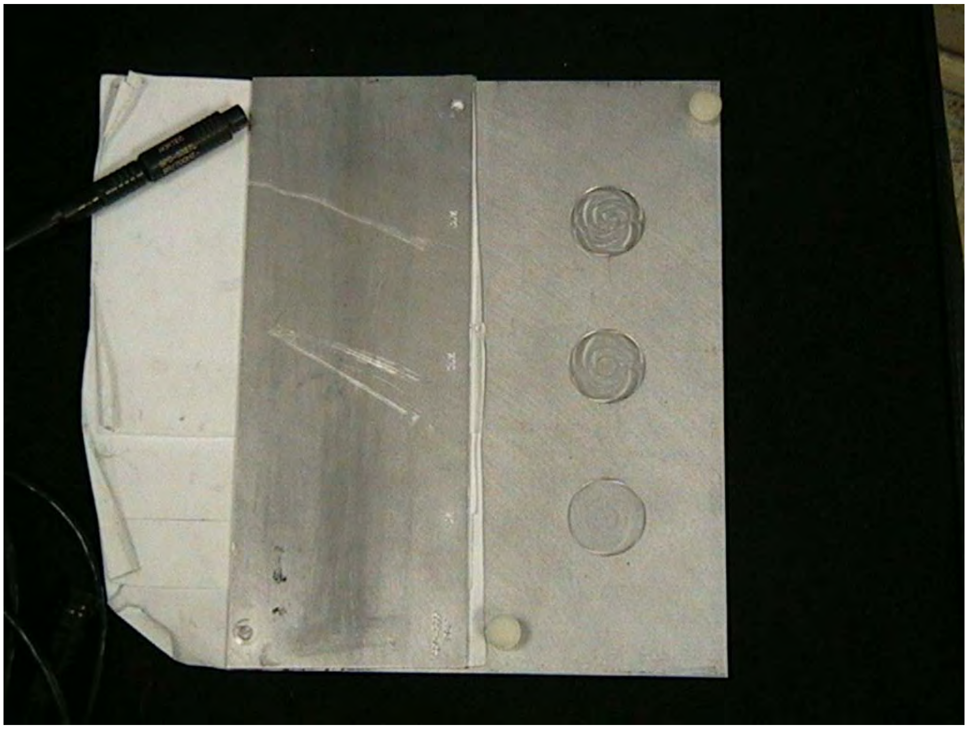


Figure 90. Spot Probe and Standard

Figure 91. Demonstration of the 100-Hz to 80-kHz Spot Probe

Instrument settings, which are determined by a number of factors, including material composition, number and thickness of material layers, and desired signal penetration depth, are all important factors when performing eddy-current-based inspections. Table 23 shows the machine settings used during these inspections. The results of these inspections are presented in tabular and pictorial format in the subsequent sections for the left wing, right wing, horizontal stabilizer, and vertical stabilizer.

Table 23. Nondestructive Inspection Machine Settings

Method	Frequency	Mode	Gain
Sliding probe	1-6 kHz	Single frequency	75 dB
Spot probe	1-3 kHz, 500 Hz-1.5 kHz	Dual frequency	75 dB

During these inspections, the airplane structure limited which techniques could be implemented on the spars to be inspected. Due to the width of the horizontal and vertical stabilizer spars, the front and rear wing spars, and the outboard 75% of the wing main spar, the spot probe inspections could not be performed because of edge effect. The inboard quarter of the wing main spars were the only areas with sufficient width to support spot probe inspections. Difficulties were also encountered with the sliding probe on the vertical stabilizer, where the presence of vortex generators caused large uninspectable areas. Since the intent of the inspections was to identify flaws in the structure prior to teardown, the vortex generators were removed and the inspection proceeded as planned. Another limitation, based on the structure of the Piper Navajo Chieftain, occurred with the sliding probe. The sliding probe can only detect cracks parallel to and up to 30 degrees off axis from its scan direction. For the wing spars, the only permissible scan direction, due to edge effect considerations, was parallel to the load path. Since typical fatigue cracks grow perpendicular to the load path, they could not be detected by this inspection procedure.

3.3.1 Left Wing Results.

The upper and lower surfaces of the front, main, and rear spars of the left wing were inspected for surface and subsurface cracks and areas of corrosion using the sliding probe and spot probe. All indications were further investigated during the postdisassembly NDI and microscopic examination. Table 24 lists all crack and corrosion indications encountered during the alternative NDI of the left wing. Figures 92 and 93 show the locations of the indications found on the lower and upper surfaces of the left wing, respectively. Figure 94 shows the screen display for the crack indication located on the left wing rear spar upper cap WS 217. This figure clearly shows the difference between a liftoff signal, a signal from a rivet, and a crack indication.

Table 24. Nondestructive Inspection Indications on the Left Wing

NDI Method	Indication	Location
Sliding probe	Crack	Left wing main spar lower cap WS 197.25
Sliding probe	Crack	Left wing rear spar lower cap WS 197.25
Sliding probe	Crack	Left wing rear spar upper cap WS 217

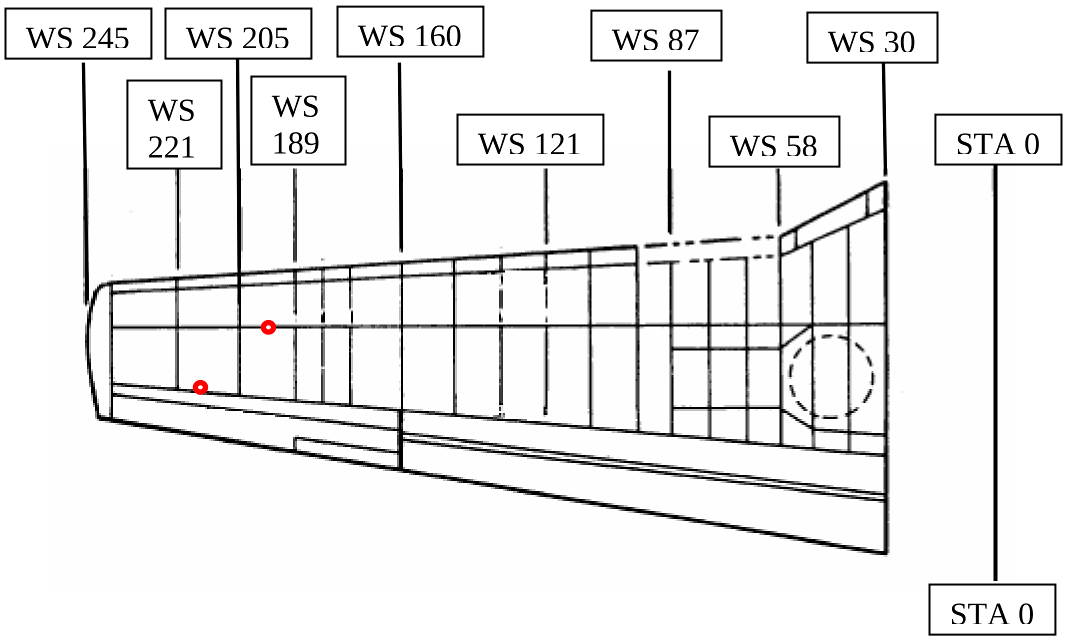


Figure 92. Left Wing Lower Surface NDI Indications

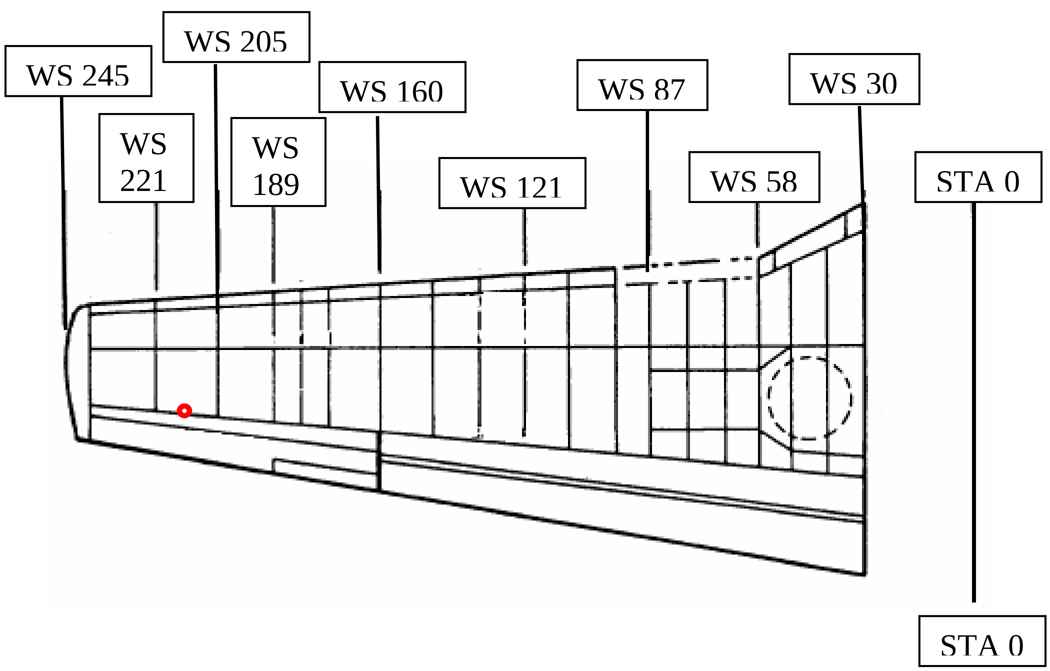


Figure 93. Left Wing Upper Surface NDI Indications

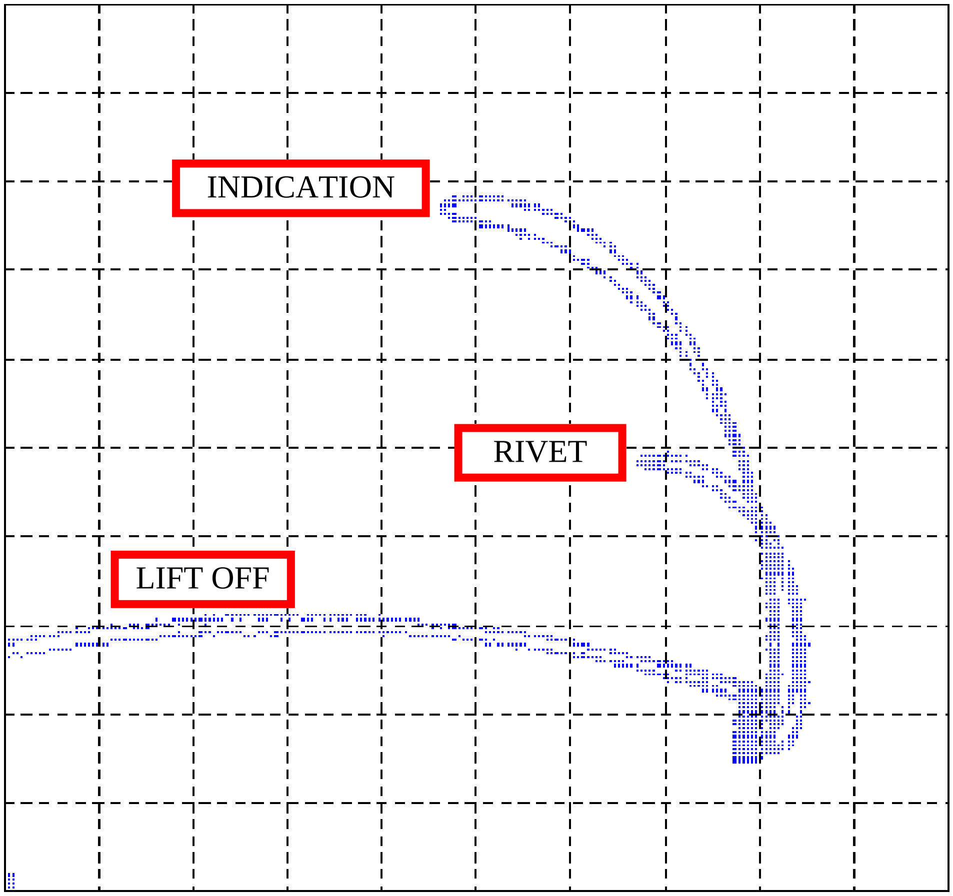


Figure 94. Left Wing Rear Spar Upper Cap WS 217 Indication Screen Display

3.3.2 Right Wing Results.

As with the left wing, the upper and lower surfaces of the right wing main, front, and rear spars were inspected by sliding probe and spot probe. All noted indications were examined microscopically using the postdisassembly NDI. Table 25 shows all indications found on the right wing. Figures 95 and 96 show the locations of the indications found on the lower and upper surfaces of the right wing, respectively. Figures 97 and 98 show the screen displays for the crack indications located on the right wing front spar cap WS 184.5 and the main spar lower cap WS 232.2, respectively. These figures clearly show the difference between a liftoff signal, a signal from a rivet, and a crack indication.

Table 25. Nondestructive Inspection Indications on the Right Wing

NDI Method	Indication	Location
Sliding probe	Crack	Right wing front spar lower cap WS 184.5
Sliding probe	Crack	Right wing main spar lower cap WS 198.5
Sliding probe	Crack	Right wing main spar lower cap WS 232.2
Sliding probe	Crack	Right wing main spar lower cap WS 233.5
Sliding probe	Crack	Right wing main spar upper cap WS 110

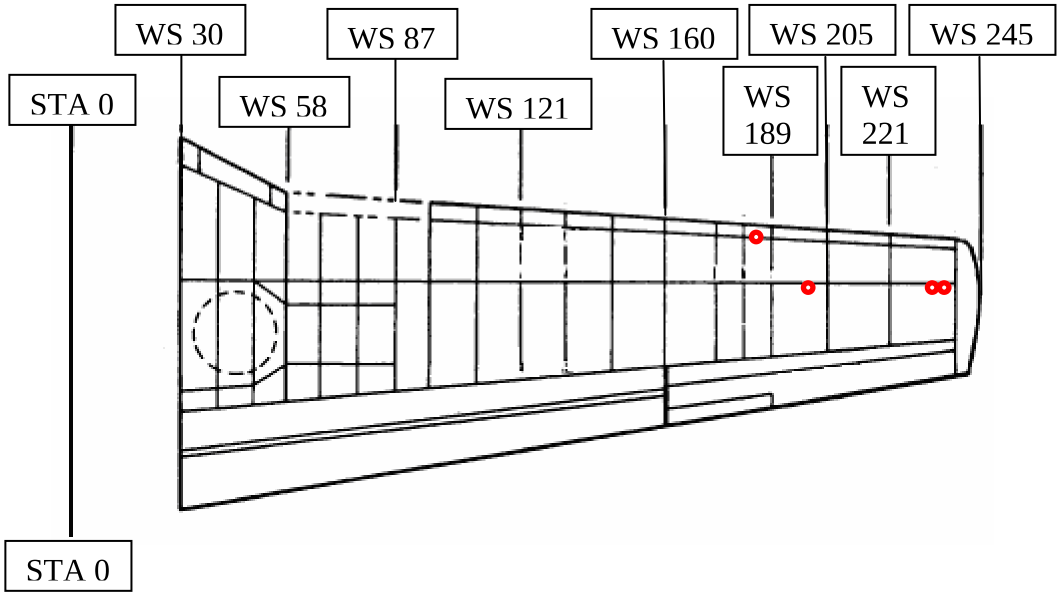


Figure 95. Right Wing Lower Surface NDI Indications

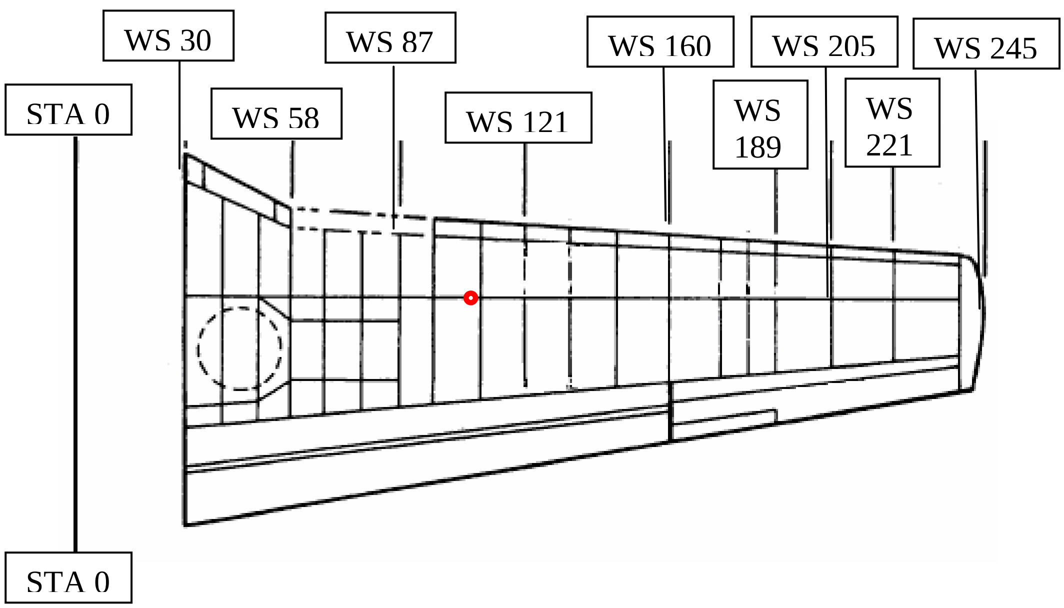


Figure 96. Right Wing Upper Surface NDI Indications

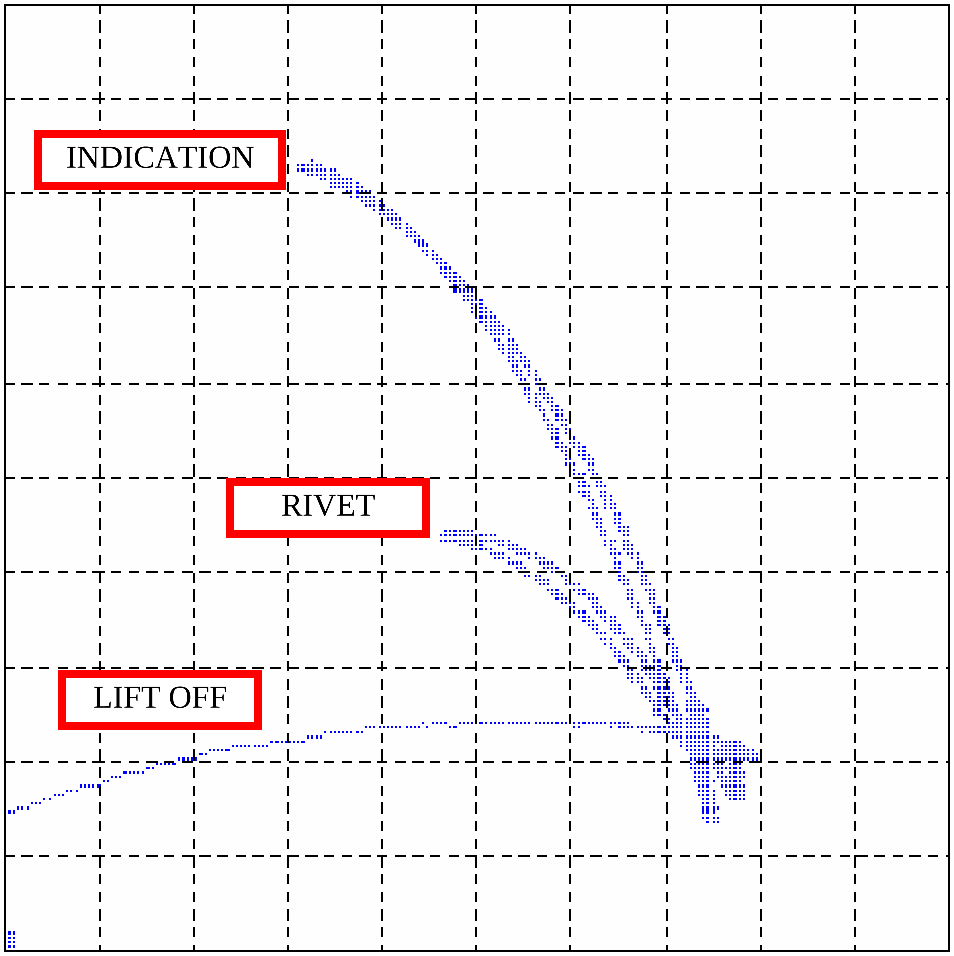


Figure 97. Right Wing Front Spar Lower Cap WS 184.5 Indication Screen Display

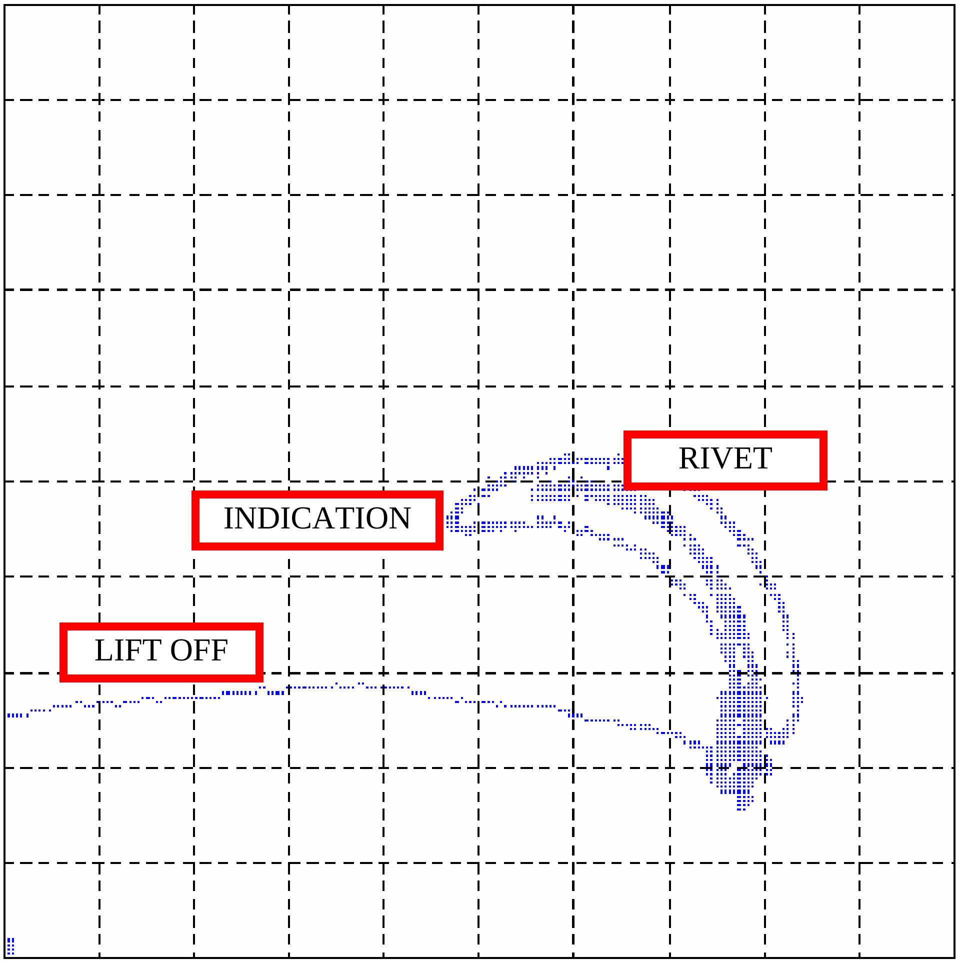


Figure 98. Right Wing Main Spar Lower Cap WS 232.2 Indication Screen Display

3.3.3 Horizontal Stabilizer Results.

The upper and lower surfaces of the front and rear spars of the horizontal stabilizer were inspected for cracks and corrosion using the sliding probe. All indications were noted and further investigated for validity during the postdisassembly NDI and microscopic examination phase of this project. Table 26 shows the indications reported during these inspections and includes information on the inspection method, type of indication, and the airplane location where indications were found. Figures 99 and 100 illustrate the locations of indications on the upper and lower surfaces of the horizontal stabilizer, respectively.

Table 26. Nondestructive Inspection Indications on the Horizontal Stabilizer

Method	Indication	Location
Sliding probe	Crack	Left horizontal stabilizer rear spar lower cap BL 12.5
Sliding probe	Crack	Left horizontal stabilizer rear spar lower cap BL 12.5
Sliding probe	Crack	Left horizontal stabilizer rear spar lower cap BL 39.5
Sliding probe	Crack	Right horizontal stabilizer rear spar lower cap BL 68
Sliding probe	Crack	Left horizontal stabilizer front spar lower cap BL 98-104.75
Sliding probe	Crack	Right horizontal stabilizer rear spar upper cap BL 34.5-37.25
Sliding probe	Crack	Right horizontal stabilizer rear spar upper cap BL 83.25-92
Sliding probe	Crack	Left horizontal stabilizer rear spar upper cap BL 90-92
Sliding probe	Crack	Left horizontal stabilizer front spar upper cap BL 100.25-104.75

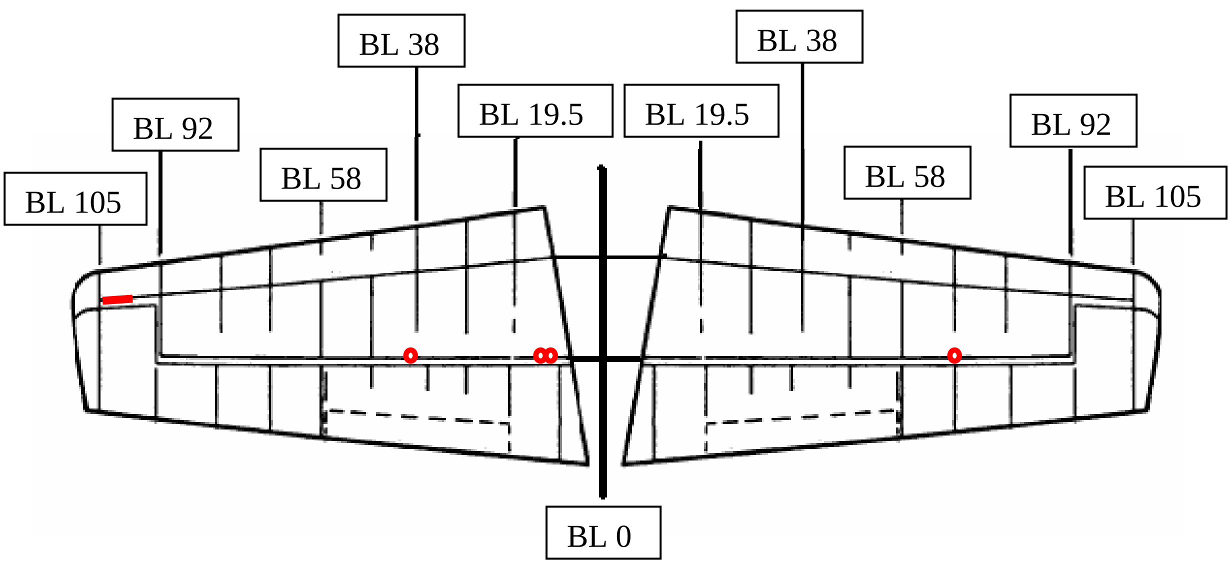


Figure 99. Horizontal Stabilizer Lower Surface NDI Indications

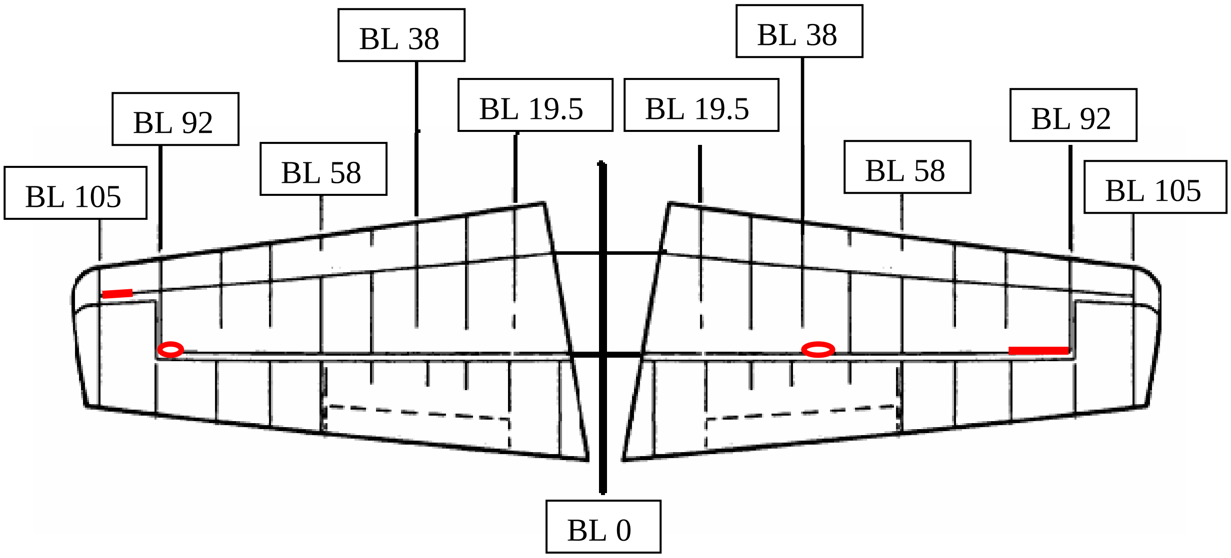


Figure 100. Horizontal Stabilizer Upper Surface NDI Indications

<u>3.3.4 Vertical Stabilizer Results</u>.

The left and right surfaces of the front and rear spars of the vertical stabilizer were inspected for cracks and corrosion using the sliding probe. All indications were noted and further examined during the postdisassembly NDI and microscopic examination portion of this project. Table 27 shows the indications reported during these inspections and includes information on the inspection method, type of indication, and the airplane location where indications were found. Figure 101 illustrates the locations of indications on the right surfaces of the vertical stabilizer. No indications were found on the left surface of the vertical stabilizer.

Table 27. Nondestructive Inspection Indications on the Vertical Stabilizer

Method	Indications	Location
Sliding probe	Crack	Right vertical stabilizer front spar cap WL 24.25
Sliding probe	Crack	Right vertical stabilizer front spar cap WL 27.25

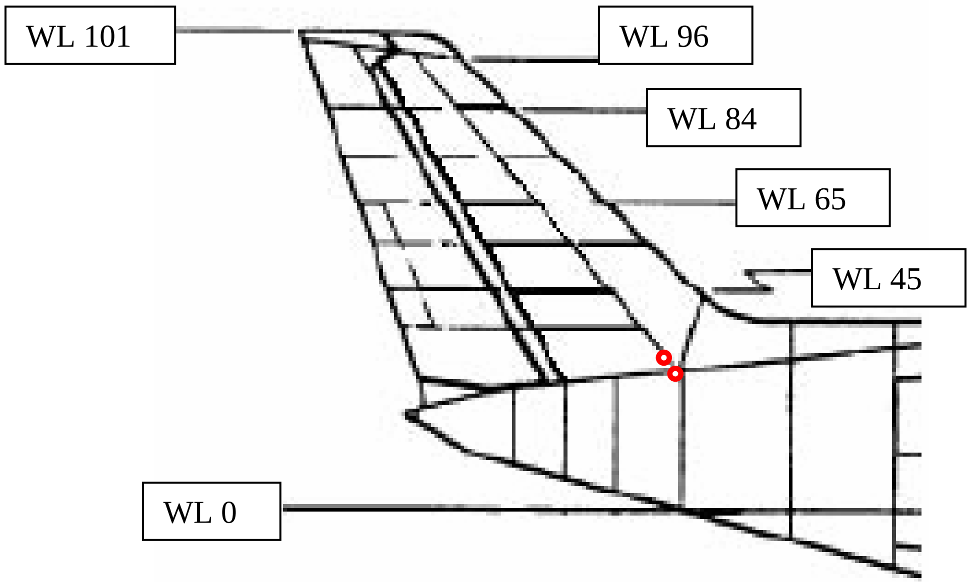


Figure 101. Vertical Stabilizer Right Side NDI Indications

3.4 TEARDOWN EVALUATION.

Following the completion of the systems' components inspections and the assessment of primary airplane structure using alternative NDI techniques, the teardown evaluation of the airplane began. Teardown evaluation involves the detailed disassembly of the major airplane sections to remove primary structure and areas with damage found during the inspections, paint removal and etching of the structure to aid in damage identification and characterization, postdisassembly NDI to identify damage requiring characterization, microscopic examination to identify and characterize all damage, and fractographic analysis to determine failure modes of selected cracks.

3.4.1 Teardown Evaluation Process.

The teardown evaluation is divided into five major tasks: detailed disassembly of the airplane sections, parts processing, postdisassembly NDI, microscopic examination, and fractographic analysis. At the conclusion of the systems' inspections and the structural assessment using alternative NDI techniques, the airplane was completely disassembled to allow full access to the primary structure and defects identified during the previous inspections. To assist in damage detection and characterization, the parts were then processed by removing the paint and etching the part surface. Next, the parts are subjected to a postdisassembly NDI to assist in the identification of all defects in the structure. All defects are then characterized during the microscopic examination. Corrosion type, area, and severity are determined during the microscopic examinations as well as crack lengths. After that, fractographic analysis used to determine failure mode, such as fatigue, stress corrosion cracking, for selected cracks.

3.4.1.1 Detailed Disassembly Procedure.

During the detailed disassembly of the major airplane sections, all primary structure is removed from the airplane using disassembly methods that minimize the potential for damage to the surround structure, systems' components, and wiring. All parts, when removed from the airplane, are subjected to a visual inspection to identify damage such as cracks, corrosion, and wear. The parts are then tagged to aid with identification. All part tags include: part number, part name, and airplane location. These parts are then assessed to determine if they will be subjected to a full microscopic examination. Areas examined meet one or more of the following criteria:

- Critical areas prescribed in the airworthiness inspections for selected structural locations
- Critical areas determined to be uninspectable during the development of airworthiness inspections for selected structural locations
- Areas assessed using alternative NDI techniques
- Areas with visible damage found during the inspection phase or during the postdisassembly visual inspection
- Any other areas of interest determined by the industry contacts or NIAR personnel

3.4.1.2 Parts Processing.

To aid with damage detection on parts selected for a full microscopic examination, paint is removed using Type V size 20/30 plastic media. The parts are then etched using a sodium hydroxide base. Paint removal and etching are not performed on cracks identified during the inspection phase or during the postdisassembly visual inspection, as media blasting and etching can damage the fracture face, removing evidence of crack failure mode. In this case, cracks are extracted from the part and set aside for microscopic/fractographic analysis, while the remainder of the part is subjected to the normal process.

3.4.1.3 Postdisassembly NDI.

Following the detailed disassembly of the Piper Navajo Chieftain, comprehensive NDI, using fluorescent liquid penetrant and magnetic particle inspection, was performed on all primary structure and all parts found to have defects during disassembly. These inspections were performed to aid in the identification of all defects present in the structure. The liquid penetrant inspections were performed according to ASTM 1417, using Type 1, Method A, Level 3 sensitivity, water washable penetrant. The magnetic particle inspections were performed according to ASTM 1444, using longitudinal and circular magnetism. All defects recorded during these inspections were noted and further examined and characterized during the microscopic examinations.

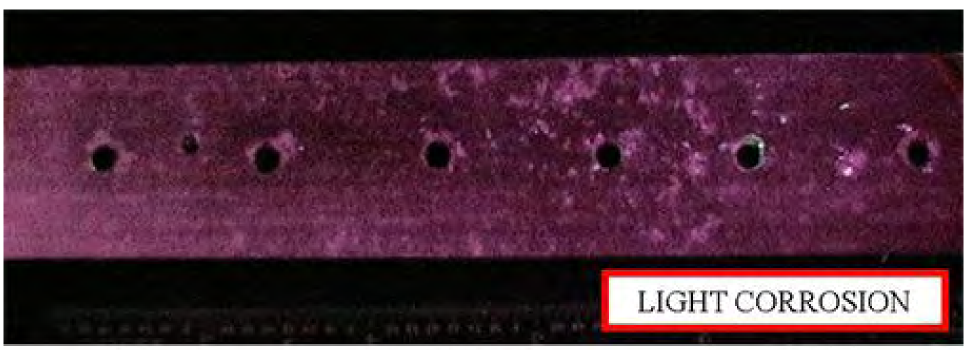
3.4.1.4 Microscopic Examination.

All primary structure was examined with the aid of a 7-45 power optical microscope following the postdisassembly NDIs. The lengths of all cracks were noted, as well as the surface area and depth of each area of corrosion. The severity of the corrosion was classified by the percentage of the thickness loss as follows:

- Light—0-2% thickness loss
- Light-moderate—2%-5% thickness loss
- Moderate—5%-7% thickness loss
- Moderate-severe—7%-10% thickness loss
- Severe— >10% thickness loss

Corrosion effects were classified by percentage of thickness loss as it seemed most representative of the actual damage done by corrosion. Typically, corrosion is classified by pit depth; however, for the purposes of this study, pit depth is not representative of the severity of damage caused. The different classification levels used in this program are unique to this research and are not representative of other studies. NIAR recommends that these classification levels be used for future studies. It is important to note that corrosion is not assumed as uniform throughout the entire area, but that it represents the maximum depth of corrosion in that area.

During the teardown evaluations, light and light-moderate corrosion was found to be widespread over the airplane. This is typical of an airplane of this age and operational environment history. Examples of light and light-moderate corrosion are shown in figures 102 through 105, as opposed to providing a photograph of each occurrence on the airplane.



Light Corr - UV

Figure 102. Fluorescent Liquid Penetrant Indication of Light Corrosion

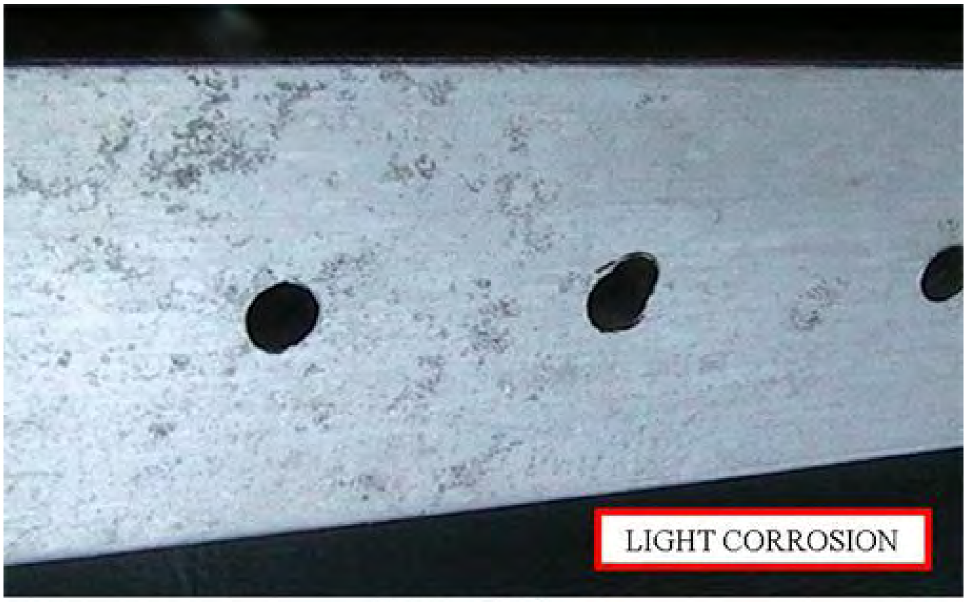


Figure 103. Light Corrosion Under Visible Light

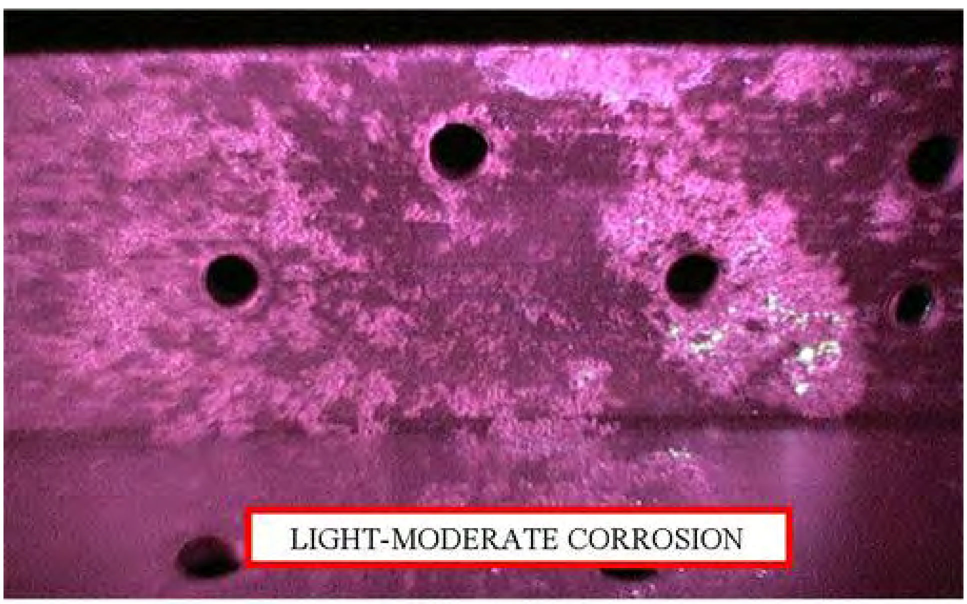


Figure 104. Fluorescent Liquid Penetrant Indication of Light-Moderate Corrosion

Figure 105. Light-Moderate Corrosion Under Visible Light

3.4.2 Teardown Evaluation Results.

During the teardown evaluation, 697 parts were examined in an effort to locate and characterize all areas of damage on the Piper Navajo Chieftain. Three hundred and three defects were identified and characterized during this process on the left wing, right wing, fuselage, horizontal stabilizer, and vertical stabilizer. One hundred and sixty-one cracks were identified, 104 areas of corrosion, 28 instances of wear, and 10 areas of other damage. On the left wing, 27 cracks, 39 areas of corrosion, 12 areas of wear, and 1 area of other damage was recorded; and 21 cracks, 23 areas of corrosion, 11 instances of wear, and 2 occurrences of other damaged were documented on the right wing. The teardown evaluation of the fuselage revealed 79 cracks, 33 areas of corrosion, 5 occurrences of wear, and 4 instances of other damage. On the horizontal stabilizer, 7 cracks, 4 areas of corrosion, and 1 area of other damage was noted; and 27 cracks, 5 areas of corrosion, and 2 areas of other damage were documented and characterized on the vertical stabilizer.

3.4.2.1 Left Wing.

Of the 196 parts subjected to the teardown evaluation from the left wing, 79 defects were noted and characterized. There were 27 cracks, 39 areas of corrosion, 12 areas of wear, and 1 damaged area that were identified on the left wing. Light to light-moderate scattered corrosion was observed on the surface of many of the left wing main spar cap components; and three areas of wear were identified on the front spar, and one on the rear spar. Numerous cracks were also identified on the left wing bulkhead wing fillets; and multiple areas of wear were observed on the rear spar assembly lower aft angle. Four areas of corrosion were also noted on the nacelle front lower floor.

3.4.2.1.1 Left Wing Front Spar.

The structural stackup for the left wing front spar is shown in figure 106. All defects identified on the left wing front spar during the teardown evaluation are listed in table 28. Two areas of wear found on the left wing front spar attachment point is shown in figure 107, and macroscopic views of both areas are shown in figures 108 and 109. The maximum thickness loss due to these areas of wear was 2%. An additional area of wear was found on the left wing front spar web at WS 62.5, as shown in figure 110. A macroscopic view of this area of wear, which caused a maximum localized thickness loss of 83%, is shown in figure 111. One crack was found on the left wing front spar web at WS 147.5, as shown in figure 112. The fluorescent liquid penetrant indication is shown in figure 113. A microscopic view of this 0.233-inch crack is shown in figure 114.

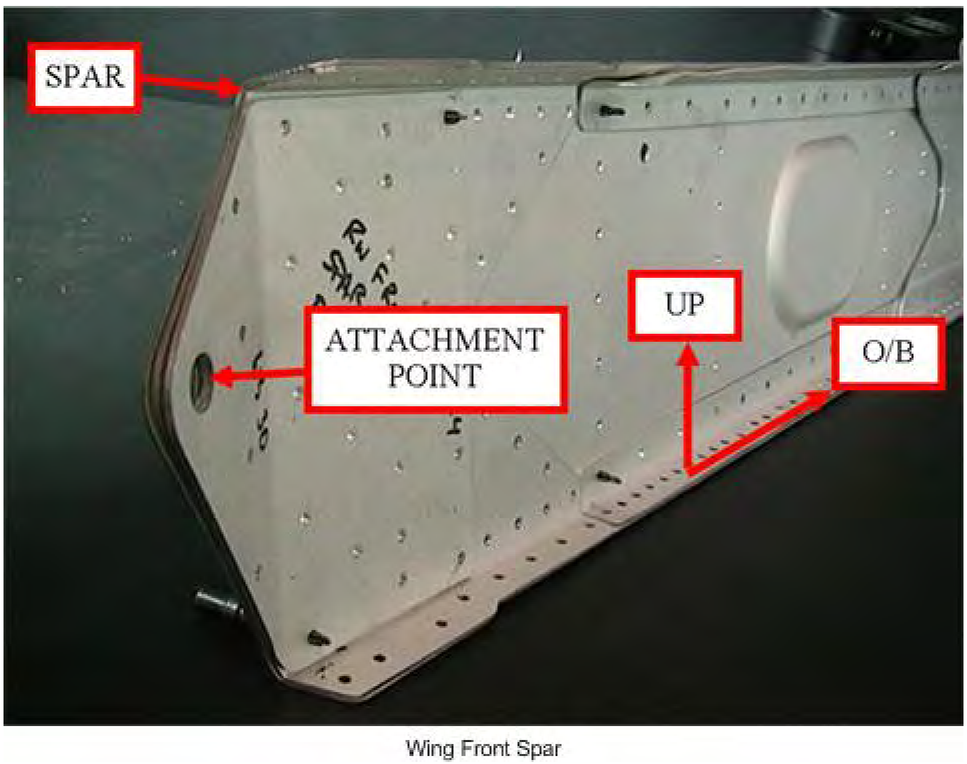


Figure 106. Structural Stackup of the Left Wing Front Spar

Table 28. Inspection Results From the Left Wing Front Spar

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing front spar attachment point, figure 107	Wear	WS 30	0.5 inch	Corrosion indication	Wear 2% thickness loss	108
	Wear	WS 30	0.239 inch	Corrosion indication	Wear 2% thickness loss	109
Left wing front spar web, figure 110	Wear	WS 62.5	0.364 inch by 1.122 inches	No Indication	Wear 83% thickness loss	111
Left wing front spar web, figure 112	Crack	WS 147.5	0.233 inch	Crack indication	Bend radii	113 114

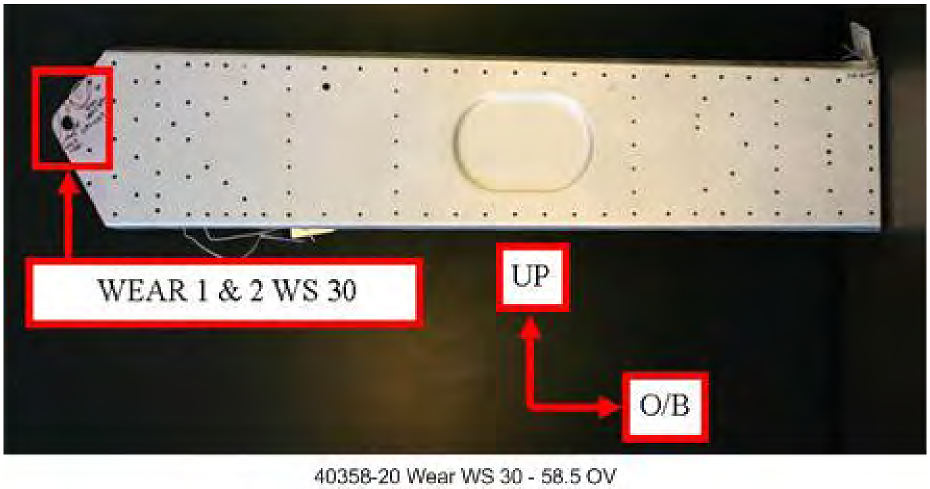


Figure 107. Location of Wear 1 and 2 on the Left Wing Front Spar Attachment Point WS 30

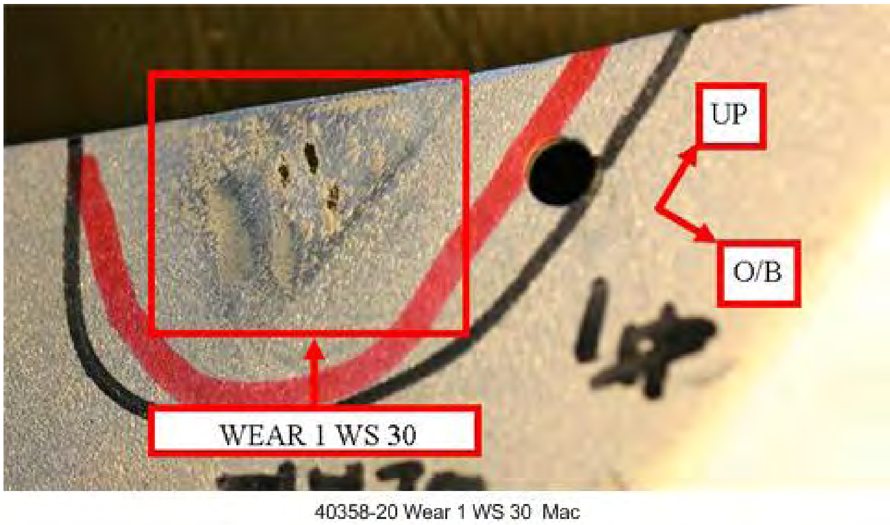


Figure 108. Macroscopic View of Wear 1 on the Left Wing Front Spar Attachment Point WS 30

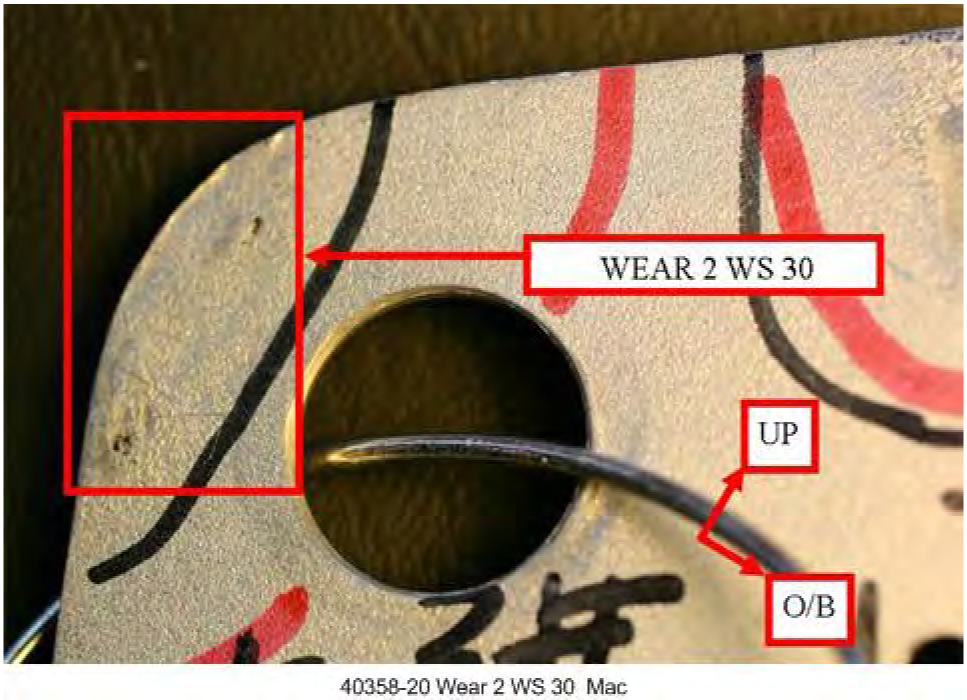


Figure 109. Macroscopic View of Wear 2 on the Left Wing Front Spar Attachment Point WS 30

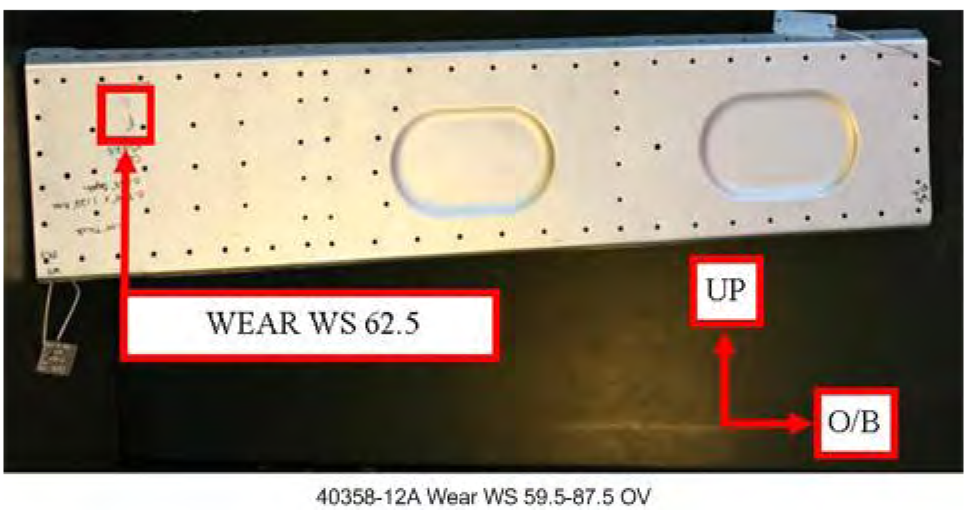
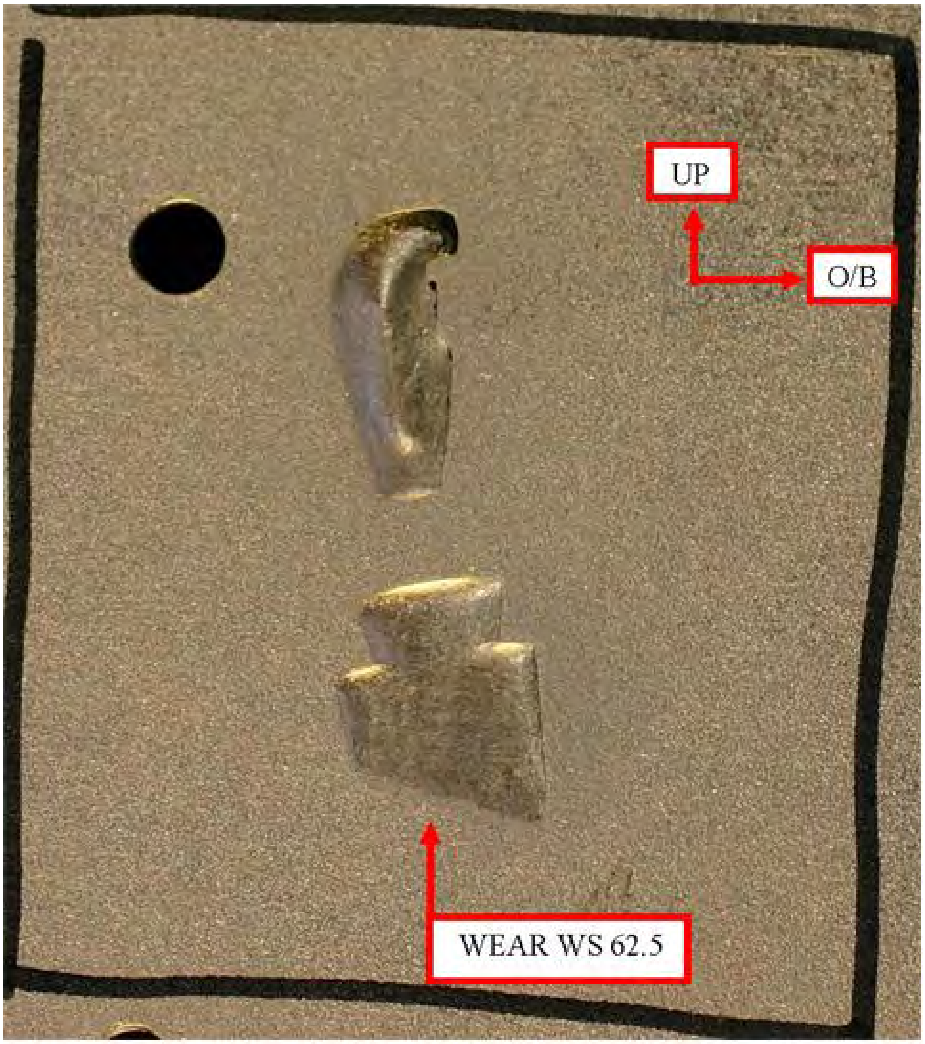
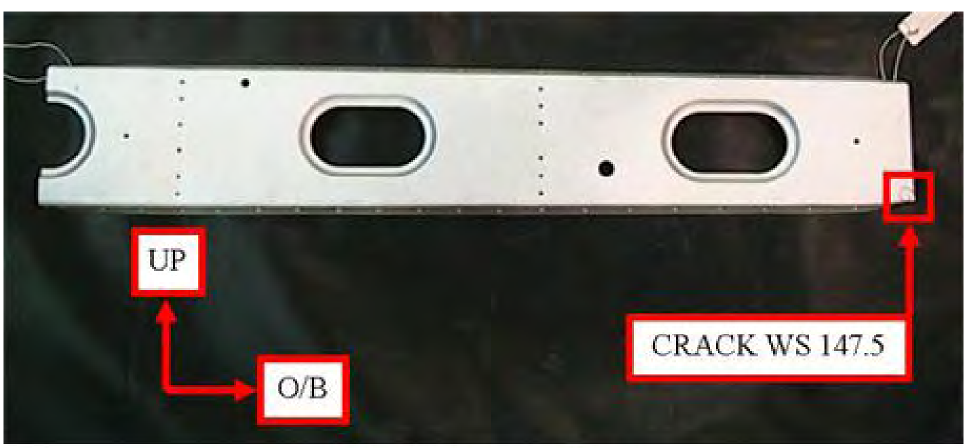


Figure 110. Location of Wear on the Left Wing Front Spar Web WS 62.5



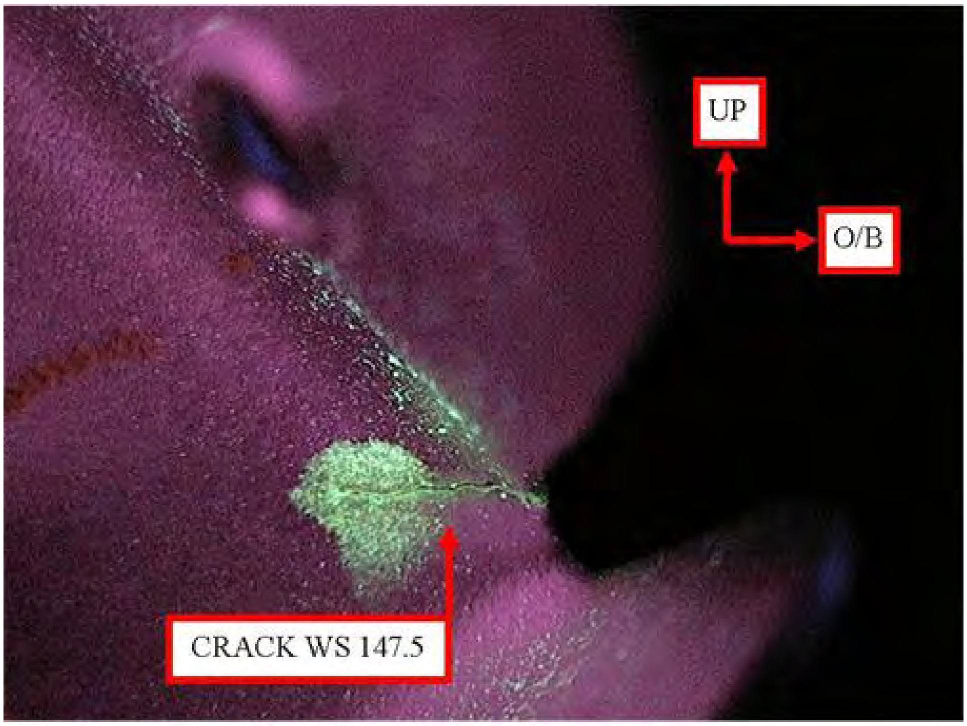
40358-12A Wear WS 59.5-87.5 Mac

Figure 111. Macroscopic View of Wear on the Left Wing Front Spar Web WS 62.5



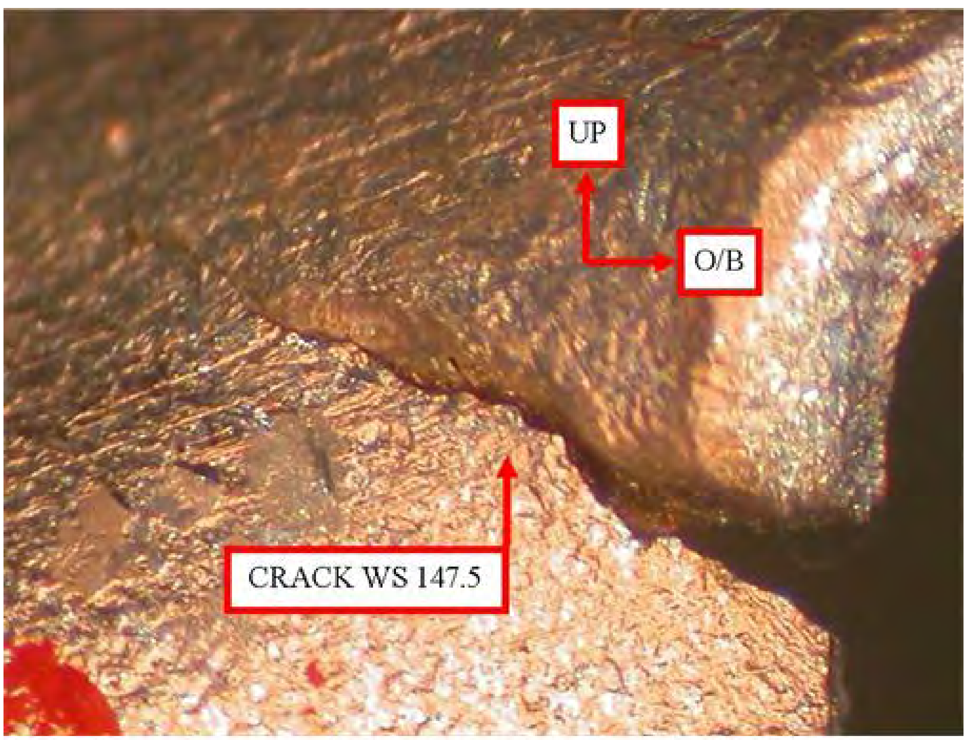
40358-12 CRK WS 116.5-147.5 OV

Figure 112. Location of Crack on the Left Wing Front Spar Web WS 147.5



40358-12 CRK WS116.5-147.5 FLP

Figure 113. Fluorescent Liquid Penetrant Indication of Crack on the Left Wing Front Spar Web WS 147.5

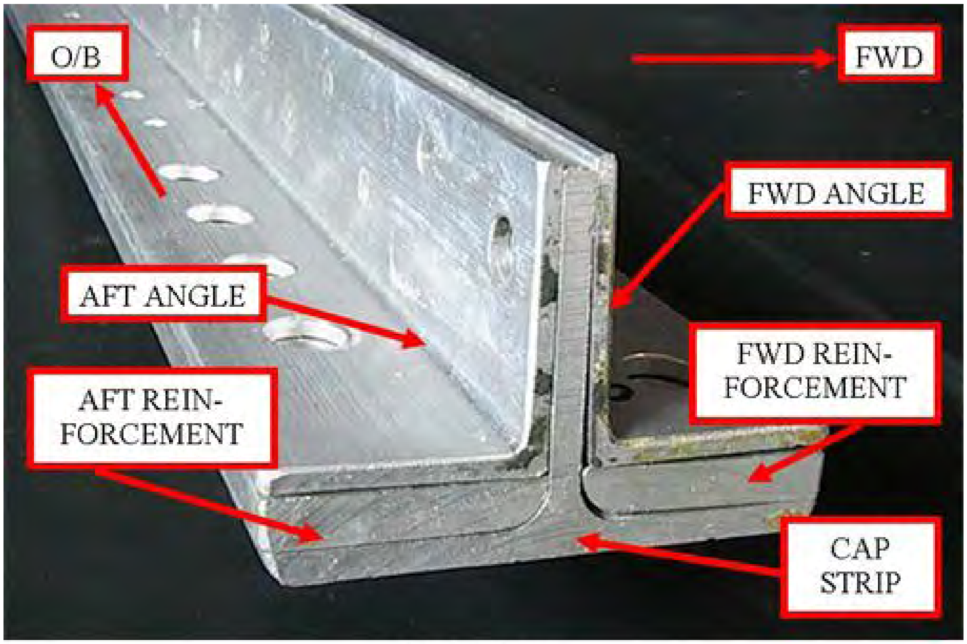


40358-12 CRK WS116.5-147.5 Mic

Figure 114. Microscopic View of Crack on the Left Wing Front Spar Web WS 147.5

3.4.2.1.2 Left Wing Main Spar.

Figure 115 shows the structural stackup of the left wing main spar upper and lower cap assemblies. Seventeen areas of corrosion were found on the left wing main spar upper and lower cap assemblies. Six of these areas were documented on the upper main spar cap assembly, and the remaining 11 were found on the lower cap assembly. Table 29 provides a complete characterization of all defects found on the left wing main spar.



Main Spar Stackup

Figure 115. Structural Stackup of the Left Wing Main Spar Upper and Lower Cap Assemblies

Table 29. Inspection Results from the Left Wing Main Spar

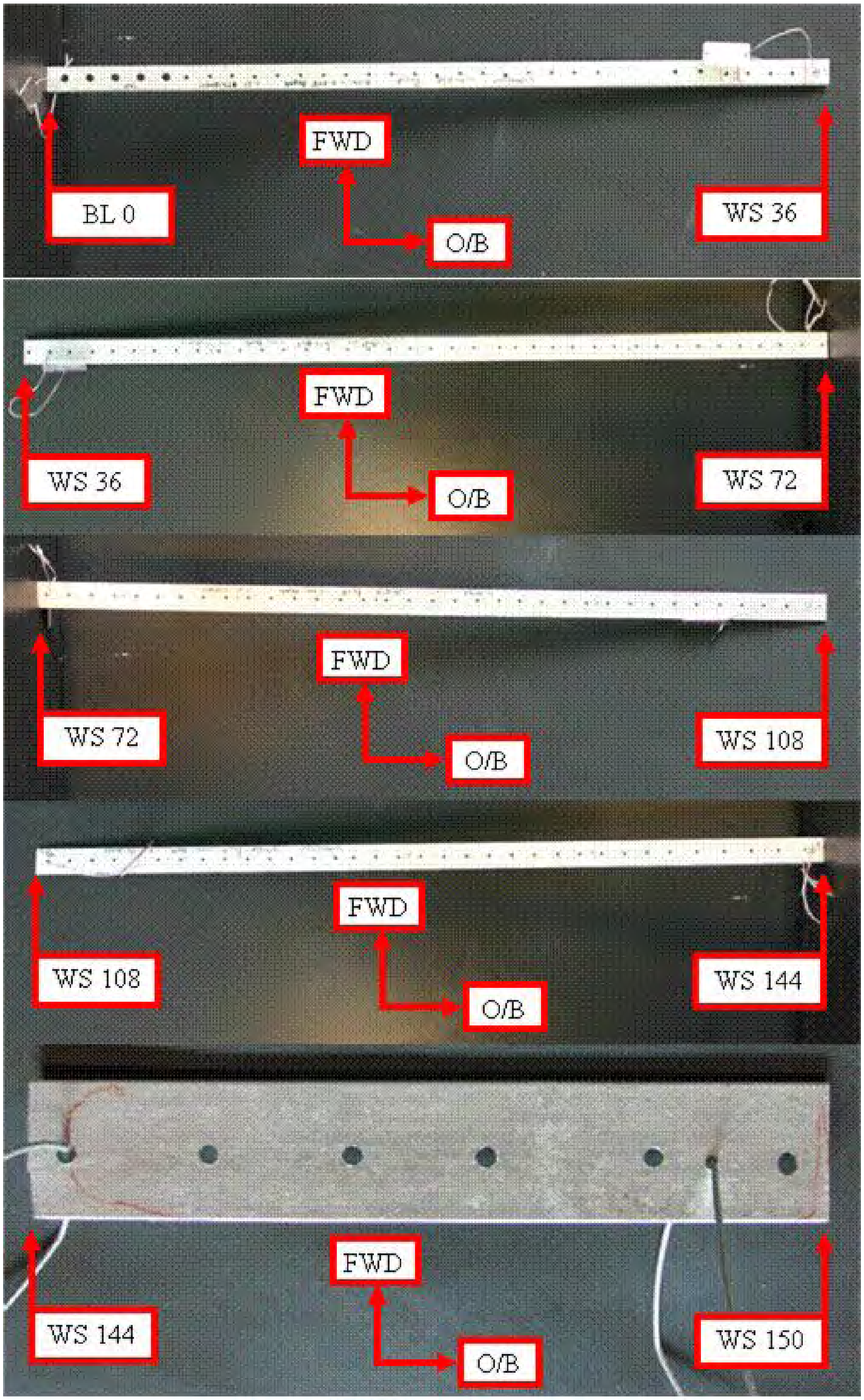
Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing main spar upper aft reinforcement, figure 116	Corrosion	BL 0 through WS 150	Scattered over entire part	Corrosion indication	Light corrosion less than 1% thickness loss	103
Left wing main spar upper aft angle, figure 117	Corrosion	BL 0 through WS 99	Scattered over entire part	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
Left wing main spar upper fwd angle, figure 118	Corrosion	WS 27.25	0.133 inch by 0.147 inch	Corrosion indication	Light corrosion 1% thickness loss	103

Table 29. Inspection Results From the Left Wing Main Spar (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing main spar upper cap strip, figure 119	Corrosion	WS 28 through WS 36	8 inches by 2.5 inches	Corrosion indication	Light-moderate corrosion 3.6% thickness loss	105
Left wing main spar upper fwd reinforcement, figure 120	Corrosion	WS 36 through WS 108	Scattered over entire part	Corrosion indication	Light corrosion less than 1% thickness loss	106
Left wing main spar upper cap strip, figure 121	Corrosion	WS 72 through WS 212.5	Scattered over entire part	Corrosion indication	Light corrosion 1.5% thickness loss	103
Left wing main spar lower aft reinforcement, figure 122	Corrosion	BL 0 through WS 108	Scattered over entire part	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
	Corrosion	BL 21 through BL 22.6	1.5 inches by 1 inch	Corrosion indication	Moderate-severe corrosion 7.5% thickness loss	123
	Corrosion	WS 34.3 through WS 35	0.7 inches by 1 Inch	Corrosion indication	Moderate corrosion 6% thickness loss	124
	Corrosion	WS 40.2 through WS 41.4	1.2 inches by 0.75 inch	Corrosion indication	Moderate corrosion 6% thickness loss	125
	Corrosion	WS 56.9 through WS 57.9	1 inch by 0.75 inch	Corrosion indication	Moderate-severe corrosion 7% thickness loss	126
	Corrosion	WS 62.25 through WS 63.25	1 inch by 0.5 inch	Corrosion indication	Moderate-severe corrosion 7% thickness loss	127
Left wing main spar lower fwd reinforcement, figure 128	Corrosion	BL 0 through WS 72	Scattered over entire part	Corrosion indication	Light corrosion 1.4% thickness loss	103
Figure 129	Corrosion	BL 0 through WS 72	Scattered over entire part	Corrosion indication BL 0 through WS 36	Light corrosion 2% thickness loss	103
	Corrosion	WS 52.5 through WS 58	5.5 inches by 2 inches	Corrosion indication	Moderate-severe corrosion 9% thickness loss	130
Left wing main spar lower cap strip, figure 131	Corrosion	WS 34.4 through WS 144	Scattered over entire part	Corrosion indication	Light corrosion less than 1 % thickness loss	103
	Corrosion	Around rivet holes WS 168 through WS 174	5.0 inches by 1.2 inches	Corrosion indication	Light corrosion less than 1% thickness loss	103

Examples of light and light-moderate corrosion are shown for reference in figures 102 through 105. Scattered light corrosion was identified on the left wing main spar upper aft reinforcement from BL 0 through WS 150. An overview of this area is shown in figure 116. Figure 117 shows an overview of the upper aft angle from BL 0 through WS 99. Scattered light-moderate corrosion was reported on the upper aft angle with a maximum localized thickness loss of 3%. Figure 118 shows the location of light corrosion observed on the upper forward angle at WS 27.25. This corrosion covered an area of 0.02 square inch and caused a maximum thickness loss of 1%.

A 20-square-inch area of light-moderate corrosion was observed on the left wing main spar upper cap strip from WS 28 to 36. The location of this area of corrosion, which caused a localized thickness loss of 3.6% on the cap strip is shown in figure 119. Light corrosion, causing a thickness loss of less than 1% was noted scattered over the surface of the upper forward reinforcement from WS 36 to 108. An overview of this area is shown in figure 120. Another area of light corrosion was found scattered across the surface of the upper cap strip from WS 72 to 212.5, as shown in figure 121. This corrosion caused a maximum thickness loss of 1.5%.



LW MS Upper Aft Reinforce Corr WS 0-150 OV

Figure 116. Overview of Left Wing Main Spar Upper Aft Reinforcement BL 0 Through WS 150

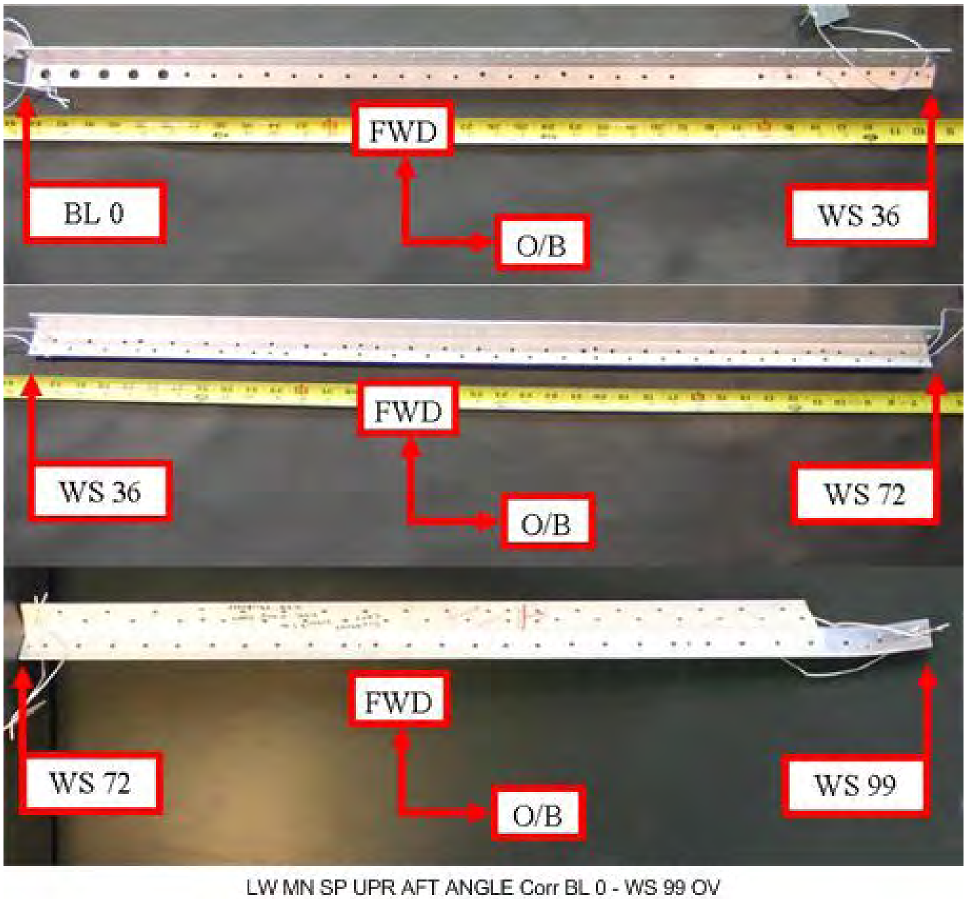


Figure 117. Overview of Left Wing Main Spar Upper Aft Angle BL 0 Through WS 99

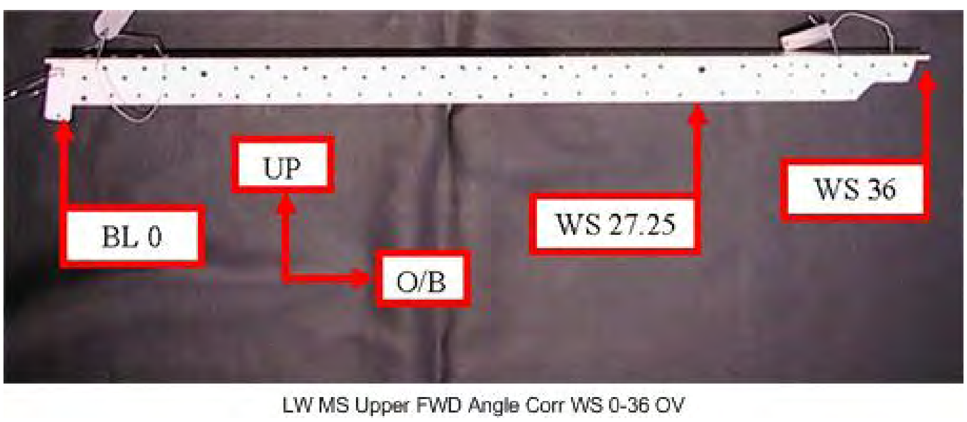


Figure 118. Location of Light Corrosion on the Left Wing Main Spar Upper Forward Angle WS 27.25

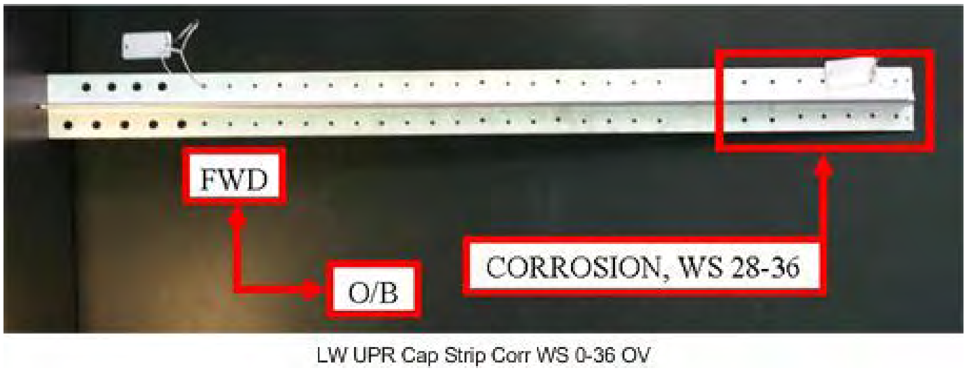


Figure 119. Location of Light-Moderate Corrosion on the Left Wing Main Spar Upper Cap Strip
WS 28 Through WS 36

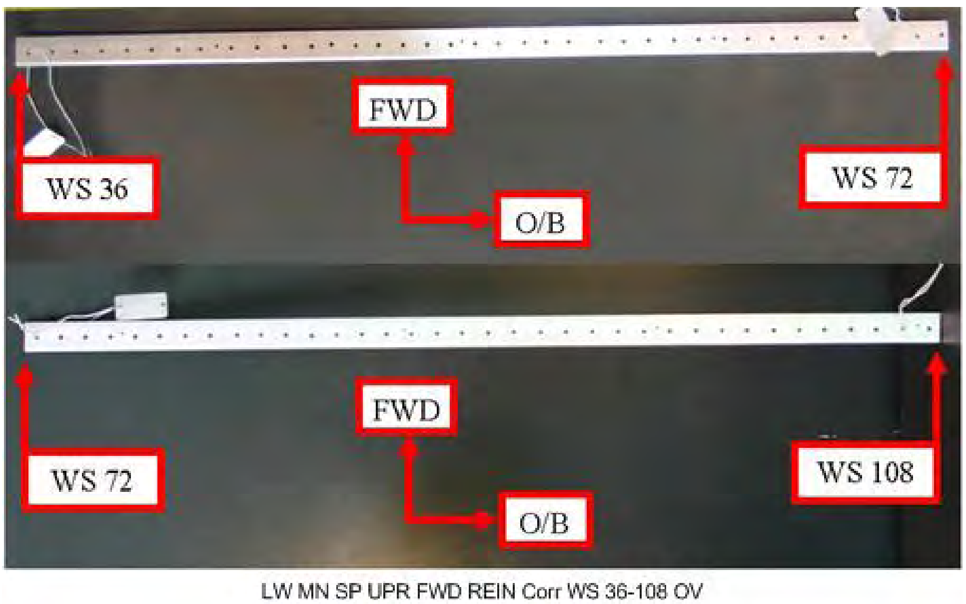


Figure 120. Overview of Left Wing Main Spar Upper Forward Reinforcement
WS 36 Through WS 108

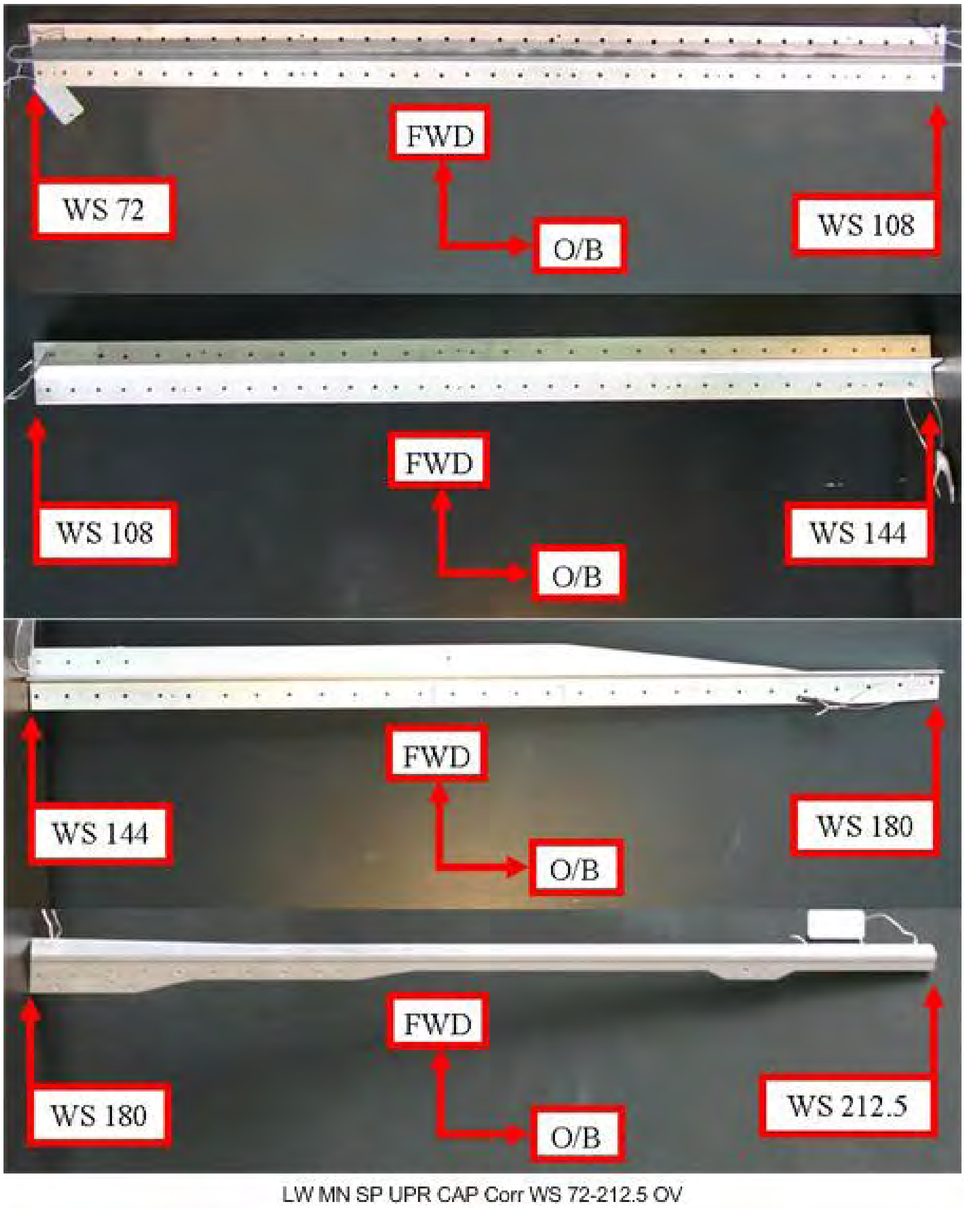


Figure 121. Overview of Left Wing Main Spar Upper Cap Strip WS 72 Through WS 212.5

Six areas of corrosion were documented on the left wing main spar lower aft reinforcement. The location of each area of corrosion is shown in figure 122. Scattered light to light-moderate corrosion was noted from BL 0 to WS 108, which caused a maximum localized thickness loss of 4%. A 1.5-square-inch area of moderate-severe corrosion was found from BL 21 to BL 22.6 and is shown in figure 123. This corrosion resulted in a 7.5% localized reduction in thickness. From WS 34.3 to WS 35, a 0.7-square-inch area of moderate corrosion, which caused a thickness loss of 6%, was recorded. A macroscopic view of this area of corrosion is shown in figure 124. Figure 125 shows a 0.9-square-inch area of moderate corrosion. This area of corrosion, located from WS 40.2 to WS 41.4, caused a maximum reduction in thickness of 6%. A 0.75-square-inch area of moderate-severe corrosion, shown in figure 126, was observed on the lower aft reinforcement. This area of corrosion, located from WS 56.9 to WS 57.9, caused a localized loss in thickness of 7%. Another area of moderate-severe corrosion, causing 7% thickness loss, was observed on the lower aft reinforcement from WS 62.25 to WS 63.25. Figure 127 shows this 0.5-square-inch area of corrosion.

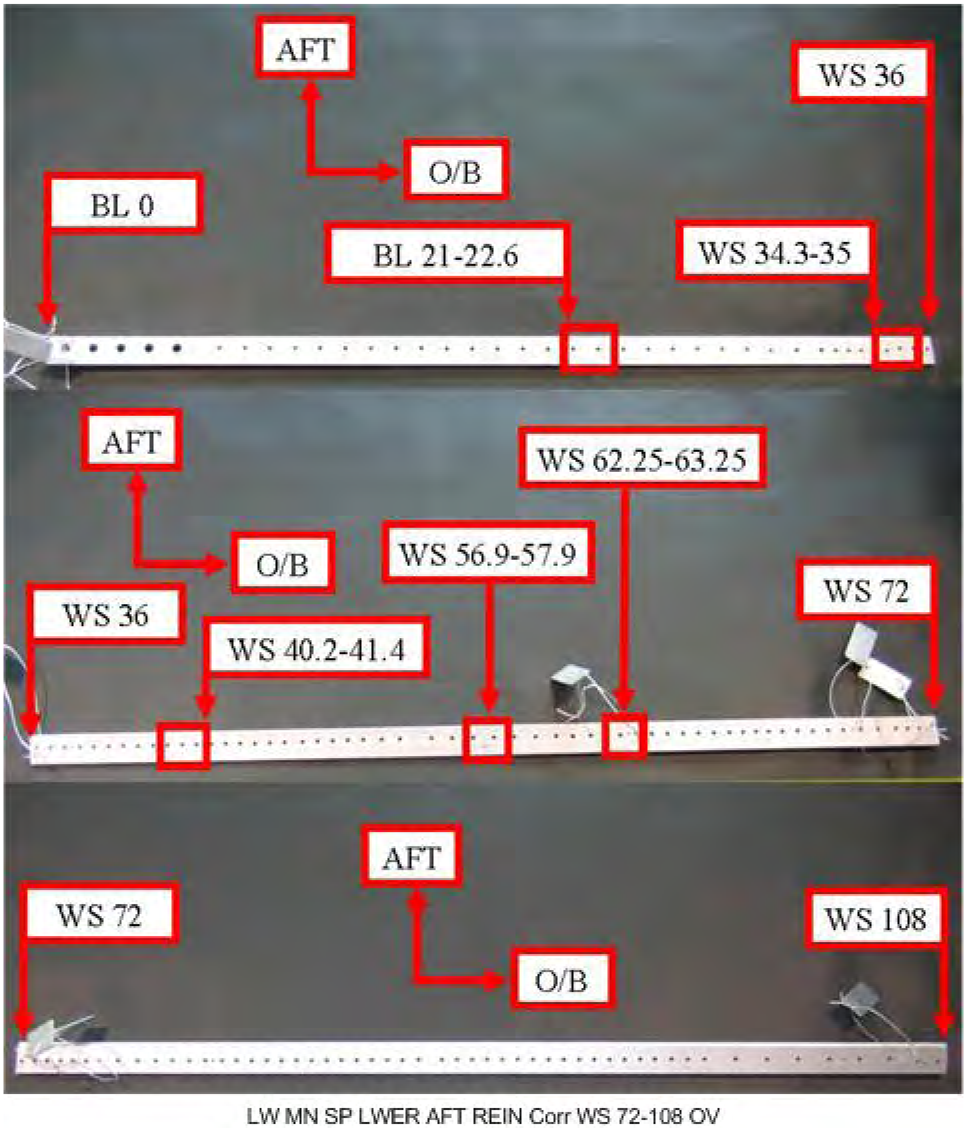
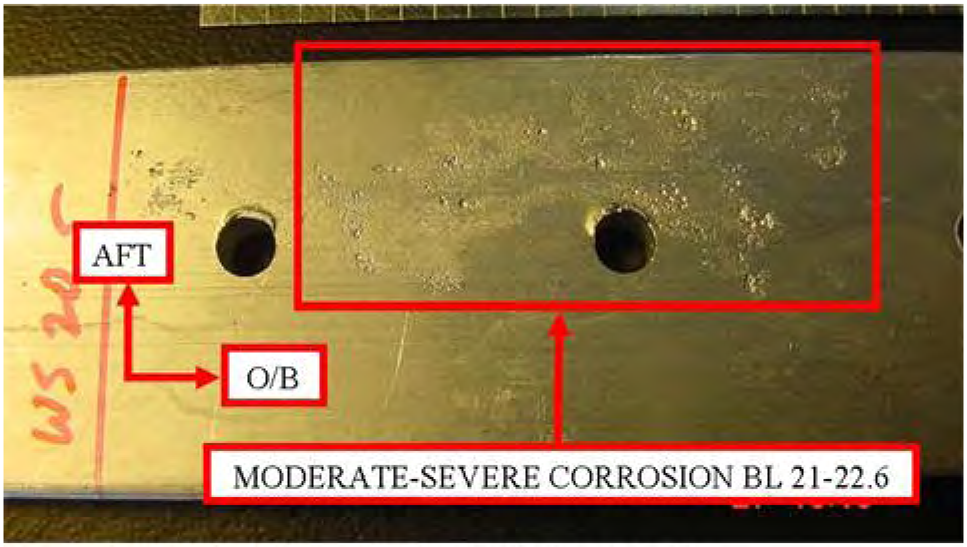
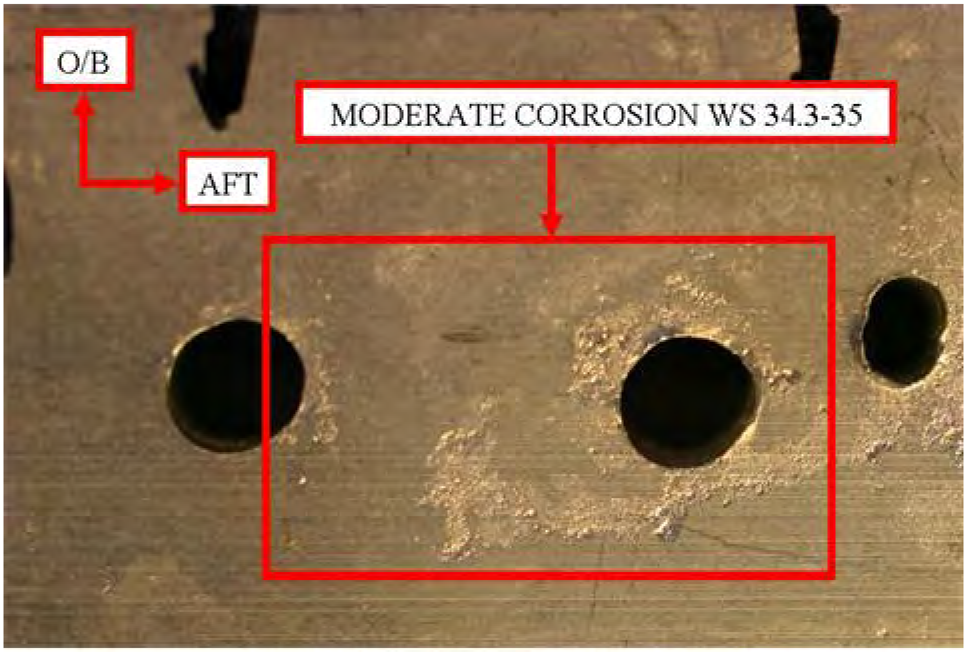


Figure 122. Locations of Moderate and Moderate-Severe Corrosion and Overview of Left Wing Main Spar Lower Aft Reinforcement BL 0 Through WS 108



LW MS Sp LWR AFT REIN Corr BL21-22.6 Mac

Figure 123. Macroscopic View of Moderate-Severe Corrosion on the Left Wing Main Spar Lower Aft Reinforcement BL 21 Through BL 22.6



LW MS SP LWR AFT REIN Corr WS34.3-35 Mac

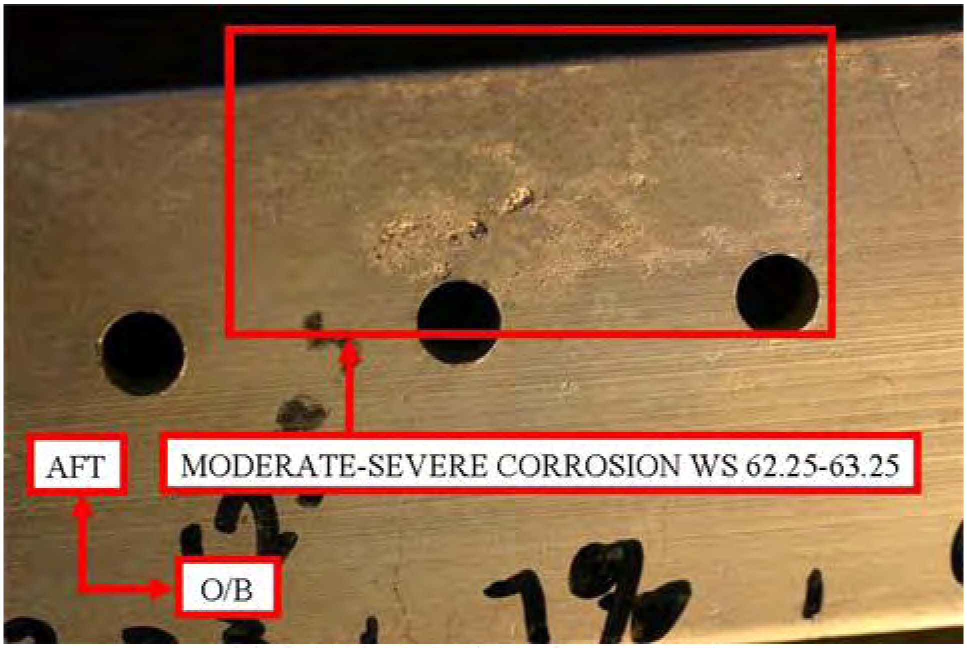
Figure 124. Macroscopic View of Moderate Corrosion on the Left Wing Main Spar Lower Aft Reinforcement WS 34.3 Through WS 35

LW MN SP LWR AFT REIN Corr WS40.2-41.4 Mac

Figure 125. Macroscopic View of Moderate Corrosion on the Left Wing Main Spar Lower Aft Reinforcement WS 40.2 Through WS 41.4

LW MN SP LWR AFT REIN Corr WS56.9-57.9 Mac

Figure 126. Macroscopic View of Moderate-Severe Corrosion on the Left Wing Main Spar Lower Aft Reinforcement WS 56.9 Through WS 57.9



LW MN SP LWR AFT REIN Corr WS62.25-63.25 Mac

Figure 127. Macroscopic View of Moderate-Severe Corrosion on the Left Wing Main Spar Lower Aft Reinforcement WS 62.25 Through WS 63.25

Figure 128 shows an overview of light scattered corrosion on the left wing main spar lower forward reinforcement from BL 0 to WS 72. This corrosion caused a maximum thickness loss of 1.4%. Light scattered corrosion, resulting in a reduction in thickness of 2%, was also observed on the left wing main spar lower aft angle from BL 0 to WS 72, as shown in figure 129. An area of moderate-severe corrosion, shown in figure 130, was also located on the lower aft angle from WS 52.5 to WS 58. This area covered 11-square inches and caused a maximum reduction in thickness of 9%. Light corrosion was also scattered across the surface of the main spar lower cap strip from WS 34.4 to WS 144, as shown in figure 131. An area of corrosion covering 6 square inches was noted on the lower cap strip around fastener holes from WS 168 to WS 174 and is shown in figure 132. Both areas of corrosion located on the left wing main spar lower cap strip were classified as light corrosion causing less than 1% localized thickness loss.

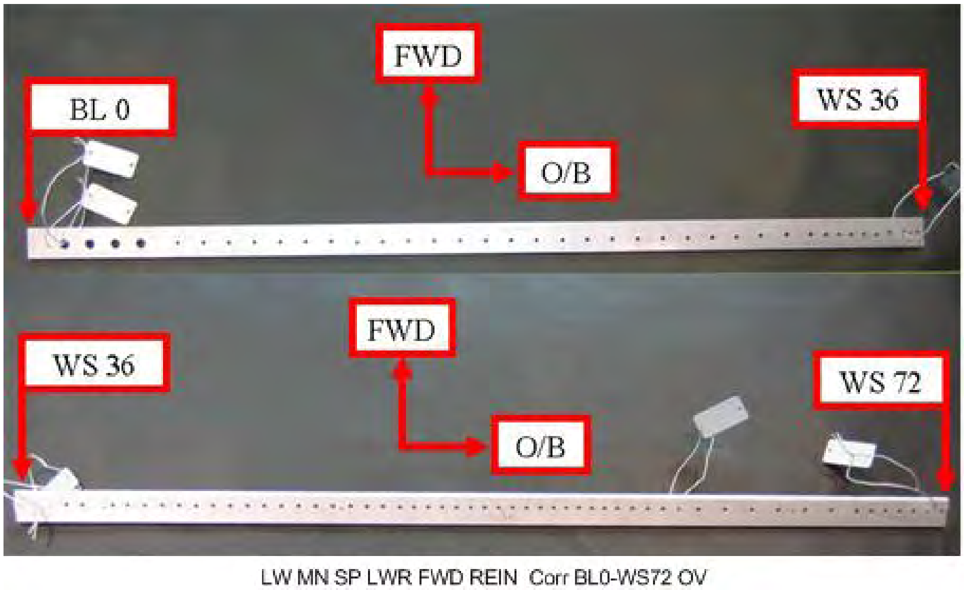


Figure 128. Overview of Left Wing Main Spar Lower Forward Reinforcement BL 0 Through WS 72

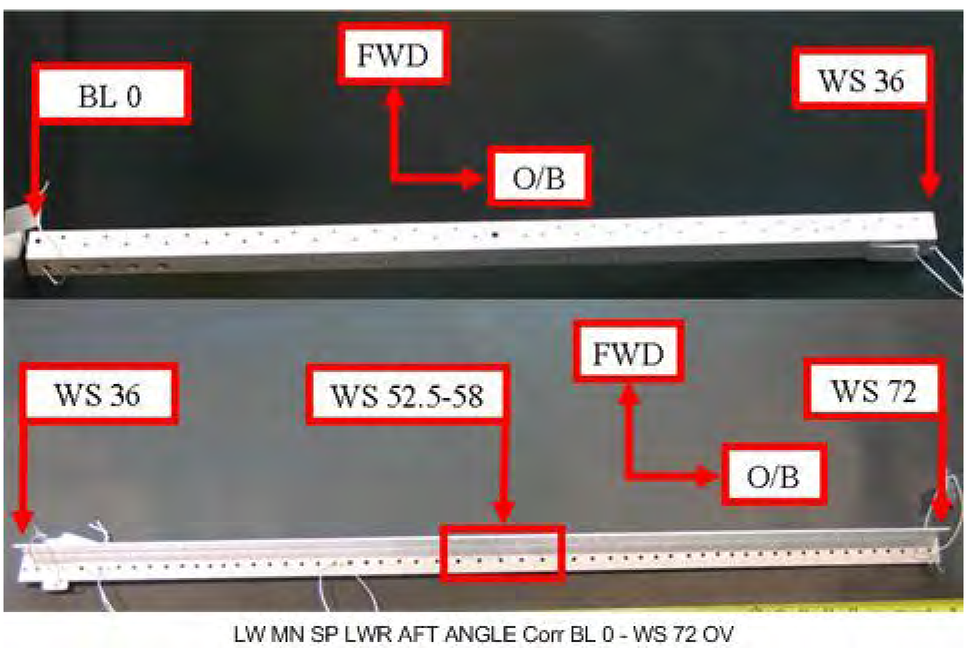
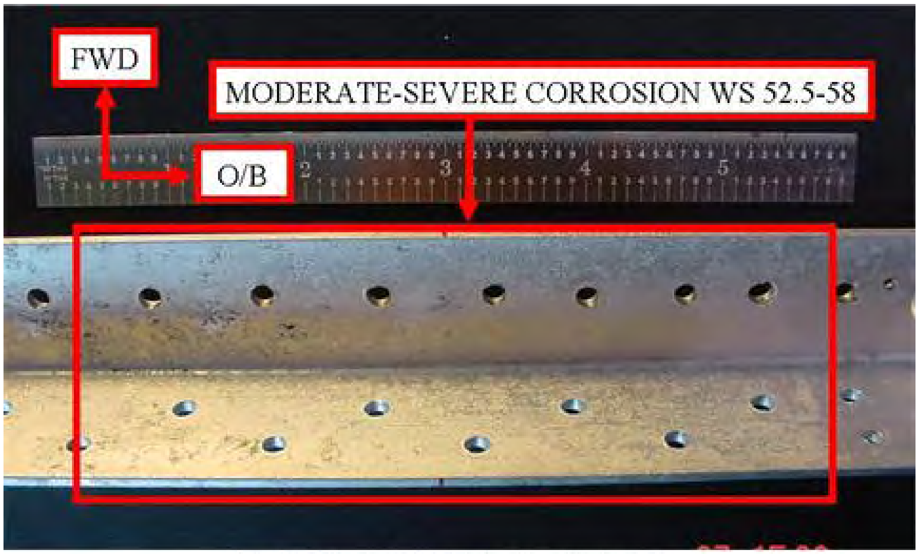
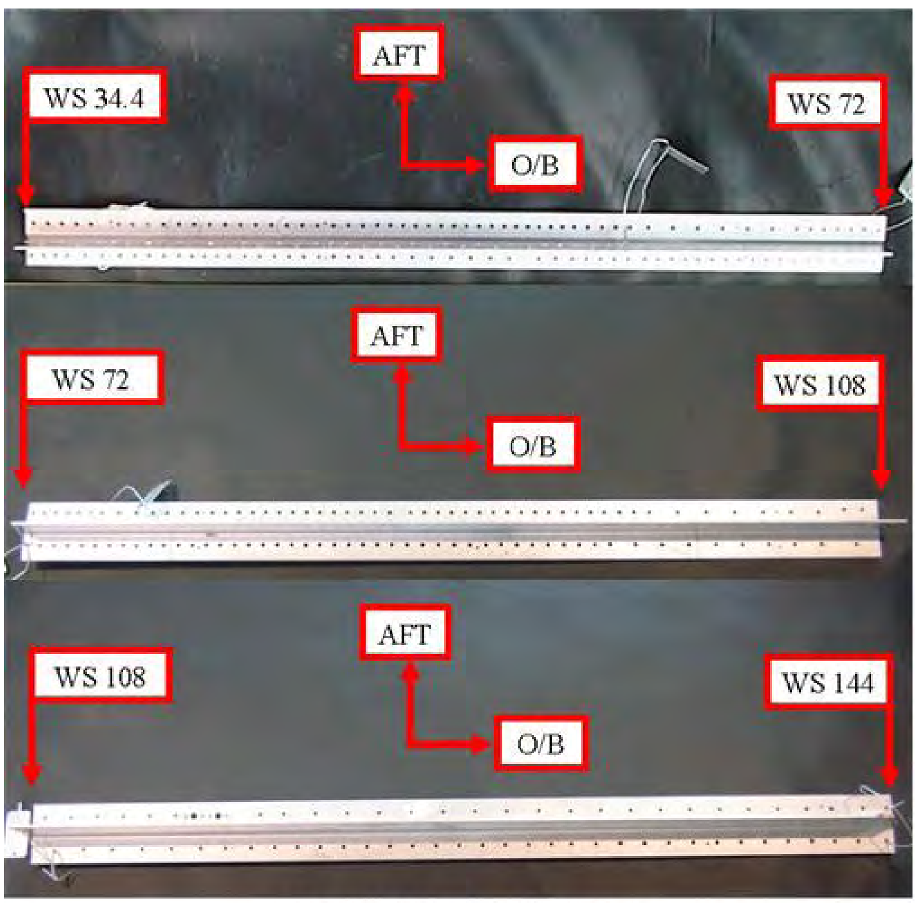


Figure 129. Location of Light Scattered Corrosion and Overview of Left Wing Main Spar Lower Aft Angle BL 0 Through WS 72



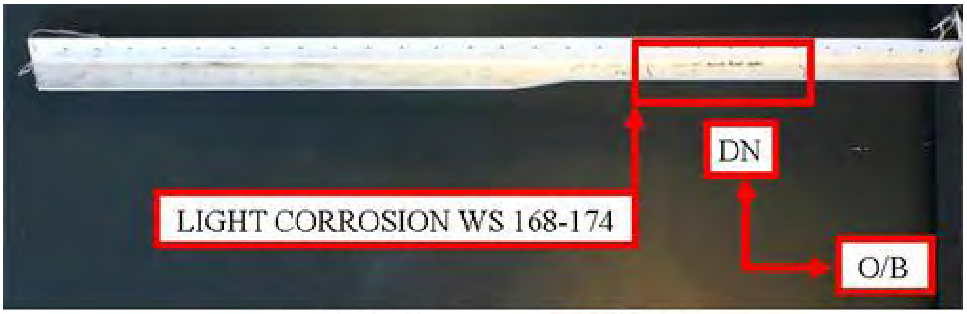
LW MN SP LWR AFT ANGLE Corr WS56.75 Mac

Figure 130. Macroscopic View of Moderate-Severe Corrosion on the Left Wing Main Spar Lower Aft Angle WS 52.5 Through WS 58



LW MN SP LWR Cap Corr WS 34-144 OV

Figure 131. Overview of Left Wing Main Spar Lower Cap Strip WS 34.4 Through WS 144

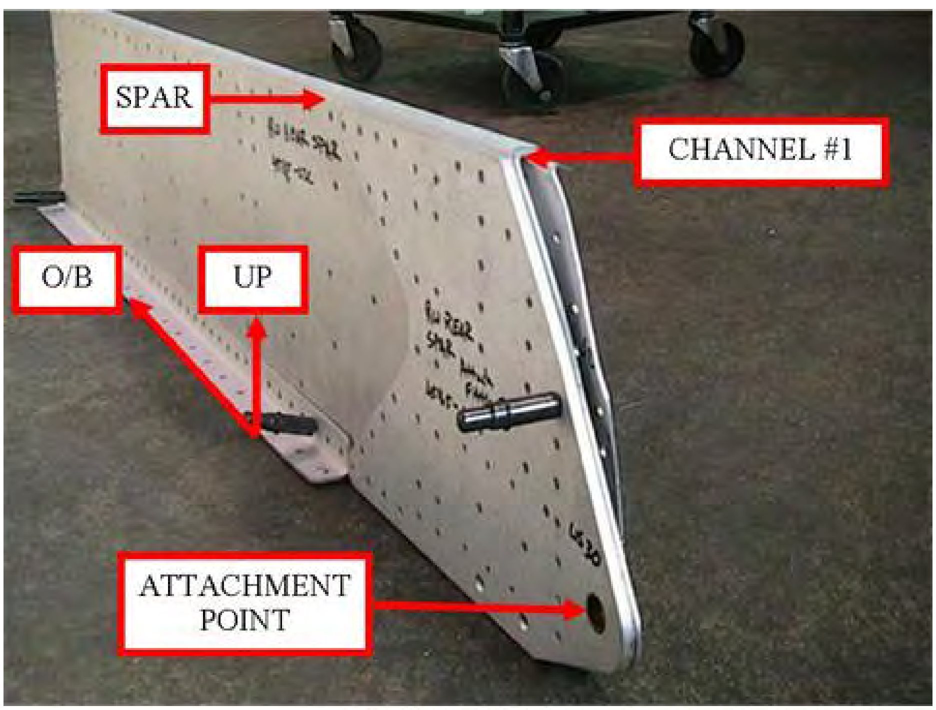


LW MN SP LWR Cap Corr WS 144-180 OV

Figure 132. Location of Light Corrosion on the Left Wing Main Spar Lower Cap Strip Around Rivet Holes WS 168 Through WS 174

3.4.2.1.3 Left Wing Rear Spar.

Figure 133 shows the structural stackup of the left wing rear spar, and table 30 provides detailed information on all defects found on the rear spar. Figure 134 shows the location of an area of wear observed on the left wing rear spar channel 1. Light-moderate corrosion, which resulted in a thickness loss of 3%, was found scattered across this channel from WS 30 to WS 64.5. The wear, shown in figure 135, caused a thickness loss of 3% and covered a surface area of 1.13 square inches. The location of a 0.293-inch-crack on the left wing rear spar web is shown in figure 136, while a macroscopic view of the crack is shown in figure 137. Figure 138 shows the fluorescent liquid penetrant indication of this crack, and figure 139 shows the microscopic view of this crack.



Wing Rear Spar Stackup

Figure 133. Structural Stackup of the Left Wing Rear Spar

Table 30. Inspection Results From the Left Wing Rear Spar

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing rear spar channel 1, figure 134	Corrosion	WS 30 through WS 64.5	Scattered over entire part	Corrosion indication	Light-moderate corrosion 3% thickness loss	134
	Wear	WS 30	0.95 inch by 1.187 inches	Corrosion indication	Wear 3% thickness loss	135
Left wing rear spar web, figure 136	Crack	WS 221	0.293 inch	Crack indication	Surface crack	137 138 139

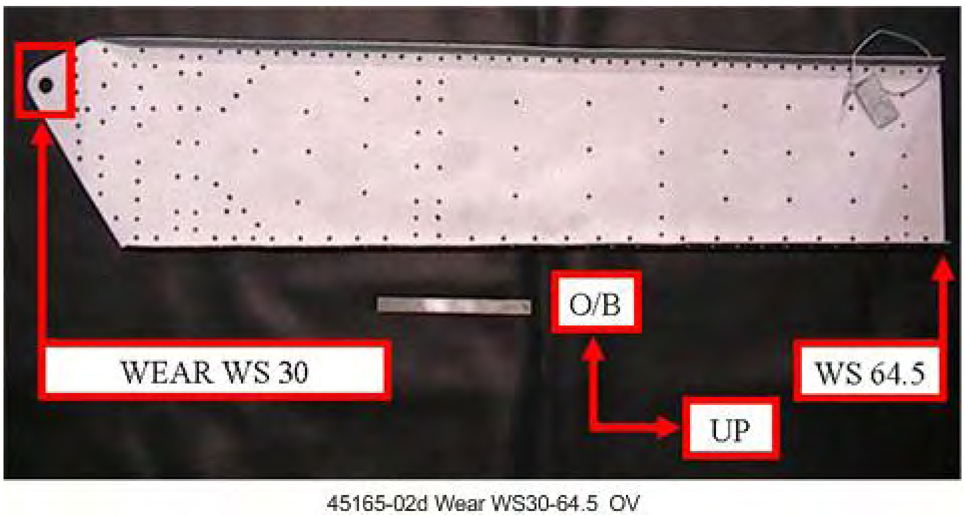


Figure 134. Location of Wear and Overview of Left Wing Rear Spar Channel 1 WS 30 Through WS 64.5

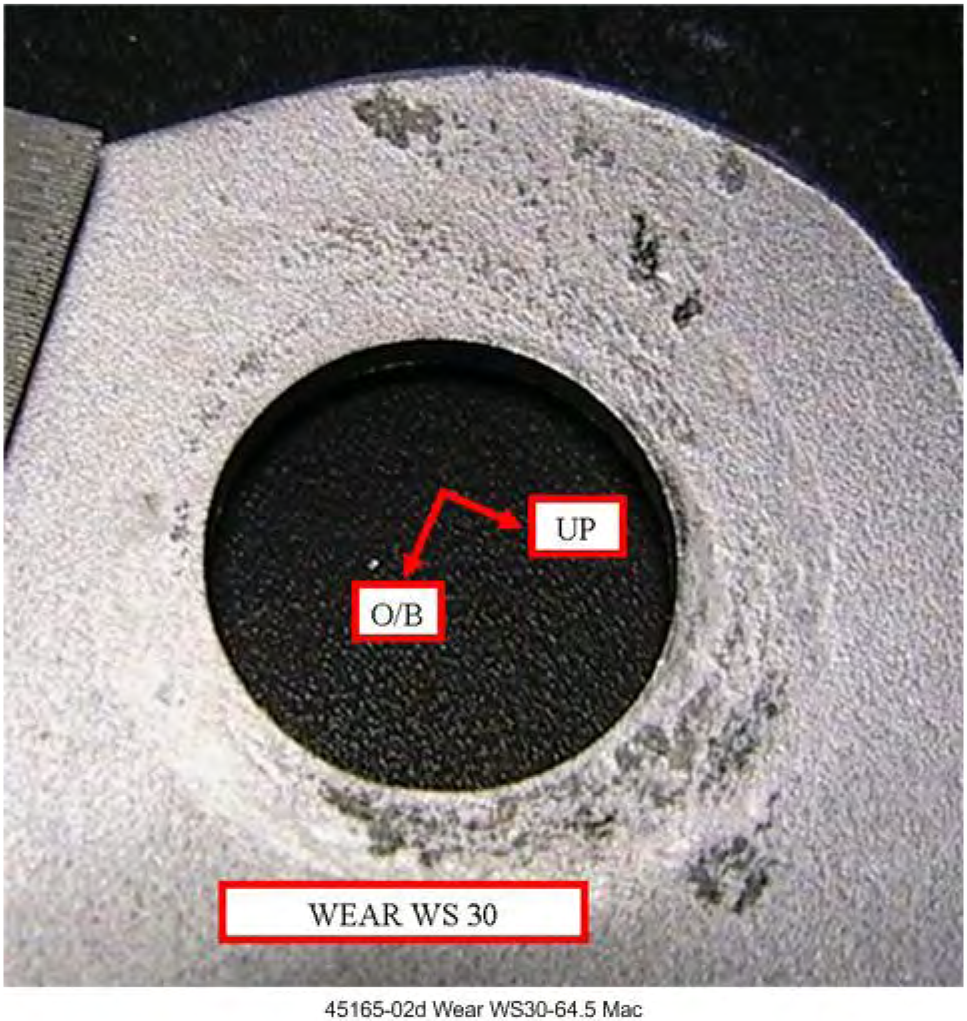


Figure 135. Macroscopic View of Wear on the Left Wing Rear Spar Channel 1 WS 30

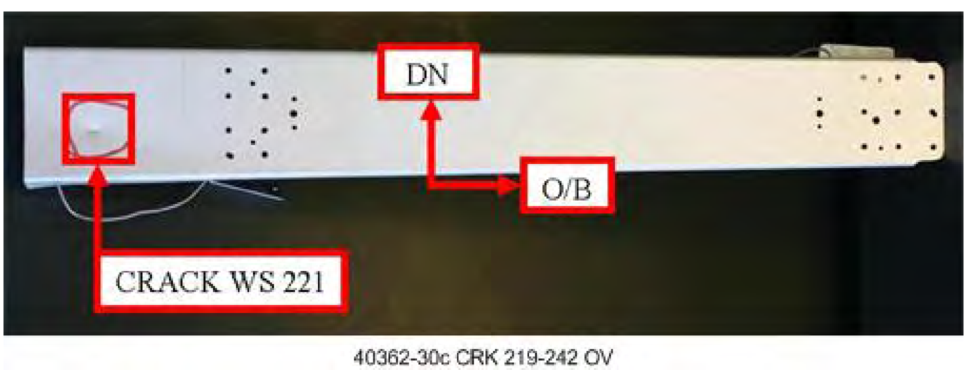


Figure 136. Location of Crack on the Left Wing Rear Spar WS 221

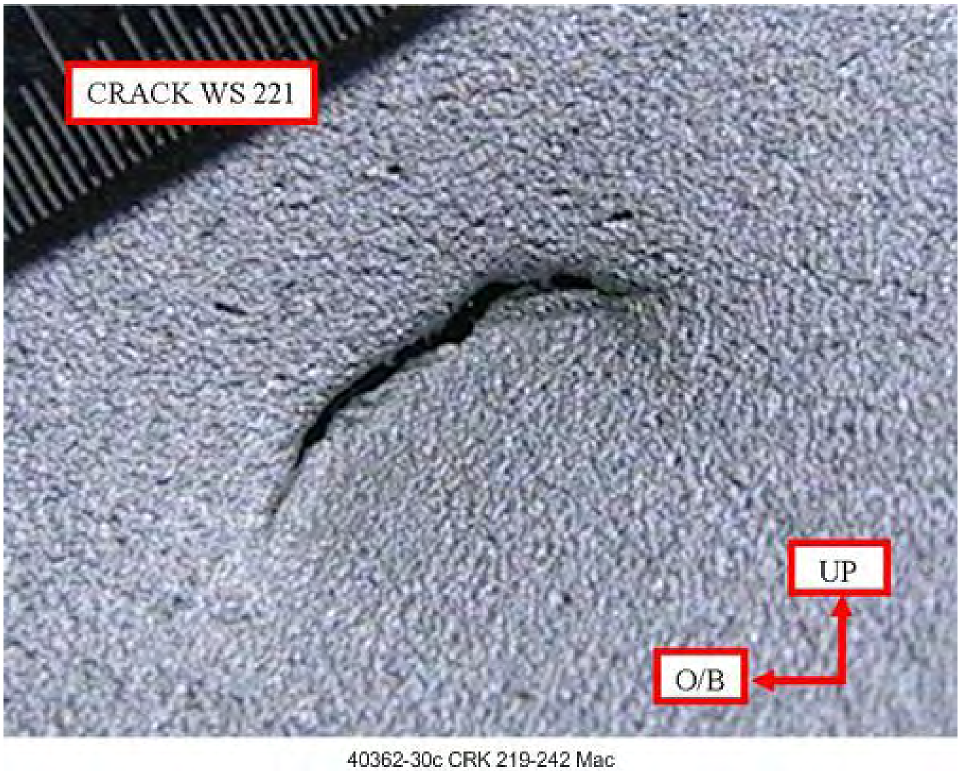


Figure 137. Macroscopic View of Crack on the Left Wing Rear Spar WS 221

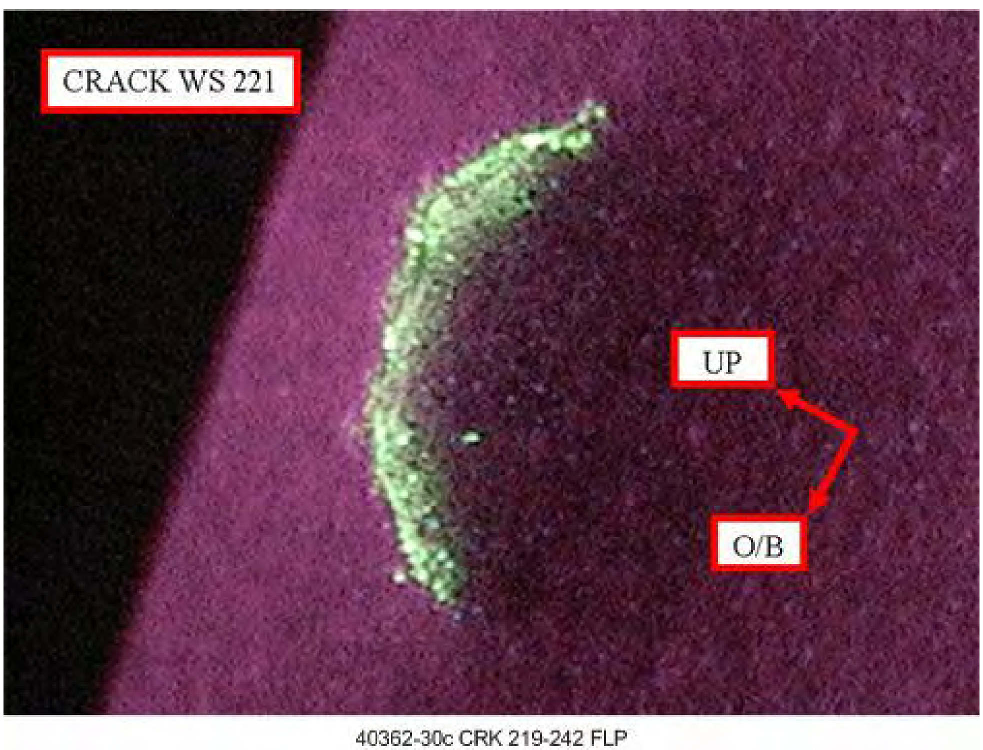


Figure 138. Fluorescent Liquid Penetrant Indication of Crack on the Left Wing Rear Spar WS 221

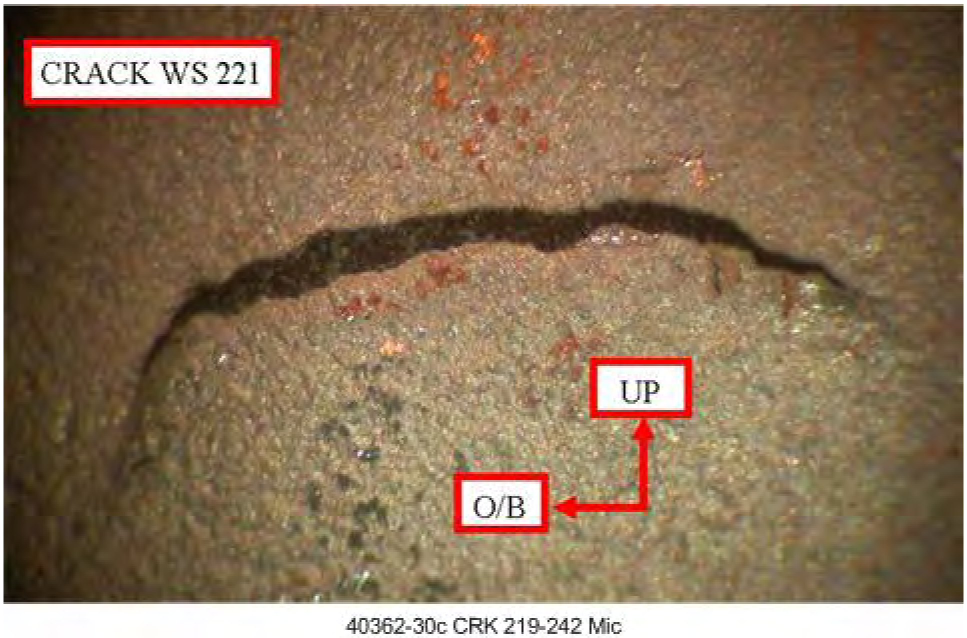


Figure 139. Microscopic View of Crack on the Left Wing Rear Spar WS 221

3.4.2.1.4 Left Wing Various Indications.

Numerous indications were identified on the left wing in areas other than the front, main, and rear spars. The detailed characterization of these defects is shown in table 31.

Table 31. Inspection Result From the Remainder of the Left Wing

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing plate-wing panel reinforcement assembly, figure 140	Corrosion	BL 18.5	0.687 inch by 0.804 inch	Corrosion indication	Moderate corrosion 6% thickness loss	141
	Corrosion	BL 23.25	1.85 inches by 0.132 inch	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
	Corrosion	BL 29.75	2.81 inches by 0.567 inch	Corrosion indication	Severe corrosion 12% thickness loss	142
	Corrosion	WS 30	1.81 inches by 0.475 inch	Corrosion indication	Severe corrosion 13.5% thickness loss	143
Left wing aft main gear door hinge bracket, figure 144	Crack	WS 30	0.273 inch	Crack indication	Bend radii	145 146

Table 31. Inspection Result From the Remainder of the Left Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing bulkhead wing fillet, figure 147	Crack	WS 30 FS 181	0.423 inch	Crack indication	Surface crack	148 149
	Crack	WS 30 FS 181	0.226 inch	Crack indication	Bend radii	150 151
	Multiple cracks	WS 30 FS 181	0.157 inch	Crack indication	Hole crack	152 153
	Multiple cracks	WS 30 FS 181	0.386 inch	Crack indication	Hole crack	154 155
Left wing bulkhead wing fillet, figure 156	Crack	WS 30 FS 191	0.77 inch	Crack indication	Bend radii	157 158 159
	Multiple cracks	WS 30 FS 191	0.756 inch	Crack indication	Bend radii	160 161 162
	Crack	WS 30 FS 191	0.349 inch	Crack indication	Bend radii	163 164
	Crack	WS 30 FS 191	0.374 inch	Crack indication	Surface crack	165
Left wing main gear door angle, figure 166	Crack	WS 32	0.235 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	167
	Crack	WS 32	0.231 inch	Not inspected[1]	Fatigue	168 169
Left wing rear spar assembly lower aft angle, figure 170	Wear	WS 38	8.5 inches by 0.29 inch	No indication	Wear 54% thickness loss	171
	Wear	WS 44	6 inches by 0.15 inch	No indication	Wear 17.7% thickness loss	172
	Wear	WS 64.5	7.3 inches by 0.11 inch	No indication	Wear 32% thickness loss	173
	Corrosion	FS 148	Scattered over entire part	Corrosion indication	Light corrosion 1.1% thickness loss	103
Left wing rib assembly, figure 174	Crack from object impacting existing repair	WS 39.5 FS 161	1.525 inches	Crack indication	Hole radii	175 176
Left wing rib, figure 177	Damage	WS 49	Entire part	Not inspected		178
Left wing main gear aft fitting assembly, figure 179	Crack	WS 58.5	0.225 inch	Crack indication		180 181 182

Table 31. Inspection Result From the Remainder of the Left Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing gear well plate 5, figure 183	Corrosion	WS 58.5 through WS 87.5	Scattered over entire part	Corrosion indication	Light corrosion 1.5% thickness loss	103
Left nacelle front locker floor, figure 184	Corrosion	WS 65.25 through WS 83	17 inches by 4.5 inches	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
	Corrosion	WS 66.75 through WS 77	10 inches by 2 inches	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
	Crack	WS 73.5	0.65 inch	Crack indication	Surface	185
	Corrosion	WS 73.75 through WS 77	2.5 inches by 1.5 inches	Corrosion indication	Severe corrosion 15.6% thickness loss	186
	Corrosion	WS 74.5 through WS 76	1 inches by 2 inches	Corrosion indication	Severe corrosion 22% thickness loss	187
Left wing engine nacelle bulkhead assembly, figure 188	Corrosion	WS 65.69	33 inches by 12 inches inboard side of part	Corrosion indication	Light corrosion 2% thickness loss	103
Left wing nacelle inboard angle longitudinal bulkhead, figure 189	Crack	WS 66	1.695 inches	Not inspected[1]	Fatigue	190 191 192
Left wing rear spar rib assembly, figure 193	Corrosion	WS 67.5	5.25 inches by 1 inch (both sides of part)	No indication	Light corrosion 1% thickness loss	103
Left wing bulkhead assembly rear spar main gear support, figure 194	Corrosion	WS 70	3 inches by 1.58 inches	Corrosion indication	Light corrosion 1.3% thickness loss	103
Left wing nacelle plate assembly firewall shear, figure 195	Wear	WS 72	1.245 inches by 0.304 inch	No indication	Wear 20% thickness loss	196
	Wear	WS 72	4 inches by 0.194 inch	No indication	Wear 25% thickness loss	197
Left wing trunnion landing gear aft, figure 198	Crack	WS 77.5	0.175 inch	Crack indication	Bend radii	199 200

Table 31. Inspection Result From the Remainder of the Left Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left wing nacelle air filter bracket, figure 201	Crack	WS 85	0.496 inch	Not inspected[1]	Fatigue	202 203
Left wing plate assembly main spar rib attach, figure 204	Wear	WS 87	0.941 inch by 2.23 inches	No indication	Wear 66% thickness loss	205
Left wing main spar rib attachment wing panel assembly (1 of 2), figure 206	Crack	WS 87.5	1.040 inches	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	207 208
Left wing main spar rib attachment wing panel assembly (2 of 2), figure 209	Crack	WS 87.5	0.904 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	216
Left wing flap track, figure 111	Multiple cracks	WS 87.5	0.044 inch	Crack indication	Hole crack	212 213 214
Left wing rear spar assembly rib assembly, figure 215	Corrosion	WS 87.5	30 inches by 12 inches scattered inboard side of part	Corrosion indication	Light corrosion 2% thickness loss	103
Left wing rear spar bulkhead assembly, figure 216	Corrosion	WS 87.5 FS 161	Scattered whole part	Corrosion indication	Light corrosion 1% thickness loss	103
Left wing rear spar rib assembly, figure 217	Corrosion	WS 87.5	Scattered whole part	No indication	Light corrosion 2% thickness loss	103
Left nacelle bulkhead longitudinal outboard, figure 218	Corrosion	WS 89 FS 154 through FS 185.5	Scattered inboard side of part	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
	Crack	WS 89 FS 184	0.13 inch	Crack indication	Hole crack	219 220
	Crack	WS 89 FS 185.5	0.60 inch	Crack indication	Surface crack	221 222

Table 31. Inspection Result From the Remainder of the Left Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left nacelle bulkhead longitudinal outboard, figure 223	Corrosion	WS 89 FS 185 through FS 211.5	Scattered over entire part	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
	Multiple cracks	WS 89 FS 207.25	0.195 inch	Crack indication	Hole crack	224 225
Left wing nacelle outboard angle-longitudinal bulkhead, figure 226	Corrosion	WS 89 FS 154 through FS 185.5	Scattered over entire part	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
	Crack	WS 89 FS 177	0.211 inch	Crack indication	Surface crack	226 227
Left wing engine nacelle bulkhead assembly, figure 229	Corrosion	WS 89.31	32.5 inches by 12 inches inboard side of part	Corrosion indication	Light corrosion 2% thickness loss	103
Left wing aileron bracket front left, figure 230	Corrosion	WS 174.5	0.55 inch by 0.55 inch	No indication	Light corrosion 1.5% thickness loss	103
Left wing rear spar lower aft angle, figure 231	Crack	FS 161	3.5 inches	Crack indication	Surface crack	232
	Wear	FS 161	11 inches by 0.2 inch	No indication	Wear 100% thickness loss	233
Left wing rear spar lower fwd angle, figure 234	Wear	FS 167	22.75 inches by 0.211 inch	No indication	Wear 13% thickness loss	235

[1]Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 140 shows the location of light-moderate, moderate, and severe corrosion on the left wing plate wing panel reinforcement assembly, part number 44710-00A. A moderate area of corrosion, shown in figure 141, was identified at BL 18.5. This area of corrosion covered a surface area of 0.55 square inch and caused a localized reduction in thickness of 6%. One area of light-moderate corrosion was identified on the reinforcement assembly at BL 23.25, covered a surface area of 0.24 square inch, and resulted in a 3% maximum localized thickness loss. A 1.59-square-inch area of severe corrosion was found at BL 29.75 and is shown in figure 142. A maximum localized thickness loss of 12% was caused by this area of corrosion. Another area of severe corrosion was noted at WS 30, which caused a localized reduction in thickness of 13.5%. This area of corrosion, shown in figure 143, covered 0.86 square inch. The location of a crack found on the left wing aft main gear door hinge bracket, part number 40560-05, is shown in figure 144. This crack, measuring 0.273 inch in length, occurred at WS 30. A macroscopic view of this crack is shown in figure 145, and a microscopic view of the crack is shown in figure 146.

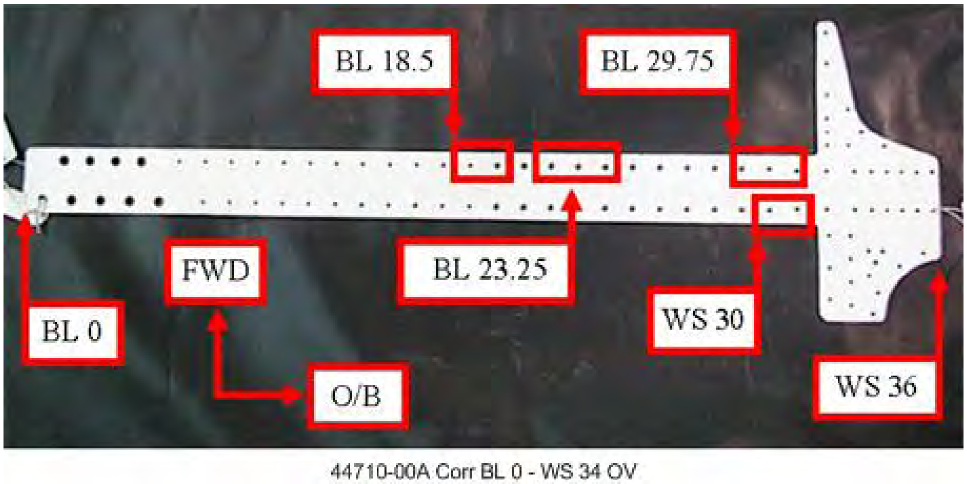


Figure 140. Locations of Light-Moderate, Moderate, and Severe Corrosion on the Left Wing Plate Wing Panel Reinforcement Assembly BL 0 Through WS 36

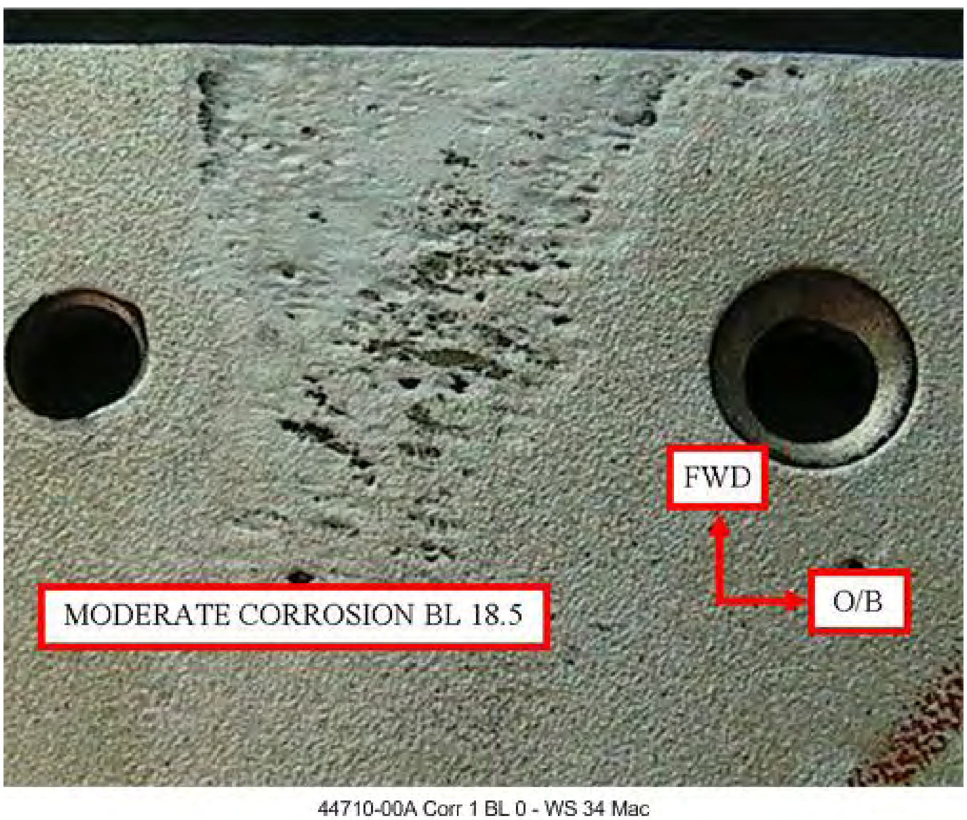


Figure 141. Macroscopic View of Moderate Corrosion on the Left Wing Plate Wing Panel Reinforcement Assembly BL 18.5

Figure 142. Macroscopic View of Severe Corrosion on the Left Wing Plate Wing Panel Reinforcement Assembly WS 29.75

Figure 143. Macroscopic View of Severe Corrosion on the Left Wing Plate Wing Panel Reinforcement Assembly WS 30

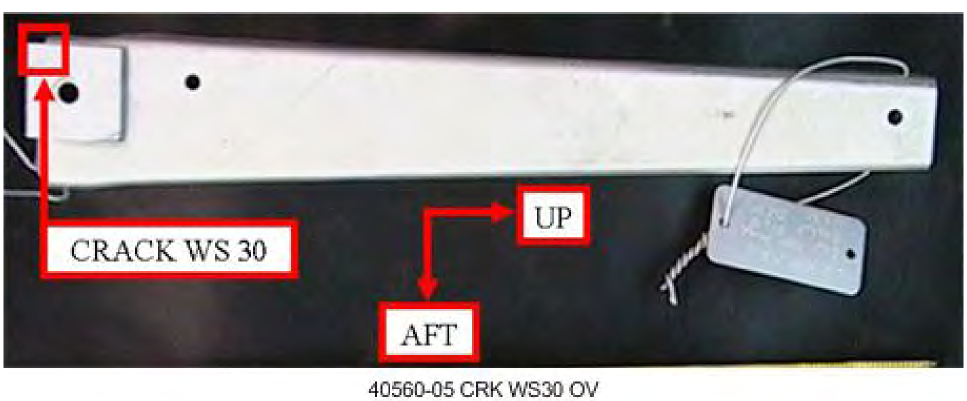


Figure 144. Location of Crack on the Left Wing Aft Main Gear Door Hinge Bracket WS 30

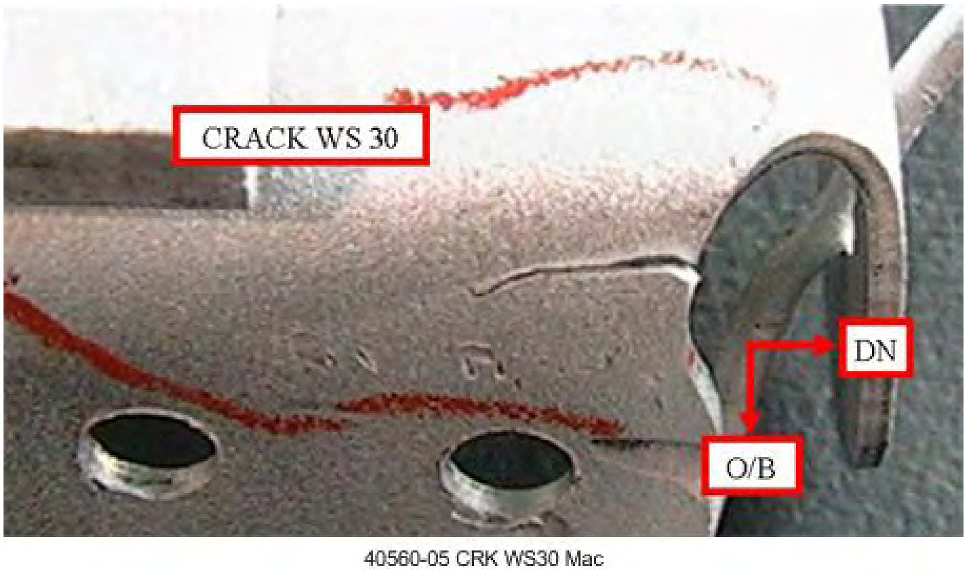


Figure 145. Macroscopic View of Crack on the Left Wing Aft Main Gear Door Hinge Bracket WS 30

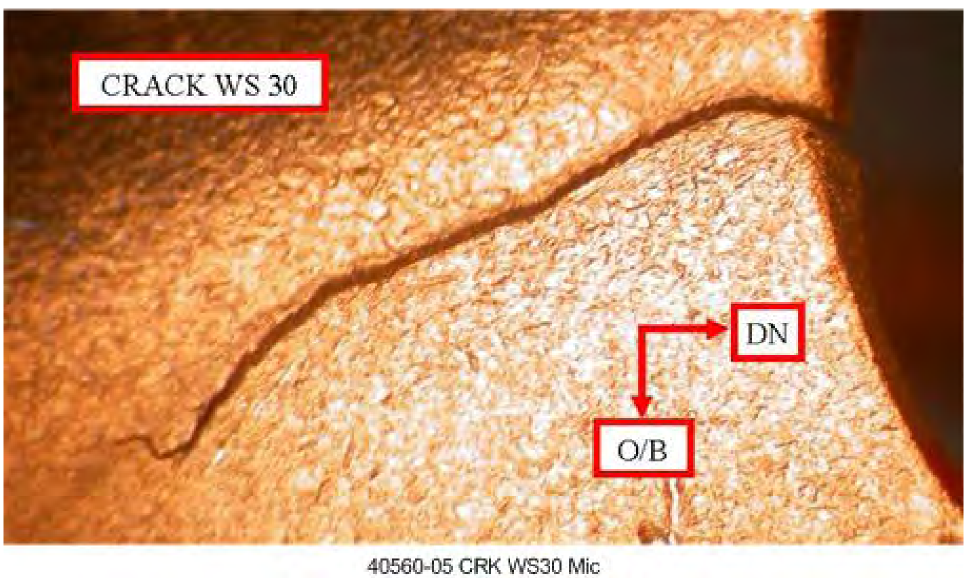
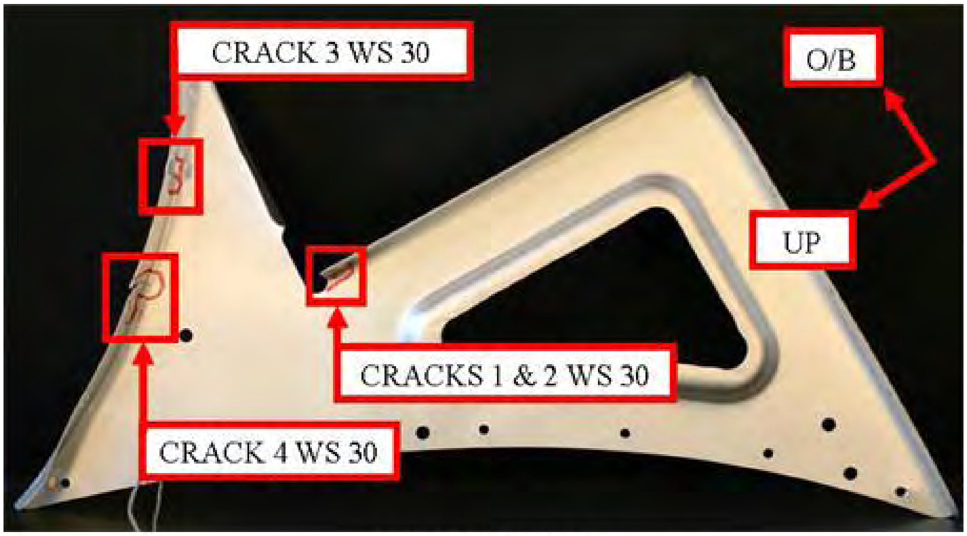


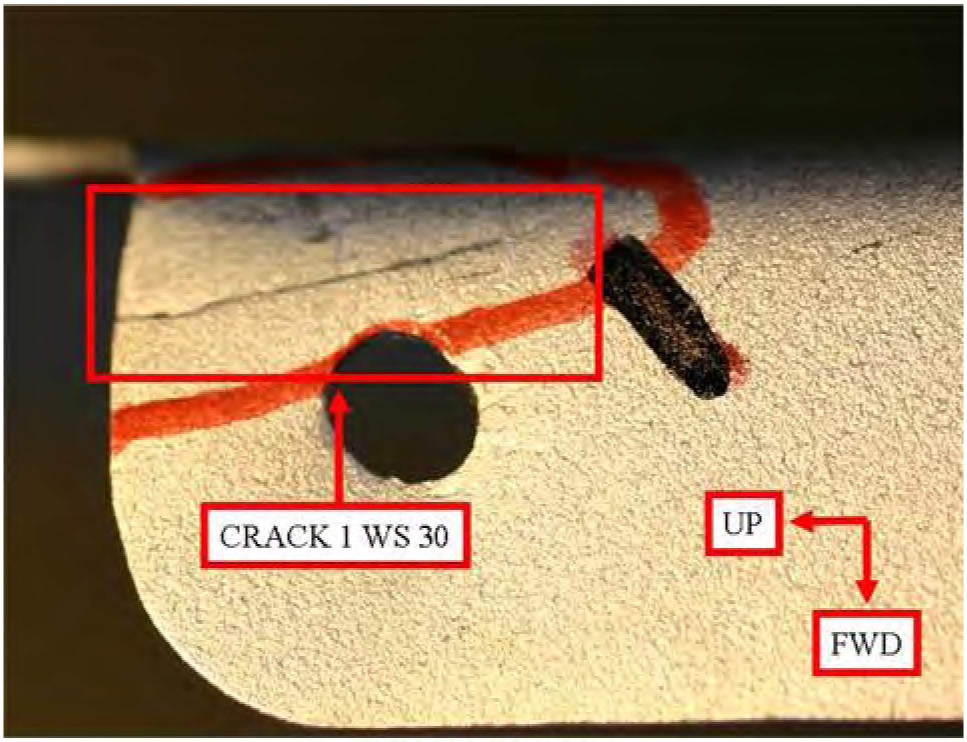
Figure 146. Microscopic View of Crack on the Left Wing Aft Main Gear Door Hinge Bracket WS 30

The location of four cracks on the left wing bulkhead wing fillet, part number 43646-04 located at WS 30 FS 181, is shown in figure 147. A macroscopic view of crack 1, measuring 0.423 inch, is shown in figure 148, while a microscopic view of this crack is shown in figure 149. Crack 2, which measures 0.226 inch, is shown macroscopically in figure 150 and microscopically in figure 151. Multiple cracks, grouped together and identified as crack 3, were found emanating from a fastener hole, as shown in figure 152. A microscopic view of these cracks, which measured 0.157 inch, is shown in figure 153. Multiple cracks, grouped together and identified as crack 4, are shown macroscopically in figure 154. A microscopic view of these cracks, which measured 0.386 inch, is shown in figure 155.



43646-4 CRK FS181 OV

Figure 147. Location of Cracks on the Left Wing Bulkhead Wing Fillet WS 30 FS 181



43646-4 CRK 1 FS181 Mac

Figure 148. Macroscopic View of Crack 1 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181

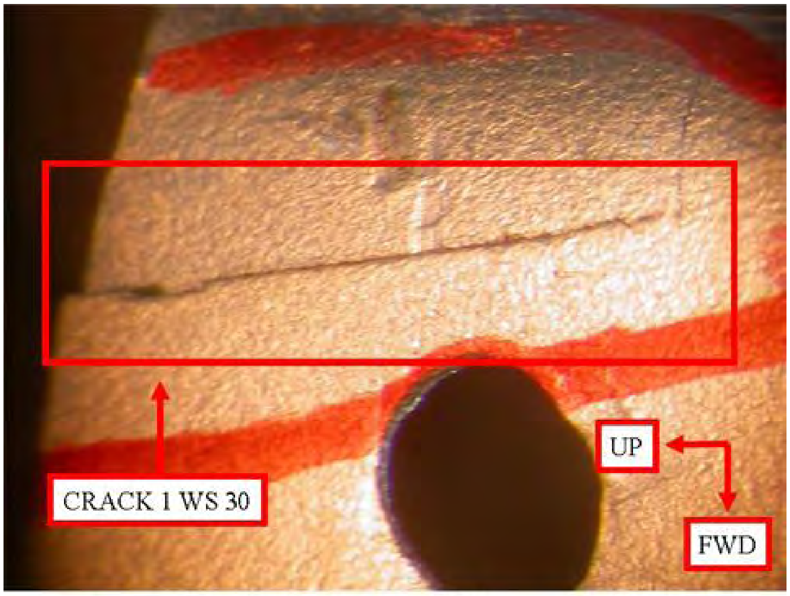
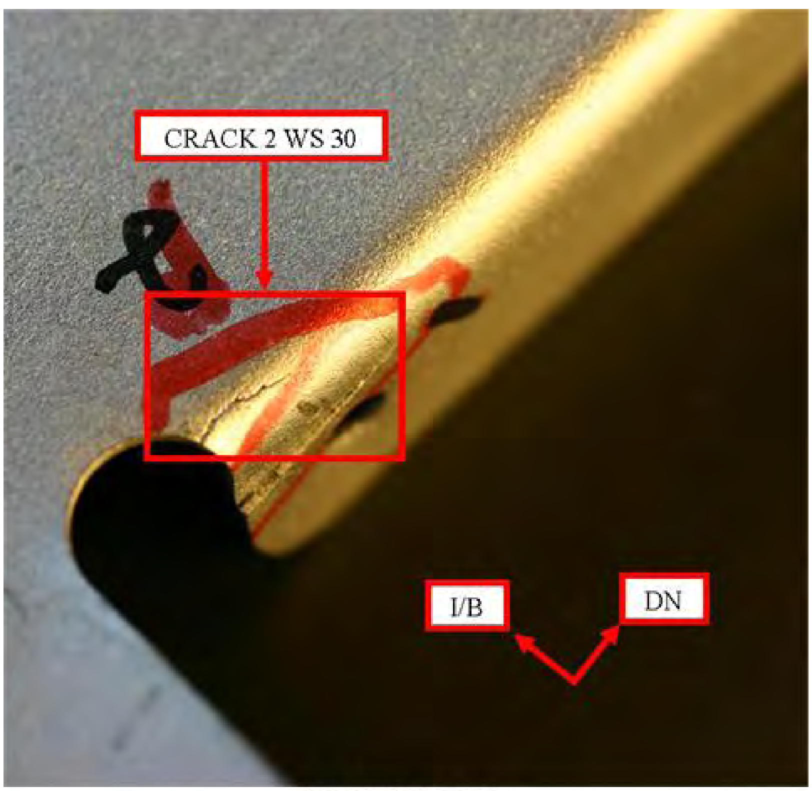


Figure 149. Microscopic View of Crack 1 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181



I/B = Inboard

Figure 150. Macroscopic View of Crack 2 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181

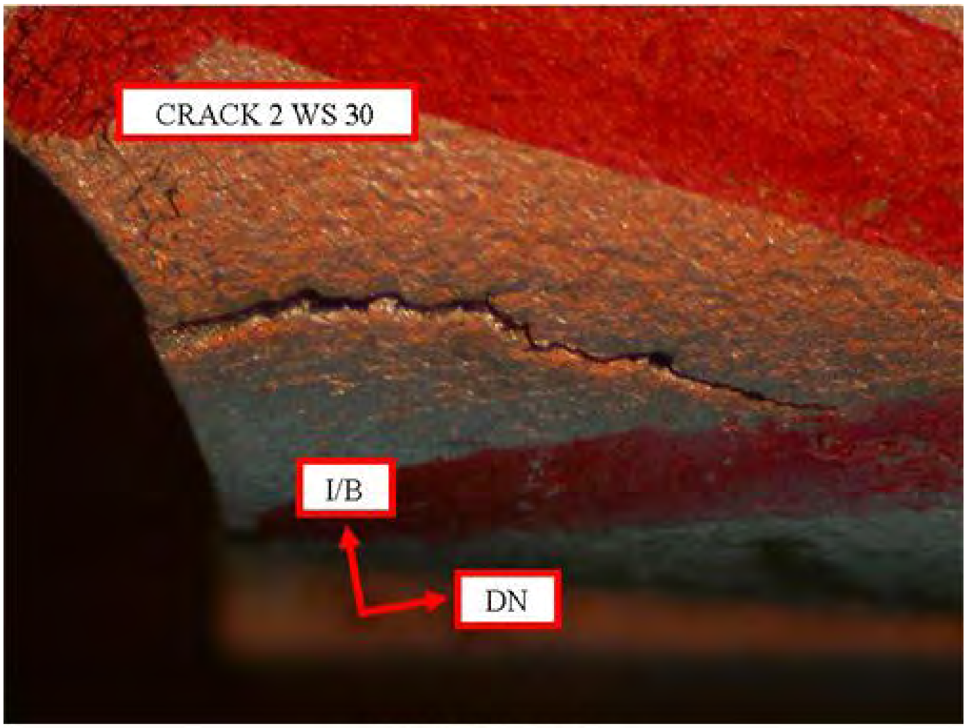


Figure 151. Microscopic View of Crack 2 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181

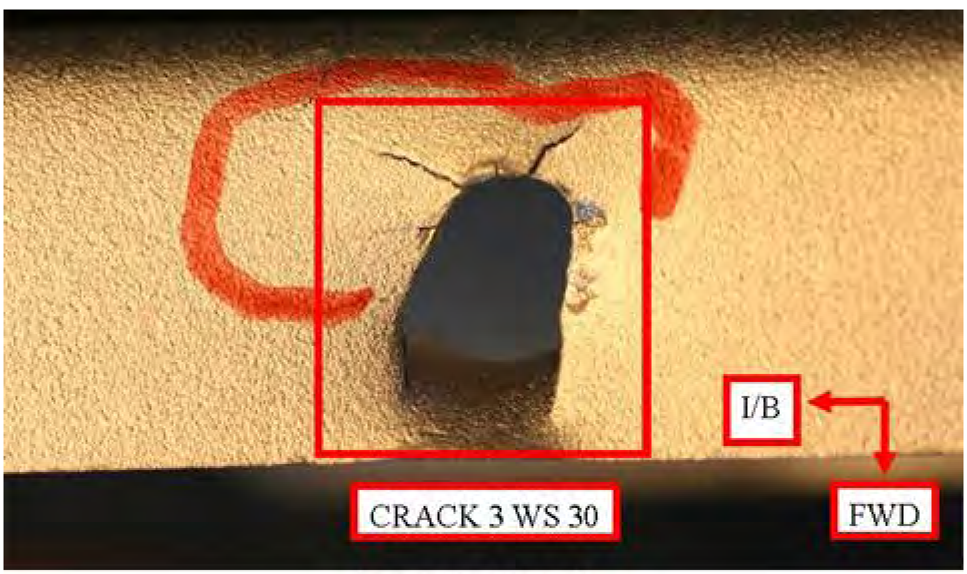
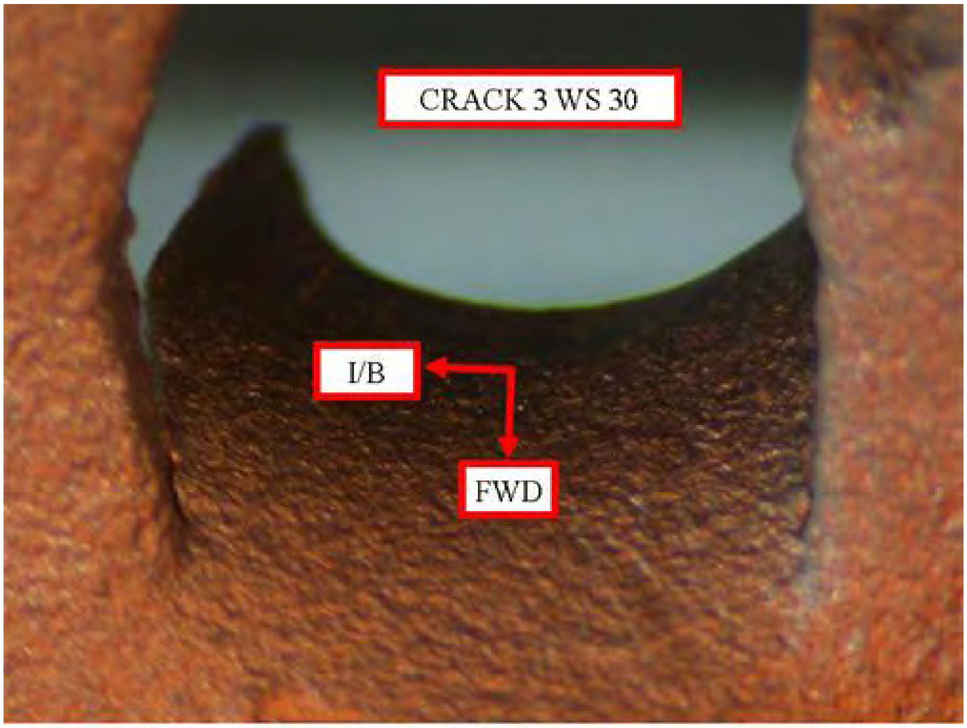


Figure 152. Macroscopic View of Crack 3 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181



43646-4 CRK 3 FS181 Mic

Figure 153. Microscopic View of Crack 3 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181

43646-4 CRK 4 FS181 Mac

Figure 154. Macroscopic View of Crack 4 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181

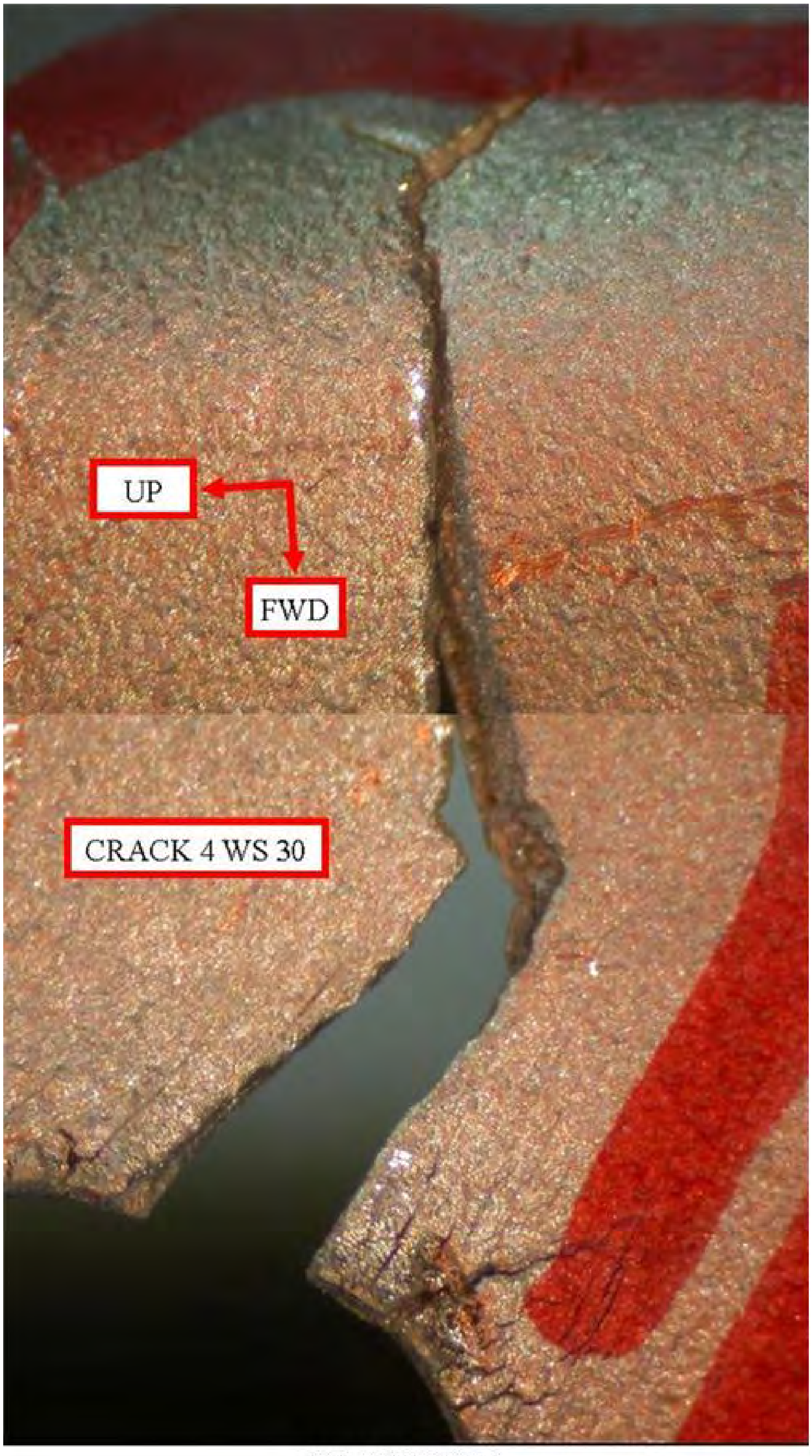
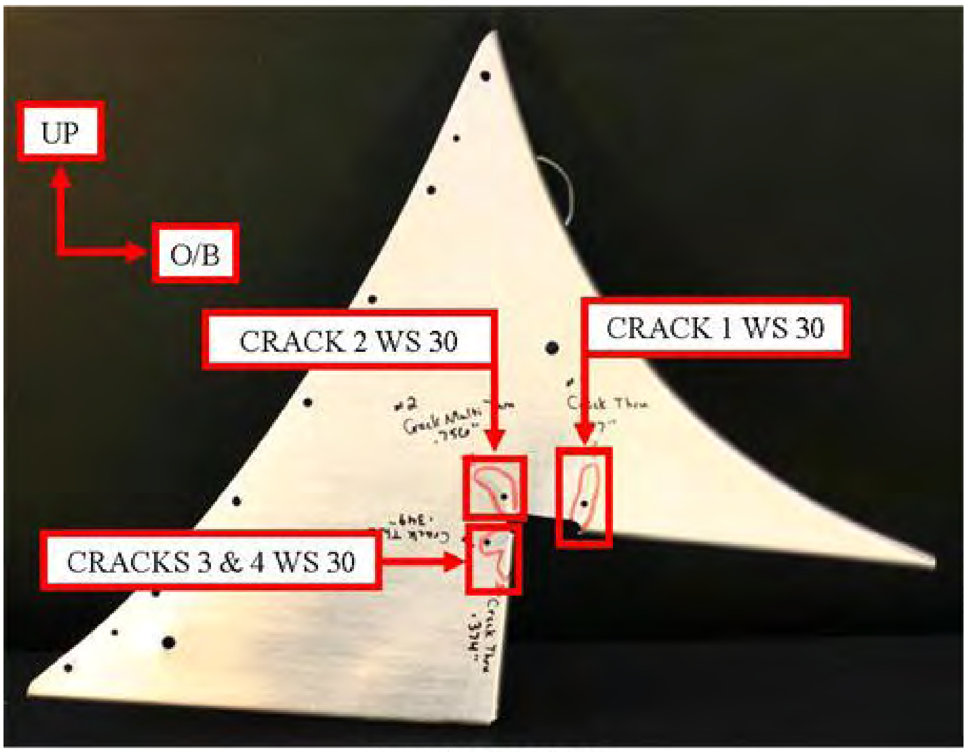


Figure 155. Microscopic View of Crack 4 on the Left Wing Bulkhead Wing Fillet WS 30 FS 181

The location of four cracks on the left wing bulkhead wing fillet at WS 30 FS 191.7, part number 43646-04, is shown in figure 156. A macroscopic view of crack 1, which originated from both sides of a fastener hole and grew to a final total length of 0.77 inch, is shown in figure 157. Figures 158 and 159 show microscopic views of both branches of the crack. The macroscopic view of multiple cracks, grouped together and identified as crack 2, is shown in figure 160. This crack also occurs on both sides of a fastener hole and has a total length of 0.756 inch. Microscopic views of this crack are shown in figures 161 and 162. Figure 163 provides the macroscopic view for cracks 3 and 4. A microscopic view of crack 3, which measured 0.349 inch, is shown in figure 164, and the microscopic view of crack 4, which measured 0.374 inch, is shown in figure 165.



43649-004 CRK FS 191 OV

Figure 156. Location of Cracks on Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

Figure 157. Macroscopic View of Crack 1 on the Left Wing Bulkhead Wing Fillet
WS 30 FS 191.7

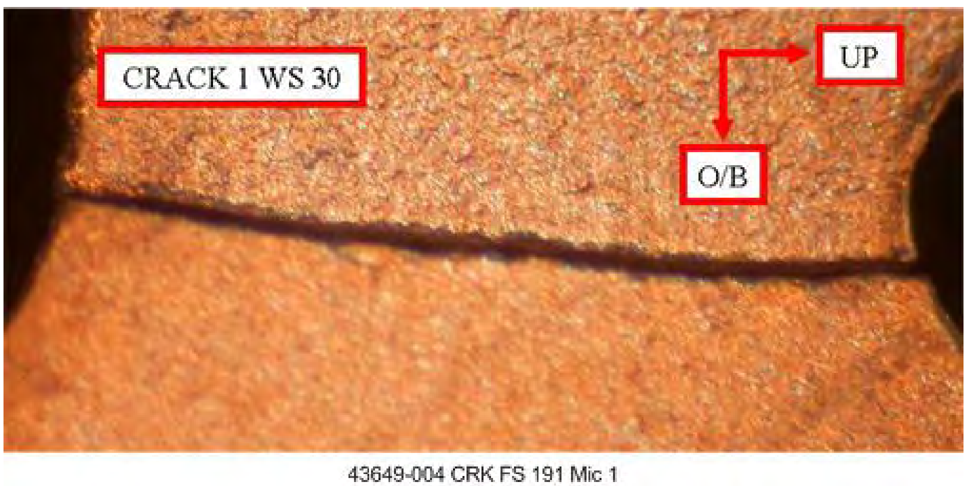
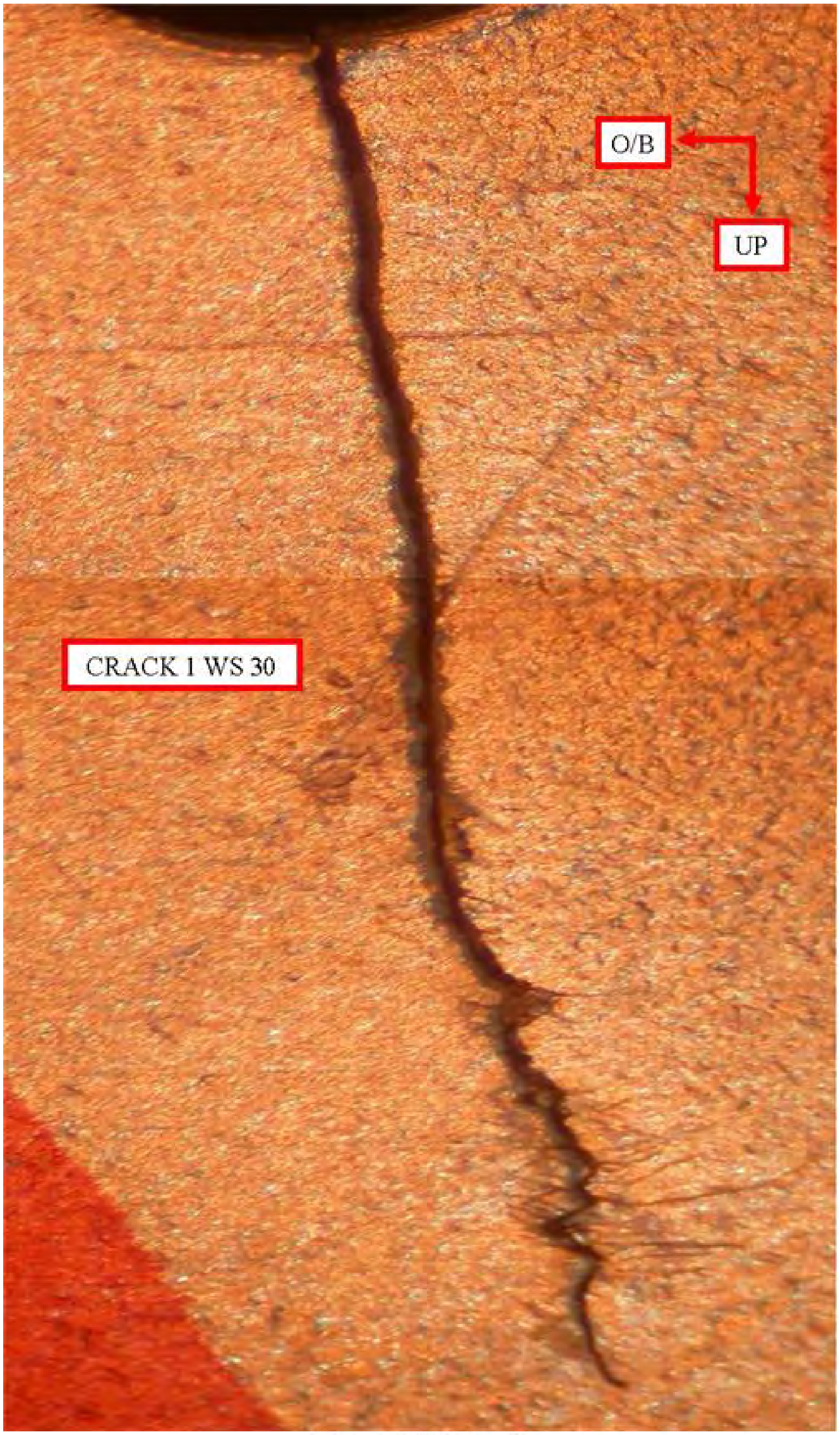


Figure 158. Microscopic View of Crack 1 Branch 1 on the Left Wing Bulkhead Wing Fillet
WS 30 FS 191.7



43649-004 CRK FS 191 Mic 2

Figure 159. Microscopic View of Crack 1 Branch 2 on the Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

Figure 160. Macroscopic View of Crack 2 on the Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

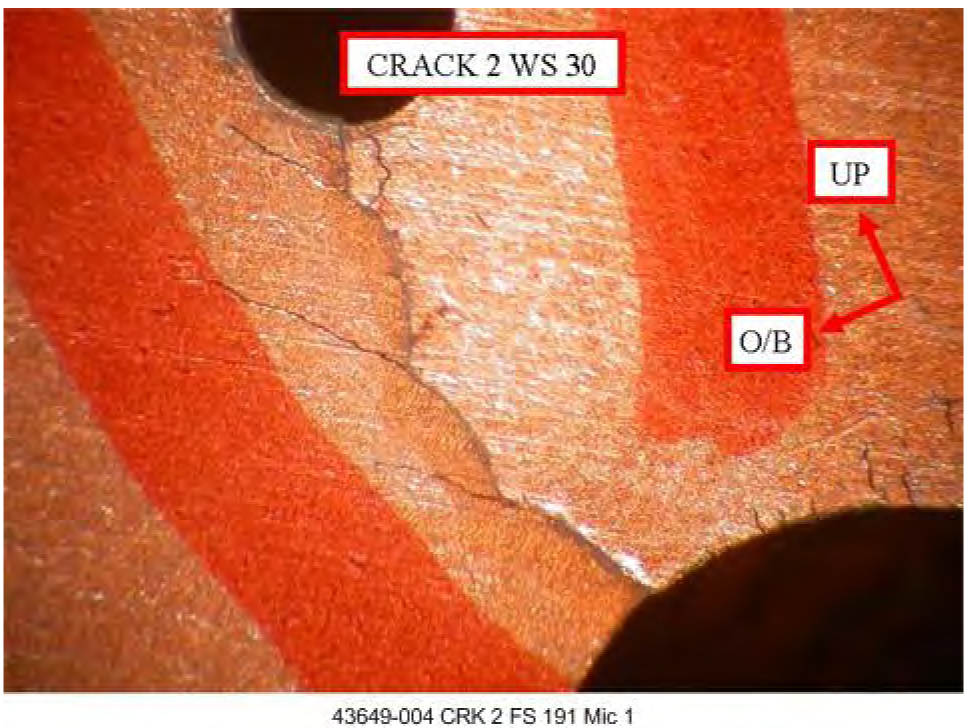


Figure 161. Microscopic View of Crack 2 Branch 1 on the Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

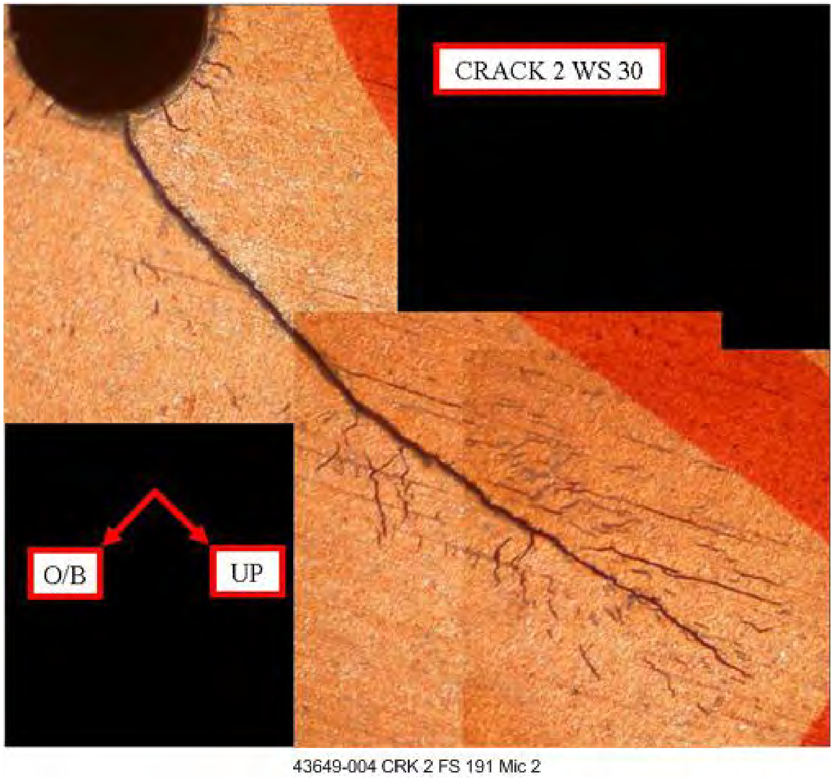


Figure 162. Microscopic View of Crack 2 Branch 2 on the Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

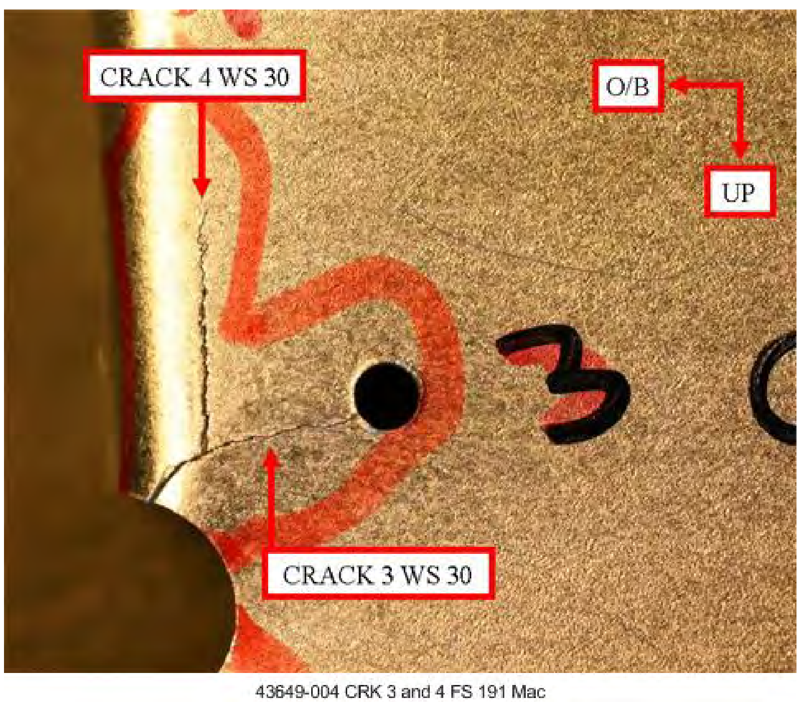


Figure 163. Macroscopic View of Cracks 3 and 4 on the Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

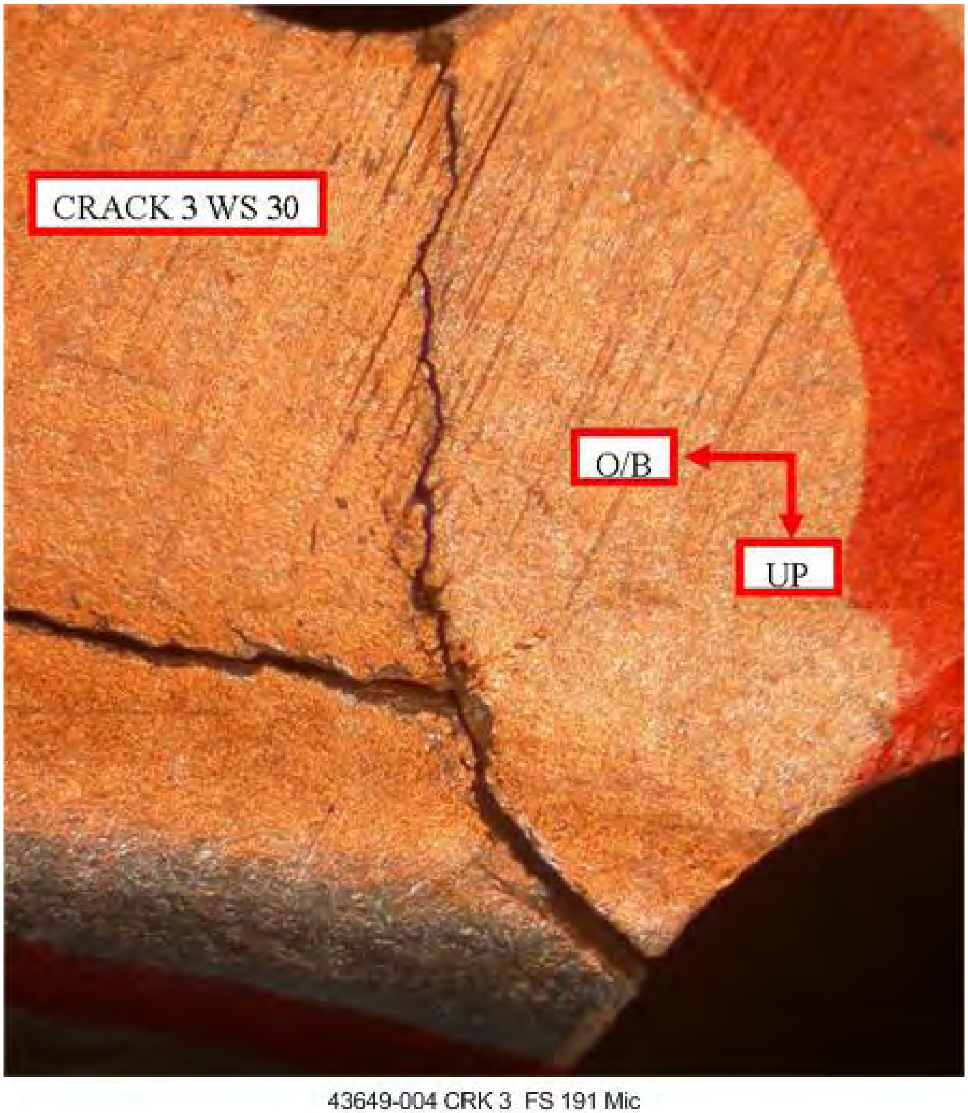


Figure 164. Microscopic View of Crack 3 on the Left Wing Bulkhead Wing Fillet
WS 30 FS 191.7

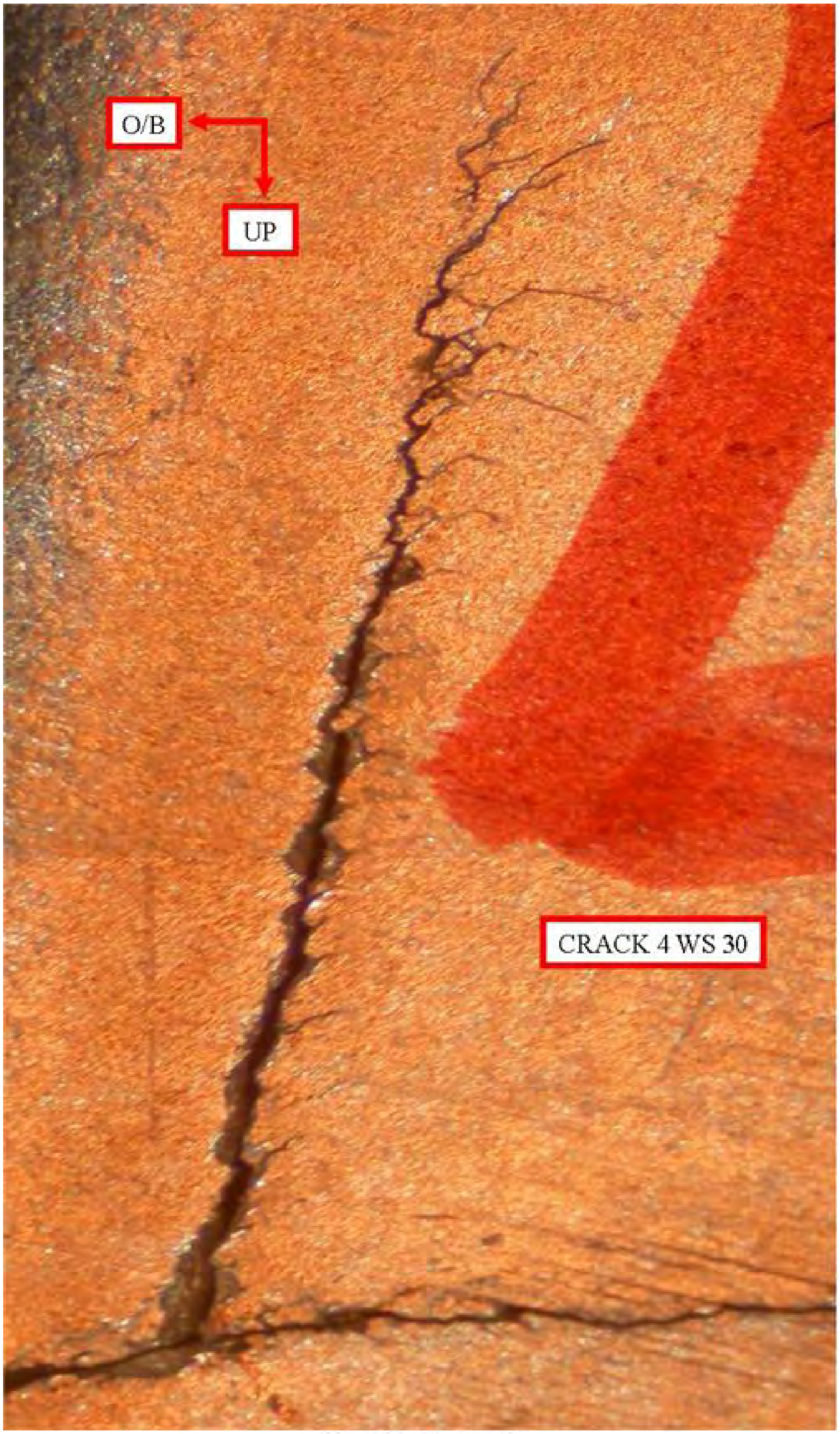
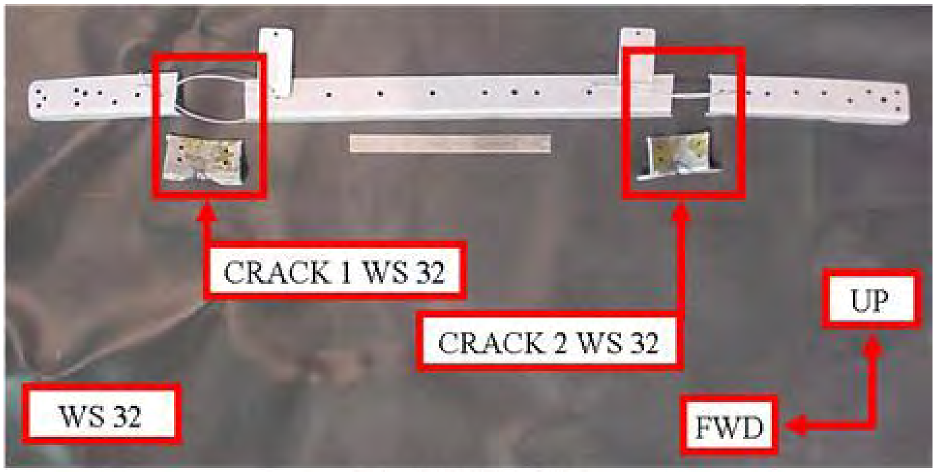


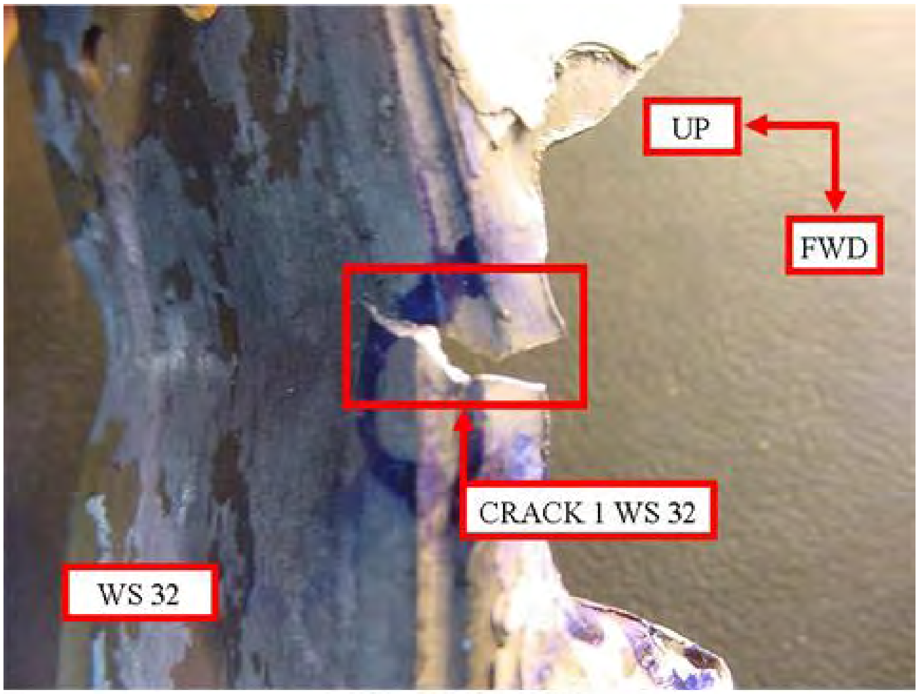
Figure 165. Microscopic View of Crack 4 on the Left Wing Bulkhead Wing Fillet WS 30 FS 191.7

Figure 166 provides the location of crack 1 and 2 on the left wing main gear door angle, part number 45955-00, at WS 32. These cracks were identified during the inspections, and were removed prior to the fluorescent liquid penetrant process. Figure 167 shows a macroscopic view of crack 1, which measures 0.235 inch, and figure 168 shows a macroscopic view of crack 2, measuring 0.231 inch. The fracture face for crack 2 is shown in figure 169. From an analysis of the fracture faces for crack 1 and 2, it was determined that the failure mode for crack 2 was fatigue. The failure mode for crack 1 could not be determined due to extensive smearing of the fracture face.



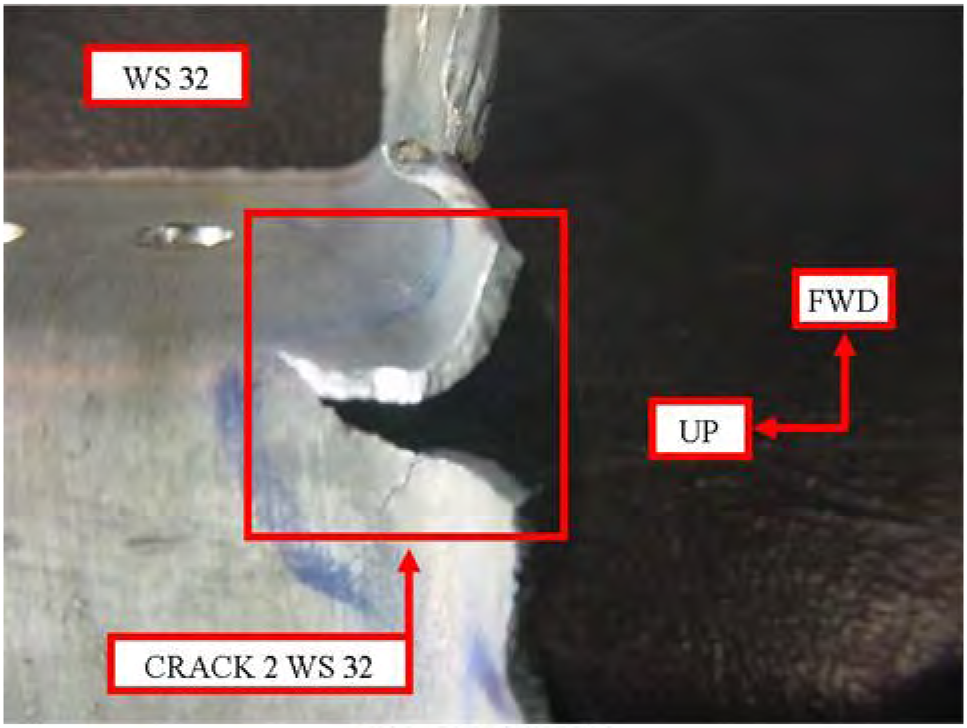
45955-00 CRK WS 32 OV

Figure 166. Location of Cracks 1 and 2 on the Left Wing Main Gear Door Angle WS 32



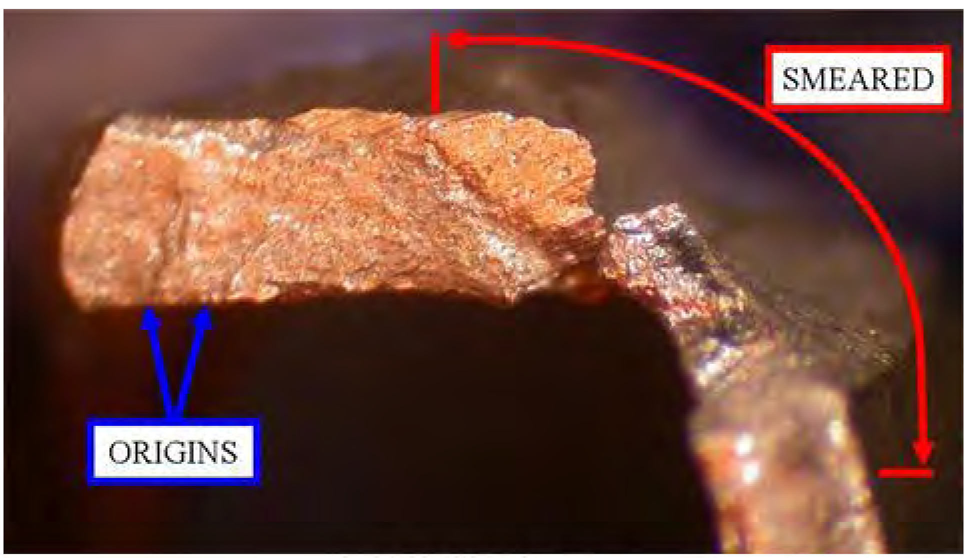
45955-00 CRK 1 WS 32 Mac

Figure 167. Macroscopic View of Crack 1 on the Left Wing Main Gear Door Angle WS 32



45955-00 CRK 2 WS 32 Mac

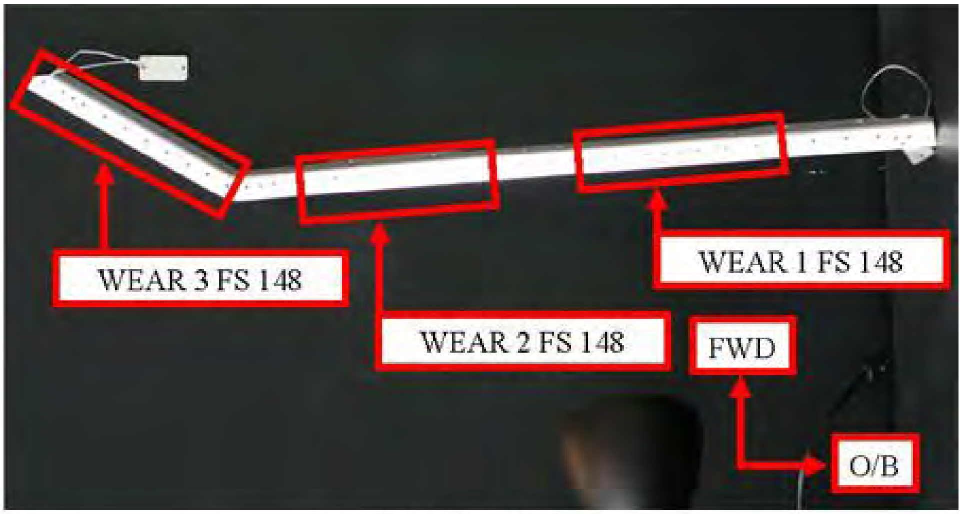
Figure 168. Macroscopic View of Crack 2 on the Left Wing Main Gear Door Angle WS 32



45955-00 CRK 2 WS 32 Frac

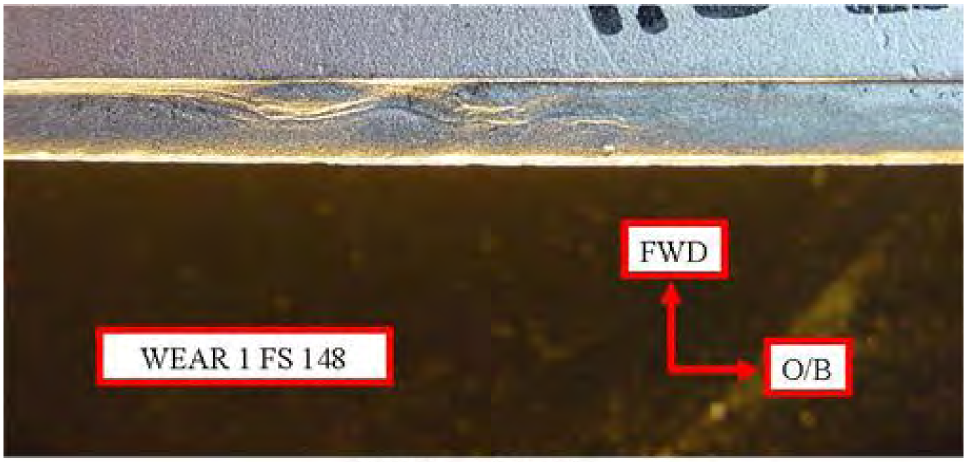
Figure 169. Fracture Face Crack 2 on the Left Wing Main Gear Door Angle WS 32

Figure 170 shows the location of three areas of wear on the left wing rear spar assembly lower aft angle, part number 45838-00. This entire part also had light scattered corrosion across its surface. Figure 171 shows a 2.47-square-inch area of wear, located at WS 38 FS 148. This area of wear resulted in a reduction in localized thickness of 54%. Another area of wear, identified as wear 2 and shown in figure 172, was located at WS 44 FS 148. This 0.9-square-inch area of wear resulted in a localized thickness loss of 17.7%. The final area of wear, documented on the lower aft angle, occurred at WS 64.5 FS 148, covered a surface area of 0.8 square inch, and caused a localized reduction in thickness of 32%. Wear 3 is shown in figure 173.



45838-00 Wear FS 148 OV

Figure 170. Location of Wear 1, 2, and 3 and Overview of Left Wing Rear Spar Assembly Lower Aft Angle FS 148



45838-00 Wear 1 FS 148 Mac

Figure 171. Macroscopic View of Wear 1 on the Left Wing Rear Spar Assembly Lower Aft Angle WS 38 FS 148

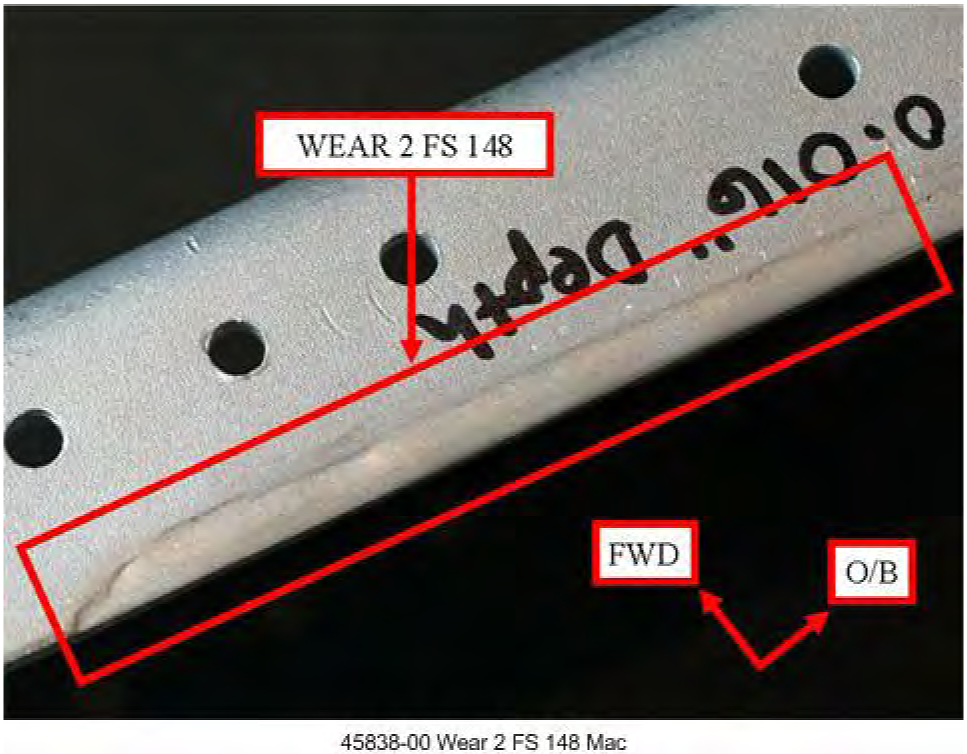


Figure 172. Macroscopic View of Wear 2 on the Left Wing Rear Spar Assembly Lower Aft Angle WS 44 FS 148

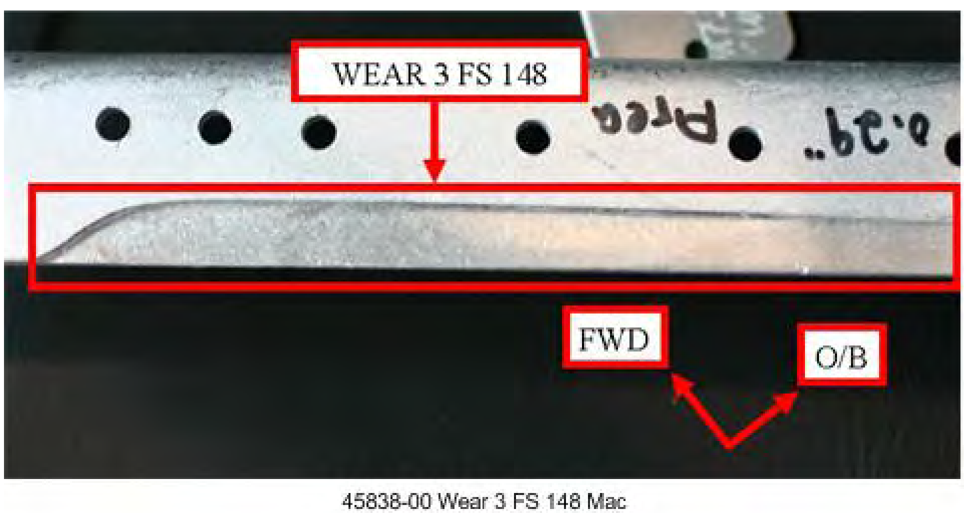
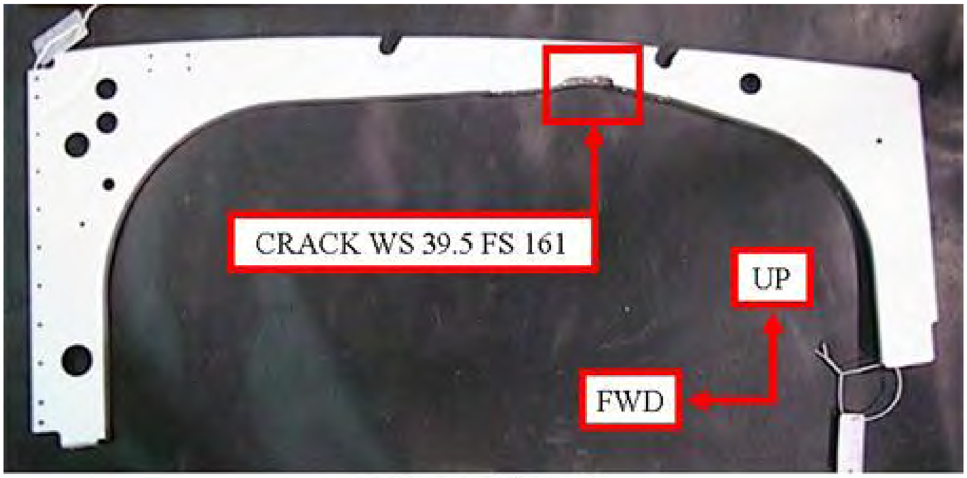


Figure 173. Macroscopic View of Wear 3 on the Left Wing Rear Spar Assembly Lower Aft Angle WS 64.5 FS 148

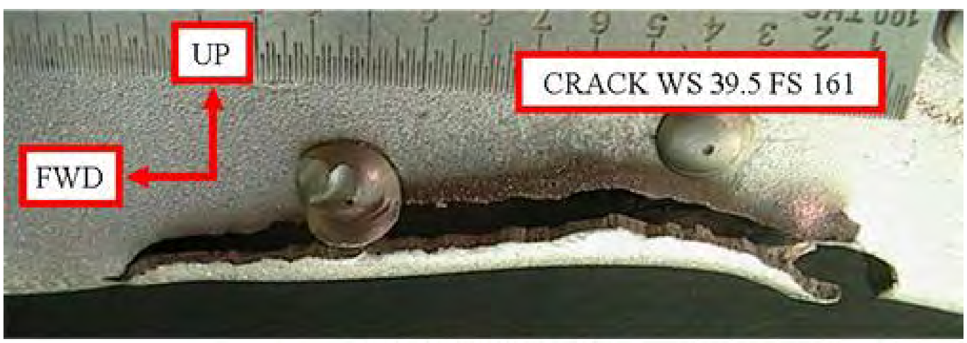
Figure 174 shows the location of a crack on the left wing rib assembly, part number 74099-06. A macroscopic view of the crack, which measures 1.525 inches and is located at WS 39.5 FS 161, is shown in figure 175, and the fluorescent liquid penetrant indication of this crack is shown in figure 176. Figure 177 shows a damaged section of the left wing rib, part number 40423-12. A macroscopic view of the damage to the left wing rib is shown in figure 178. The location of a

0.225-inch crack on the left wing main gear aft fitting assembly, part number 40294-00A, is shown in figure 179. A macroscopic view of this crack, which occurred at WS 58.5, is shown in figure 180, and the fluorescent liquid penetrant indication is shown in figure 181. Figure 182 shows the microscopic view of this crack. Light scattered corrosion was found on the left wing gear well plate 5, part number 40500-00 from WS 58.5 to WS 87.5. An overview providing the location of this corrosion, which caused a localized maximum thickness loss of 1.5%, is shown in figure 183.



74099-06 CRK WS 39.5 OV

Figure 174. Location of Crack on the Left Wing Rib Assembly WS 39.5 FS 161



74099-06 CRK WS 39.5 Mac

Figure 175. Macroscopic View of Crack on the Left Wing Rib Assembly WS 39.5 FS 161

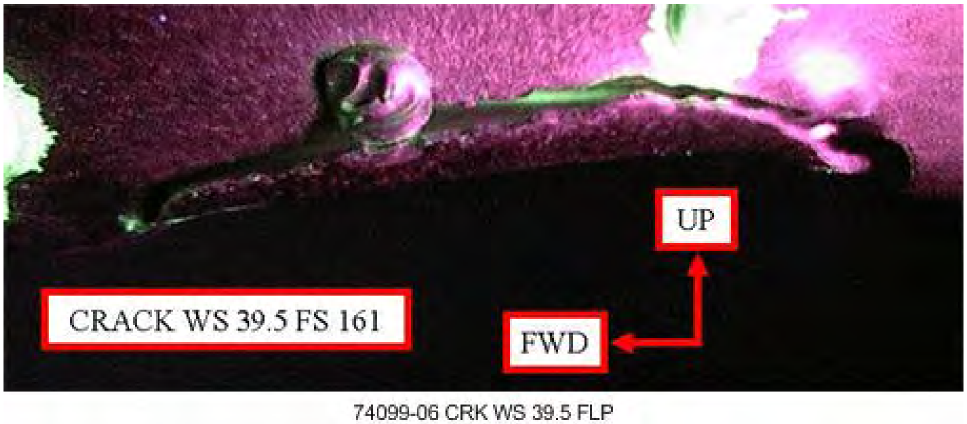


Figure 176. Fluorescent Liquid Penetrant Indication of Crack on the Left Wing Rib Assembly WS 39.5 FS 161

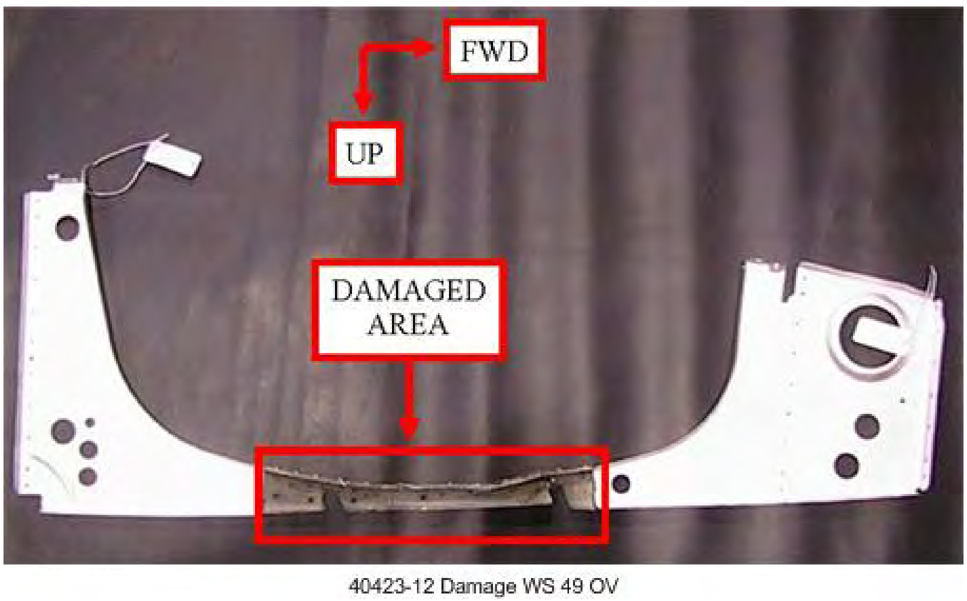
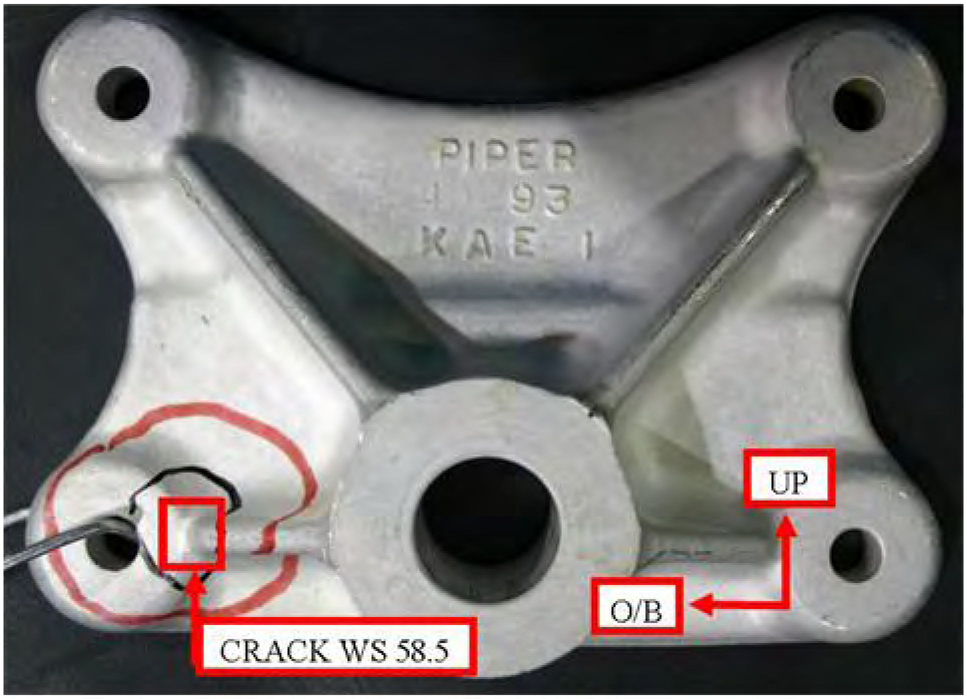


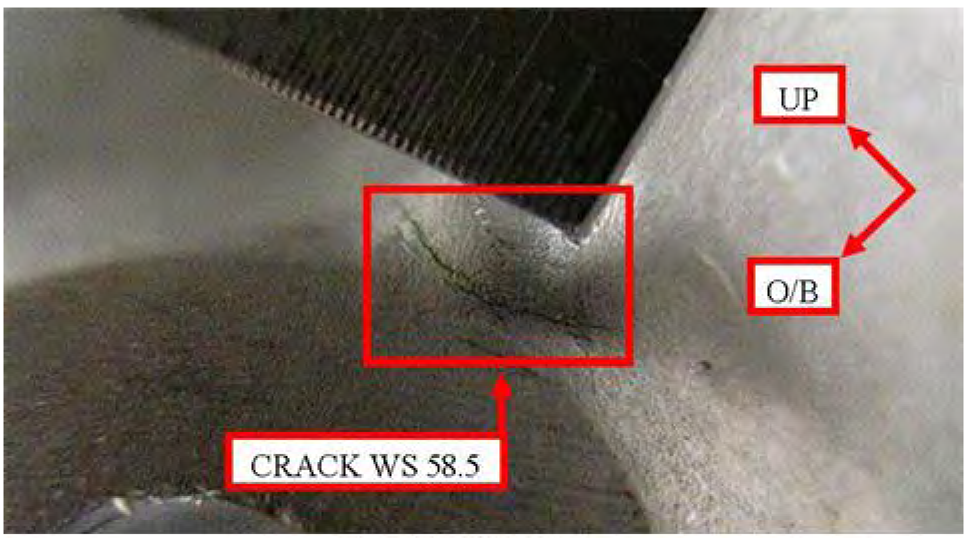
Figure 177. Overview of Damage on the Left Wing Rib WS 49

Figure 178. Macroscopic View of the Damage on the Left Wing Rib WS 49



40294-00A CRK WS 58.5 OV

Figure 179. Location of Crack on the Left Wing Main Gear Aft Fitting Assembly WS 58.5



40294-00A CRK WS 58.5 Mac

Figure 180. Macroscopic View of the Left Wing Main Gear Aft Fitting Assembly WS 58.5

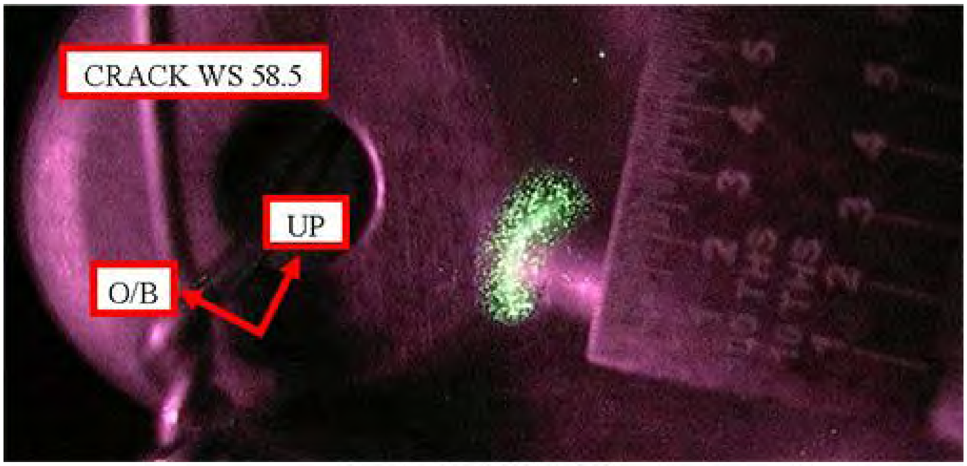


Figure 181. Fluorescent Liquid Penetrant Indication on the Left Wing Main Gear Aft Fitting Assembly WS 58.5

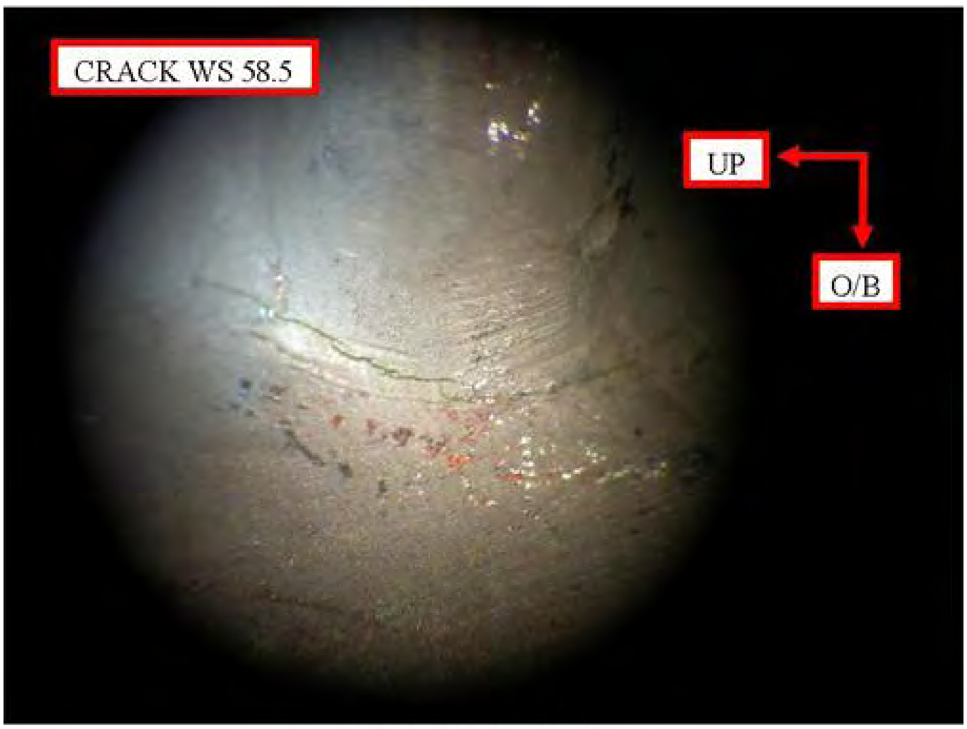


Figure 182. Microscopic View of Crack on the Left Wing Main Gear Aft Fitting Assembly WS 58.5

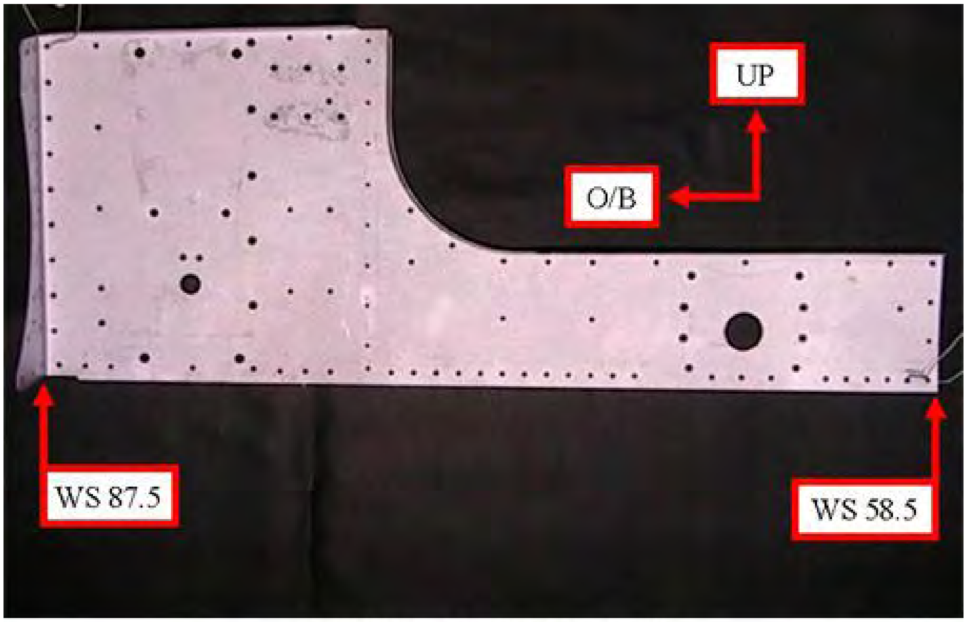
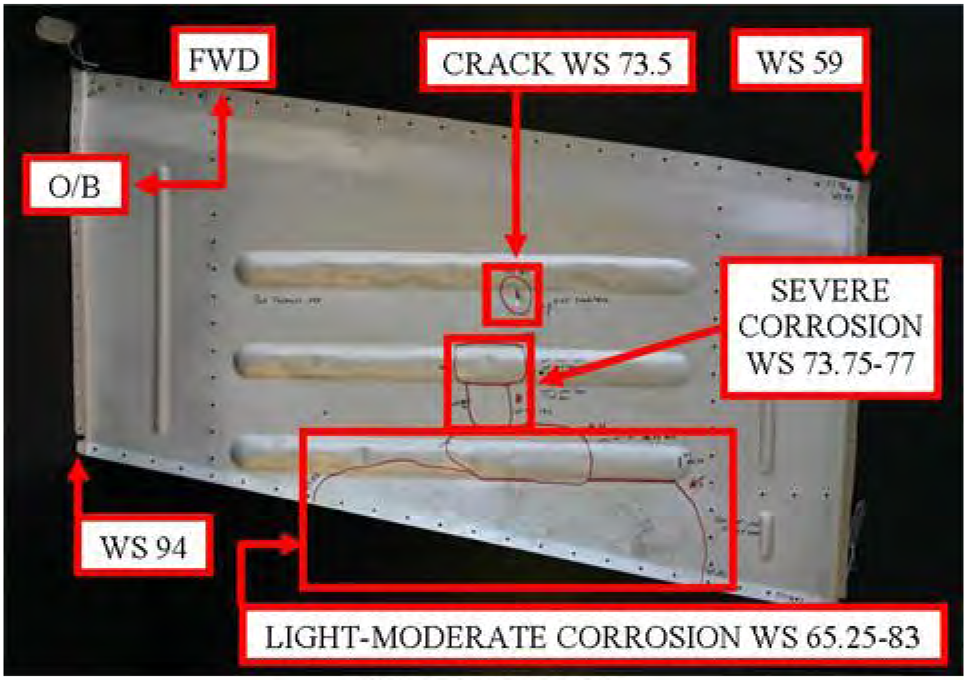


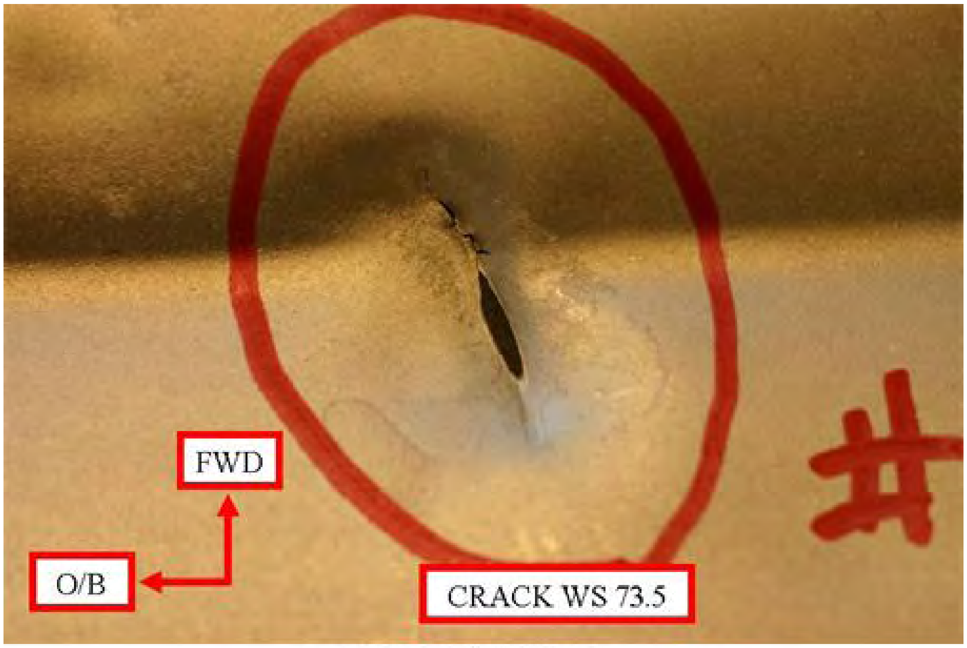
Figure 183. Overview of Left Wing Gear Well Plate 5 WS 58.5 Through WS 87.5

The location of a crack, light-moderate, and severe areas of corrosion found on the left nacelle front locker floor, part number 44493-04, is shown in figure 184. A 76.5-square-inch area of light-moderate corrosion, which caused a localized reduction in thickness of 3%, was located from WS 65.25 to WS 83, and an additional 20-square-inch area of light-moderate corrosion, also causing a thickness loss of 3%, was documented from WS 66.75 to WS 77. Figure 185 shows a macroscopic view of a 0.65-inch-long crack located at WS 73.5. The macroscopic view of a 3.75-square-inch area of severe corrosion, causing a localized reduction in thickness of 15.6%, is shown in figure 186. This area of corrosion was located from WS 73.75 to WS 77 on the locker floor. Another area of severe corrosion, located from WS 74.5 to WS 76, was documented on the front locker floor. This 2-square-inch area of corrosion, shown in figure 187, caused a localized reduction in thickness of 22%. Figure 188 shows an overview of scattered light corrosion on the left wing engine nacelle bulkhead assembly, part number 44437-04, located at WS 65.69. This corrosion resulted in a localized thickness loss of 2%.



44493-04 Corr WS 59-94 OV

Figure 184. Location of Crack, Light-Moderate, and Severe Corrosion on the Left Nacelle Front Locker Floor WS 59 Through WS 94



44493-04 CRK WS 59-94 Mac

Figure 185. Macroscopic View of Crack on the Left Nacelle Front Locker Floor WS 73.5

Figure 186. Macroscopic View of Severe Corrosion on the Left Nacelle Front Locker Floor WS 73.75 Through WS 77

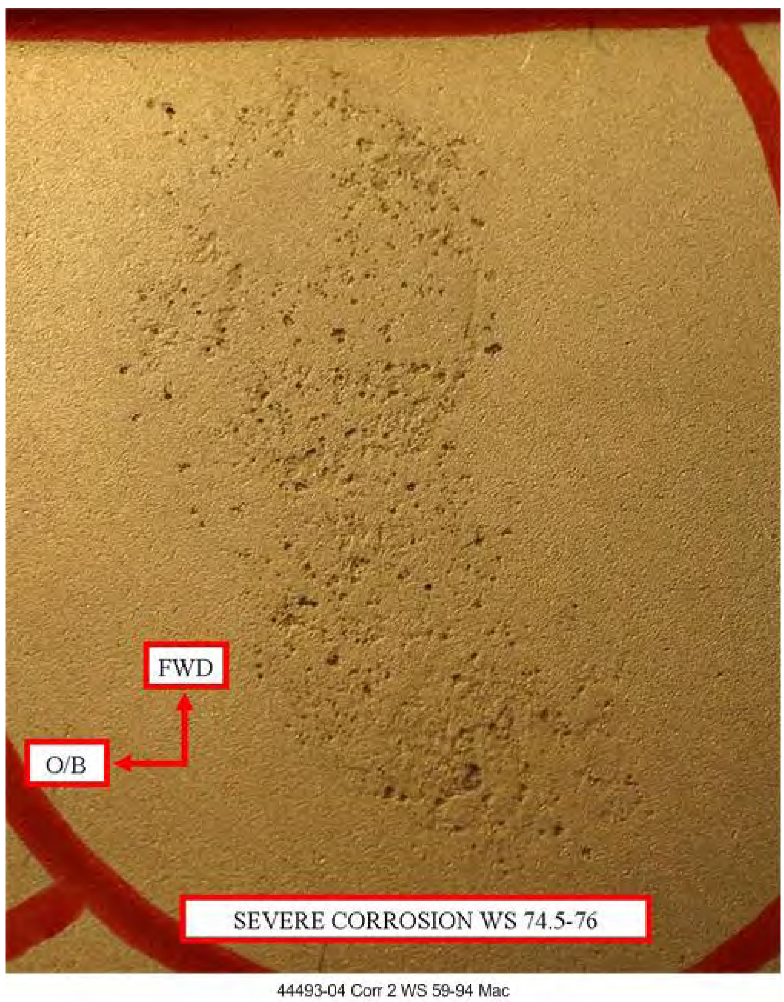


Figure 187. Macroscopic View of Severe Corrosion on the Left Nacelle Front Locker Floor WS 74.5 Through WS 76

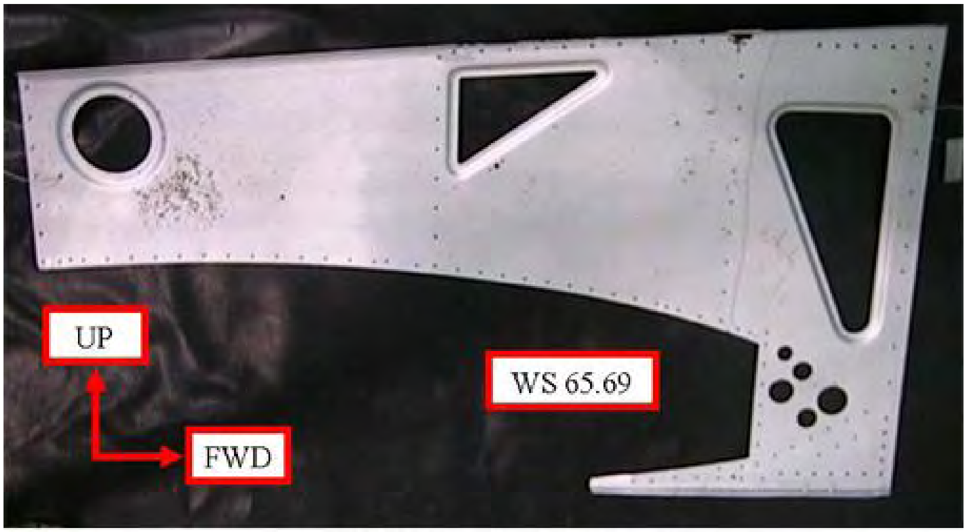


Figure 188. Overview of Left Wing Engine Nacelle Bulkhead Assembly WS 65.69

The location of a crack in the left wing nacelle inboard angle longitudinal bulkhead, part number 41632-00, is shown in figure 189. A macroscopic view of this crack, which measures 1.695 inches, is shown in figure 190, and the facture face of this crack, located at WS 66, is shown in figure 191. A fractograph, shown in figure 192, confirms the failure mode of this crack is fatigue. This fractorgraph provides a view of the fracture face features, in this case fatigue striations, and the crack growth direction.

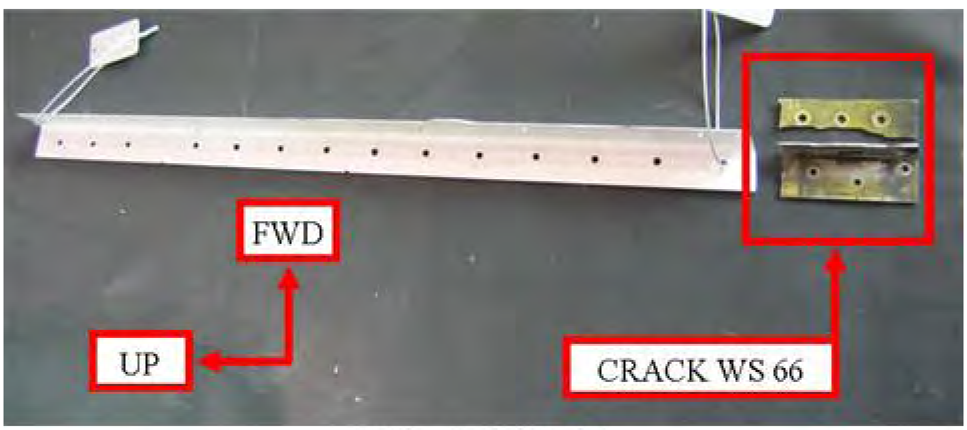


Figure 189. Location of Crack on the Left Wing Nacelle Inboard Angle Longitudinal Bulkhead WS 66

Figure 190. Macroscopic View of Crack on the Left Wing Nacelle Inboard Angle Longitudinal Bulkhead WS 66

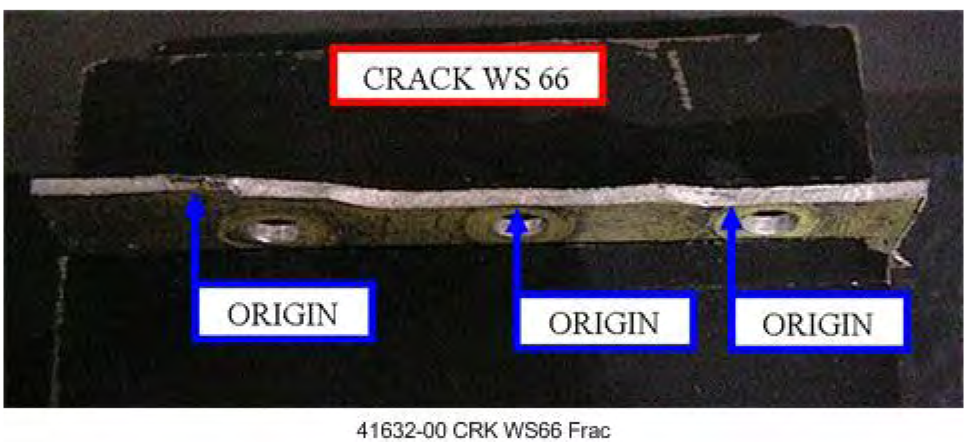


Figure 191. Fracture Face of Crack on the Left Wing Nacelle Inboard Angle Longitudinal Bulkhead WS 66

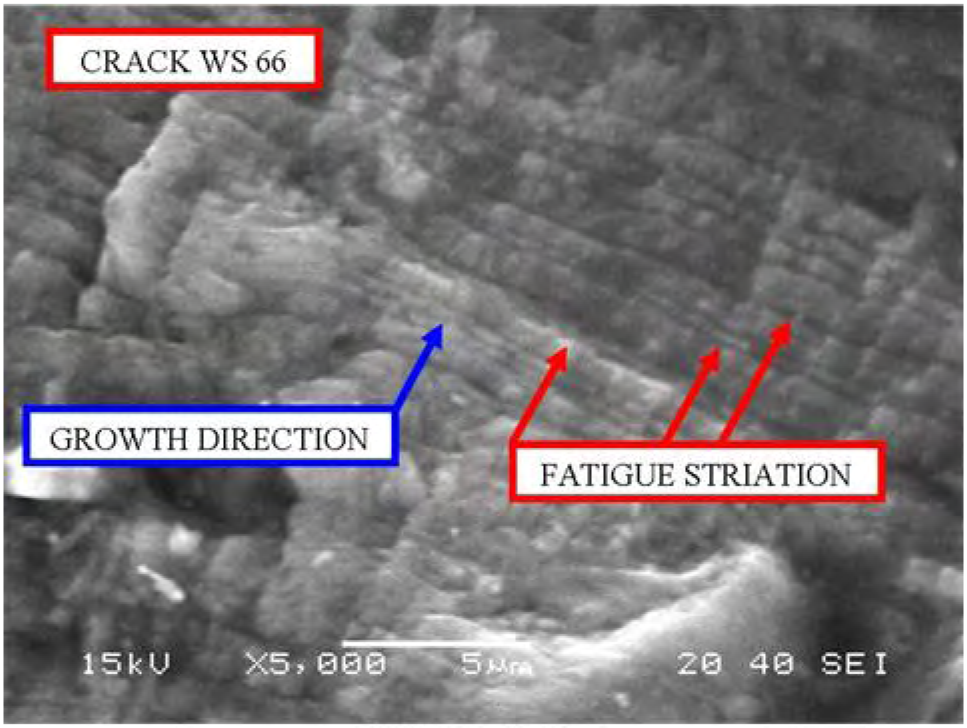
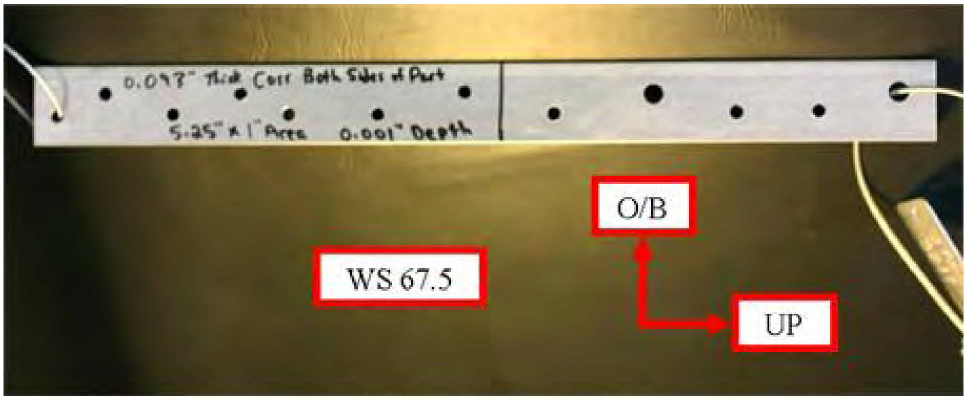


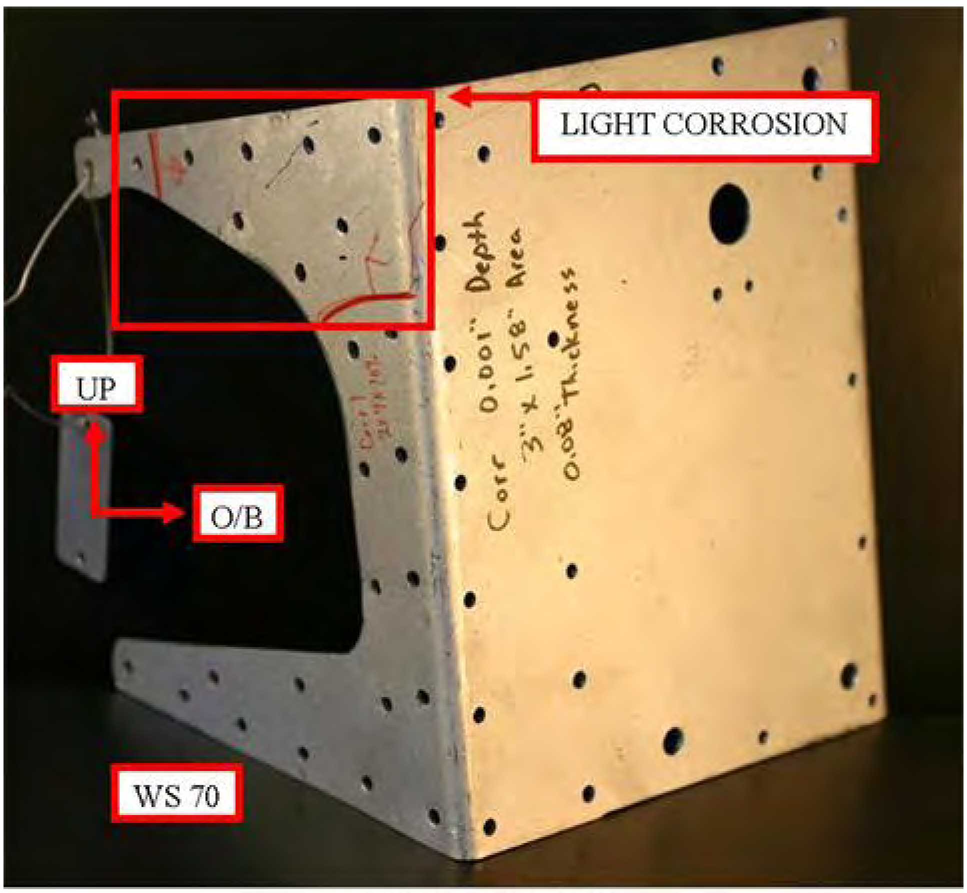
Figure 192. Fractograph of the Crack on the Left Wing Nacelle Inboard Angle Longitudinal Bulkhead WS 66

Figure 193 shows the location of light corrosion on the left wing rear spar rib assembly, part number 40419-24. This 5.25-square-inch area of corrosion, located at WS 67.5, caused a maximum thickness loss of 1%. An area of light corrosion, causing a maximum reduction in thickness of 1.3%, was observed on the left wing bulkhead assembly rear spar main gear support, part number 45501-00, at WS 70. Figure 194 shows the location of this 4.74-square-inch area of corrosion. Two areas of wear were observed on the left wing nacelle plate assembly firewall shear, part number 40671-00A, at WS 72, as shown in figure 195. A macroscopic view of wear 1, which covered a surface area of 0.38 square inch and caused a localized reduction in thickness of 20%, is shown in figure 196. Figure 197 shows a macroscopic view of wear 2, which covered a surface area of 0.78 square inch and resulted in a maximum thickness loss of 25%.



40419-24 Corr WS 67.5 OV

Figure 193. Location of Light Corrosion on the Left Wing Rear Spar Rib Assembly WS 67.5



45501-00 Corr WS 70 OV

Figure 194. Location of Light Corrosion on the Left Wing Bulkhead Assembly Rear Spar Main Gear Support WS 70

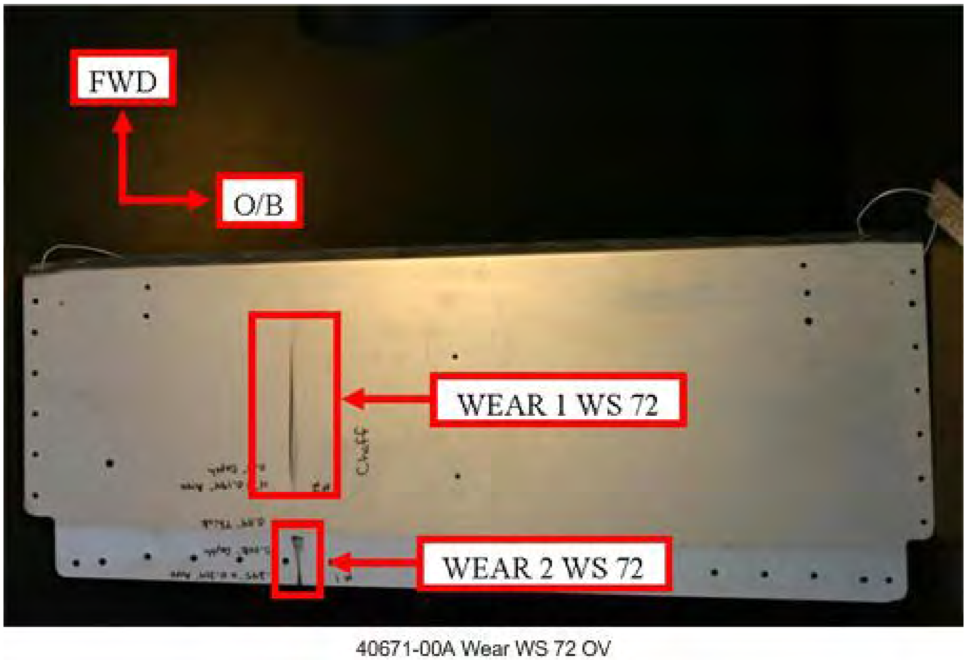


Figure 195. Location of Wear on the Left Wing Nacelle Plate Assembly Firewall Shear WS 72

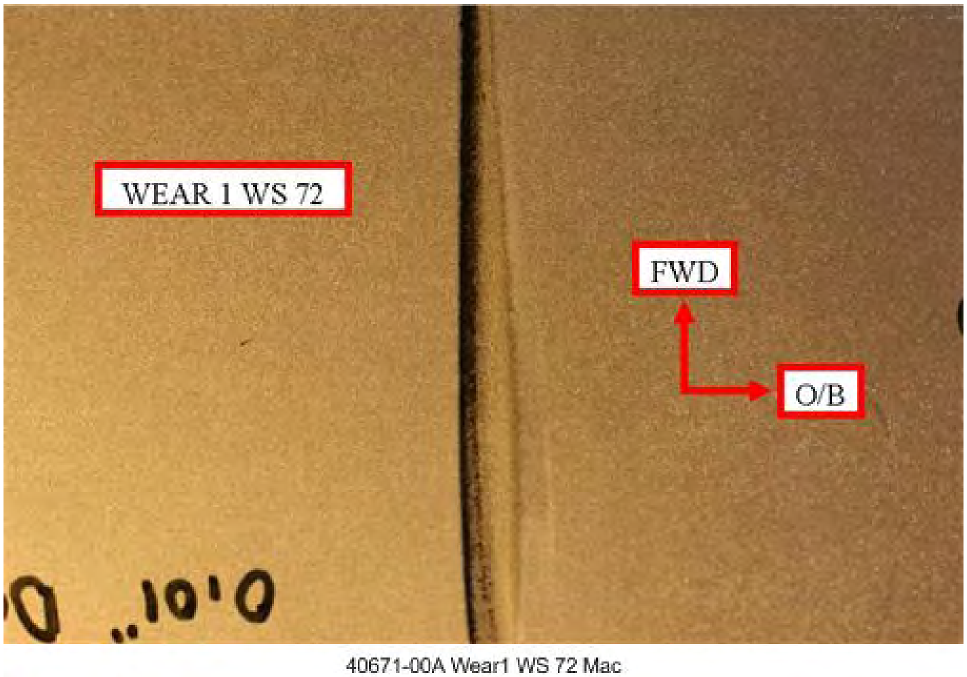


Figure 196. Macroscopic View of Wear 1 on the Left Wing Nacelle Plate Assembly Firewall Shear WS 72

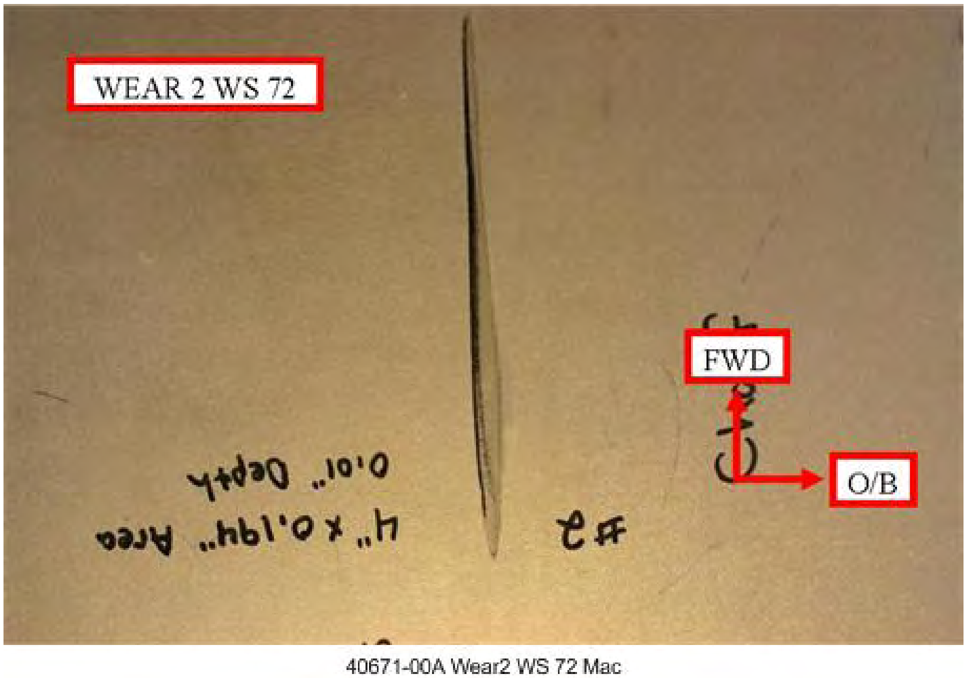
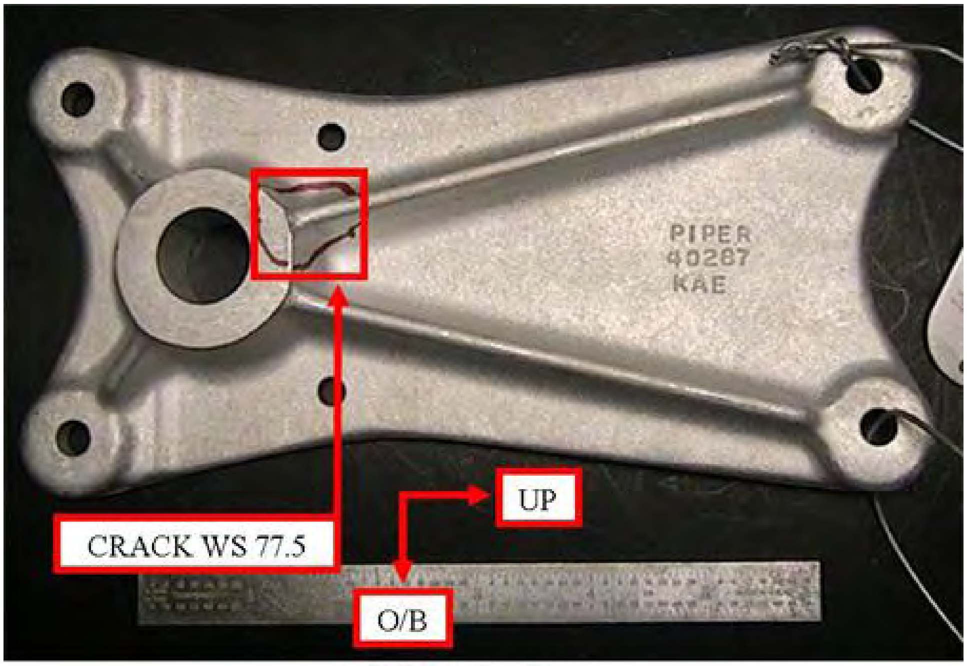


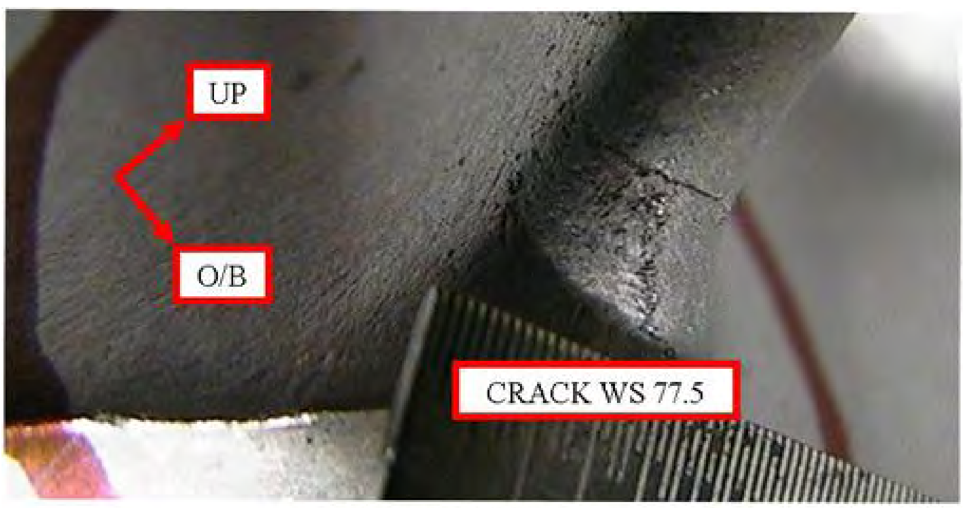
Figure 197. Macroscopic View of Wear 2 on the Left Wing Nacelle Plate Assembly Firewall Shear WS 72

The location of a crack on the left wing trunnion landing gear aft, part number 40288-00A, is shown in figure 198. A macroscopic view of this crack, which measures 0.175 inches and occurred at WS 77.5, is shown in figure 199, and the fluorescent liquid penetrant indication is shown in figure 200. Figure 201 shows the location of a crack on the left wing nacelle air filter bracket, part number 44397-00, which measured 0.496 inch in length and was determined to be caused by fatigue. A macroscopic view of this crack is shown in figure 202, and the fracture face is shown in figure 203. Figure 204 shows the location of wear on the left wing plate assembly main spar rib attachment, part number 40456-02A, at WS 87. This area of wear, shown macroscopically in figure 205, covered a surface area of 2.1 square inches and resulted in a maximum reduction in localized thickness of 66%.



40288-00A CRK WS 77.5 OV

Figure 198. Location of Crack on the Left Wing Trunnion Landing Gear Aft WS 77.5



40288-00A CRK WS 77.5 Mac

Figure 199. Macroscopic View of Crack on the Left Wing Trunnion Landing Gear Aft WS 77.5

Figure 200. Fluorescent Liquid Penetrant Indication of Crack on the Left Wing Trunnion Landing Gear Aft WS 77.5

Figure 201. Location of Left Wing Nacelle Air Filter Brack WS 85

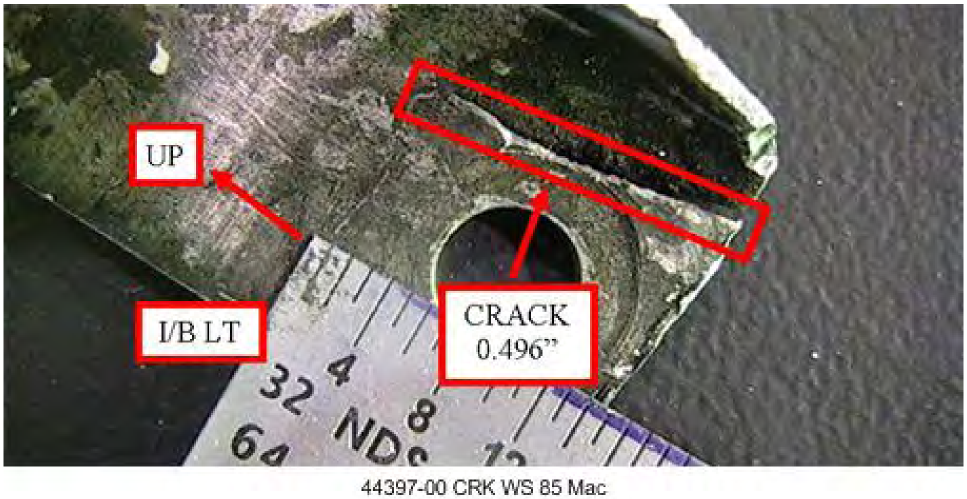
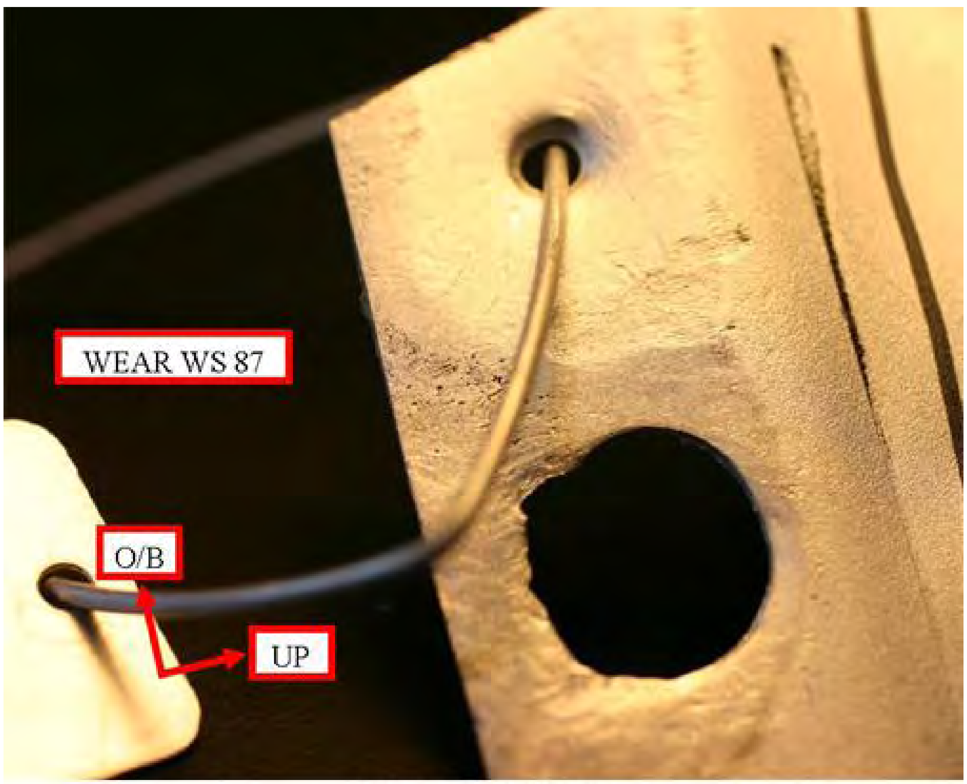


Figure 202. Macroscopic View of Crack on the Left Wing Nacelle Air Filter Bracket WS 85

Figure 203. Fracture Face of Crack on the Left Wing Nacelle Air Filter Bracket WS 85

40456-02A Wear WS 87 OV

Figure 204. Location of Wear on the Left Wing Plate Assembly Main Spar Rib Attachment WS 87



40456-02A Wear WS 87 Mac

Figure 205. Macroscopic View of Wear on the Left Wing Plate Assembly Main Spar Rib Attachment WS 87

Figure 206 shows the location of a 1.04-inch crack on the left wing main spar rib attachment wing panel assembly (1 of 2), part number 40456-02, at WS 87.5. A macroscopic view of this crack is shown in figure 207. The fracture face of this crack was examined; however, due to extensive crack face smearing, shown in figure 208, the failure mode could not be determined. A similar crack, measuring 0.904 inch, was identified on the left wing main spar rib attachment wing panel assembly (2 of 2), part number 40456-02, at WS 87.5. The location of this crack is shown in figure 209, and a macroscopic view of the crack is shown in figure 210. As with the previous crack, extensive crack face smearing prevented identification of the failure mode. The location of cracks in the left wing flap track, part number 45809-10C, is shown in figure 211. A macroscopic view of these cracks, located at WS 87.5 and measuring 0.044 inch in length, is shown in figure 212. Figure 213 shows of the fluorescent liquid penetrant indication of these cracks. A microscopic view of the cracks on the flap track is shown in figure 214.

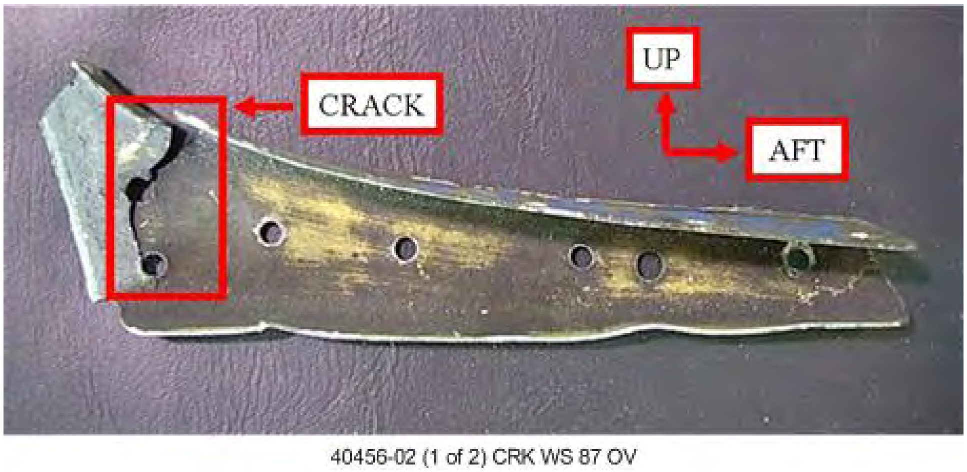
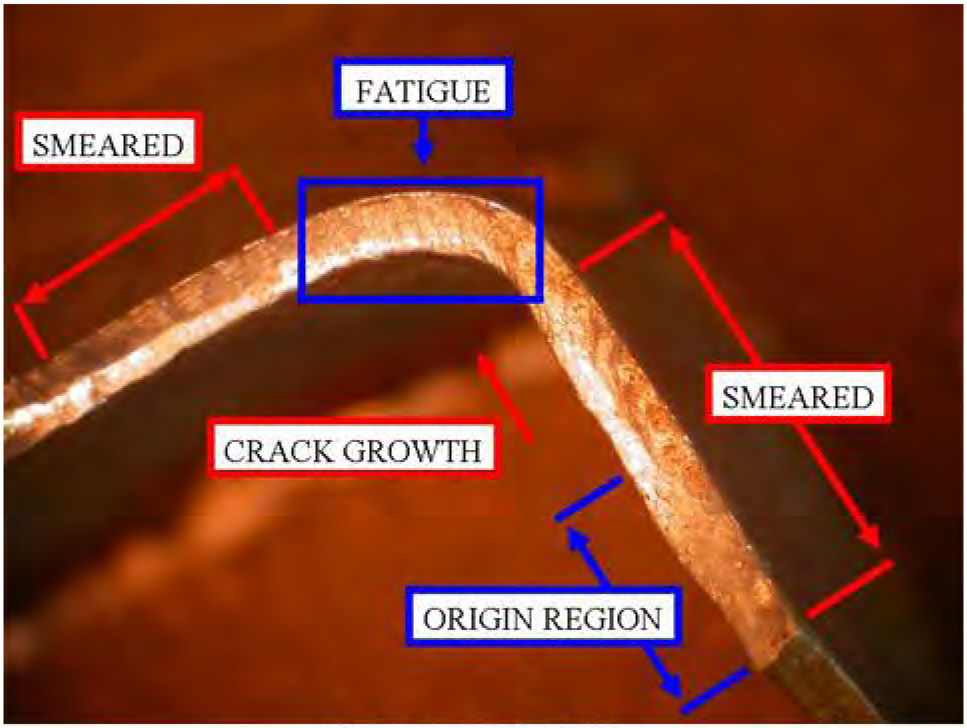


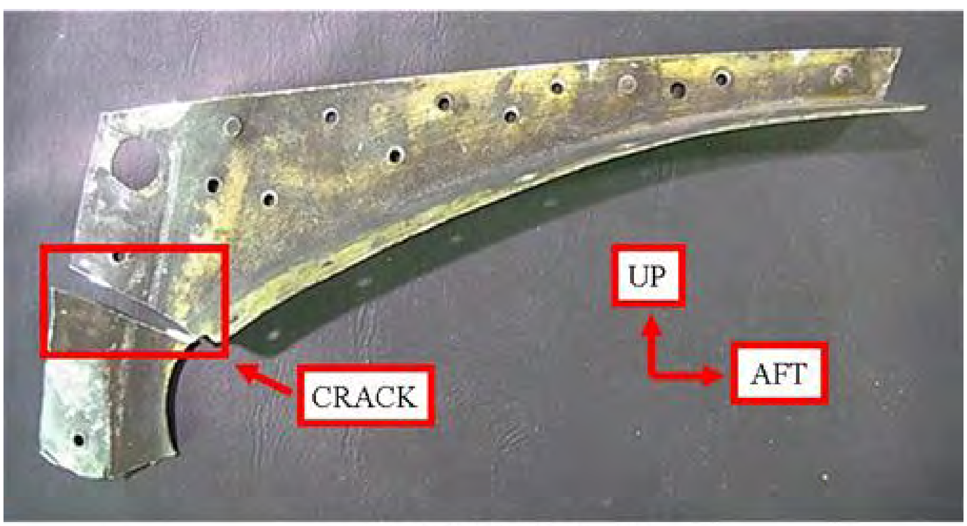
Figure 206. Location of Crack on the Left Wing Main Spar Rib Attachment Wing Panel Assembly (1 of 2) WS 87.5

Figure 207. Macroscopic View of Crack on the Left Wing Main Spar Rib Attachment Wing Panel Assembly (1 of 2) WS 87.5



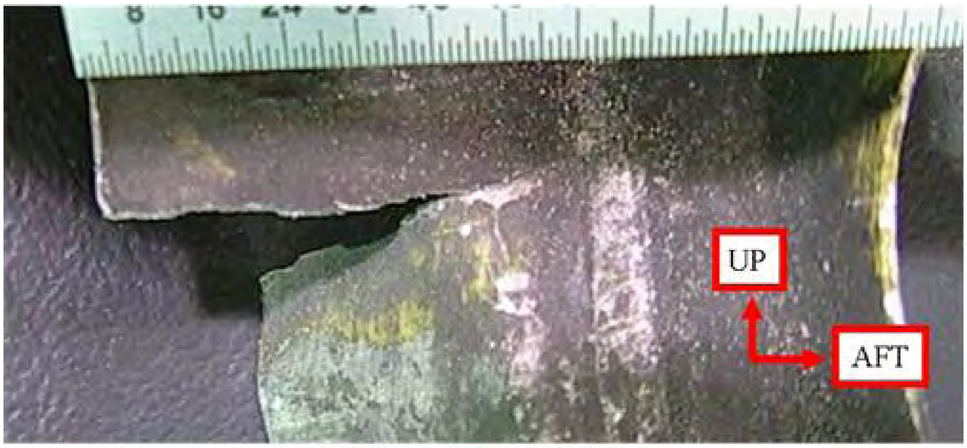
40456-02 (1 of 2) CRK WS 87 Frac

Figure 208. Fracture Face of Crack on the Left Wing Main Spar Rib Attachment Wing Panel Assembly (1 of 2) WS 87.5



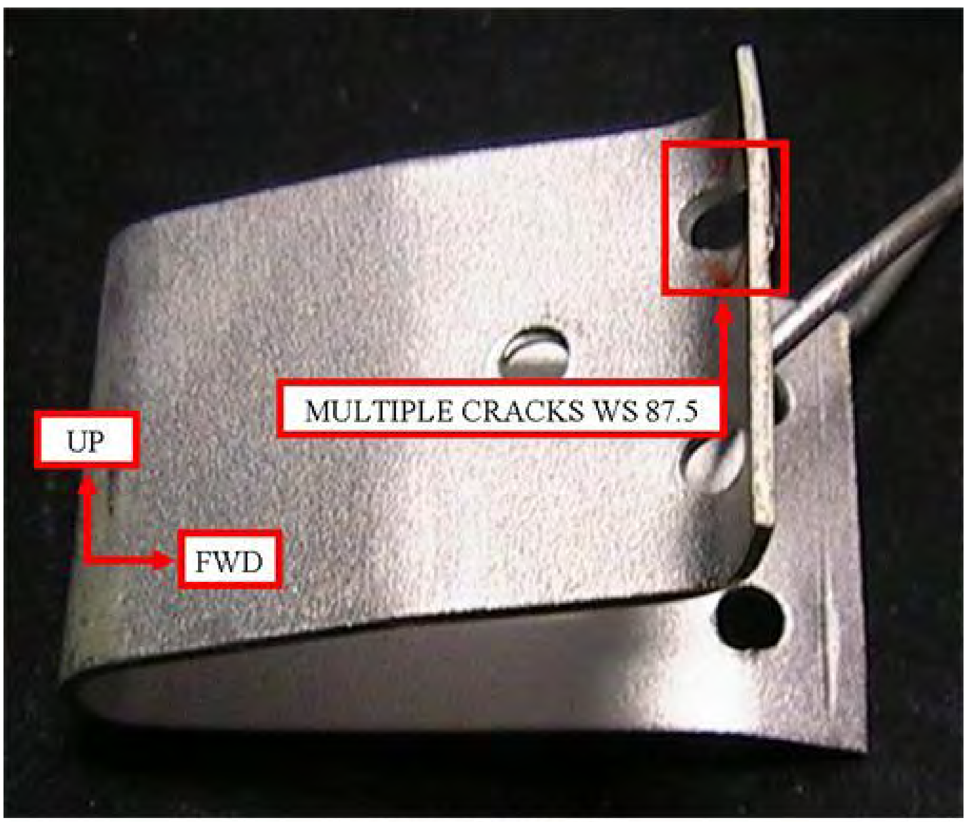
40456-02 (2 of 2) CRK WS 87.5 OV

Figure 209. Location of Crack on the Left Wing Main Spar Rib Attachment Wing Panel Assembly (2 of 2) WS 87.5



40456-02 (2 of 2) CRK WS 87.5 Mac

Figure 210. Macroscopic View of Crack on the Left Wing Main Spar Rib Attachment Wing Panel Assembly (2 of 2) WS 87.5



45809-10C CRK WS87.5 OV

Figure 211. Location of Cracks on the Left Wing Flap Track WS 87.5

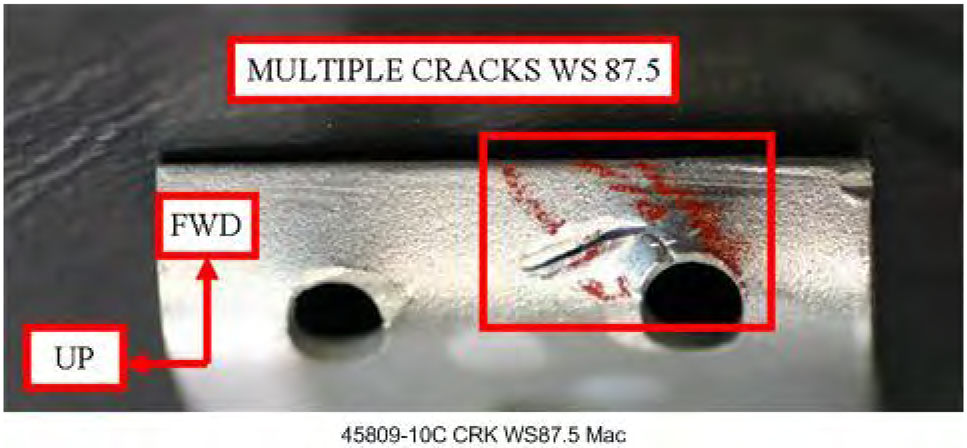


Figure 212. Macroscopic View of Cracks on the Left Wing Flap Track WS 87.5

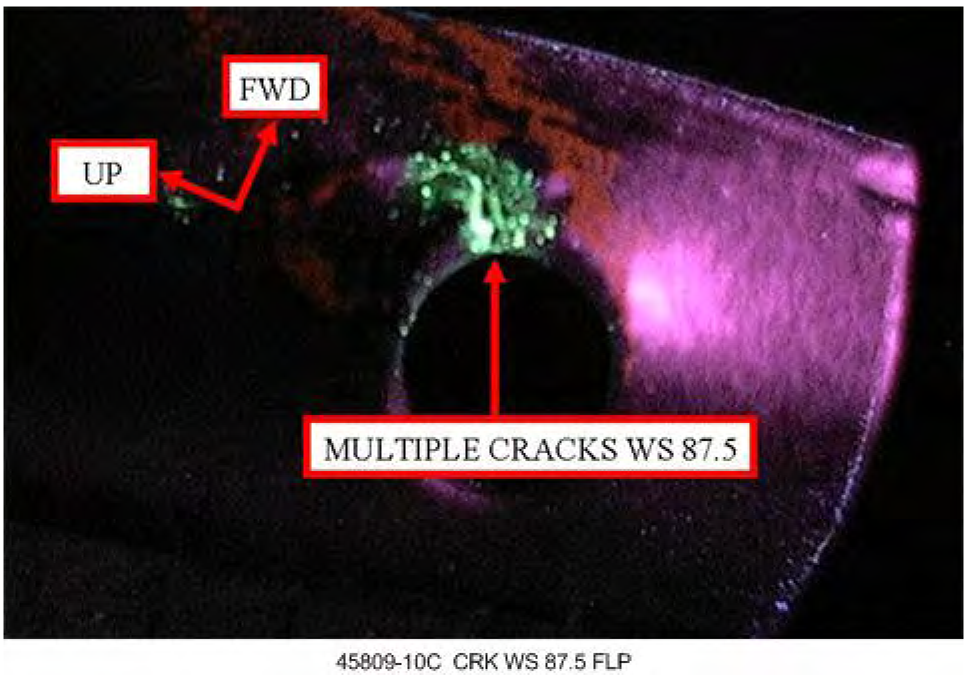
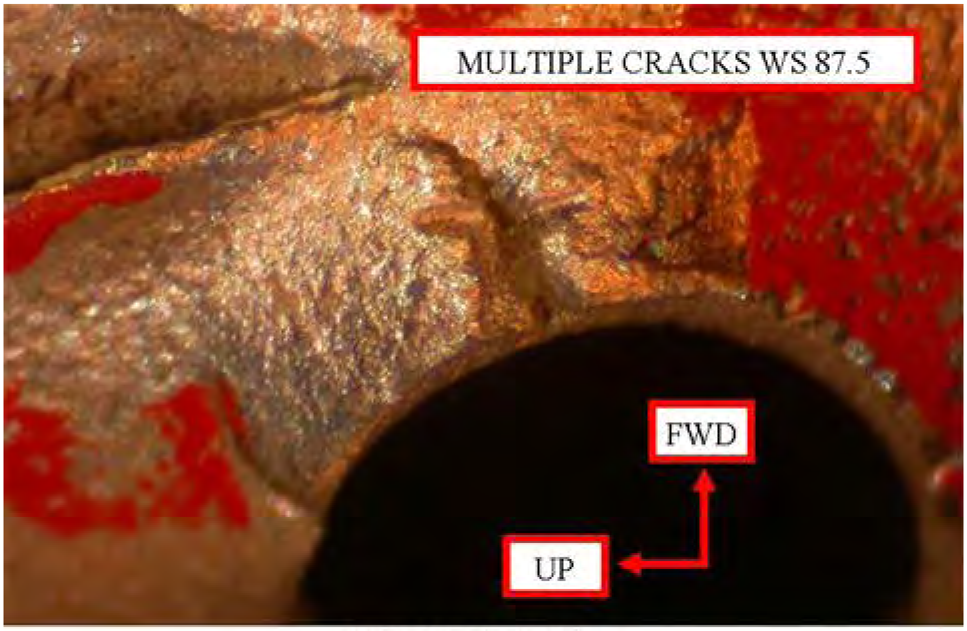


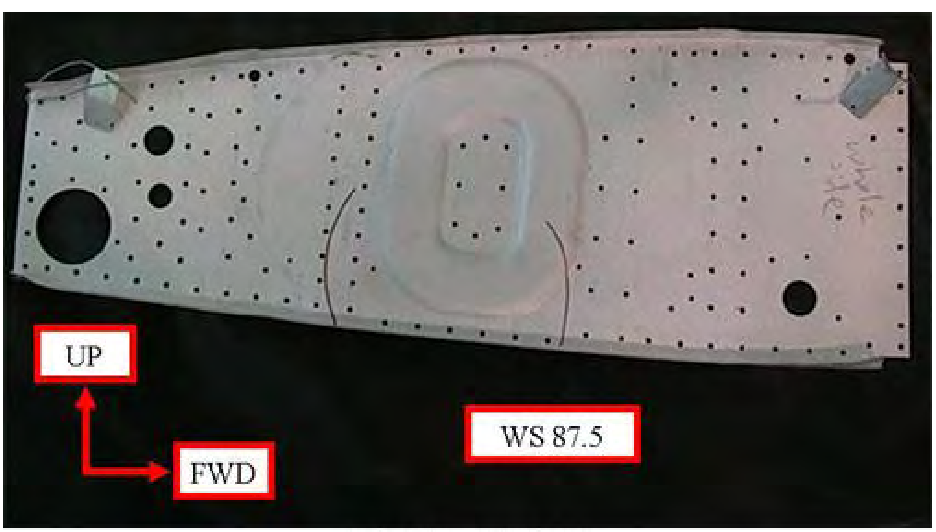
Figure 213. Fluorescent Liquid Penetrant Indication of Cracks on the Left Wing Flap Track WS 87.5



45809-10C CRK WS87.5 Mic

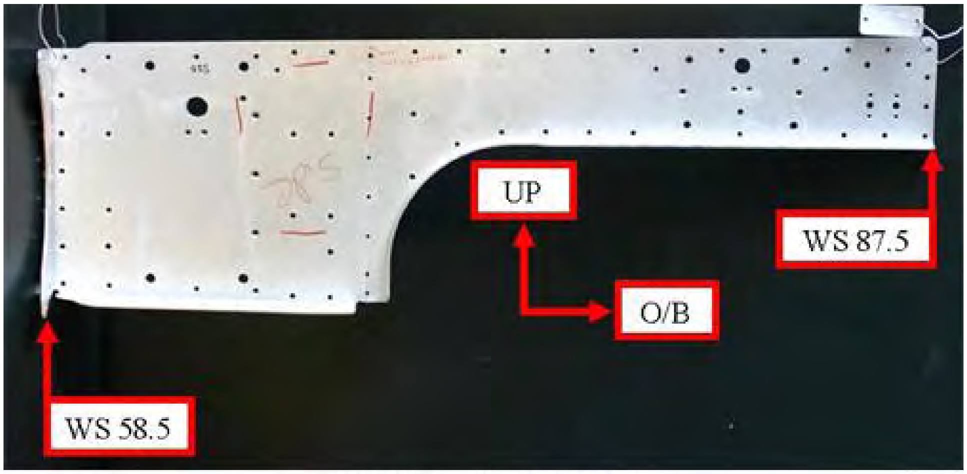
Figure 214. Microscopic View of Cracks on the Left Wing Flap Track WS 87.5

Figure 215 shows the location of light scattered corrosion on the left wing rear spar rib assembly, part number 40455-20. This light corrosion occurred at WS 87.5 and caused only a 2% localized thickness loss. Light scattered corrosion, causing a maximum thickness loss of 1%, was also observed on the left wing rear spar bulkhead assembly, part number 45501-00E. This area of corrosion, located at WS 87.5 FS 161, is shown in figure 216. Figure 217 also shows light scattered corrosion, resulting in a localized reduction in thickness of 2%, on the left wing rear spar rib assembly, part number 40455-22. This scattered corrosion was located at WS 87.5.



40455-20 Corr WS 87.5 OV

Figure 215. Location of Light Corrosion on the Left Wing Rear Spar Rib Assembly WS 87.5



45501-00E Corr WS 58.5- 87.5 OV

Figure 216. Overview of Left Wing Rear Spar Bulkhead Assembly WS 87.5 FS 161

40455-22 Corr WS 87.5 OV

Figure 217. Overview of Left Wing Rear Spar Rib Assembly WS 87.5

The location of two cracks and light-moderate corrosion on the left nacelle bulkhead longitudinal outboard, part number 44438-06A, is shown in figure 218. The light-moderate corrosion was scattered across the inboard side of the part and resulted in a maximum localized reduction in thickness of 3%. Figure 219 shows a macroscopic view of the 0.13-inch crack located at FS 184, and a microscopic view of the crack is shown in figure 220. A macroscopic view of the 0.60-inch crack, located at FS 185.5, is shown in figure 221, and a microscopic view is shown in figure 222. Figure 223 shows the location of cracks and light-moderate scattered corrosion on the left nacelle bulkhead longitudinal outboard, part number 44438-06B, which spans from WS 185.5 to FS 211.5. Light-moderate scattered corrosion was observed across the entire surface of this part, which resulted in a maximum localized reduction in thickness of 4%. A macroscopic view of multiple cracks emanating from a fastener hole at FS 207.5 is shown in figure 224, and a microscopic view of these cracks is shown in figure 225. The maximum length of these cracks was 0.195 inch.

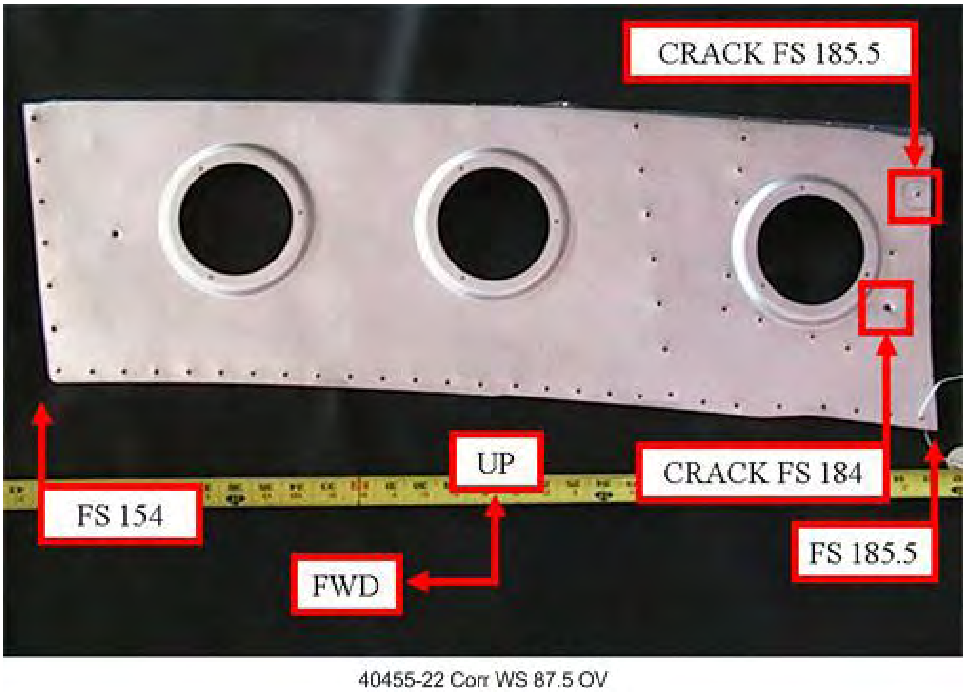


Figure 218. Location of Cracks and Overview of Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 154 Through FS 185.5

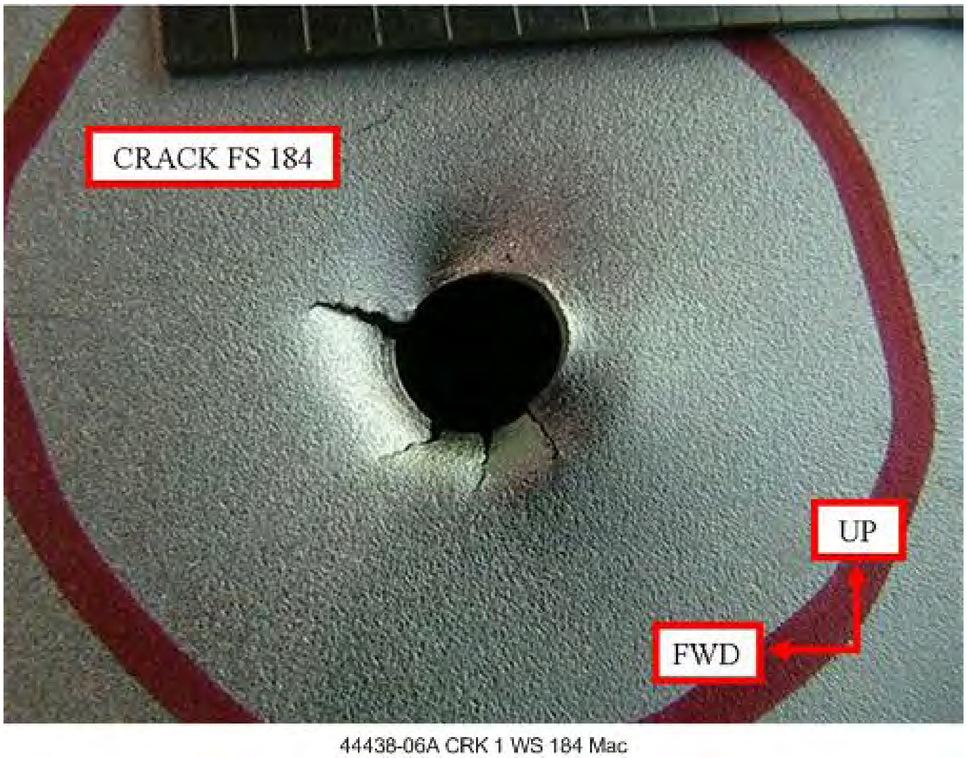


Figure 219. Macroscopic View of Crack on the Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 185.5

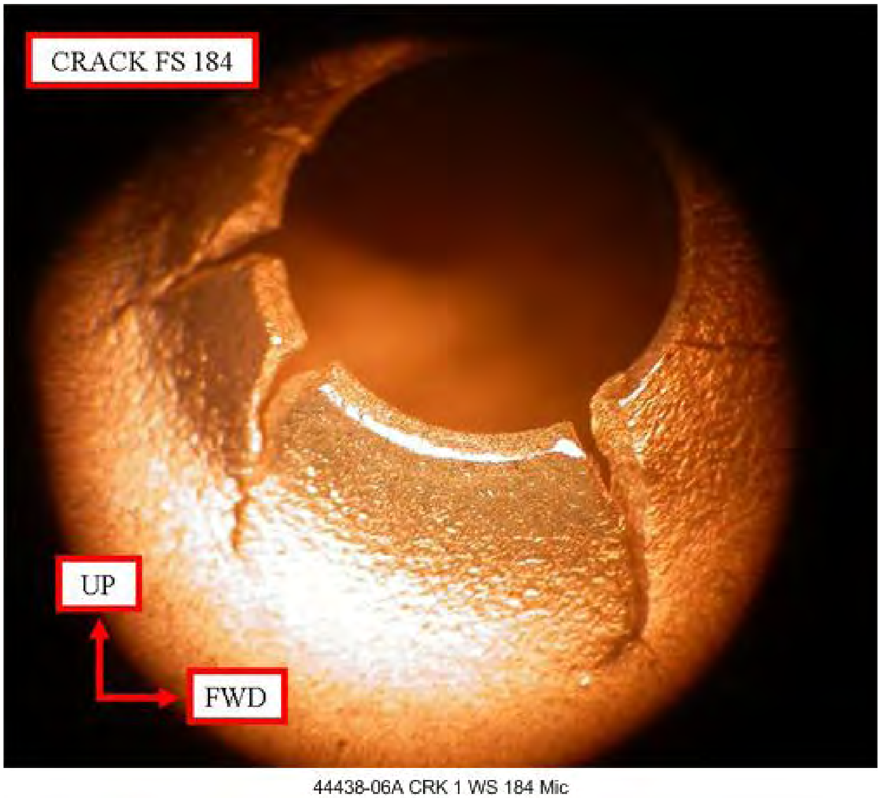


Figure 220. Microscopic View of Crack on the Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 184

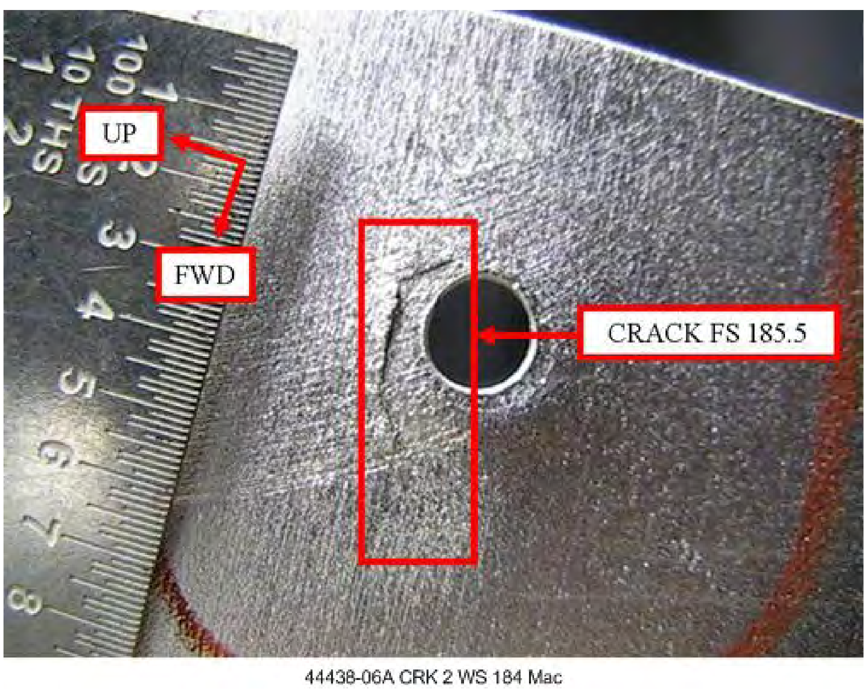
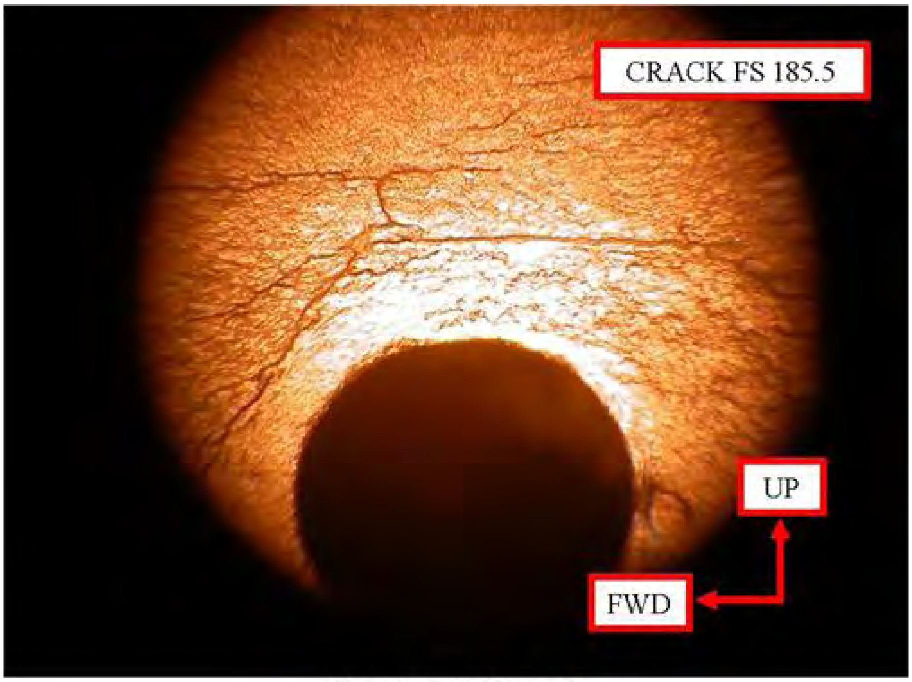
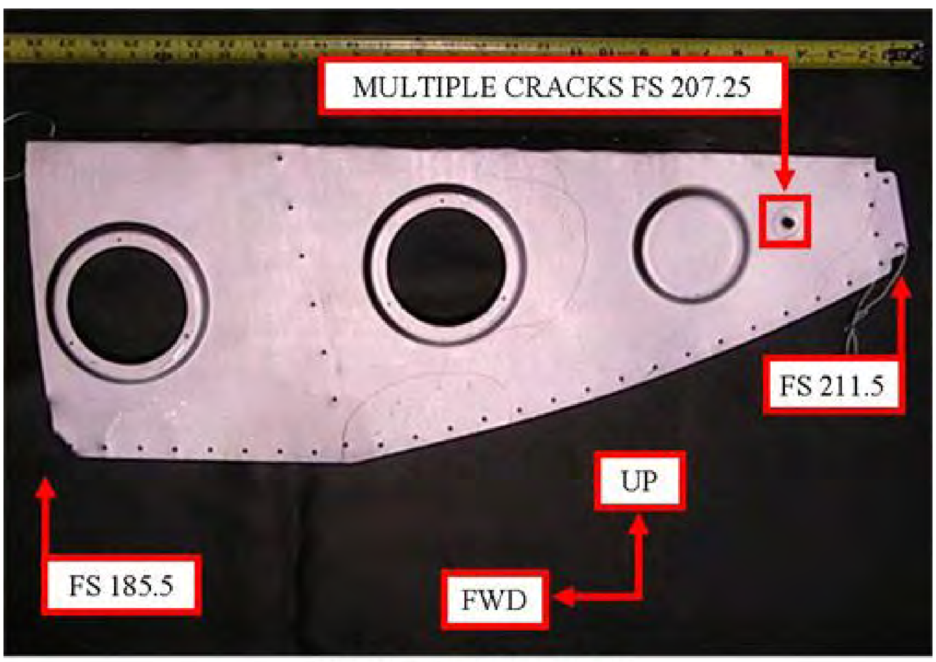


Figure 221. Macroscopic View of Crack on the Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 185.5



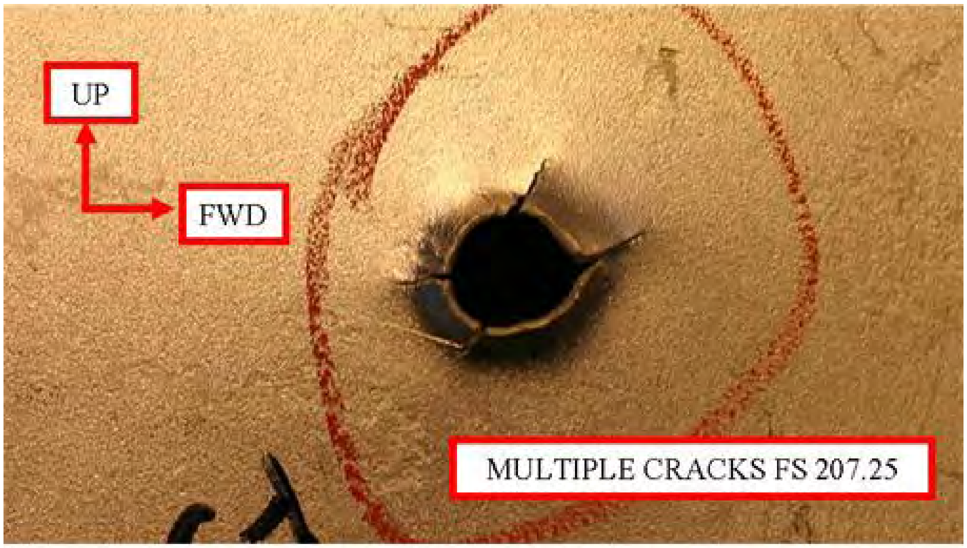
44438-06A CRK 2 WS 184 Mic

Figure 222. Microscopic View of Crack on the Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 185.5



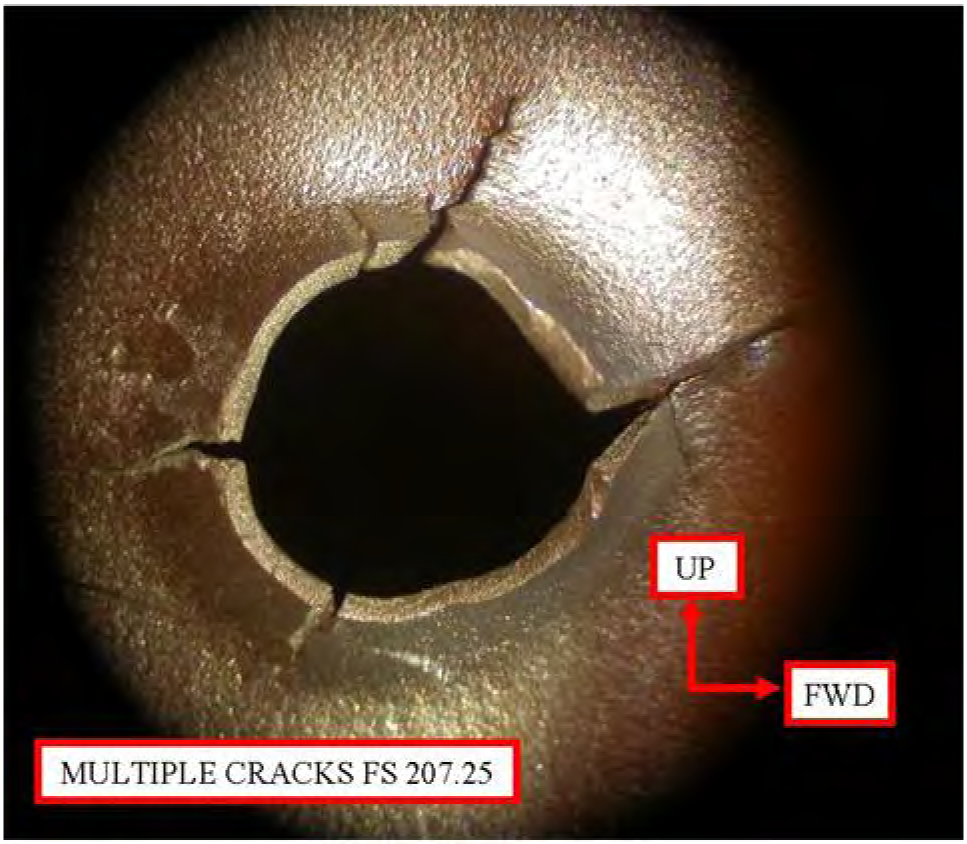
44438-06B CRK FS 185.5-211.5 OV

Figure 223. Location of Cracks and Overview Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 185.5 Through FS 211.5



44438-06B CRK FS 185.5-211.5 Mac

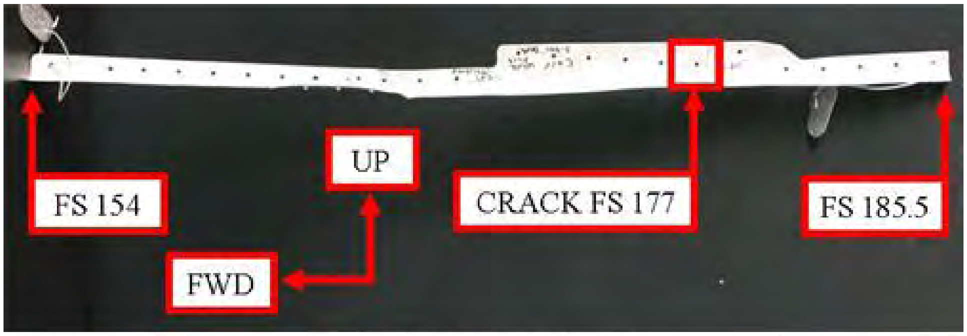
Figure 224. Macroscopic View of Cracks on the Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 207.25



44438-06B CRK FS 185.5-211.5 Mic

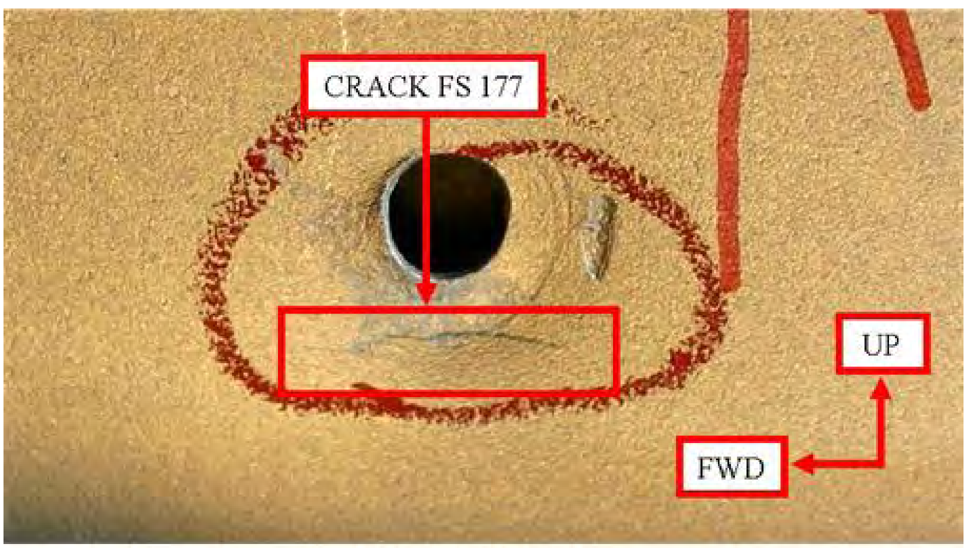
Figure 225. Microscopic View of Cracks on the Left Nacelle Bulkhead Longitudinal Outboard WS 89 FS 207.25

Figure 226 shows an overview of the location of a crack and light-moderate corrosion on the left wing nacelle outboard angle longitudinal bulkhead, part number 44438-10A. The light-moderate corrosion was scattered over the entire part and resulted in a maximum localized thickness loss of 4%. The macroscopic view of the 0.211-inch crack, located at WS 89 FS 177, is shown in figure 227, and a microscopic view of this crack is shown in figure 228.



44438-10A CRK FS 154-185.5 OV

Figure 226. Location of Crack and Overview of Light-Moderate Corrosion on the Left Wing Nacelle Outboard Angle—Longitudinal Bulkhead WS 89 FS 154 Through FS 185.5



44438-10A CRK FS 177 Mac

Figure 227. Macroscopic View of Crack on the Left Wing Nacelle Outboard Angle—Longitudinal Bulkhead WS 89 FS 177

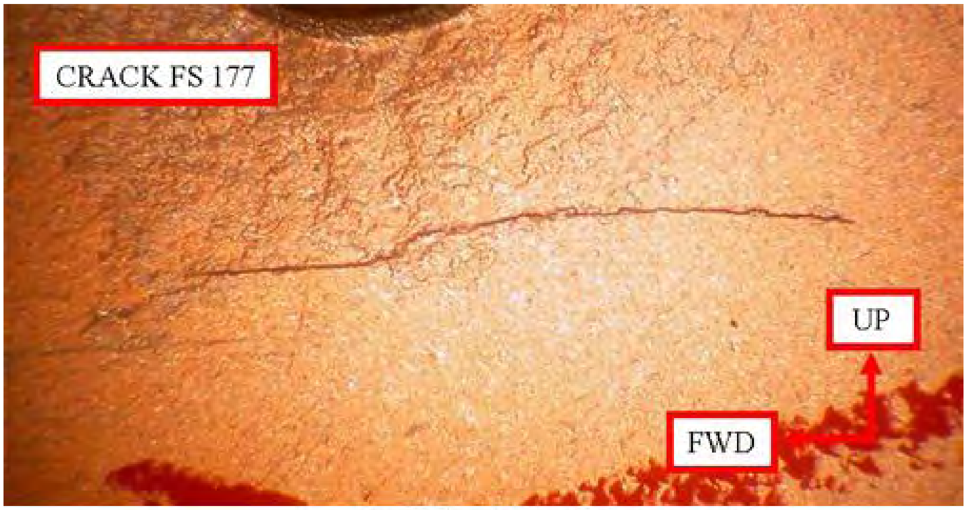


Figure 228. Microscopic View of Crack on the Left Wing Nacelle Outboard Angle—Longitudinal Bulkhead WS 89 FS 177

Figure 229 shows the location of light scattered corrosion on the left wing engine nacelle bulkhead assembly, part number 44437-04, at WS 89.31. This corrosion resulted in a maximum thickness loss of 2%. The location of light corrosion, resulting in a localized thickness loss of 1.5%, on the left wing aileron bracket front left, part number 40109-01, is shown in figure 230. The corrosion covered a surface area of 0.3 square inch at WS 174.5.

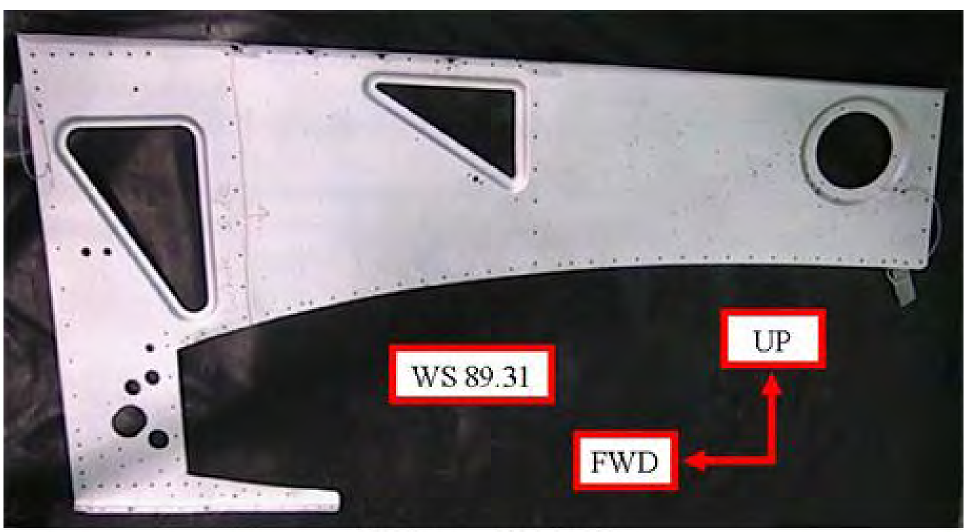
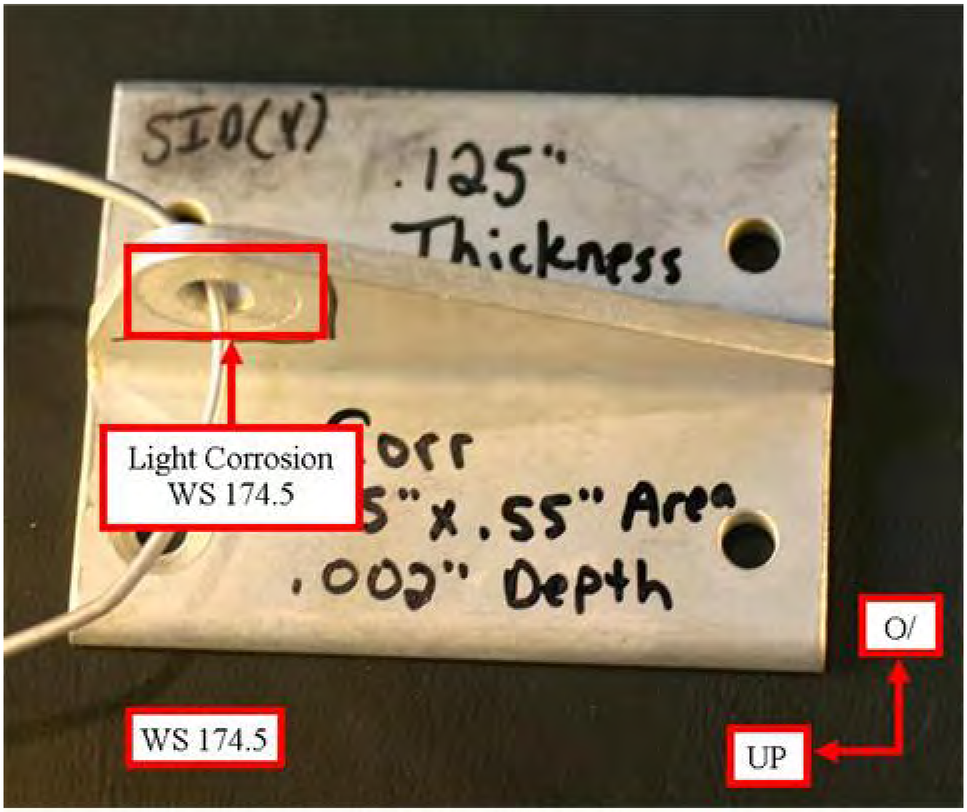


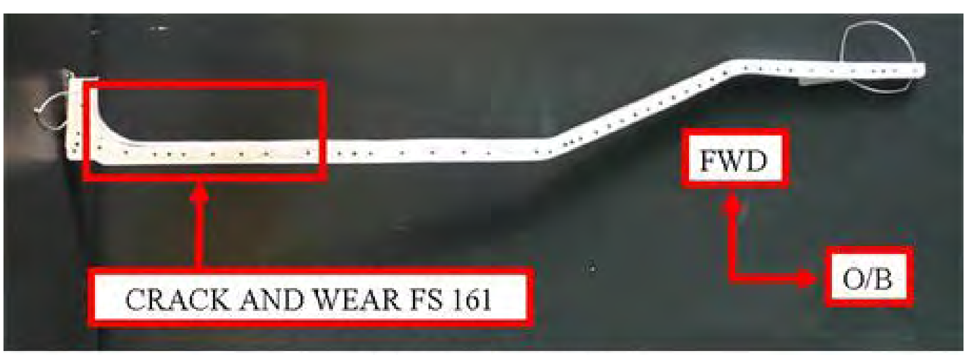
Figure 229. Location of Light Corrosion on the Left Wing Engine Nacelle Bulkhead Assembly WS 89.31



40109-01 Corr WS 174.5 OV

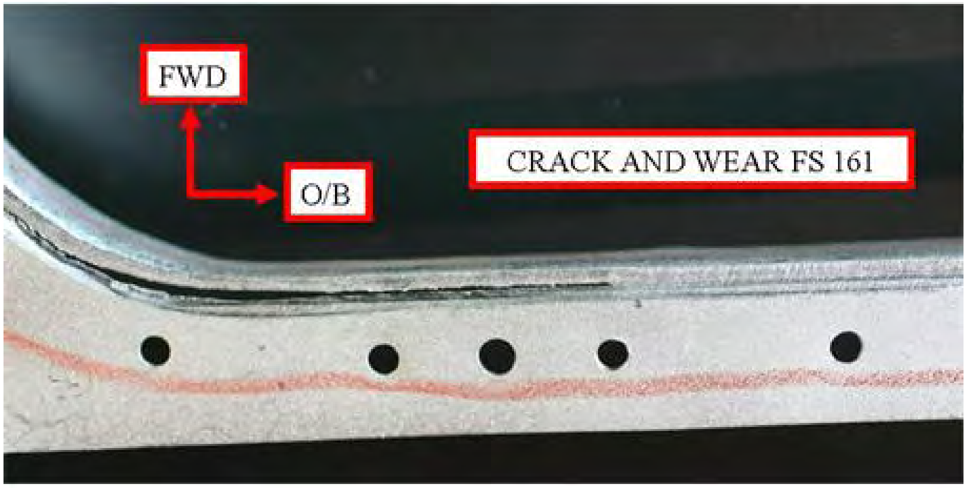
Figure 230. Location of Light Corrosion on the Left Wing Aileron Bracket Front Left WS 174.5

Figure 231 shows the location of a crack and an area of wear on the left wing rear spar lower aft angle, part number 45838-00, at FS 161. A macroscopic view of the 3.5-inch crack and 2.2-square-inch area of wear, resulting in areas of complete thickness loss, is shown in figure 232, and the fluorescent liquid penetrant indication is shown in figure 233. The location of a 4.75-square-inch area of wear on the left wing rear spar lower forward angle, part number 45824-00, is shown in figure 234. A macroscopic view of this area of wear, which caused a localized reduction in thickness of 13%, is shown in figure 235.



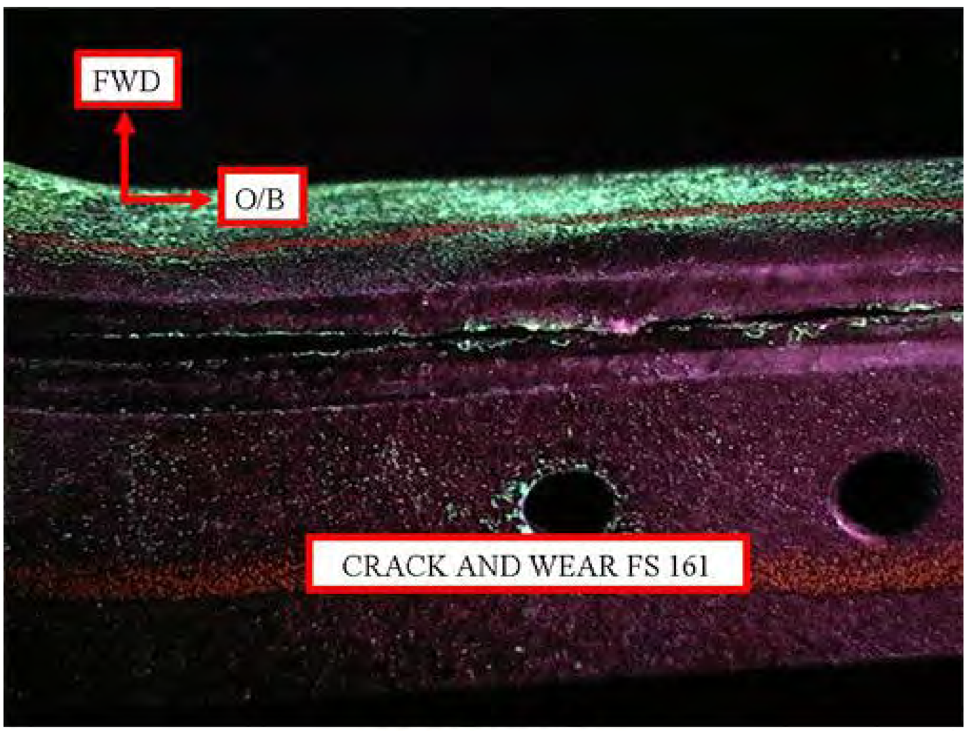
45838-00 CRK FS 161 OV

Figure 231. Location of Crack and Wear on the Left Wing Rear Spar Lower Aft Angle FS 161



45838-00 CRK FS 161 Mac

Figure 232. Macroscopic View of Crack and Wear on the Left Wing Rear Spar Lower Aft Angle FS 161



45838-00 CRK FS 161 FLP

Figure 233. Fluorescent Liquid Penetrant Indication of Crack and Wear on the Left Wing Rear Spar Lower Aft Angle FS 161

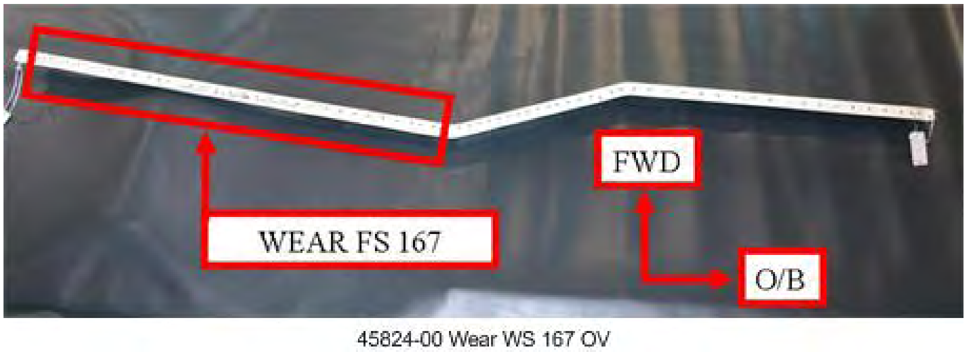


Figure 234. Location of Wear on the Left Wing Rear Spar Lower Forward Angle FS 167

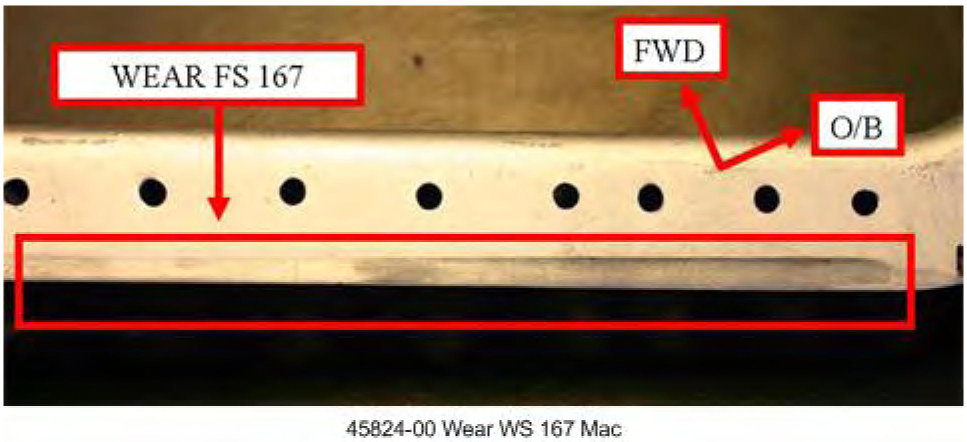


Figure 235. Macroscopic View of Wear on the Left Wing Rear Spar Lower Forward Angle FS 167

3.4.2.2 Right Wing.

Of the 186 parts inspected during the teardown evaluation phase of the right wing, 57 defects were noted and characterized. There were 21 cracks, 23 areas of corrosion, 11 instances of wear, and 2 areas of damage noted during the teardown evaluation of the right wing. No defects were identified on the right wing front or rear spars. Similar to the left wing, scattered light to light-moderate corrosion was observed frequently on the main spar cap assembly components, and numerous cracks were found on the bulkhead wing fillets. Areas of wear were observed on the right wing plate wing panel reinforcement assembly, rear spar assembly main gear forward, and the rear spar lower aft assembly. A large number of cracks and areas of corrosion were also observed on the right wing nacelle bulkhead components.

3.4.2.2.1 Right Wing Front Spar.

No defects were found during the microscopic examination of the right wing front spar.

3.4.2.2.2 Right Wing Main Spar.

Figure 236 shows the structural stackup of the right wing main spar upper and lower cap assemblies. Table 32 shows detailed characterization of each defect found on the right wing main spar. Thirteen areas of corrosion were identified on the right wing main spar, eight on the upper cap assembly, and five on the lower cap assembly. These areas of corrosion ranged in severity from light to moderate-severe. One area of wear was also noted on the right wing main spar lower aft angle.

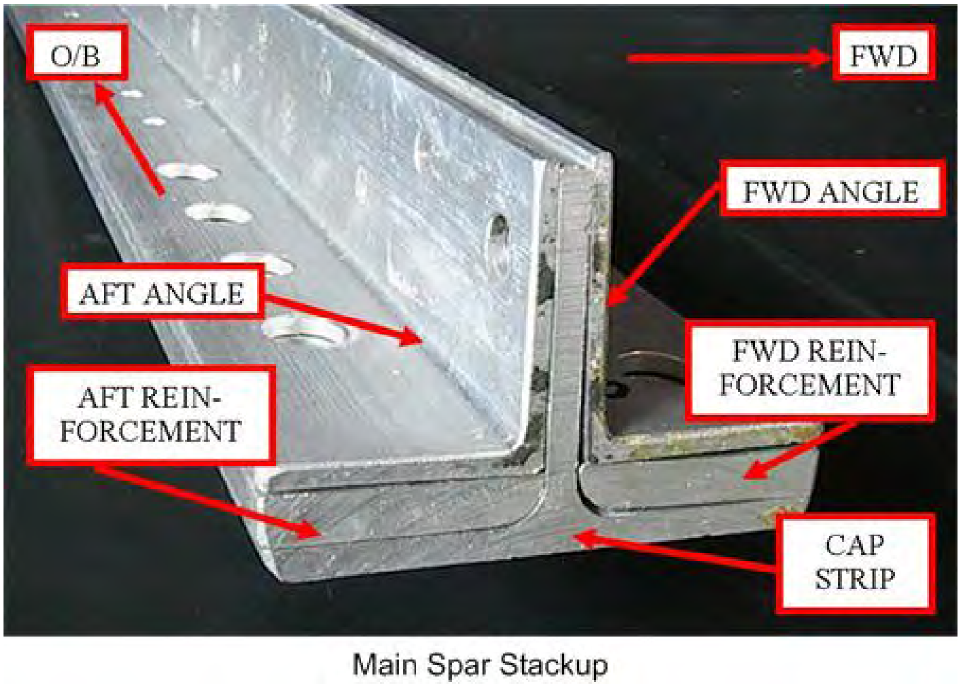


Figure 236. Structural Stackup of the Right Wing Main Spar

Table 32. Inspection Results From the Right Wing Main Spar

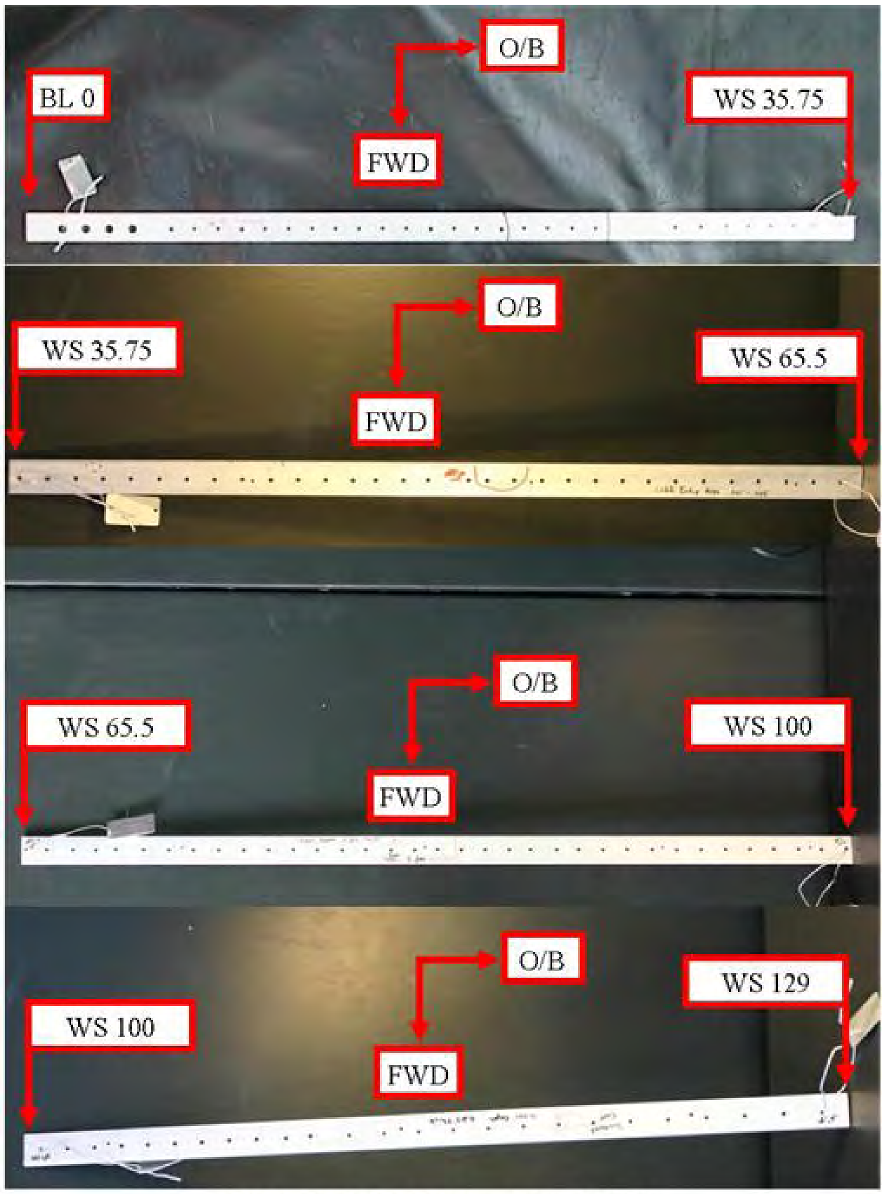
Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right wing main spar upper fwd reinforcement, figure 238	Corrosion	BL 0 through WS 129	Scattered over entire part	Corrosion indication	Light corrosion 1% thickness loss	103
Right wing main spar upper aft reinforcement, figure 239	Corrosion	WS 65.5 through WS 149.5	Scattered over entire part	Corrosion indication	Light corrosion less than 1% thickness loss	103
	Corrosion	BL 10.5 through BL 17	6.5 inches by 1.5 inches	Corrosion indication	Moderate corrosion 5.5% thickness loss	240

Table 32. Inspection Results From the Right Wing Main Spar (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
	Corrosion	BL 27.5 through BL 29.25	1.75 inches by 1 inch	Corrosion indication	Moderate corrosion 5.5% thickness loss	241
Right wing main spar upper fwd angle, figure 242	Corrosion	BL 0 through WS 65.5	Scattered over entire part	Corrosion indication	Light corrosion 1% thickness loss	103
Right wing main spar upper aft angle, figure 244	Corrosion	WS 28.25 through WS 29.75	1.5 inches by 1 inch	Corrosion indication	Moderate-severe corrosion 9% thickness loss	243 245
Right wing main spar upper cap strip, figure 246	Corrosion	WS 26.5 through WS 35.75	2.5 inches by 9 inches	Corrosion indication	Light-moderate corrosion 3.8% thickness loss	105
	Corrosion	WS 96.75 through WS 100	3.7 inches by 0.7 inch	Corrosion indication	Light corrosion less than 1% thickness loss	247
Right wing main spar lower cap strip, figure 248	Corrosion	BL 0 through WS 100	Scattered over entire part	Corrosion indication	Light-moderate corrosion 4.5% thickness loss	105
Right wing main spar lower fwd reinforcement, figure 249	Corrosion	BL 0 through WS 100	Scattered over entire part	Corrosion indication	Light corrosion less than 1% thickness loss	103
Right wing main spar lower aft reinforcement, figure 250	Corrosion	BL 0 through WS 65.5	Scattered over entire part	Corrosion indication	Light corrosion 1.4% thickness loss	103
Right wing main spar lower fwd angle, figure 151	Corrosion	BL 13.75 through BL 14.5	0.75 inch by 0.15 inch	Corrosion indication	Moderate-severe corrosion 7.3% thickness loss	252
Right wing main spar lower aft angle, figure 3-183	Corrosion	BL 0 through WS 65.5	Scattered over entire part	Corrosion indication	Light-moderate corrosion 4.4% thickness loss	3-36
	Wear	BL 0	0.78 inch by 0.68 inch	No indication	Wear 4.4% thickness loss	3-184

Fwd = Forward

Figure 237 shows an overview of light corrosion scattered over the entire right wing main spar upper forward reinforcement from BL 0 to WS 129. This corrosion caused a localized reduction in thickness of only 1%. The location of scattered light corrosion and two areas of moderate corrosion on the right wing main spar upper aft reinforcement are shown in figure 238. The scattered light corrosion covered the surface of the aft reinforcement from WS 65.5 to WS 149.5 and caused a maximum localized thickness loss of less than 1%. A macroscopic view of a 9.75-square-inch area of moderate corrosion, located from BL 10.5 to 17, is shown in figure 239. Figure 240 shows a macroscopic view of another moderate area of corrosion, covering a surface area of 1.75 square inches, on the aft reinforcement, located from BL 27.5 to BL 29.25. Both areas of corrosion caused a maximum localized reduction in thickness of 5.5%.



RW MN SP UP FWD REIN Corr BL 0 - WS 129 OV

Figure 237. Overview of Right Wing Main Spar Upper Forward Reinforcement BL 0 Through WS 129

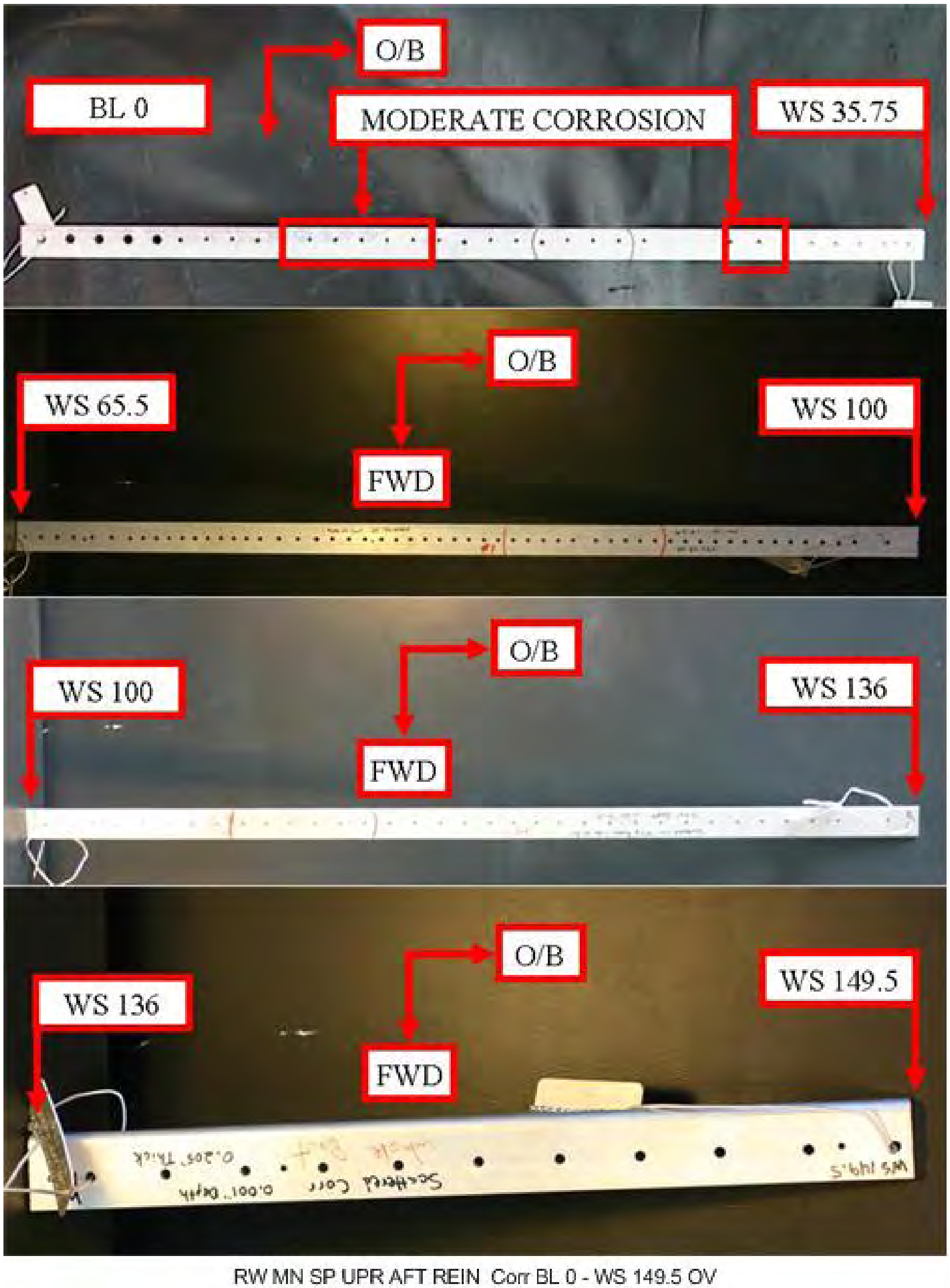
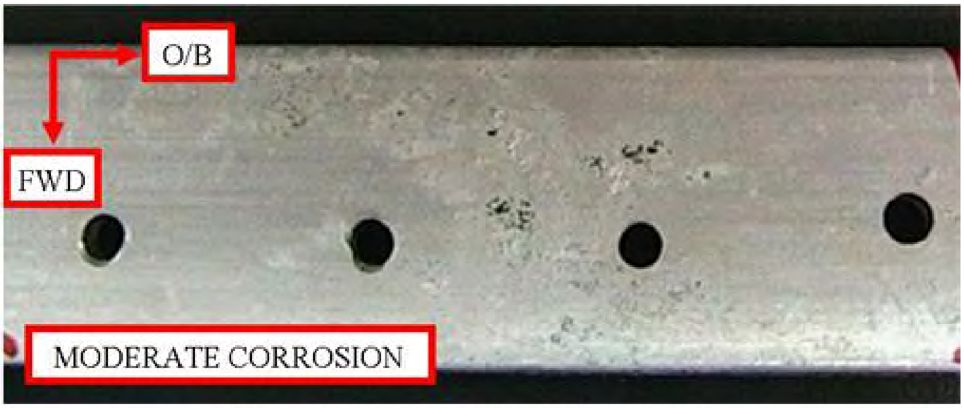
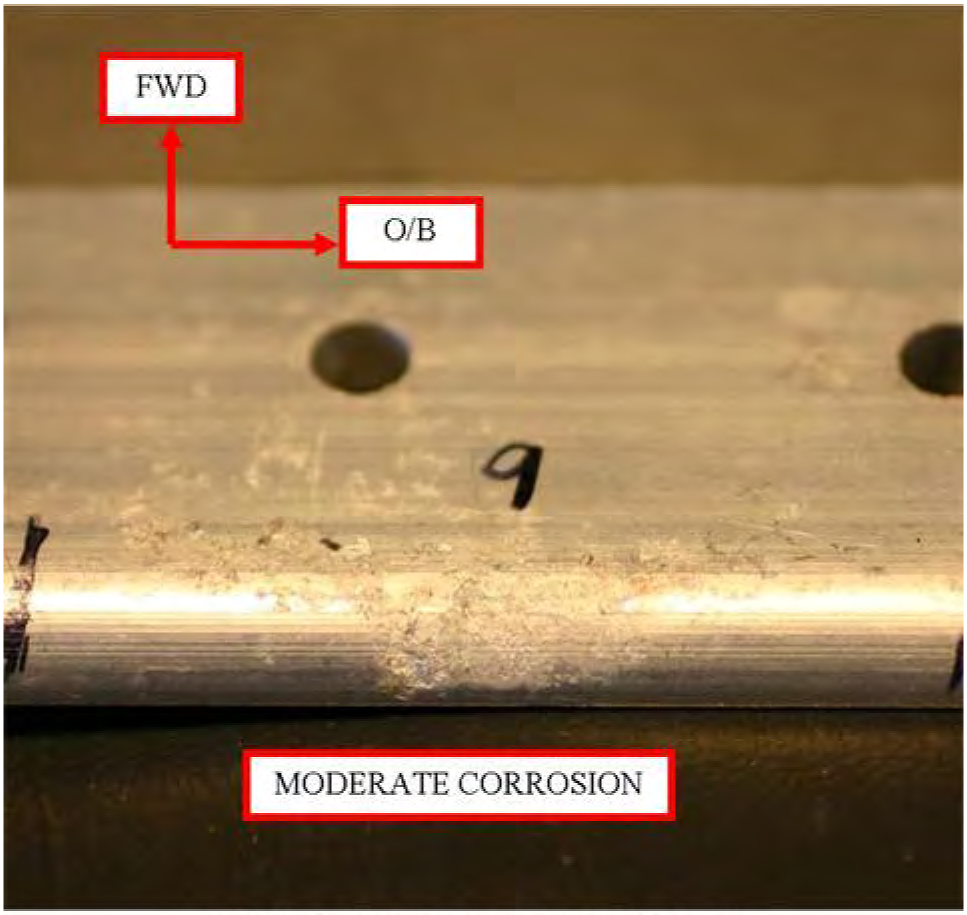


Figure 238. Location of Moderate Corrosion and Overview of Right Wing Main Spar Upper Aft Reinforcement BL 0 Through WS 149.5



RW MN SP UPR AFT REIN Corr BL 0 -WS 35.75 Mac

Figure 239. Macroscopic View of Moderate Corrosion on the Right Wing Main Spar Upper Aft Reinforcement BL 10.5 Through BL 17



RW MS Upper Aft Reinforcement Corr BL 27.5-29.25 Mac

Figure 240. Macroscopic View of Moderate Corrosion on the Right Wing Main Spar Upper Aft Reinforcement BL 27.5 Through BL 29.25

Figure 241 shows the location of light scattered corrosion on the right wing main spar upper forward angle from BL 0 to WS 65.5. This corrosion caused a localized thickness loss of 1%. The location of a 1.5-square-inch area of moderate-severe corrosion, located on the upper aft angle from WS 28.25 to WS 29.75, is shown in figure 242. A macroscopic view of the area of corrosion, which caused a maximum localized reduction in thickness of 9%, is shown in figure 243. The fluorescent liquid penetrant indication of this area of moderate-severe corrosion is shown in figure 244. The location of an area of light corrosion on the main spar upper cap strip is shown in figure 245. This 22.5-square-inch area of corrosion, located from WS 26.5 to WS 35.75, resulted in a maximum localized thickness loss of 3.8%. Figure 246 shows the location of a 2.59-square-inch area of light corrosion on the upper cap strip from WS 96.75 to WS 100, which caused a reduction in thickness of less than 1%.

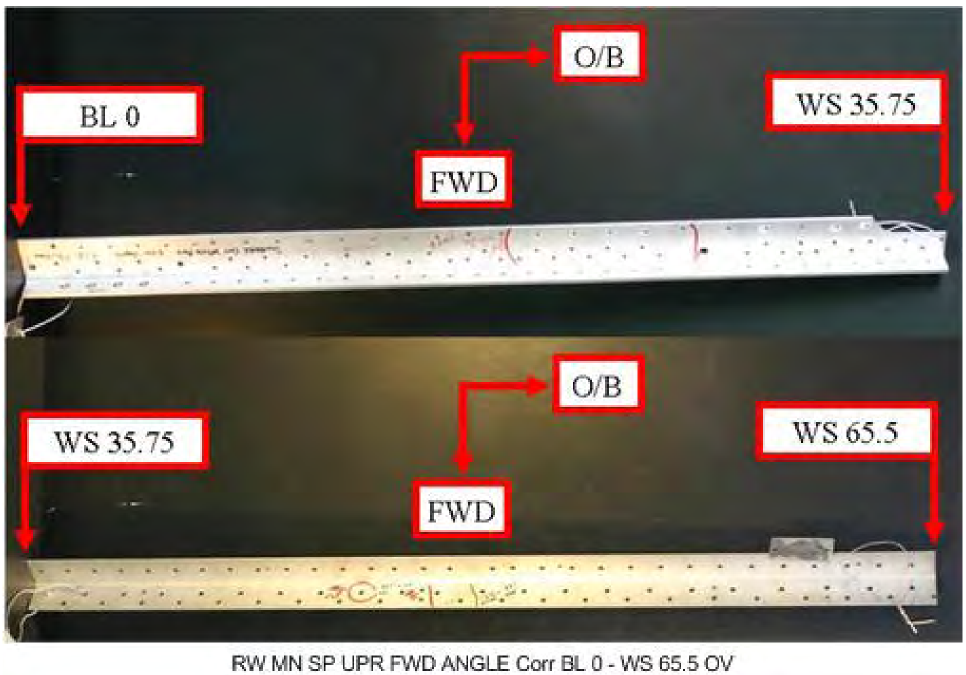


Figure 241. Location of Light Corrosion on the Right Wing Main Spar Upper Forward Angle BL 0 Through WS 65.5

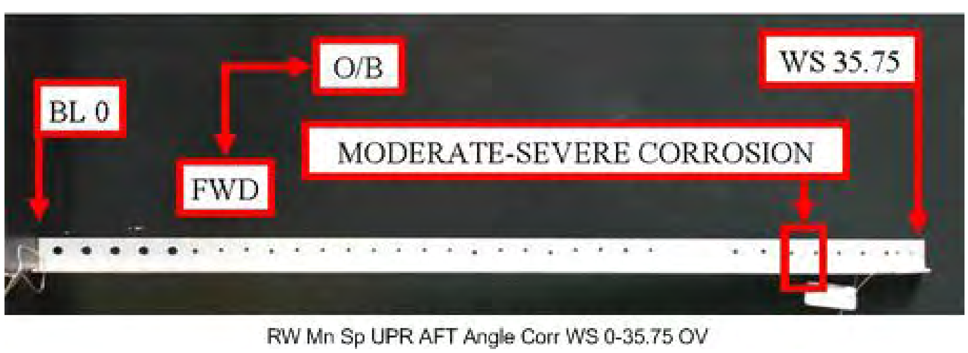


Figure 242. Location of Moderate-Severe Corrosion on the Right Wing Main Spar Upper Aft Angle WS 28.25 Through WS 29.75

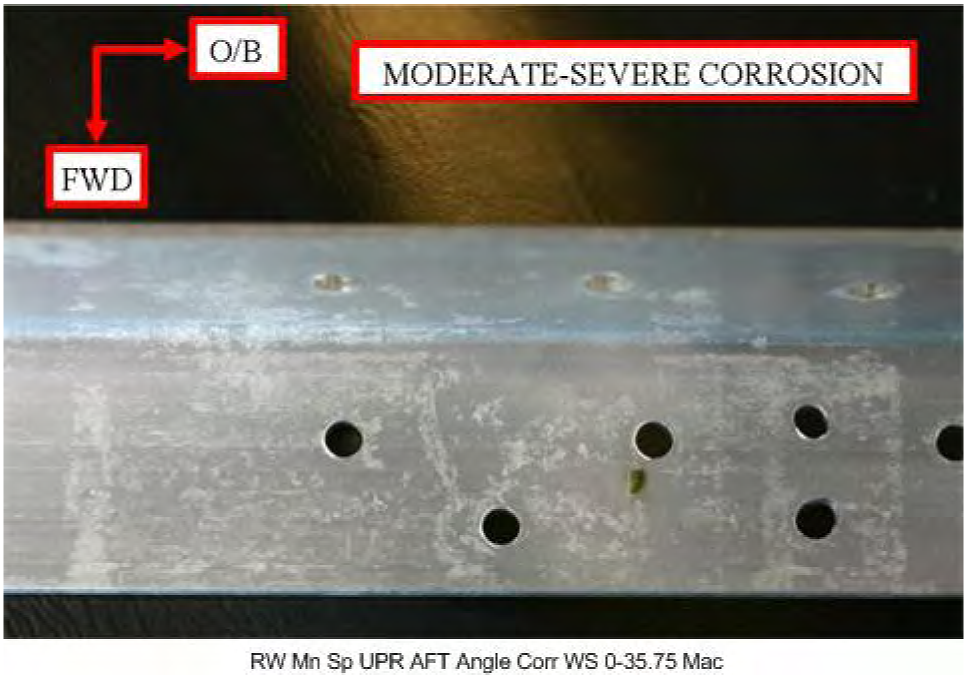


Figure 243. Macroscopic View of Moderate-Severe Corrosion on the Right Wing Main Spar Upper Aft Angle WS 28.25 Through WS 29.75

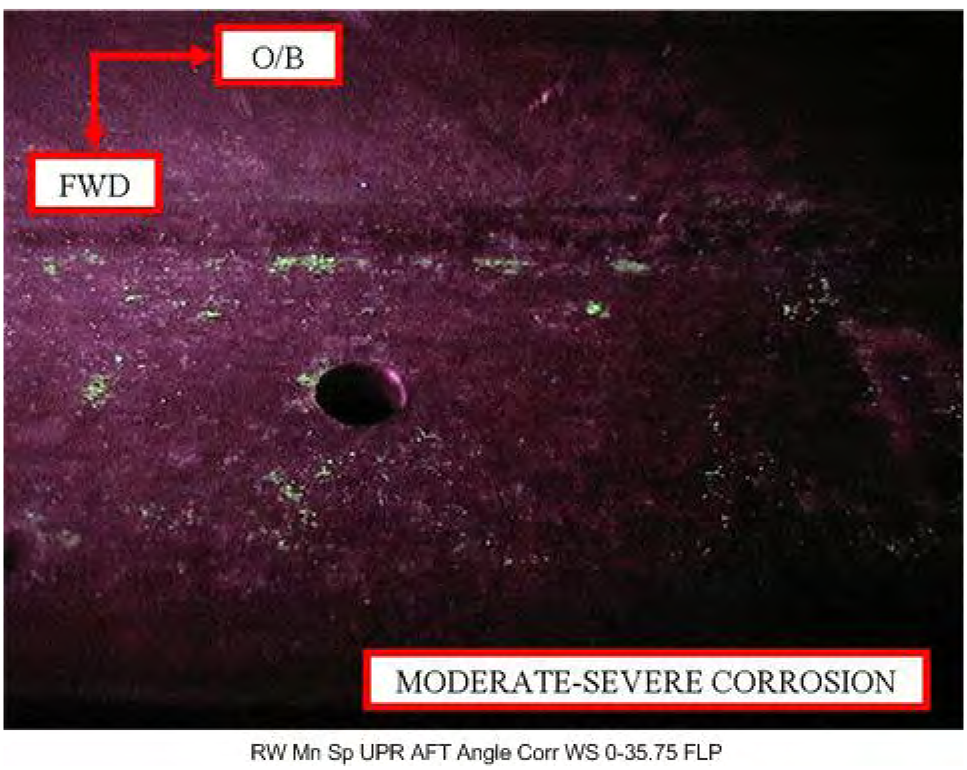
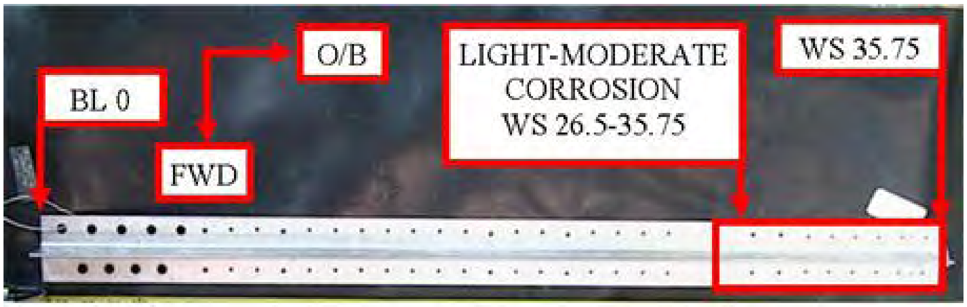
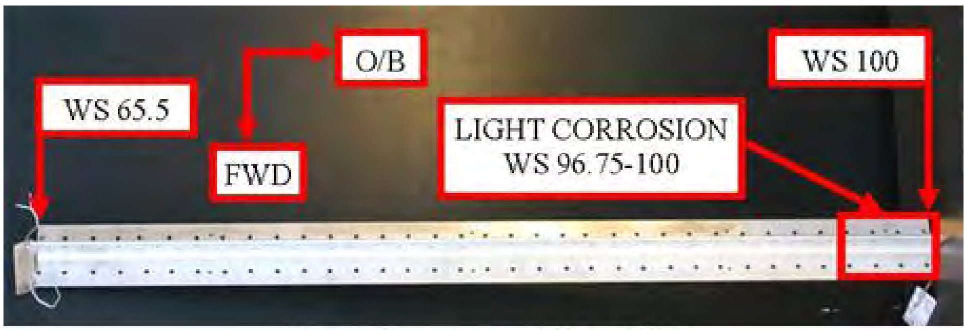


Figure 244. Fluorescent Liquid Penetrant View of Moderate-Severe Corrosion on the Right Wing Main Spar Upper Aft Angle WS 28.25 Through WS 29.75



RW Mn Sp UPR Cap Strip Corr WS 0-36 OV

Figure 245. Location of Light-Moderate Corrosion on the Right Wing Main Spar Upper Cap Strip WS 26.5 Through WS 35.75



RW MS SP Upper Cap Corr WS 65.5-100 OV

Figure 246. Location of Light Corrosion on the Right Wing Main Spar Upper Cap Strip WS 96.75 Through WS 100

The location of scattered light-moderate corrosion, located from BL 0 to WS 100, on the lower cap strip is shown in figure 247. This scattered corrosion caused a maximum localized thickness loss of 4.5%. Scattered light corrosion was also found on the lower forward reinforcement from BL 0 to WS 100, as shown in figure 248. This corrosion caused a maximum localized thickness loss of less than 1%. Scattered light corrosion, which caused a localized reduction in thickness of 1.4%, was also found on the lower aft reinforcement, scattered from BL 0 to WS 65.5, as shown in figure 249.

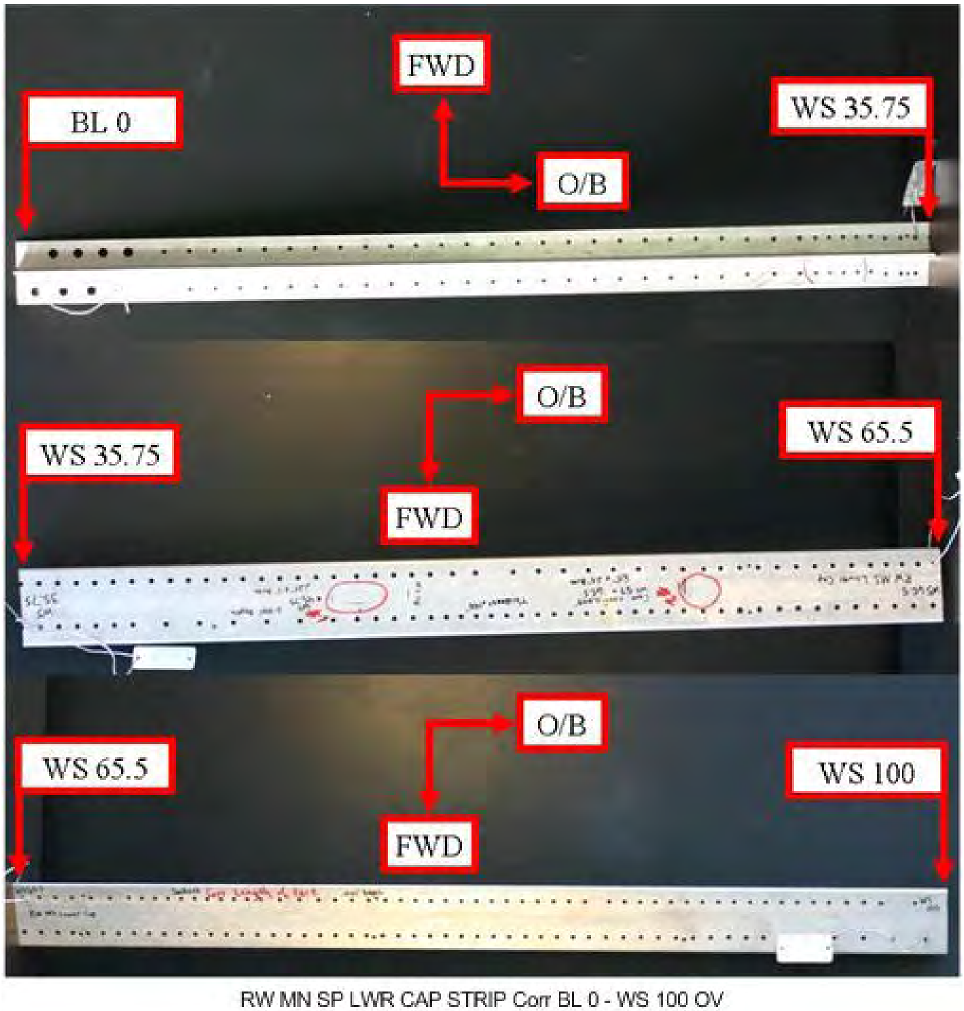


Figure 247. Location of Light-Moderate Corrosion on the Right Wing Main Spar Lower Cap Strip BL 0 Through WS 100

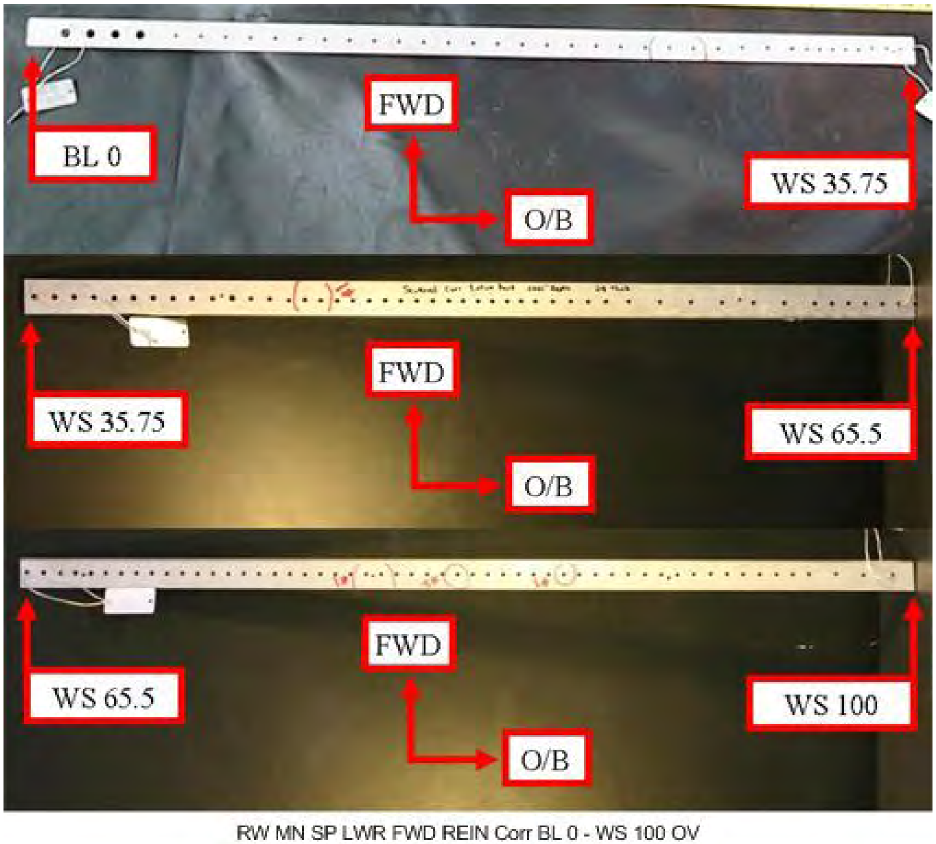


Figure 248. Location of Light-Moderate Corrosion on the Right Wing Main Spar Lower Forward Reinforcement BL 0 Through WS 100

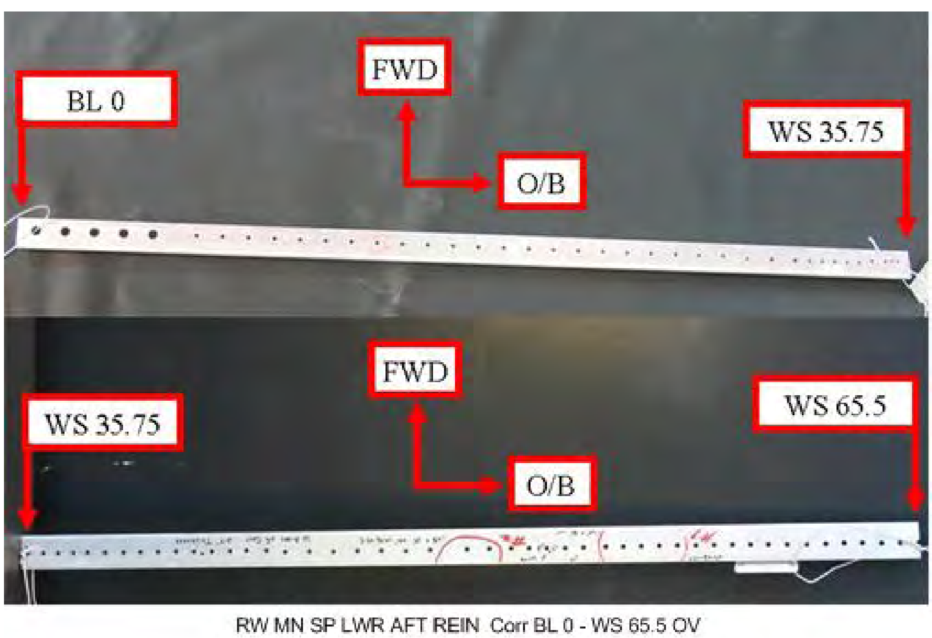


Figure 249. Location of Light-Moderate Corrosion on the Right Wing Main Spar Lower Aft Reinforcement BL 0 Through WS 65.5

A 0.11-square-inch area of moderate-severe corrosion was observed on the right wing main spar lower forward angle from BL 13.75 to BL 14.5, as shown in figure 250. This area of corrosion, shown in figure 251, caused a maximum localized thickness loss of 7.3%. Scattered light-moderate corrosion and an area of wear were found on the main spar lower aft angle from BL 0 to WS 65.5, as shown in figure 252. The light-moderate corrosion was scattered across the surface of the part from BL 0 to WS 65.5 and caused a maximum reduction in thickness of 4.4%. The 0.53 square inch of wear, shown in figure 253, was located at BL 0 and caused a maximum thickness loss of 4.4%.

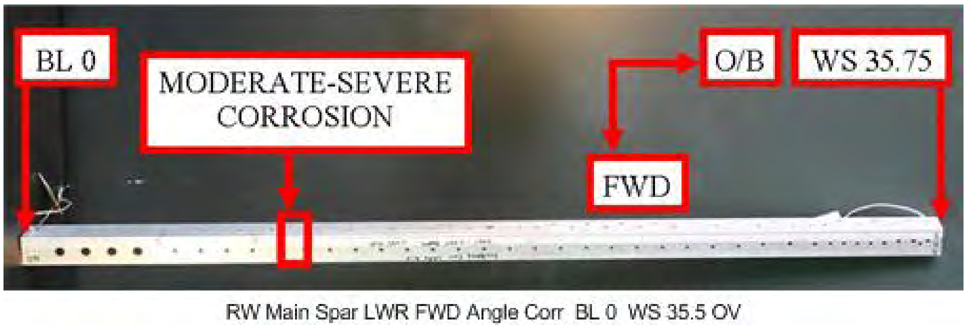


Figure 250. Location of Moderate-Severe Corrosion on the Right Wing Main Spar Lower Forward Angle BL 13.75 Through BL 14.5

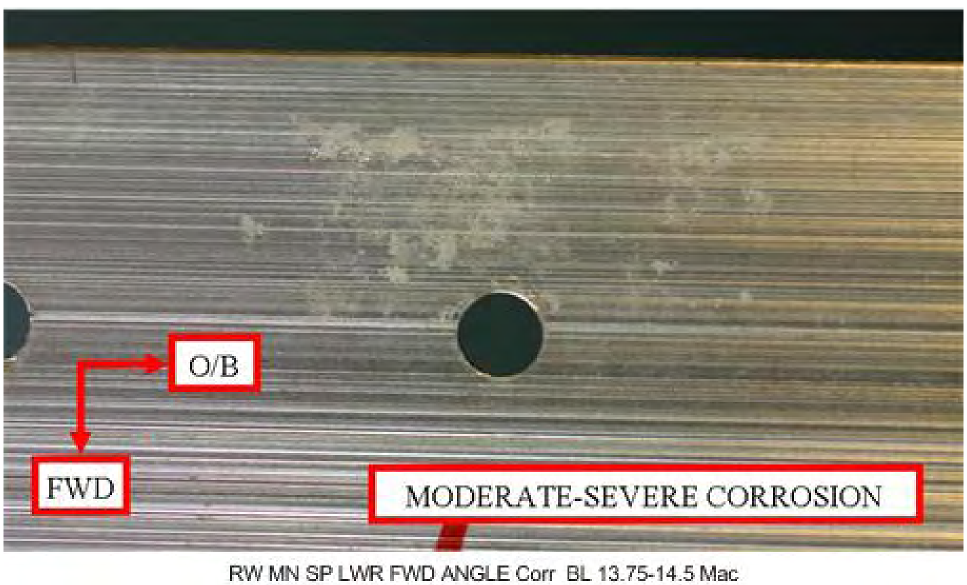


Figure 251. Macroscopic View of Moderate-Severe Corrosion on the Right Wing Main Spar Lower Forward Angle BL 13.75 Through BL 14.5

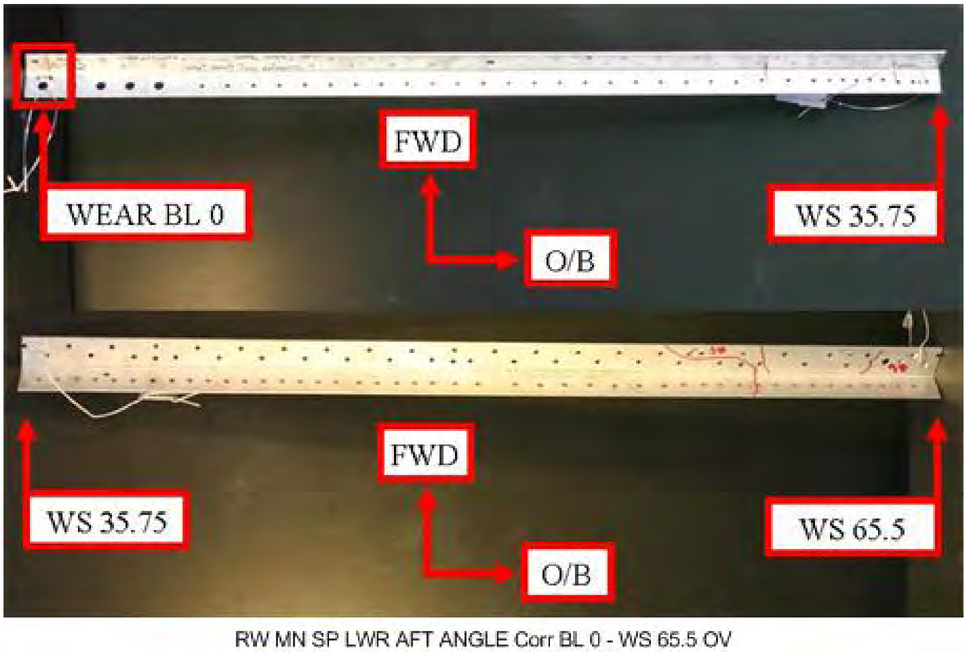


Figure 252. Location of Light-Moderate Corrosion and Wear on the Right Wing Main Spar Lower Aft Angle BL 0 Through WS 65.5

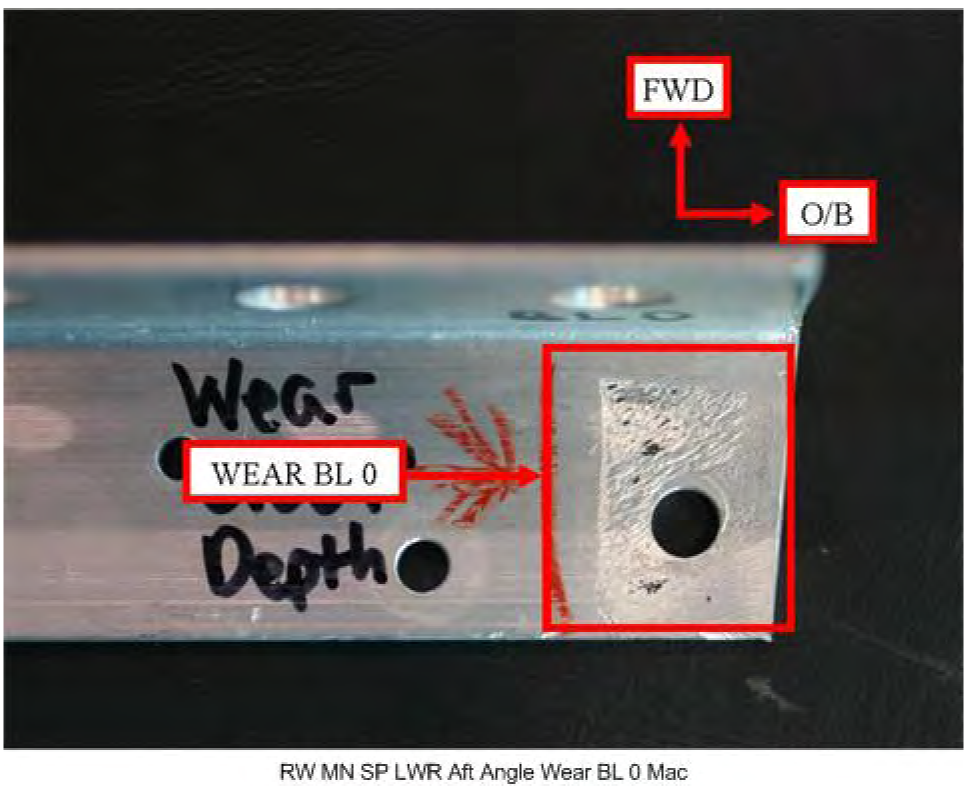


Figure 253. Macroscopic View of Wear on the Right Wing Main Spar Lower Aft Angle BL 0

3.4.2.2.3 Right Wing Rear Spar.

There were no defects found on the right wing rear spar.

3.4.2.2.4 Right Wing Various Indications.

Numerous indications were identified on the right wing in areas other than the front, main, and rear spars. The detailed characterization of these defects is shown in table 33.

Table 33. Inspection Results From the Remainder of the Right Wing

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right wing plate wing panel reinforcement assembly, figure 255	Wear	BL 17.5	0.895 inch by 0.97 inch	No indication	Wear 6.7% thickness loss	256
	Wear	BL 28 through BL 29.5	1.60 inches by 0.850 inch	No indication	Wear 9% thickness loss	257
	Wear	BL 28 through BL 29.5	1.90 inches by 0.643 inch	No indication	Wear 9% thickness loss	258
Right wing bulkhead wing fillet, figures 259 and 260	Crack	WS 30 FS 181	2.25 inches	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	261
	Crack	WS 30 FS 181	0.75 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	262
Right wing bulkhead wing fillet, figure 263	Crack	WS 30 FS 191.7	1.075 inches	Crack indication	Bend radii	264 265 266
	Crack	WS 30 FS 191.7	0.525 inch	Crack indication	Bend radii	267 268 269
	Crack	WS 30 FS 191.7	0.458 inch	Crack indication	Bend radii	270 271 272
Right wing rear spar rib assembly, figure 273	Crack	WS 39.5	0.216 inch	Crack indication	Bend radii	274 275
	Crack	WS 39.5	1.09 inches	Crack indication	Bend radii	277 278 279
Right wing rear spar rib assembly, figure 282	Damage	WS 49	Entire part	Not inspected		280 281
	Crack	WS 49	0.244 inch	Crack indication	Bend radii	283 284
	Crack	WS 49	0.242 inch	Crack indication	Hole crack	285 286

Table 33. Inspection Results From the Remainder of the Right Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right wing rear spar assembly main gear forward, figure 287	Wear	WS 49.5 through WS 59.25	9.75 inches by 0.26 inch	No indication	Wear 48% thickness loss	288
	Wear	WS 59 through WS 85.5	26.5 inches by 0.16 inch	No indication	Wear 21% thickness loss	289
Right wing rear spar lower aft assembly, figure 290	Wear	WS 54	5.25 inches by 0.114 inch	No indication	Wear 19.3% thickness loss	291
	Wear	WS 60	17 inches by 0.235 inch	No indication	Wear 44% thickness loss	281
Right wing rear spar bulkhead assembly main gear support fwd, figure 292	Corrosion	WS 58.5 through WS 87	Scattered over entire forward side of part	Corrosion indication	Light-moderate corrosion 2.5% thickness loss	105
Right wing nacelle inboard plate skin attachment, figure 293	Crack/ Wear	WS 60	0.4 inch	Crack indication	Surface crack	294 295 2966
Right wing inboard upper rivet of the engine attachment, figure 297	Sheared	WS 66	Whole rivet	Not inspected	Fatigue	298 299
Right nacelle longitudinal inboard bulkhead aft piece, figures 300 and 307	Corrosion	WS 66 FS 154 through FS 182	Scattered over entire part	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
	Crack	WS 66 FS 160.25	0.369 inch	Crack indication	Surface crack	304
	Crack	WS 66 FS 161.125	0.608 inch	Crack indication	Surface crack	307
	Crack	WS 66 FS 162.125	0.127 inch	Crack indication	Surface crack	310
	Crack	WS 66 FS 171.25	0.318 inch	Crack indication	Surface crack	313
	Crack	WS 66 FS 179.25	0.133 inch	Crack indication	Hole crack	316

Table 33. Inspection Results From the Remainder of the Right Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right nacelle inboard longitudinal bulkhead, figure 317	Corrosion	WS 66 FS 151 through FS 152	1 inch by 1.25 inches	Corrosion	Severe corrosion 22% thickness loss	318
	Corrosion	WS 66 FS 157 through FS 158.5	1.5 inches by 0.75 inch	Corrosion	Severe corrosion 22% thickness loss	319
	Corrosion	WS 66 FS 174 through FS 180	6 inches by 2.5 inch	Corrosion indication	Severe corrosion 22% thickness loss	320
	Crack	WS 66 FS 174	0.385 inch	Crack indication	Surface crack	321 322
Right nacelle bulkhead longitudinal inboard, figure 323	Corrosion	WS 66 FS 182 through FS 210.75	Scattered over entire part	Corrosion indication	Light-moderate 4% thickness loss	105
Right wing nacelle bulkhead assembly, figure 324	Corrosion	WS 66 through 89 FS 121	Scattered over entire aft side of part	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
Right wing nacelle torque tube bracket assembly, figure 325	Crack	WS 66 through 89 FS 111.5	0.21 inch	Crack indication	Surface crack	326 327 328
Right wing nacelle plate assembly firewall shear, figure 329	Wear	WS 71	3.42 inches by 0.332 inch	No indication	Wear 60% thickness loss	330
	Wear	WS 82	1.85 inches by 0.236 inch	No indication	Wear 12.5% thickness loss	331
Right wing trunnion landing gear aft, figure 332	Crack	WS 77.5	0.210 inch	Crack indication	Surface crack	333 334
	Crack	WS 77.5	0.172 inch	Crack indication	Surface crack	335 336 337
Right wing rear spar rib assembly, figure 338	Corrosion	WS 87.5	Scattered inboard side of part	Corrosion indication	Light corrosion 2% thickness loss	103

Table 33. Inspection Results From the Remainder of the Right Wing (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right wing nacelle outboard longitudinal bulkhead, figure 339	Corrosion	WS 89 FS 182.25 through FS 210.75	Scattered over entire inboard side of part	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
	Crack	WS 89 FS 210.75	2.29 inches	Crack indication	Surface crack	340
Right wing flap track, figure 341	Crack	WS 147.5	0.047 inch	Crack indication	Hole crack	342 343
Right aileron hinge, figure 344	Corrosion	WS 174.5	Over entire part	No indication	Light corrosion less than 1% thickness loss	103

[1] Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

The location of three areas of wear that were observed on the right wing panel wing panel reinforcement assembly, part number 44710-01, is shown in figure 254. A macroscopic view of wear 1, located at BL 17.5, is shown in figure 255. This 0.87-square-inch area of wear resulted in a localized reduction in thickness of 6.7%. Figure 256 shows a macroscopic view of wear 2, which covered a surface area of 1.36 square inches from BL 28 to BL 29.5 and caused a maximum localized thickness loss of 9%. Wear 3 resulted in a 9% localized thickness loss, covered 1.22 square inches, and was located from BL 28 to BL 29.5, as shown in figure 257. Two cracks were documented on the right wing bulkhead wing fillet, part number 43646-5, located at WS 30 FS 181. The location of these two cracks is shown in figure 258, and figure 259 shows an overview of the section removed for fractographic analysis of these cracks. A macroscopic view of crack 1 is shown in figure 260. This crack measured 2.25 inches and the failure mode could not be determined due to extensive crack face smearing. A macroscopic view of crack 2, which measured 0.75 inches, is shown in figure 261. The failure mode of this crack could also not be determined.

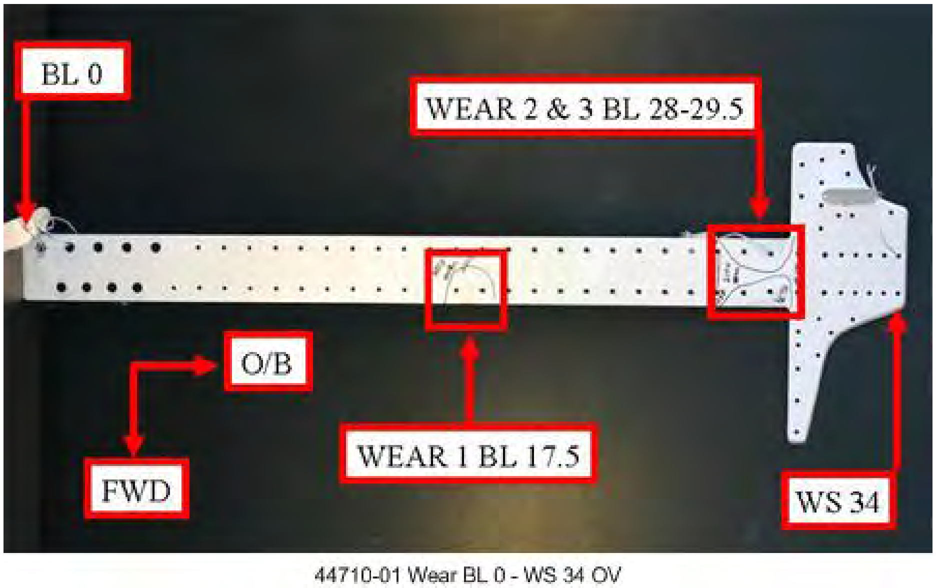


Figure 254. Location of Wear on the Right Wing Panel—Wing Panel Reinforcement Assembly BL 0 Through WS 36

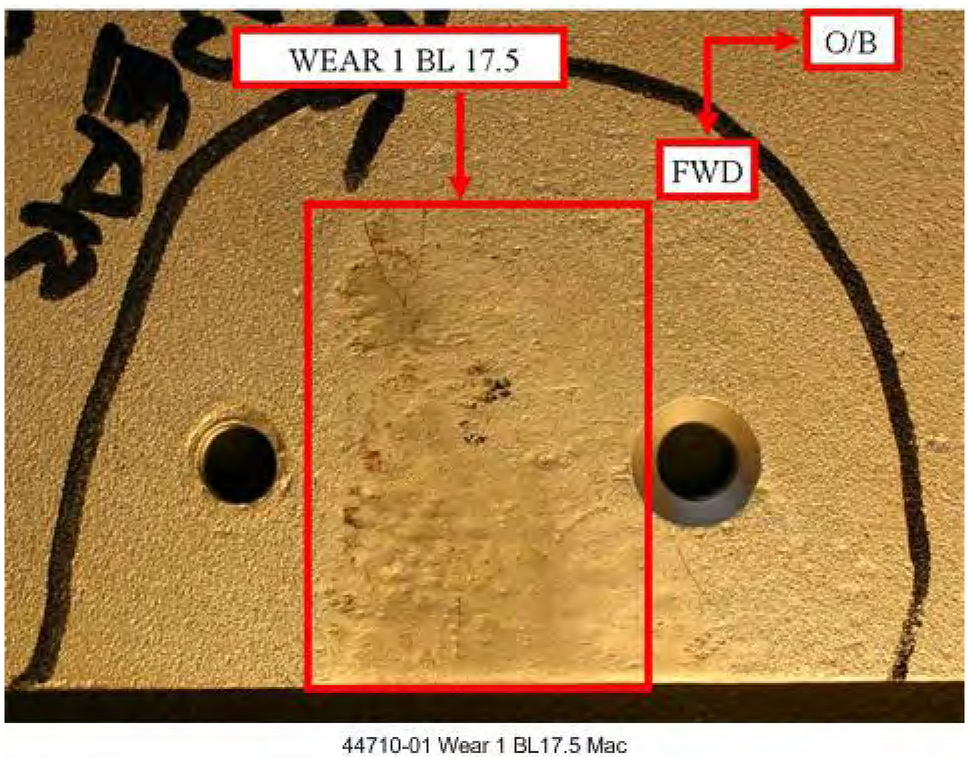


Figure 255. Macroscopic View of Wear 1 on the Right Wing Panel—Wing Panel Reinforcement Assembly BL 17.5

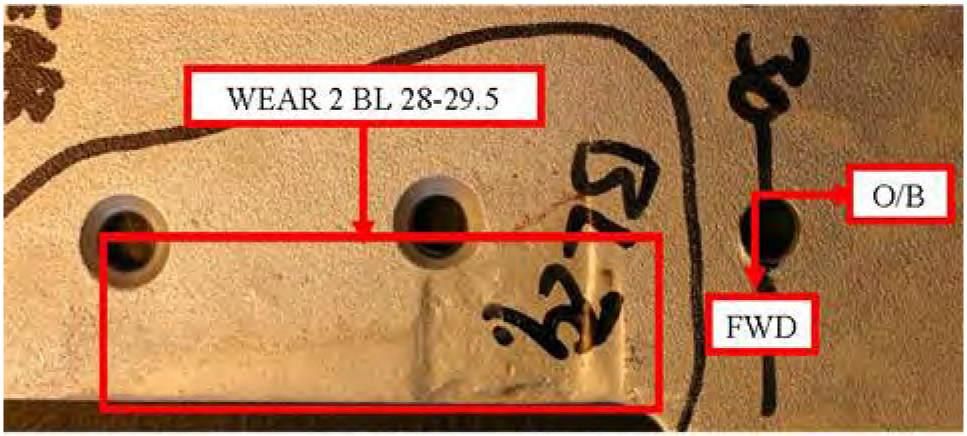


Figure 256. Macroscopic View of Wear 2 on the Right Wing Panel—Wing Panel Reinforcement Assembly BL 28 Through BL 29.5

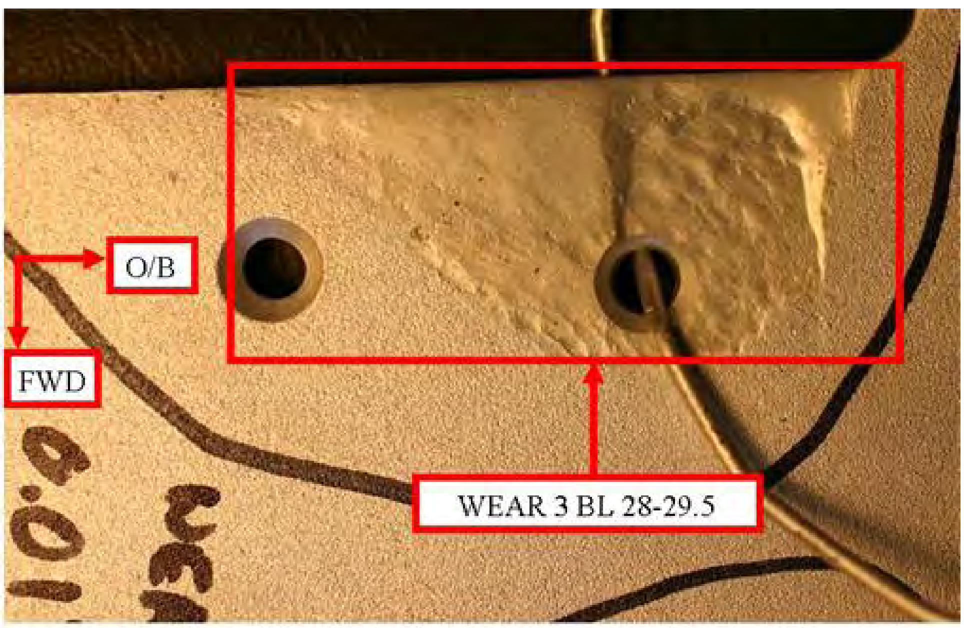
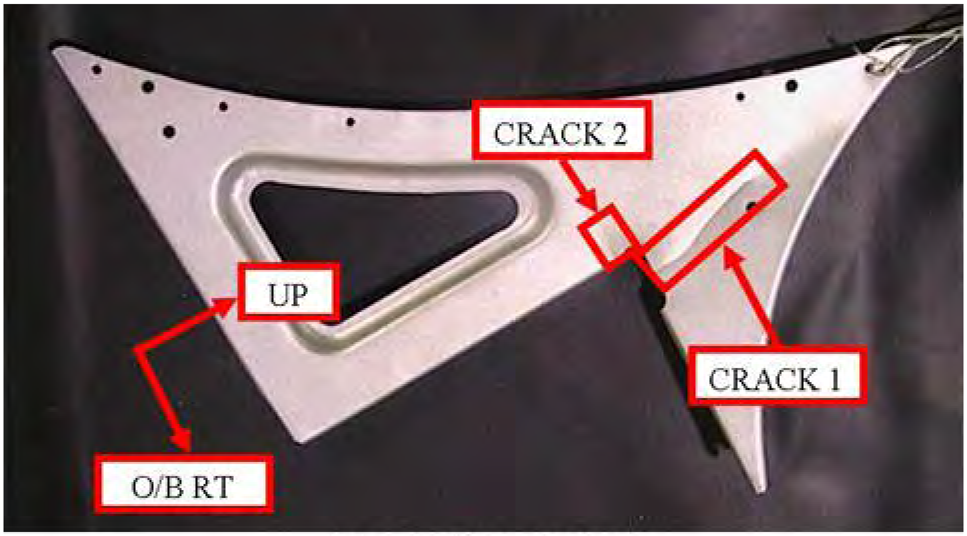
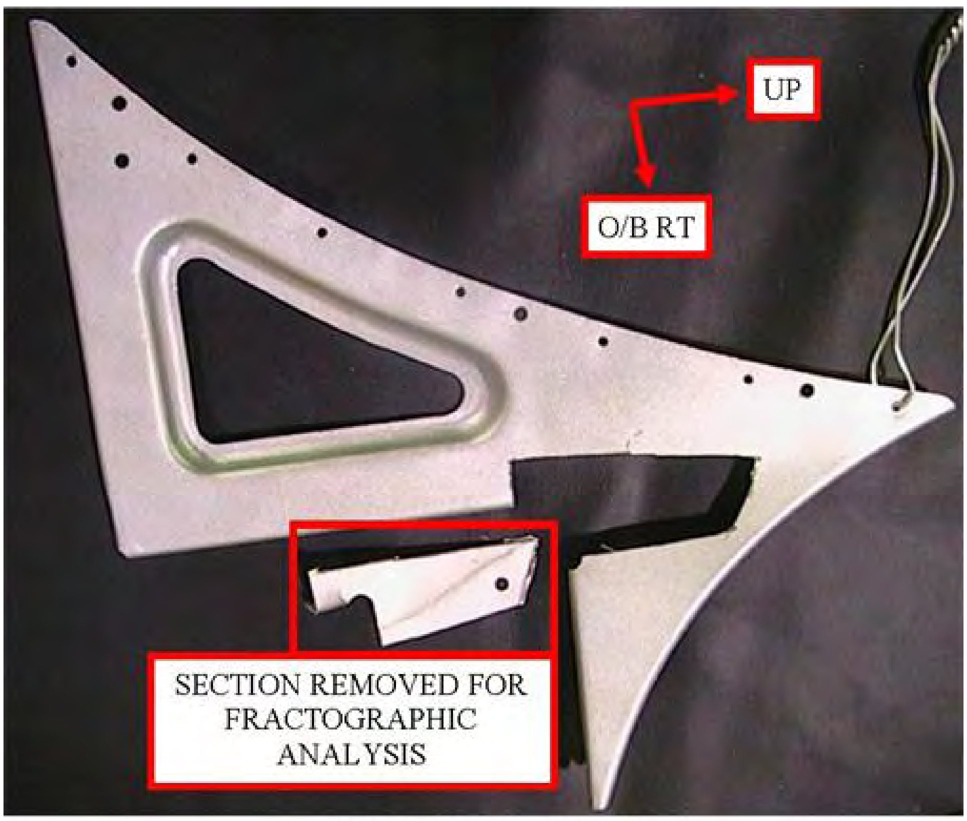


Figure 257. Macroscopic View of Wear 3 on the Right Wing Panel—Wing Panel Reinforcement Assembly BL 28 Through FS 29.5



43646-5 CRK WS 30 FS 181 OV

Figure 258. Location of Cracks 1 and 2 on the Right Wing Bulkhead Wing Fillet WS 30 FS 181



43646-5 CRK WS 30 FS 181 OV 2

Figure 259. Overview of Section of the Right Wing Bulkhead Wing Fillet Removed for Fractographic Analysis WS 30 FS 181

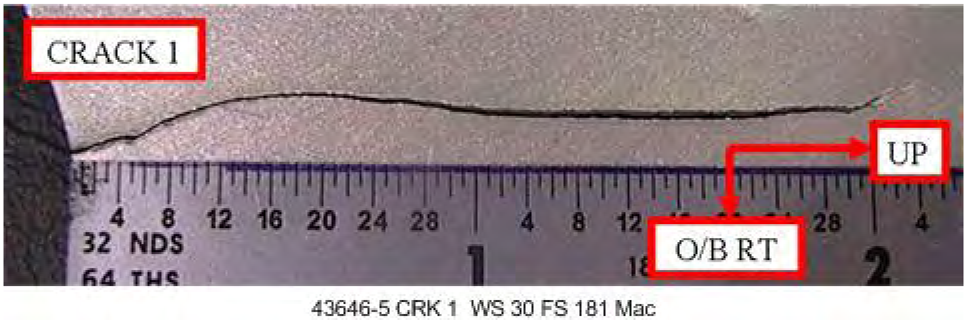
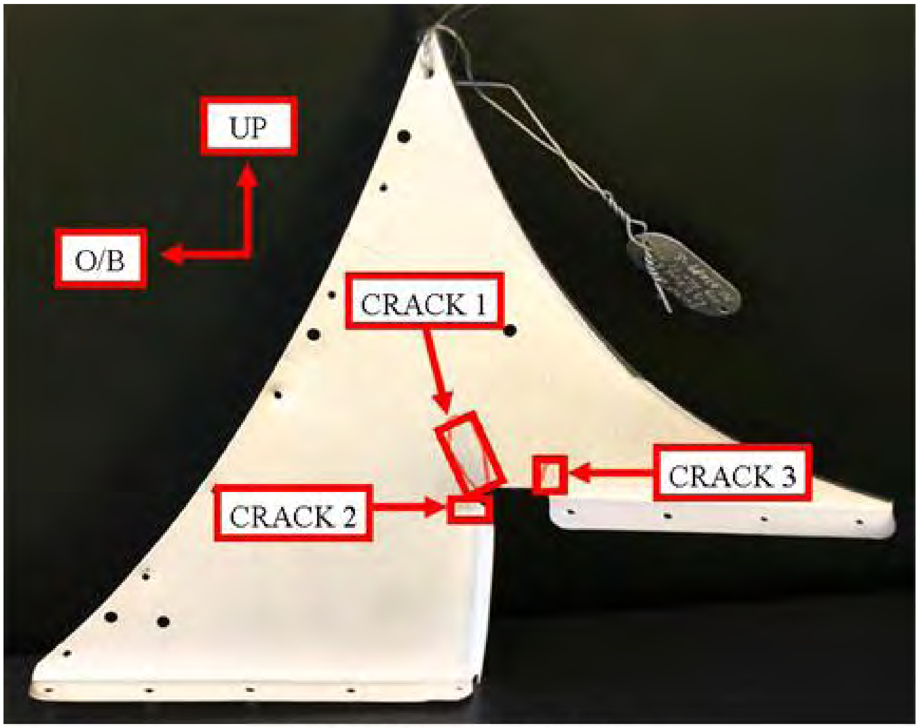


Figure 260. Macroscopic View of Crack 1 on the Right Wing Bulkhead Wing Fillet
WS 30 FS 181

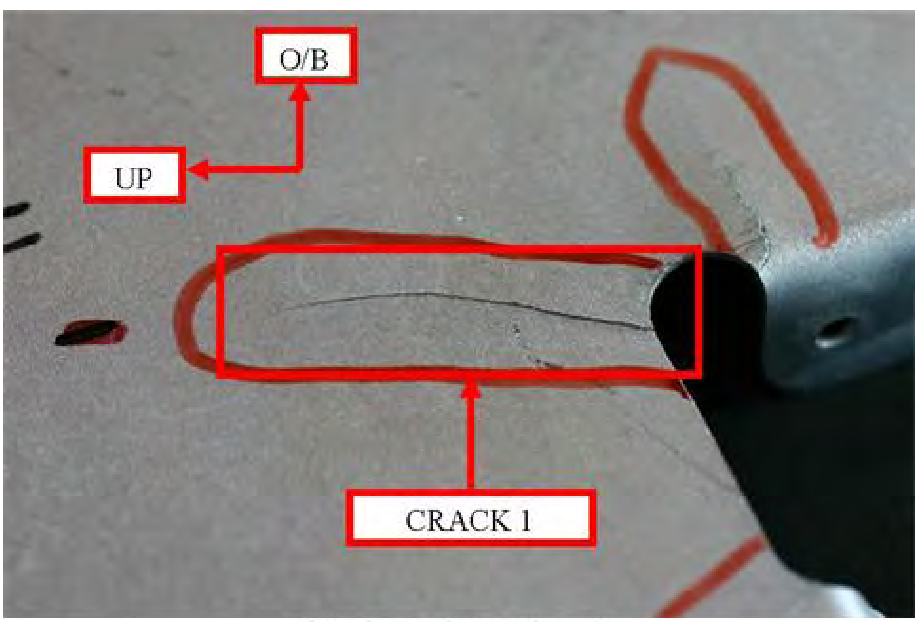
Figure 261. Macroscopic View of Crack 2 on the Right Wing Bulkhead Wing Fillet
WS 30 FS 181

The location of three cracks on the right wing bulkhead wing fillet, part number 43646-05, is shown in figure 262 at WS 30 FS 191.7. A macroscopic view of crack 1, which measured 1.075 inches, is shown in figure 263, and the fluorescent liquid penetrant indication is shown in figure 264. A microscopic view of crack 1 is shown in figure 265. A macroscopic view of crack 2 is shown in figure 266. The fluorescent liquid penetrant indication of crack 2 is shown in figure 267, and a microscopic view of crack 2, which measured 0.525 inch in length, is shown in figure 268. Crack 3, which measured 0.458 inch, is shown macroscopically in figure 269. The fluorescent liquid penetrant indication of crack 3 is shown in figure 270, and a microscopic view is shown in figure 271.



43646-5 CRK WS 30 FS 191.7 OV

Figure 262. Location of Cracks 1, 2, and 3 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7



43646-5 CRK 1 WS 30 FS 191.7 Mac

Figure 263. Macroscopic View of Crack 1 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7

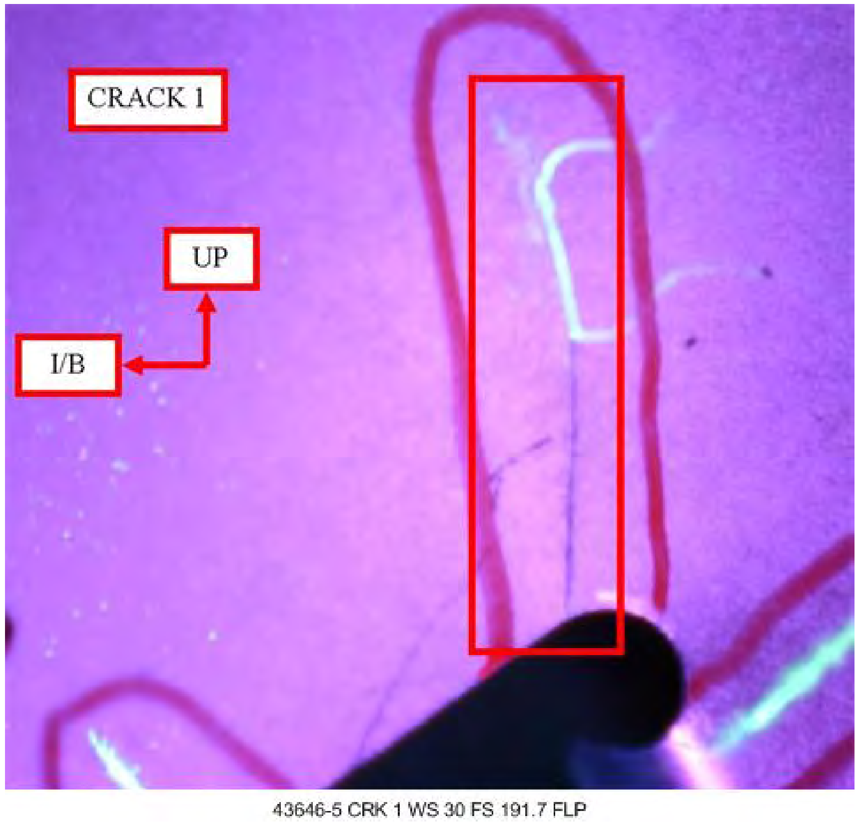


Figure 264. Fluorescent Liquid Penetrant Indication of Crack 1 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7

Figure 265. Microscopic View of Crack 1 on the Right Wing Bulkhead Wing Fillet
WS 30 FS 191.7

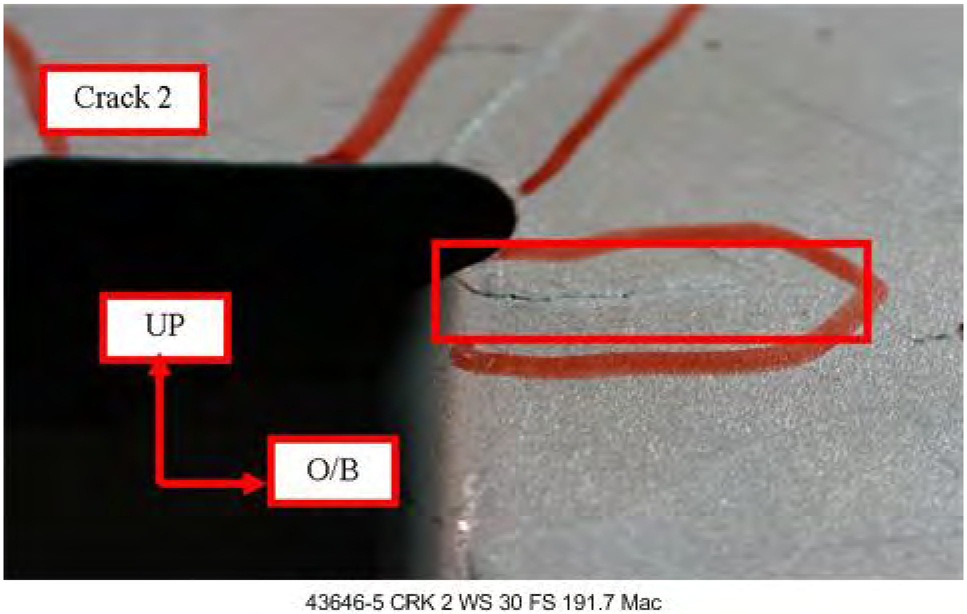


Figure 266. Macroscopic View of Crack 2 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7

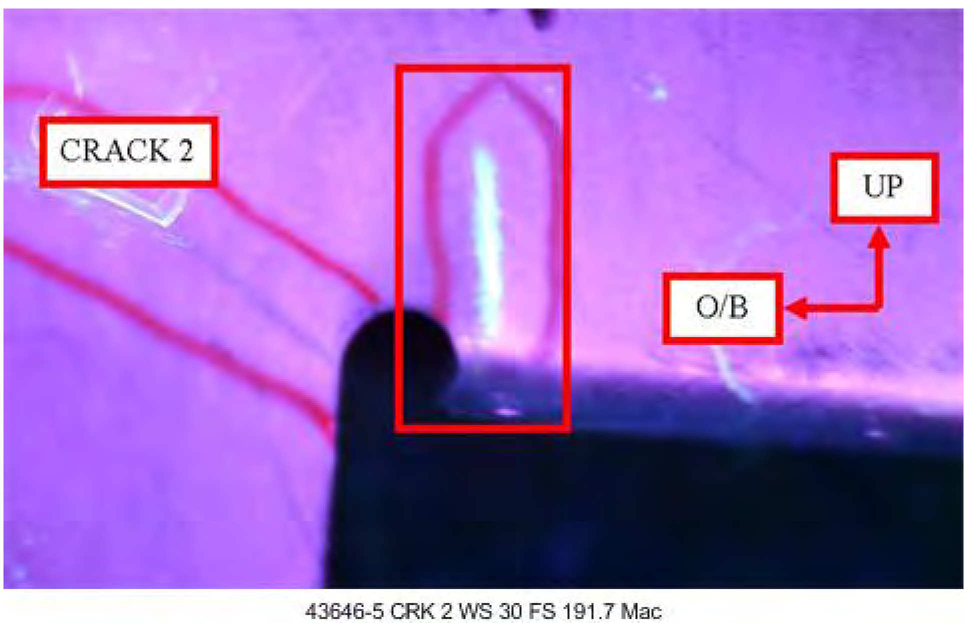
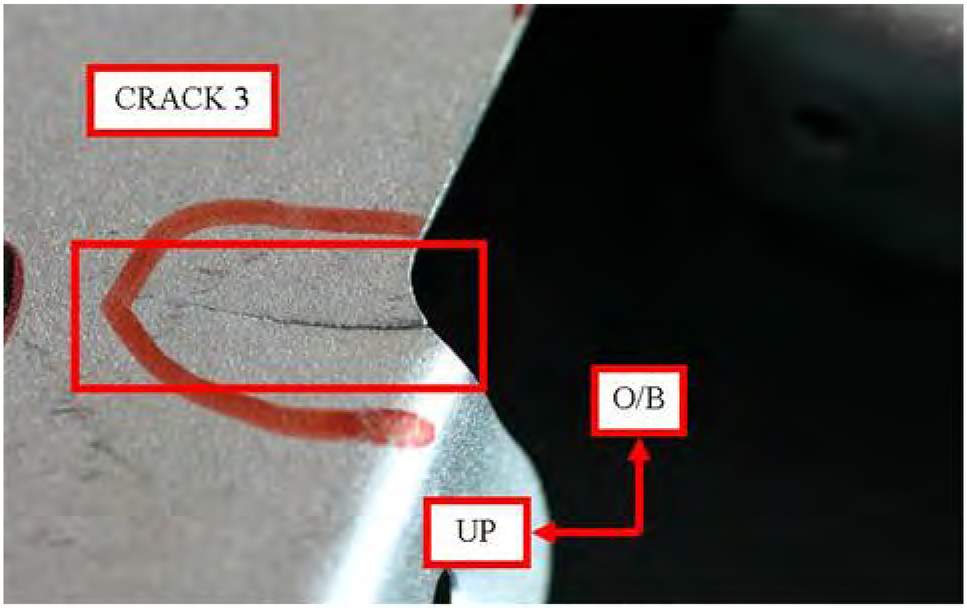


Figure 267. Fluorescent Liquid Penetrant Indication of Crack 2 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7

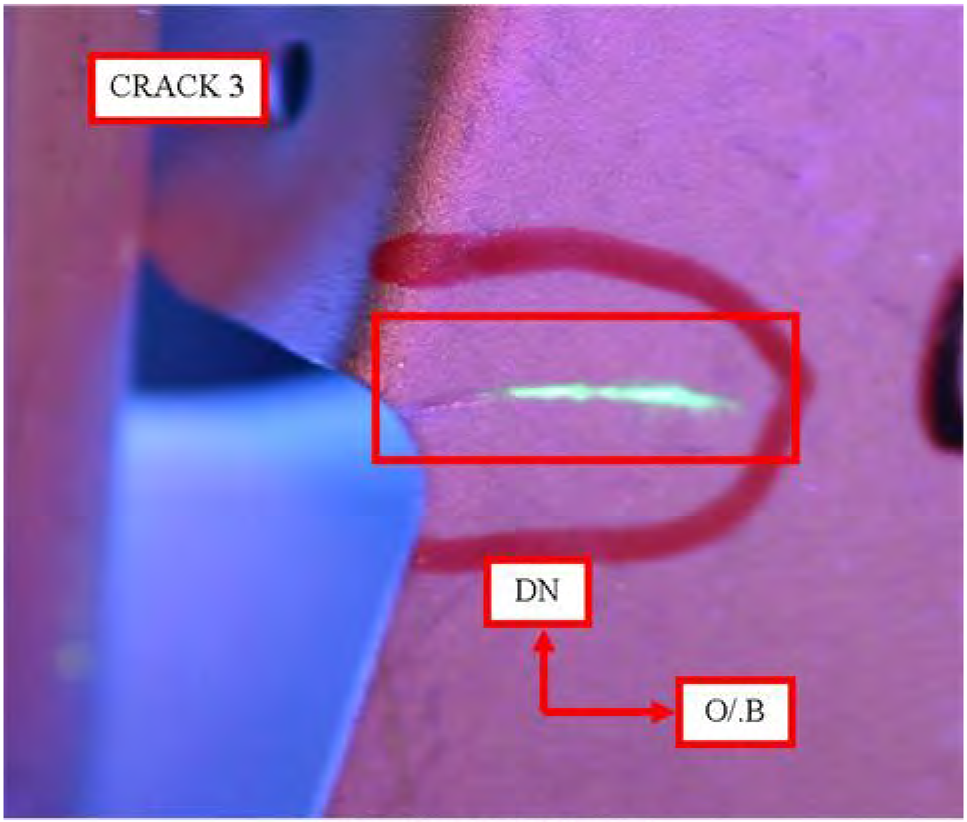
43646-5 CRK 2 WS 30 FS 191.7 Mic

Figure 268. Microscopic View of Crack 2 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7



43646-5 CRK 3 WS 30 FS 191.7 Mac

Figure 269. Macroscopic View of Crack 3 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7



43646-5 CRK 3 WS 30 FS 191.7 FLP

Figure 270. Fluorescent Liquid Penetrant Indication of Crack 3 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7

43646-5 CRK 3 WS 30 FS 191.7 Mic

Figure 271. Microscopic View of Crack 3 on the Right Wing Bulkhead Wing Fillet WS 30 FS 191.7

Figure 272 shows the location of two cracks on the right wing rear spar rib assembly at WS 39.5, part number 74099-07B. A macroscopic view of crack 1, which measures 0.216 inches, is shown in figure 273, and the fluorescent liquid penetrant indication is shown in figure 274. A microscopic view of crack 1 is shown in figure 275. A macroscopic view of crack 2 is shown in figure 276, and the fluorescent liquid penetrant indication of crack 2 is shown in figure 277. A microscopic view of this crack, which measured 1.09 inches, is shown in figure 278. An overview of damage observed on the right wing rear spar rib assembly WS 49, part number 40423-23A, is shown in figure 279. A close-up of this damage is shown in figure 280. Figure 281 shows the location of two cracks observed on the rear spar rib assembly WS 49. A macroscopic view of crack 1, which measures 0.244 inches, is shown in figure 282, and the fluorescent liquid penetrant indication of this crack is shown in figure 283. Figure 284 shows a macroscopic view of crack 2, and figure 285 shows the fluorescent liquid penetrant indication of this 0.242-inch crack.

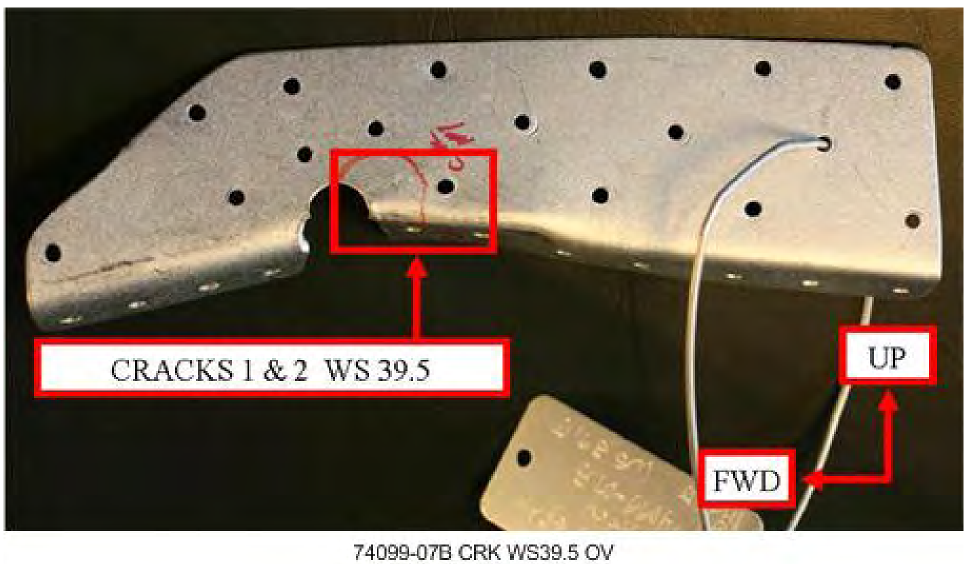


Figure 272. Location of Cracks 1 and 2 on the Right Wing Rear Spar Rib Assembly WS 39.5

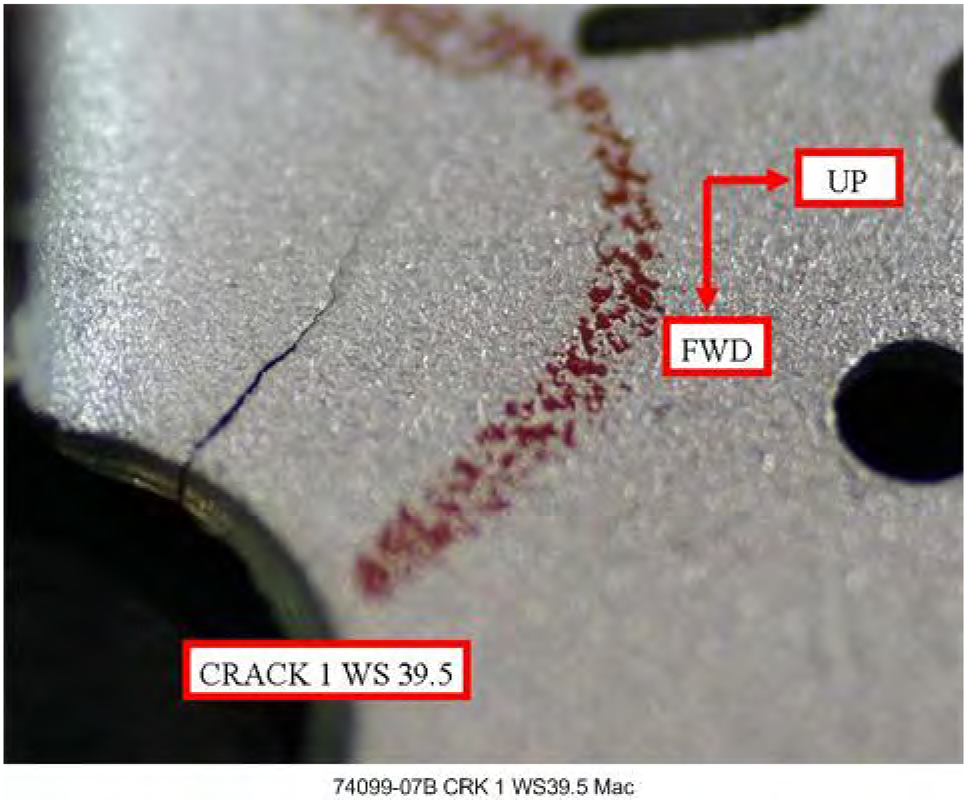


Figure 273. Macroscopic View of Crack 1 on the Right Wing Rear Spar Rib Assembly WS 39.5

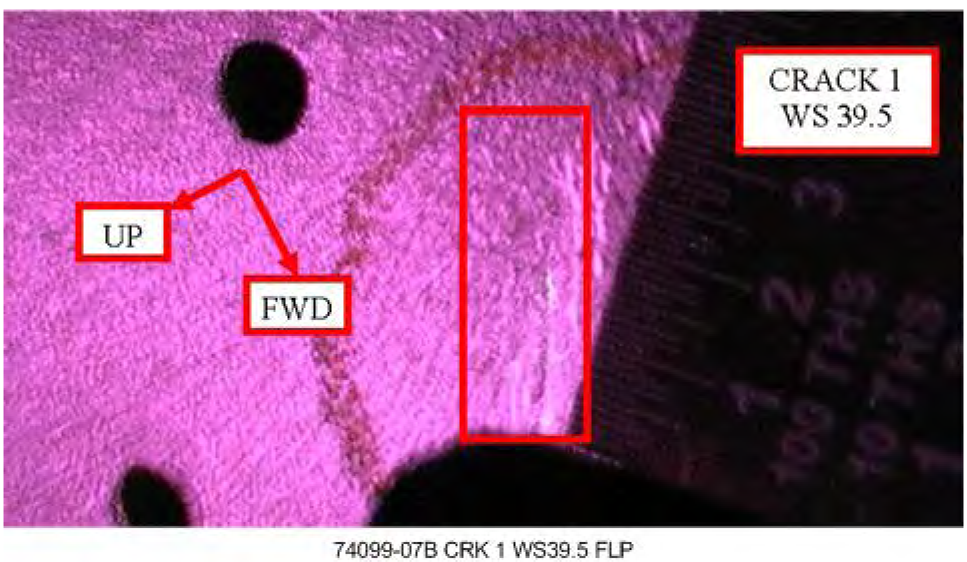


Figure 274. Fluorescent Liquid Penetrant Indication of Crack 1 on the Right Wing Rear Spar Rib Assembly WS 39.5

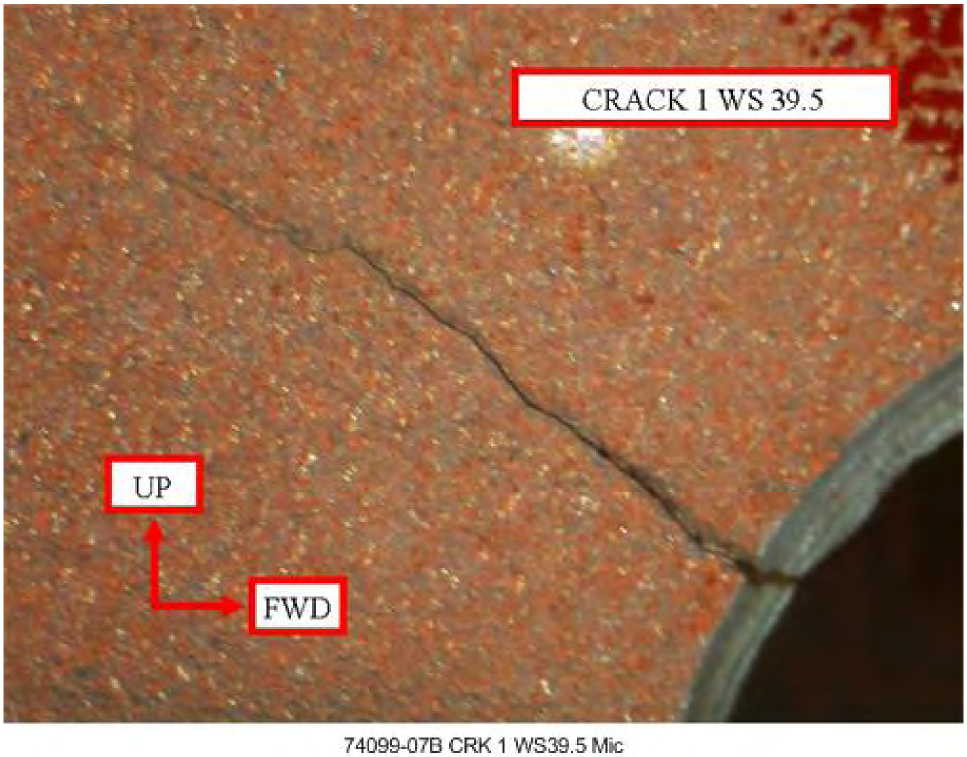


Figure 275. Microscopic View of Crack 1 on the Right Wing Rear Spar Rib Assembly WS 39.5

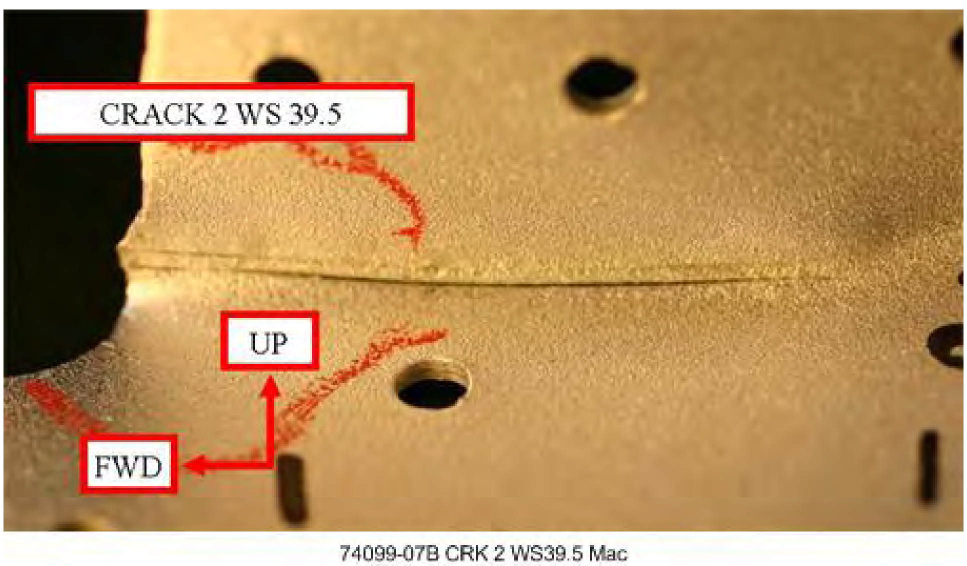


Figure 276. Macroscopic View of Crack 2 on the Right Wing Rear Spar Rib Assembly WS 39.5

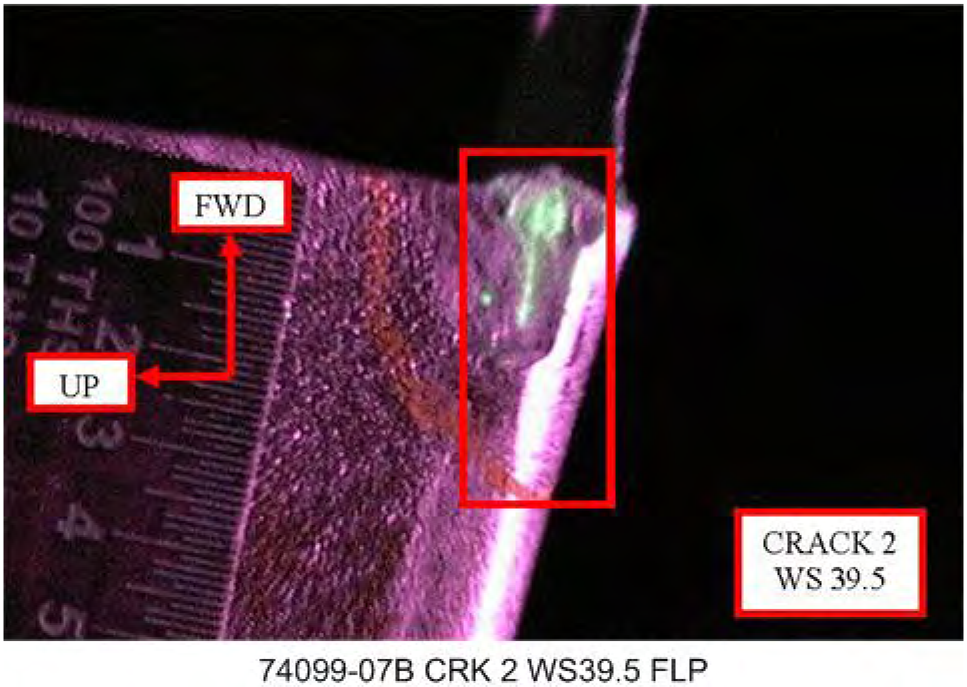


Figure 277. Fluorescent Liquid Penetrant Indication of Crack 2 on the Right Wing Rear Spar Rib Assembly WS 39.5

Figure 278. Microscopic View of Crack 2 on the Right Wing Rear Spar Rib Assembly WS 39.5

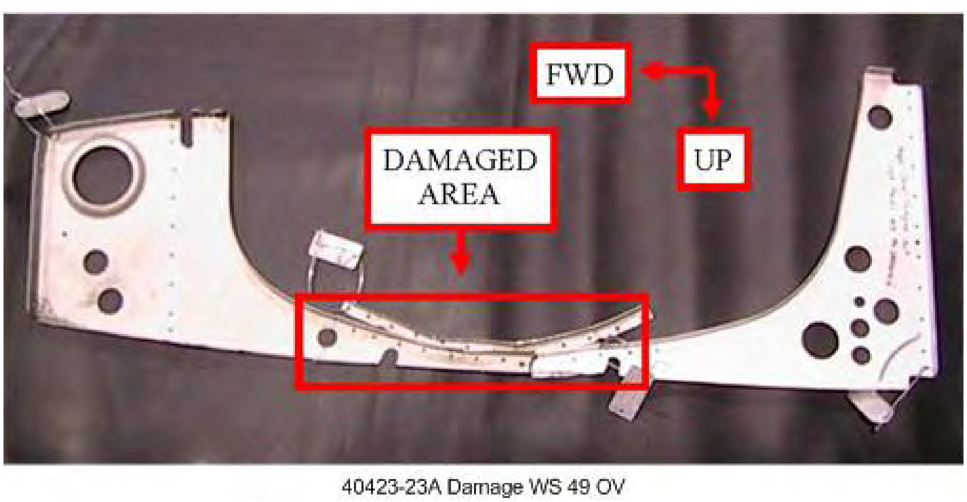


Figure 279. Overview of Damage on the Right Wing Rear Spar Rib Assembly WS 49

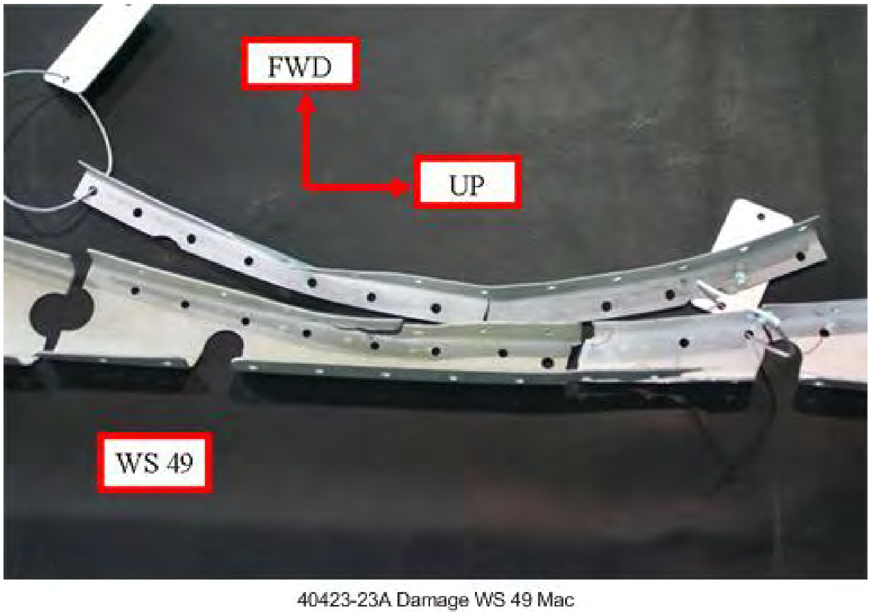


Figure 280. Close-Up of Damage on the Right Wing Rear Spar Rib Assembly WS 49

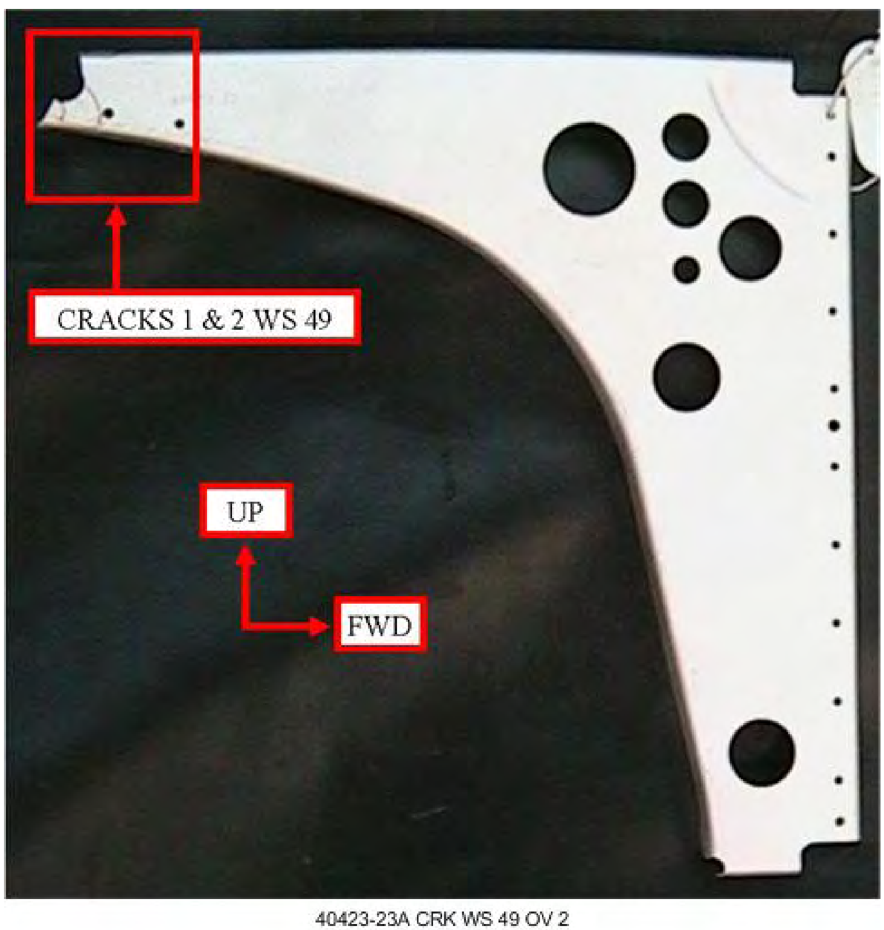


Figure 281. Location of Cracks 1 and 2 on the Right Wing Rear Spar Rib Assembly WS 49

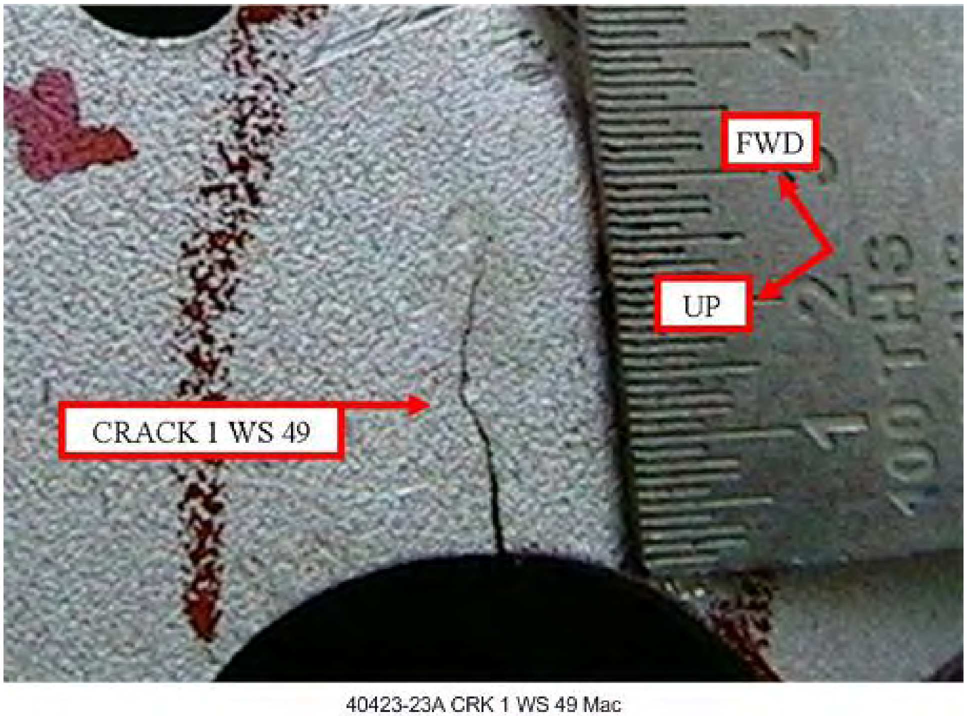


Figure 282. Macroscopic View of Crack 1 on the Right wing Rear Spar Rib Assembly WS 49

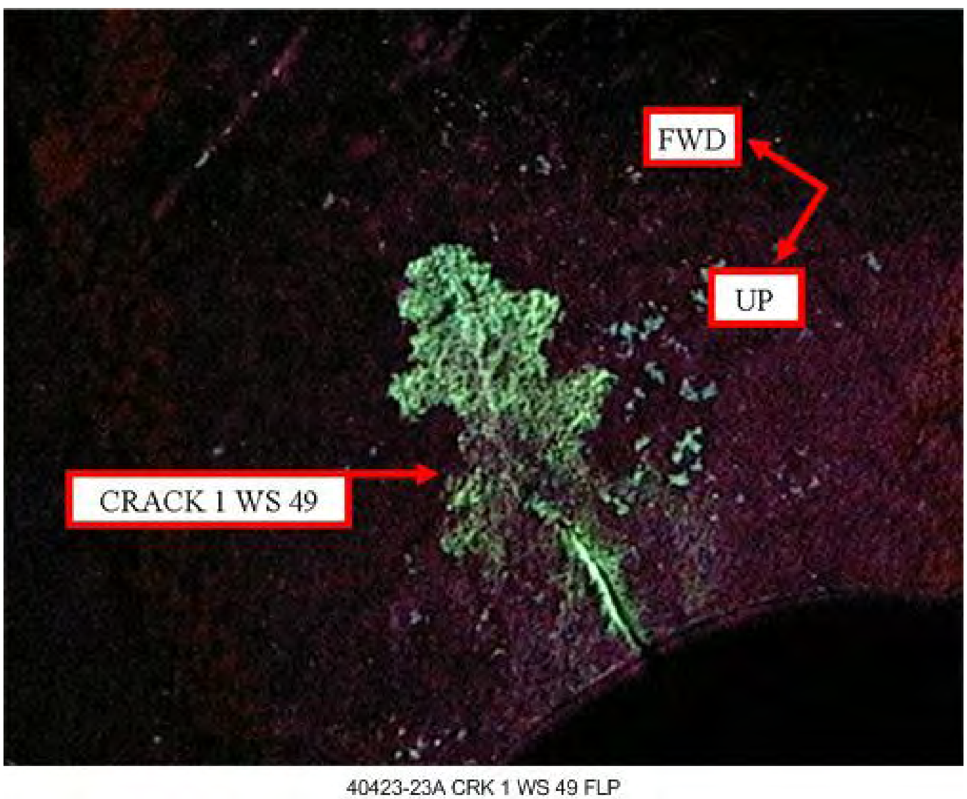


Figure 283. Fluorescent Liquid Penetrant Indication of Crack 1 on the Right Wing Rear Spar Rib Assembly WS 49

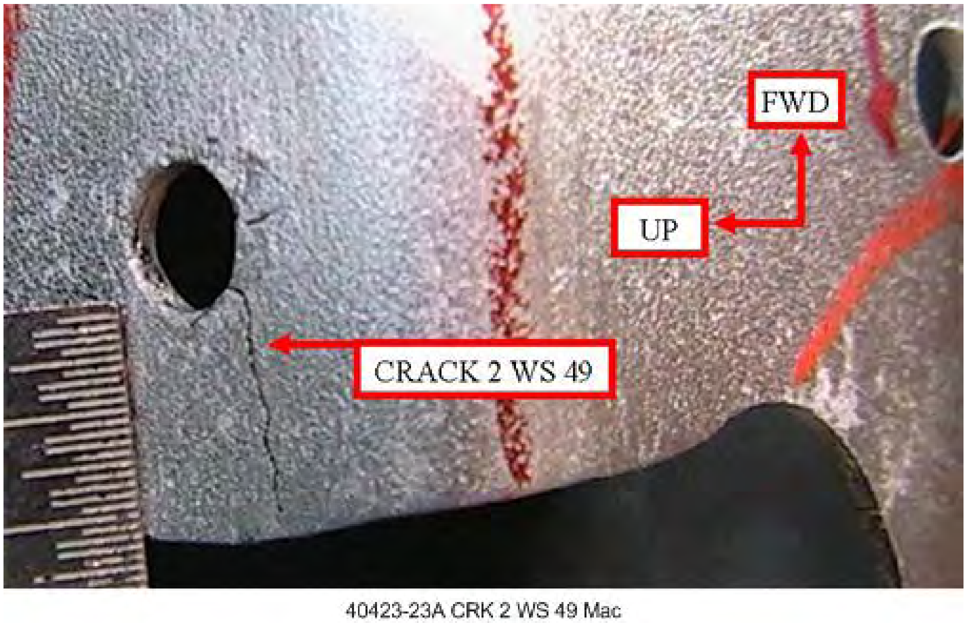


Figure 284. Macroscopic View of Crack 2 on the Right Wing Rear Spar Rib Assembly WS 49

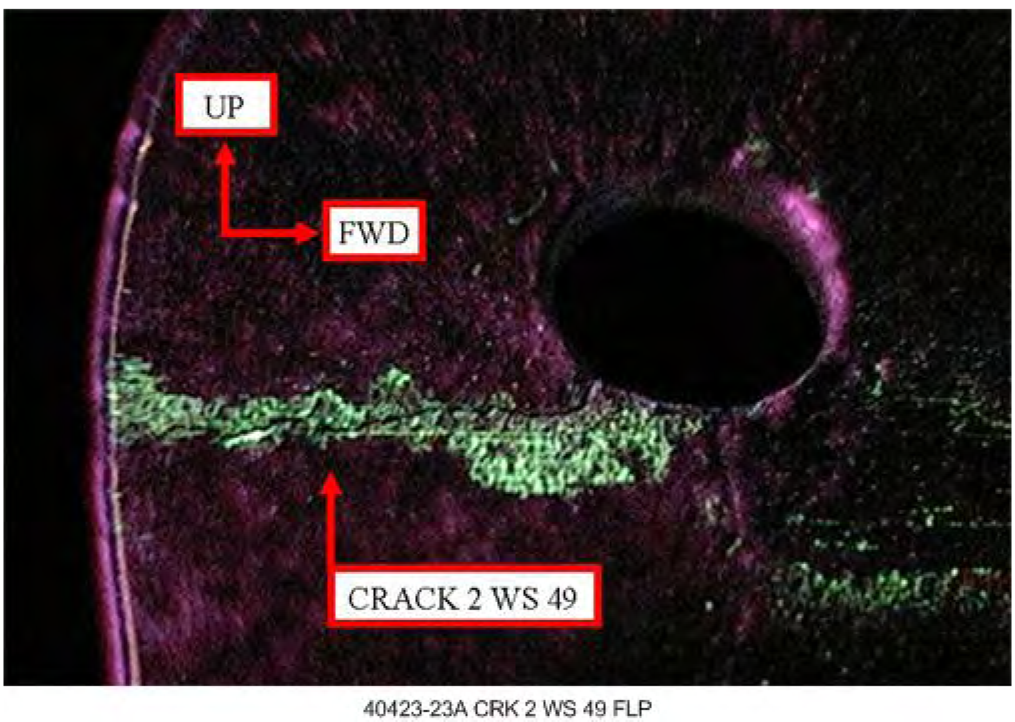
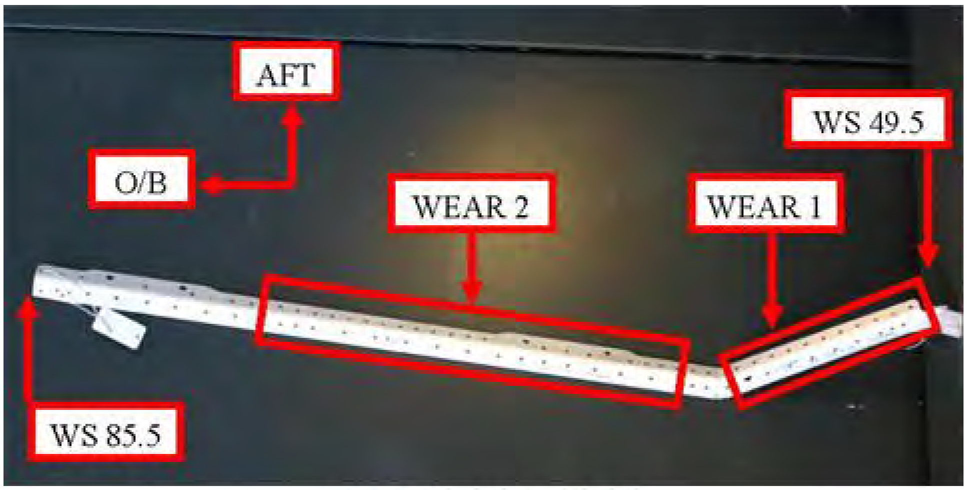


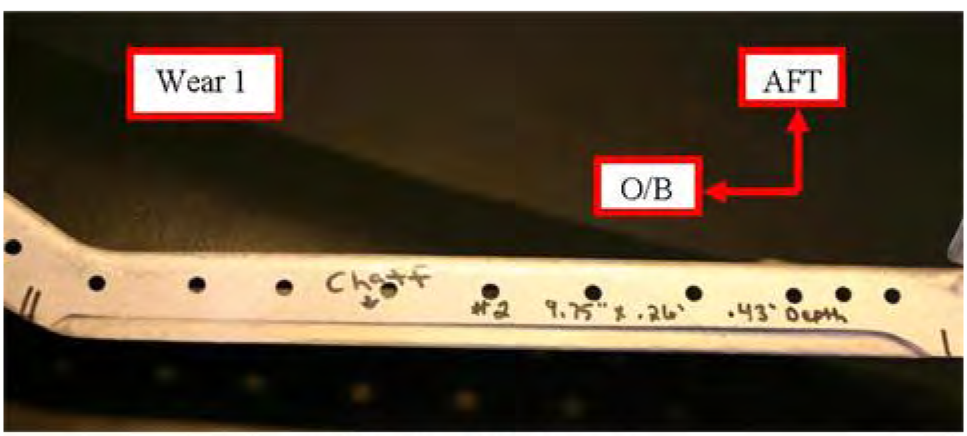
Figure 285. Fluorescent Liquid Penetrant Indication of Crack 2 on the Right Wing Rear Spar Rib Assembly WS 49

Figure 286 shows the location of wear 1 and 2 on the right wing rear spar main gear forward, part number 40500-01. Figure 287 shows a macroscopic view of wear 1. Wear 1, located from WS 49.5 to WS 59.25, covered a surface area of 2.54 square inches and caused a maximum reduction in thickness of 48%. Wear 2, shown macroscopically in figure 288, covered a surface area of 4.24 square inches and resulted in a maximum localized thickness loss of 21%. The location of two areas of wear on the right wing rear spar lower aft assembly, part number 45838-01, is shown in figure 289. The macroscopic view of both areas of wear are shown in figure 290. Wear 1 covered a surface area of 0.6 square inches and resulted in a localized reduction in thickness of 19.3%. Wear 2 encompassed an area of 4 square inches and caused a maximum localized thickness loss of 44%.



40500-01 Wear WS 49.5-85.5 OV

Figure 286. Location of Wear 1 and 2 on the Right Wing Rear Spar Assembly Main Gear Forward WS 49.5 Through WS 85.5



40500-01 Wear 1 WS 49.5-85.5 Mac

Figure 287. Macroscopic View of Wear 1 on the Right Wing Rear Spar Assembly Main Gear Forward WS 49.5 Through WS 59.25

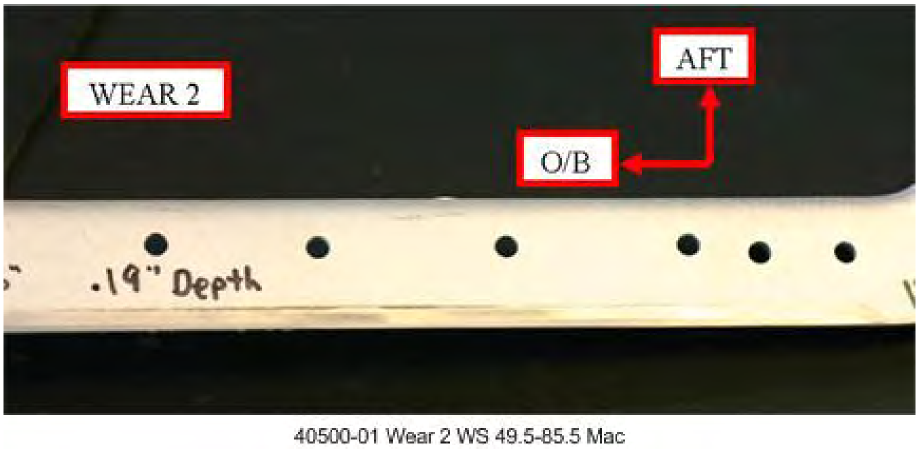


Figure 288. Macroscopic View of Wear 2 on the Right Wing Rear Spar Assembly Main Gear Forward WS 59 Through WS 85.5

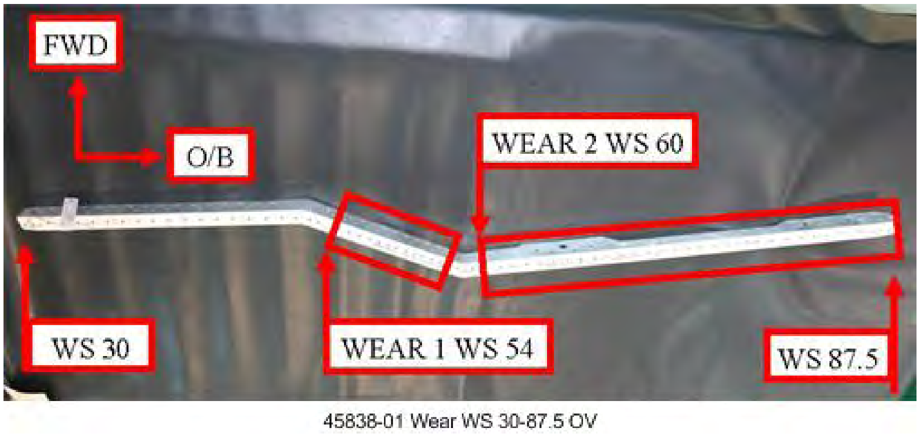


Figure 289. Location of Wear on the Right Wing Rear Spar Lower Aft Assembly WS 30 Through WS 87.5

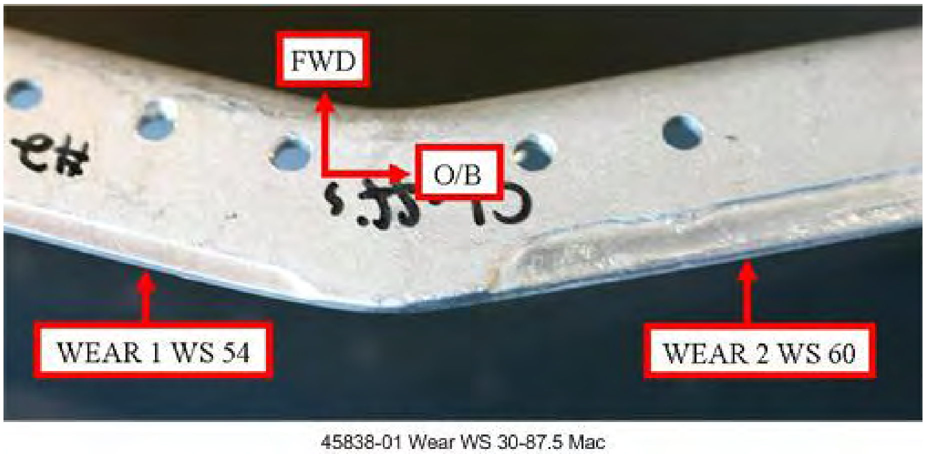


Figure 290. Macroscopic View of Wear on the Right Wing Rear Spar Lower Aft Assembly WS 54 and WS 60

The location of light-moderate scattered corrosion on the right wing rear spar bulkhead assembly main gear support forward, part number 40500-01B, is shown in figure 291. This corrosion caused a maximum localized reduction in thickness of 2.5%. Figure 292 shows the location of a crack and an area of wear on the right wing nacelle inboard plate skin attachment, part number 41663-01A, at WS 60. Figure 293 shows a macroscopic view of the crack and area of wear, and figure 294 shows the fluorescent liquid penetrant indication. A microscopic view of the crack and area of wear, measuring 0.4 inches in length, is shown in figure 295. The location of the right wing inboard upper rivet of the engine attachment is shown in figure 296, and the fracture face of this rivet is shown in figure 297. A fractograph, showing fatigue striations, is shown in figure 298.

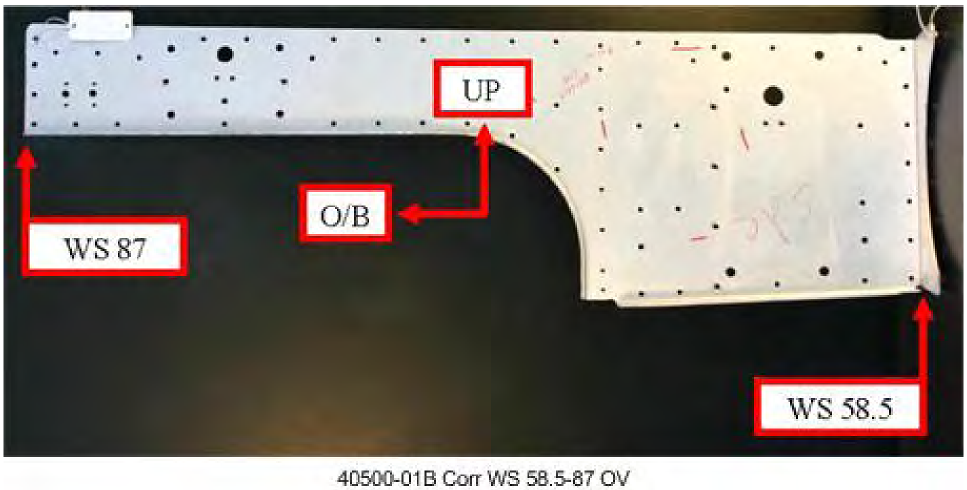
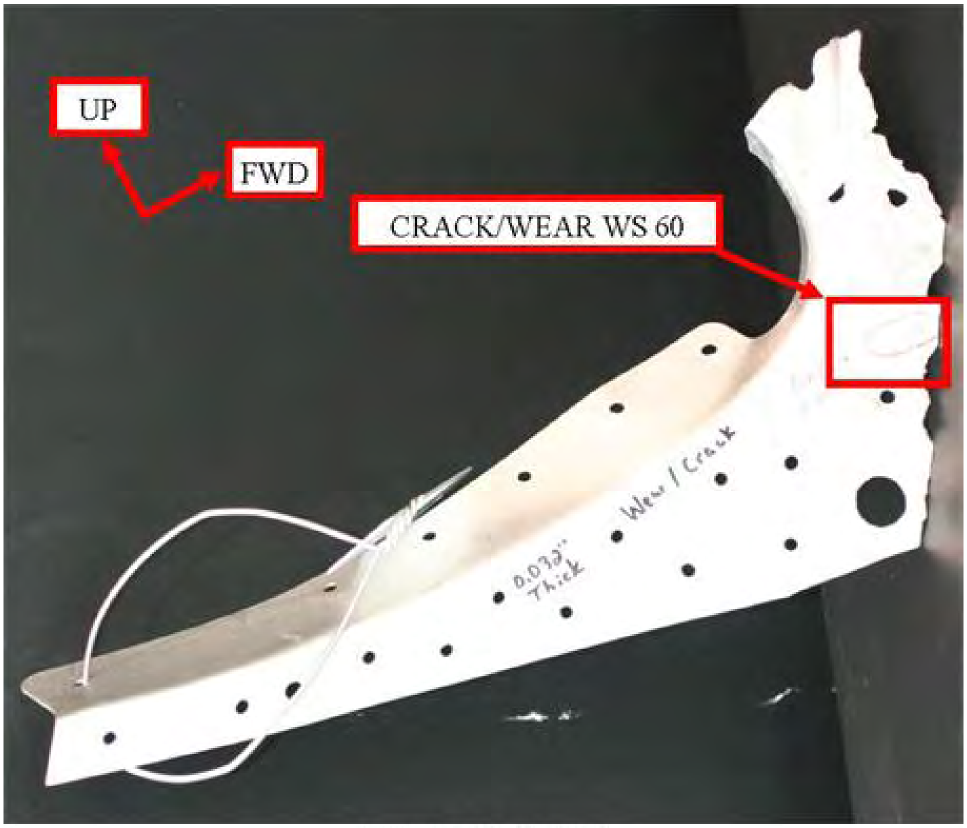
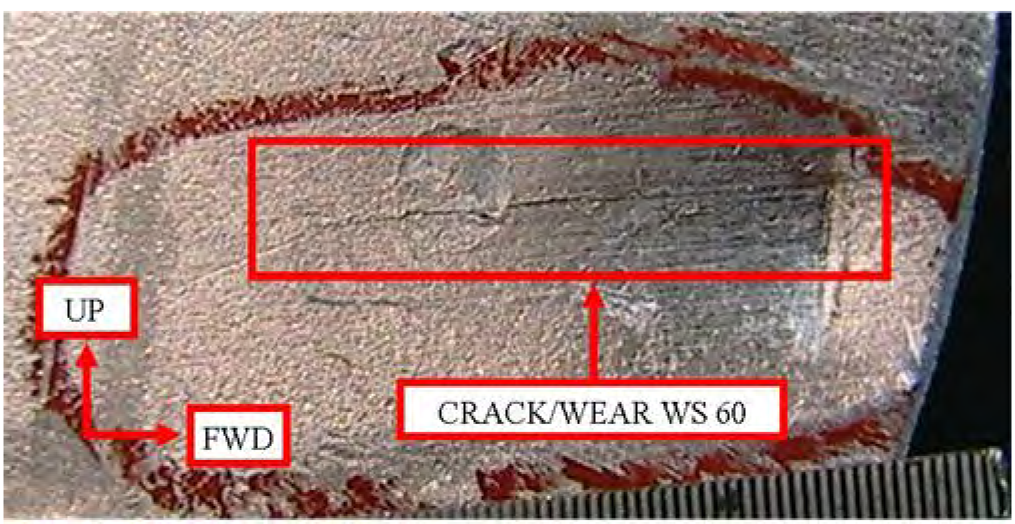


Figure 291. Location of Light-Moderate Corrosion on the Right Wing Rear Spar Bulkhead Assembly Main Gear Support Forward WS 58.5 Through WS 87



41663-01A CRK WS 60 OV

Figure 292. Location of Crack and Wear on the Right Wing Nacelle Inboard Plate Skin Attachment WS 60



41663-01A CRK WS 60 Mac

Figure 293. Macroscopic View of Crack and Wear on the Right Wing Nacelle Inboard Plate Skin Attachment

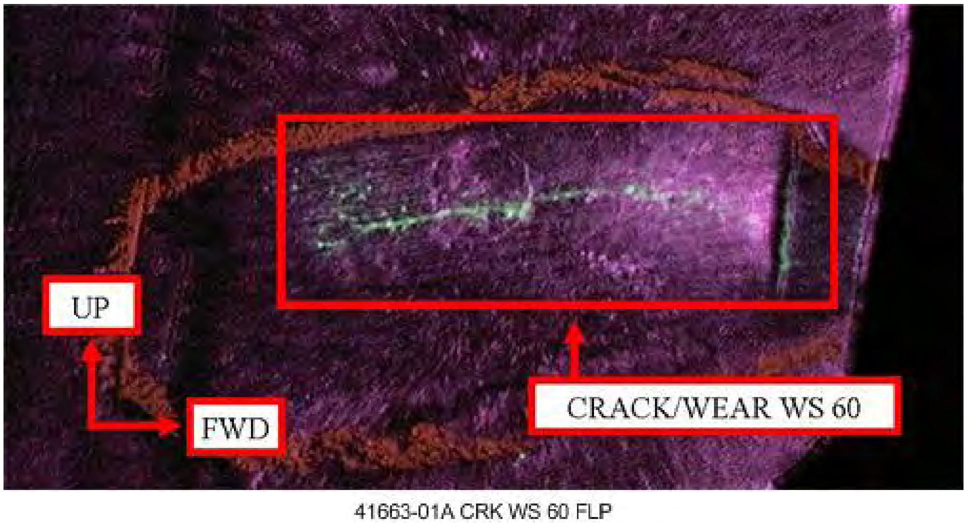


Figure 294. Fluorescent Liquid Penetrant Indication of Crack and Wear on the Right Wing Nacelle Inboard Plate Skin Attachment WS 60

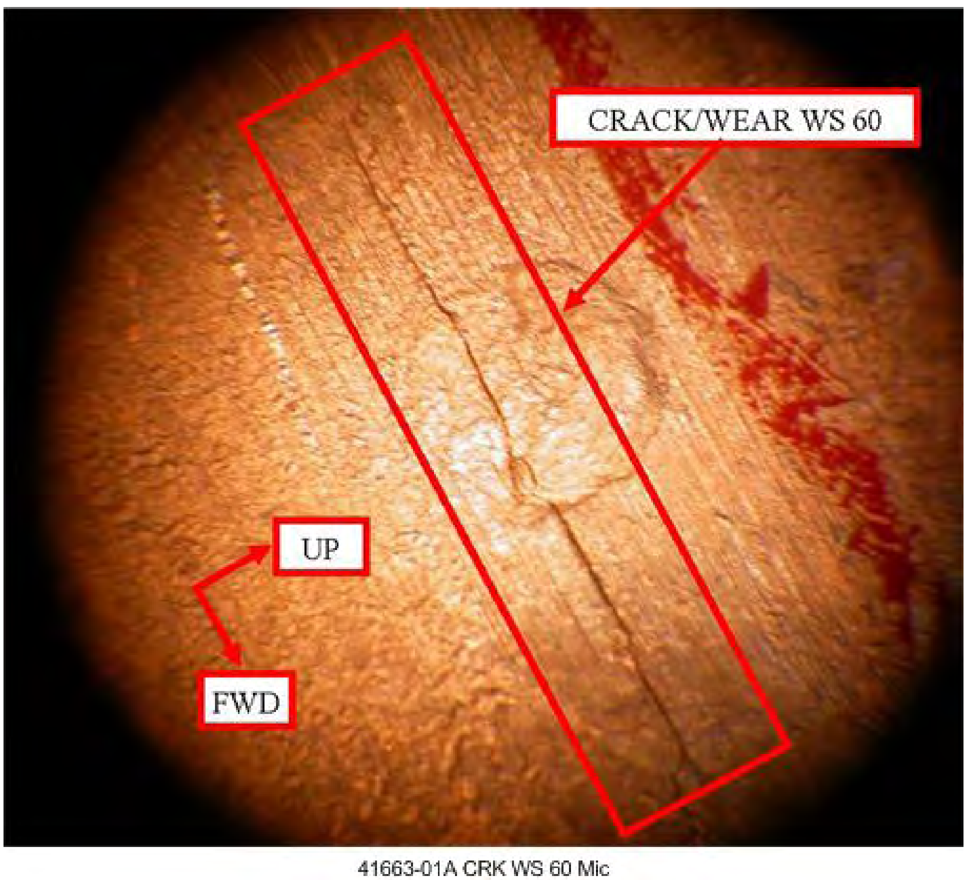


Figure 295. Microscopic View of Crack and Wear on the Right Wing Nacelle Inboard Plate Skin Attachment WS 60

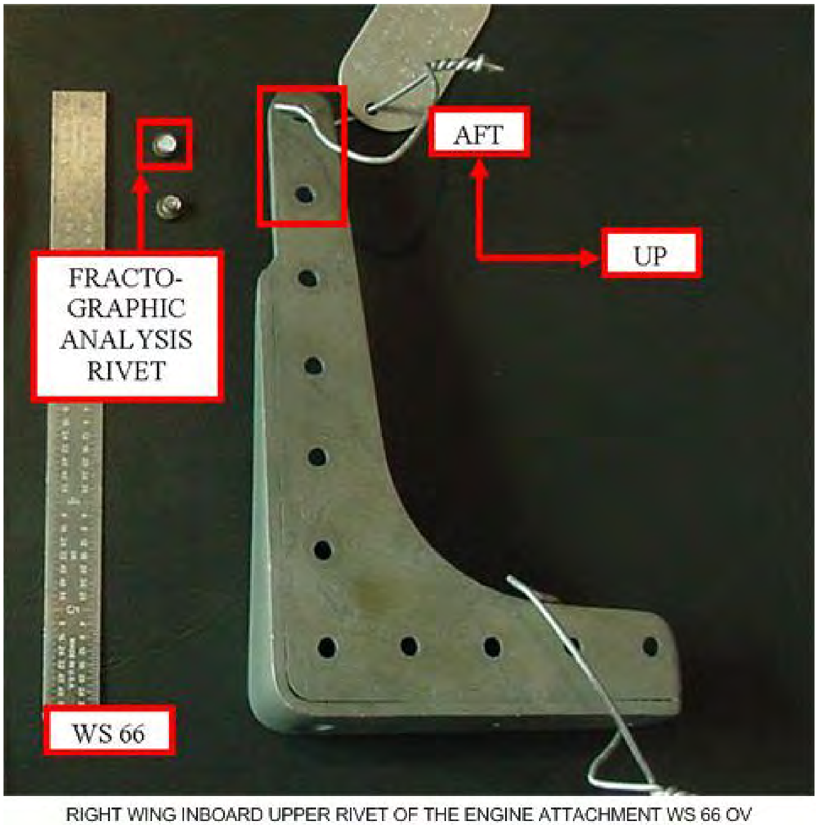


Figure 296. Location of Right Wing Inboard Upper Rivet of the Engine Attachment WS 66

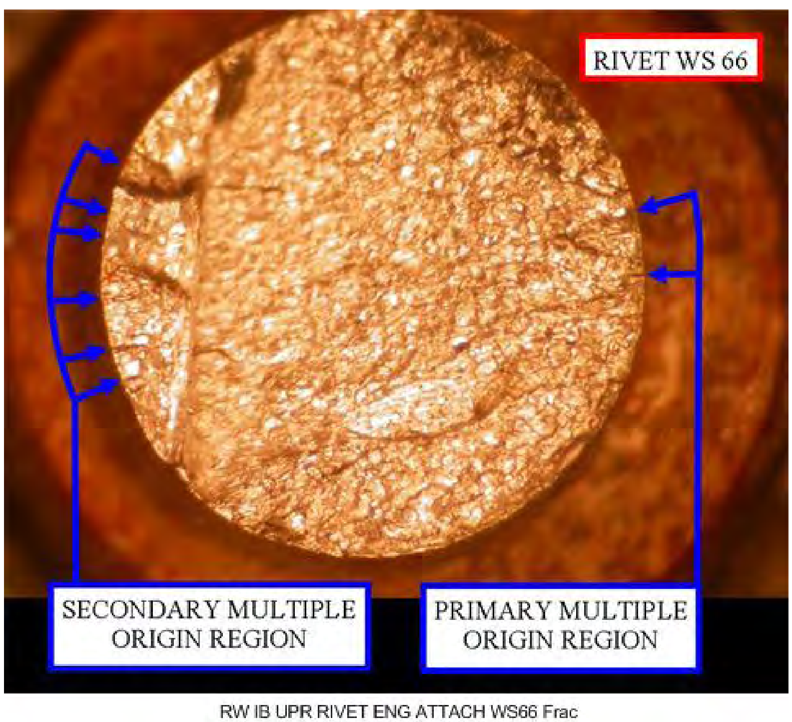


Figure 297. Fracture Face of Right Wing Inboard Upper Rivet of the Engine Attachment WS 66

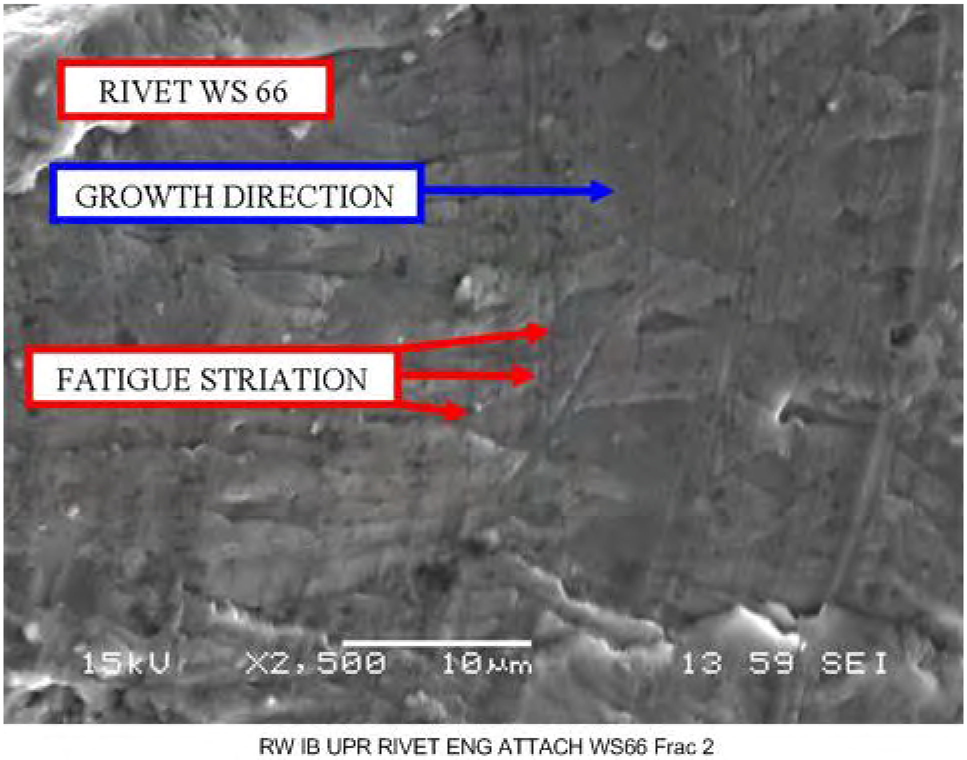
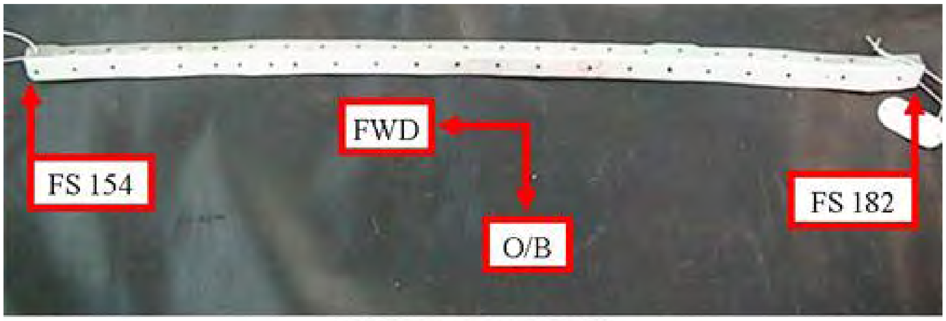


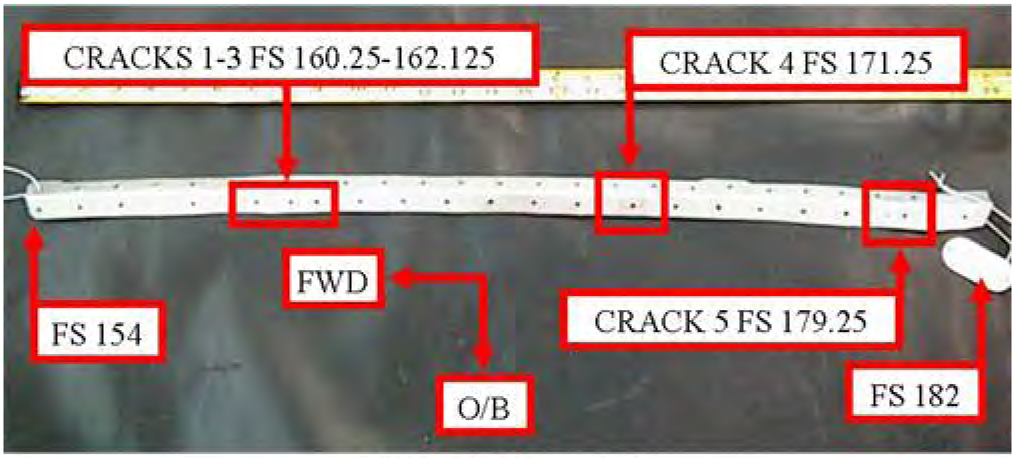
Figure 298. Fractograph of Right Wing Inboard Upper Rivet of the Engine Attachment WS 66

Figure 299 shows the location of light-moderate scattered corrosion observed on the right nacelle longitudinal inboard bulkhead aft piece, part number 44437-07, and figure 300 shows the location of five cracks found on the bulkhead aft piece. A macroscopic view of crack 1, which measured 0.369 inches in length, is shown in figure 301, and the fluorescent liquid penetrant indication is shown in figure 302. Figure 303 shows a microscopic view of crack 1, which was located at WS 66 FS 160.25 on the nacelle longitudinal inboard bulkhead aft piece. Figure 304 provides a macroscopic view of crack 2, which measured 0.608 inches in length and was located at WS 66 FS 161.25. The fluorescent liquid penetrant indication is shown in figure 305, and a microscopic view of this crack is shown in figure 306. A macroscopic view of crack 3, which measured 0.127 inch in length, on the right wing nacelle longitudinal inboard bulkhead aft piece is shown in figure 307. The fluorescent liquid penetrant indication of crack 3 is shown in figure 308, and a microscopic view of this crack is shown in figure 309. Figure 310 shows a macroscopic view of crack 4 at WS 66 FS 171.25, and figure 311 shows the fluorescent liquid penetrant indication of this 0.318-inch crack. Figure 312 shows a microscopic view of crack 4 on the inboard bulkhead aft piece. Crack 5, which measured 0.133 inches, is shown macroscopically in figure 313, and the fluorescent liquid penetrant indication is shown in figure 314. A microscopic view of crack 5, which was located at WS 66 FS 179.25 on the longitudinal inboard bulkhead aft piece, is shown in figure 315.



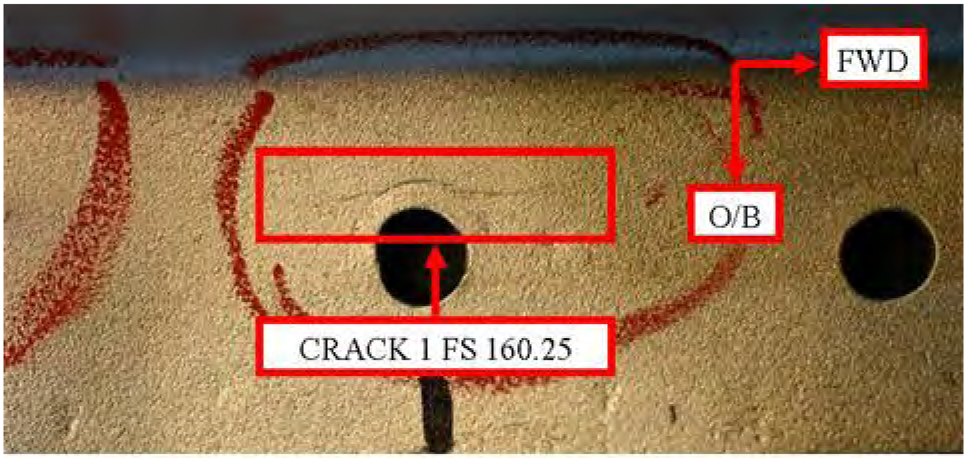
44437-07 Corr FS 154-182 OV

Figure 299. Location of Light-Moderate Corrosion on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 154 Through FS 182



44437-07 CRK FS154-182 OV

Figure 300. Location of Cracks on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 154 Through FS 182



44437-07 CRK 1 FS154-182 Mac

Figure 301. Macroscopic View of Crack 1 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 160.25

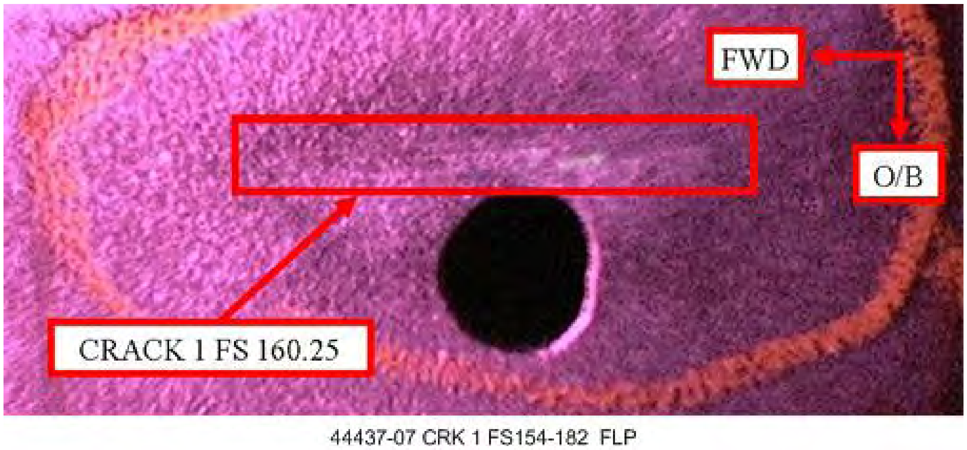


Figure 302. Fluorescent Liquid Penetrant of Crack 1 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 160.25

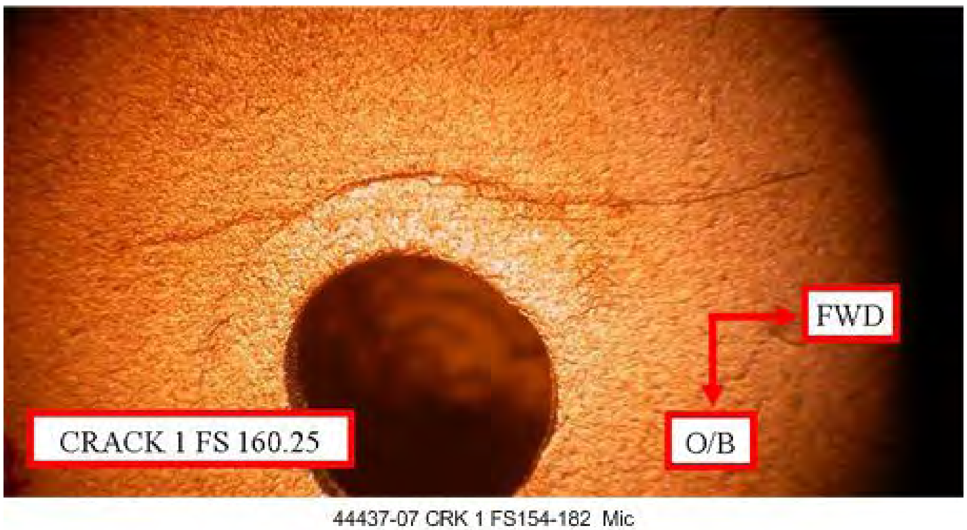


Figure 303. Microscopic View of Crack 1 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 160.25

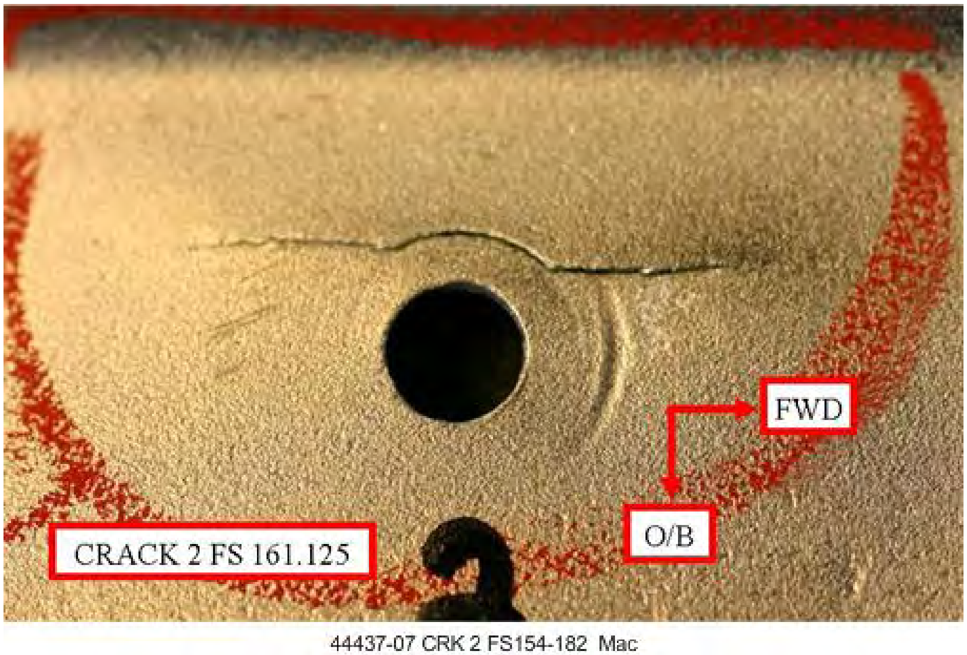


Figure 304. Macroscopic View of Crack 2 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 161.25

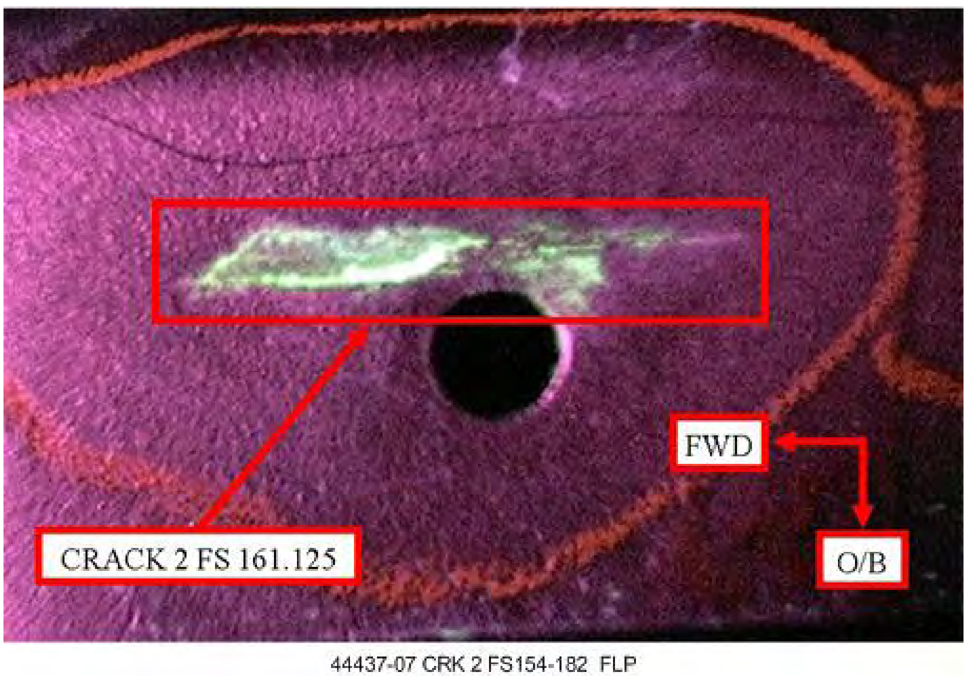


Figure 305. Fluorescent Liquid Penetrant Indication of Crack 2 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 161.25

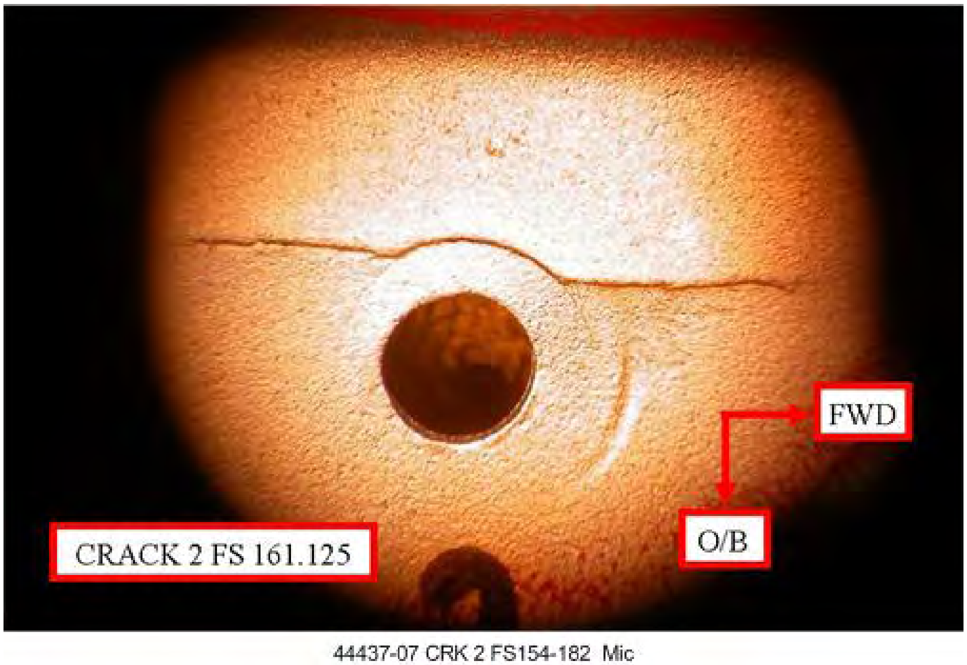


Figure 306. Microscopic View of Crack 2 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 161.25

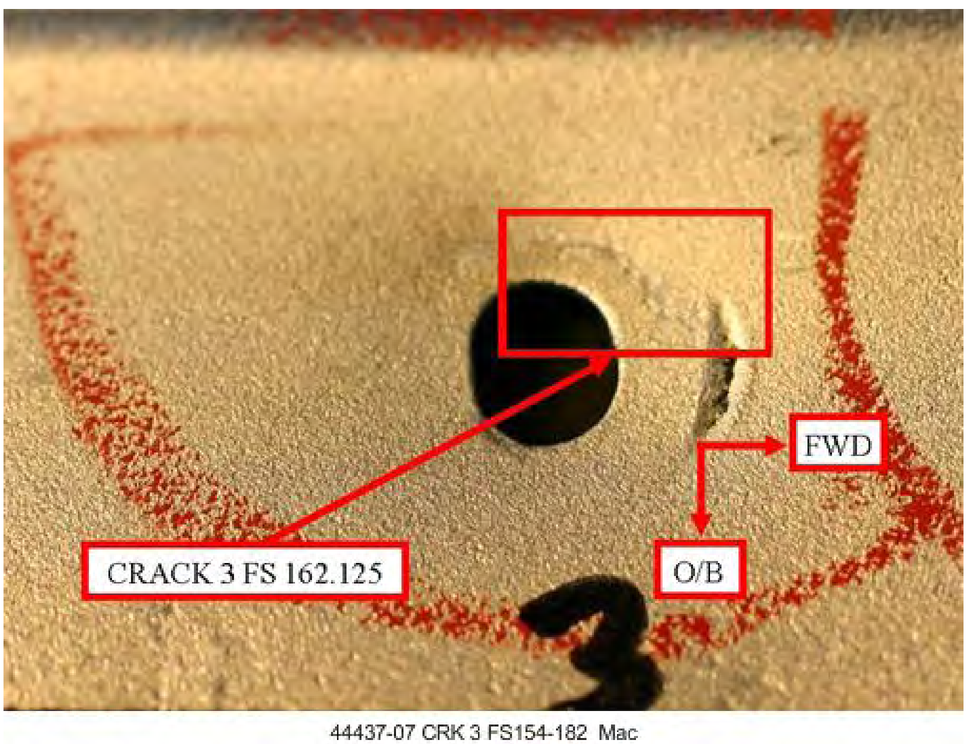
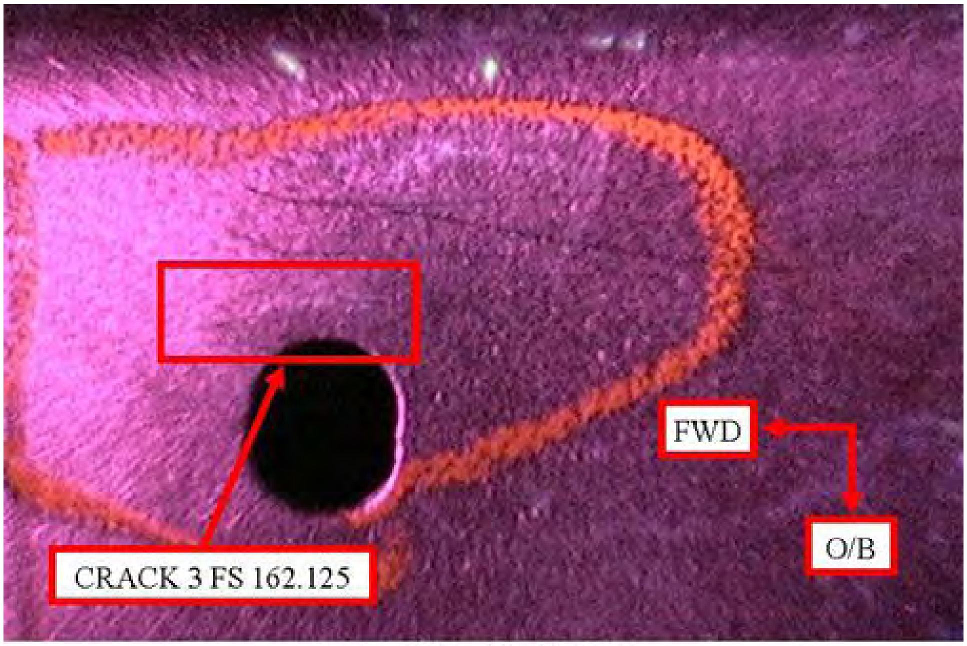
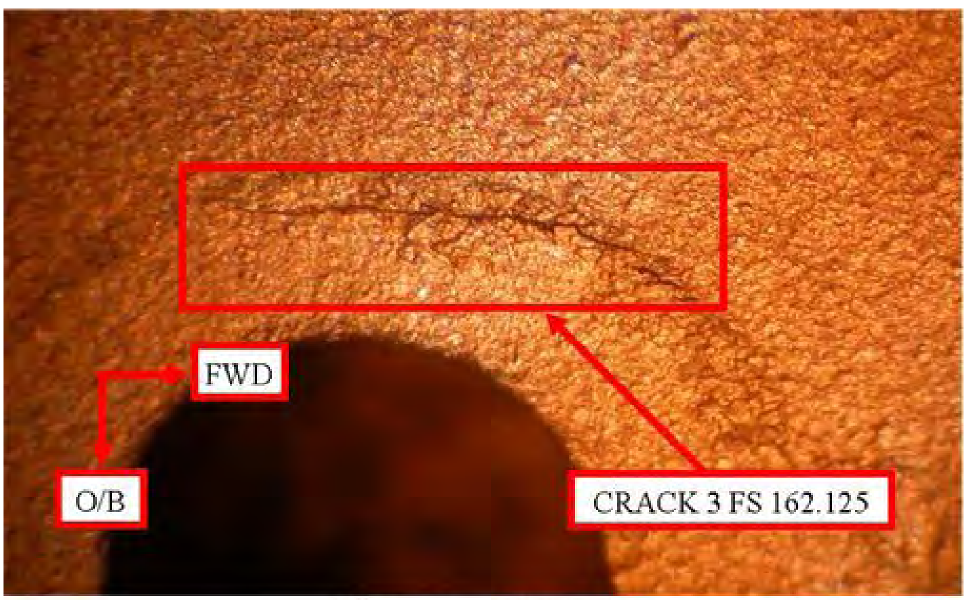


Figure 307. Macroscopic View of Crack 3 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 162.125



44437-07 CRK 3 FS154-182 FLP

Figure 308. Fluorescent Liquid Penetrant Indication of Crack 3 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 162.125



44437-07 CRK 3 FS154-182 Mic

Figure 309. Microscopic View of Crack 3 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 162.125

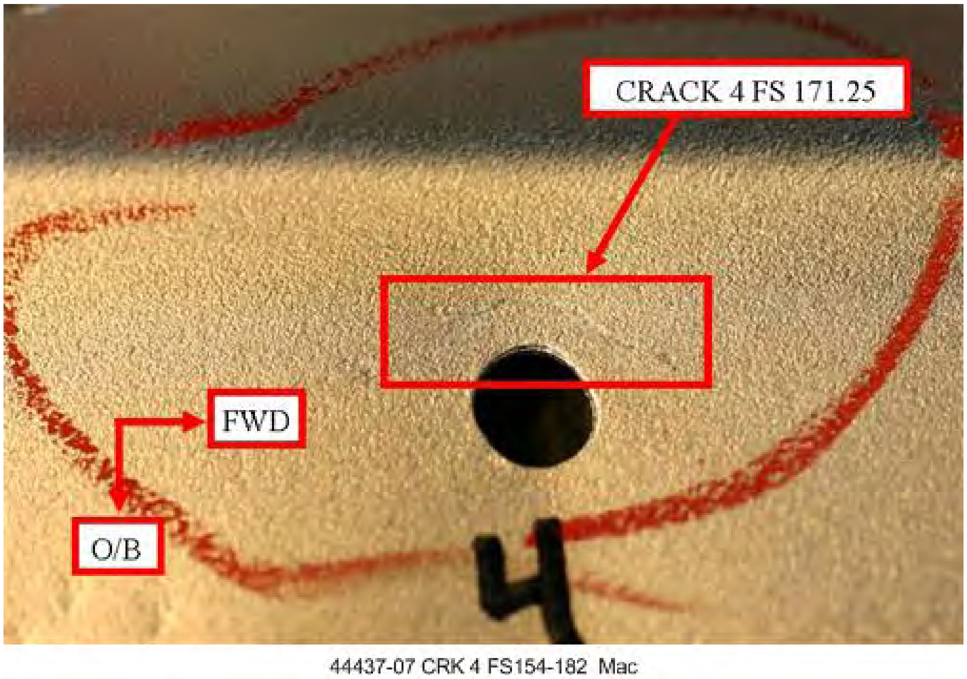


Figure 310. Macroscopic View of Crack 4 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 171.25

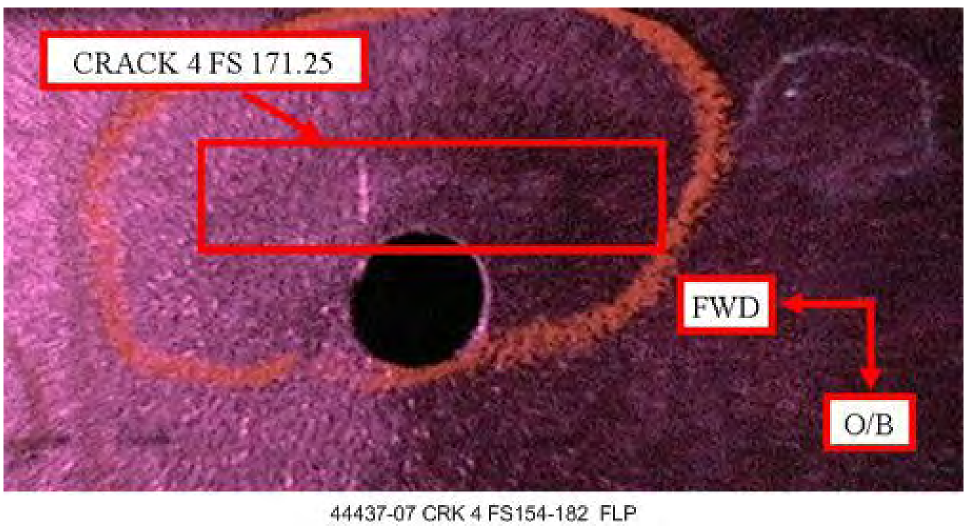


Figure 311. Fluorescent Liquid Penetrant Indication of Crack 4 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 171.25

Figure 312. Microscopic View of Crack 4 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 171.25

Figure 313. Macroscopic View of Crack 5 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 179.25

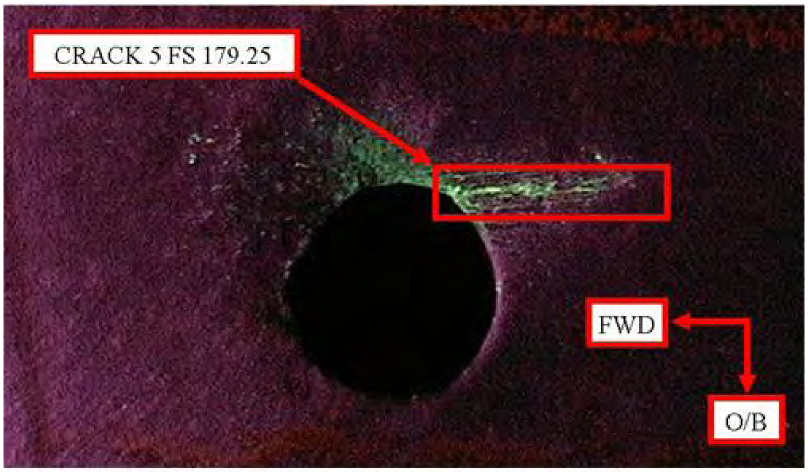
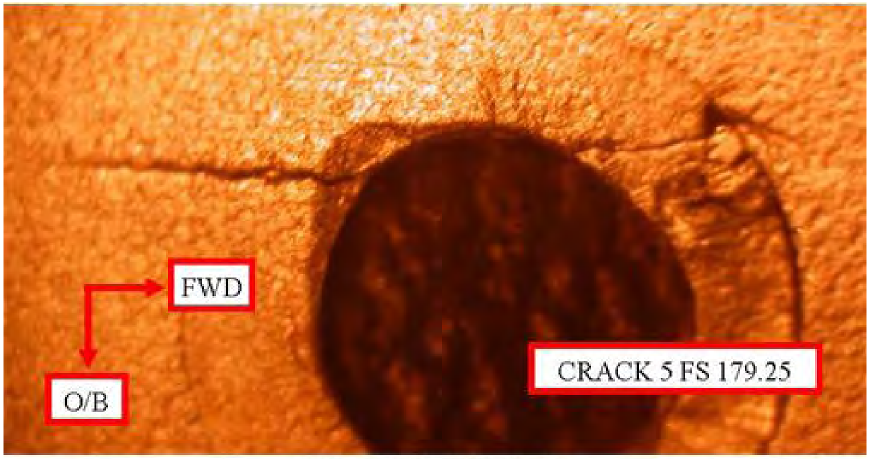


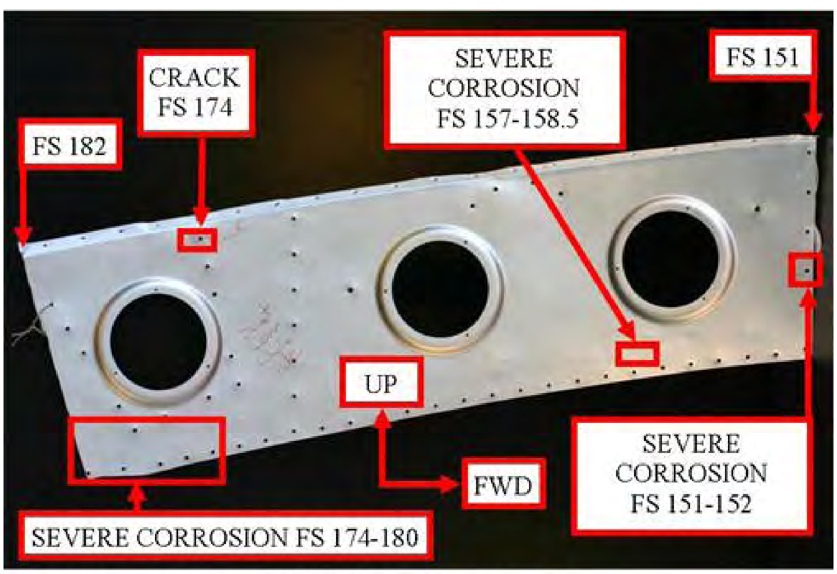
Figure 314. Fluorescent Liquid Penetrant Indication of Crack 5 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 179.25



44437-07 CRK 5 FS154-182 Mic

Figure 315. Microscopic View of Crack 5 on the Right Nacelle Longitudinal Inboard Bulkhead Aft Piece WS 66 FS 179.25

The location of a crack and three areas of severe corrosion are shown in figure 316 on the right nacelle inboard longitudinal bulkhead, part number 44437-07. Figure 317 shows a macroscopic view of severe corrosion located at WS 66 FS 151 to FS 152. This 1.25-square-inch area of corrosion caused a localized reduction in thickness of 22%. Another area of severe corrosion, located at WS 66 FS 157 to FS 158.5, is shown macroscopically in figure 318. This area of corrosion also caused a localized thickness loss of 22%. Figure 319 shows another area of severe corrosion on the right nacelle inboard longitudinal bulkhead at WS 66 FS 174 to FS 180. This 15-square-inch area of corrosion caused a maximum localized thickness loss of 22%. Figure 320 shows a macroscopic view of a 0.385-inch-crack located at WS 66 FS 174 on the nacelle inboard bulkhead, and figure 321 shows a microscopic view of this crack.



44437-07 Corr FS157-182 WS66 OV

Figure 316. Location of Crack and Severe Corrosion on the Right Nacelle Inboard Bulkhead WS 66 FS 151 Through FS 182

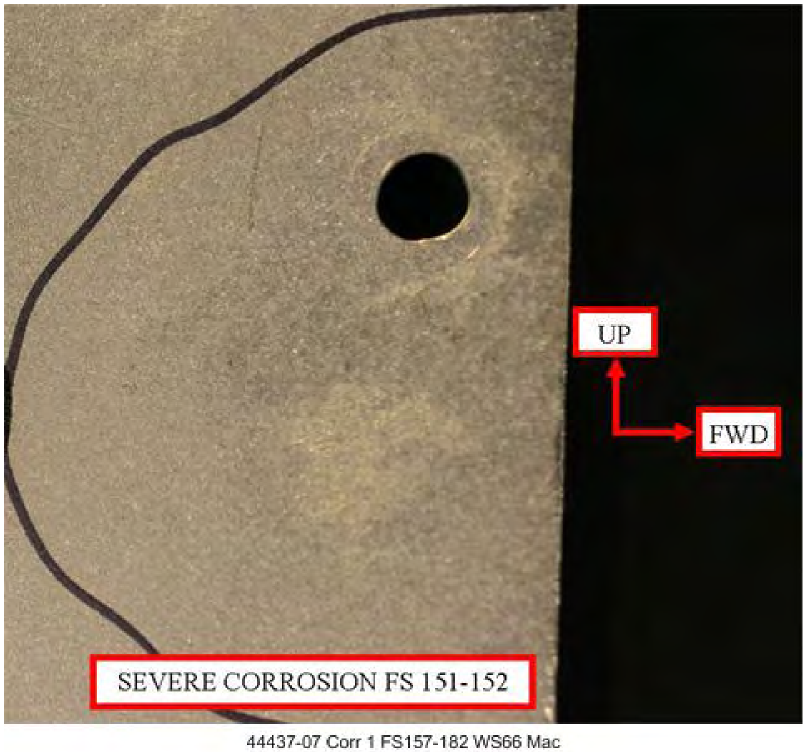


Figure 317. Macroscopic View of Severe Corrosion on the Right Nacelle Inboard Longitudinal Bulkhead WS 66 FS 151 Through FS 152

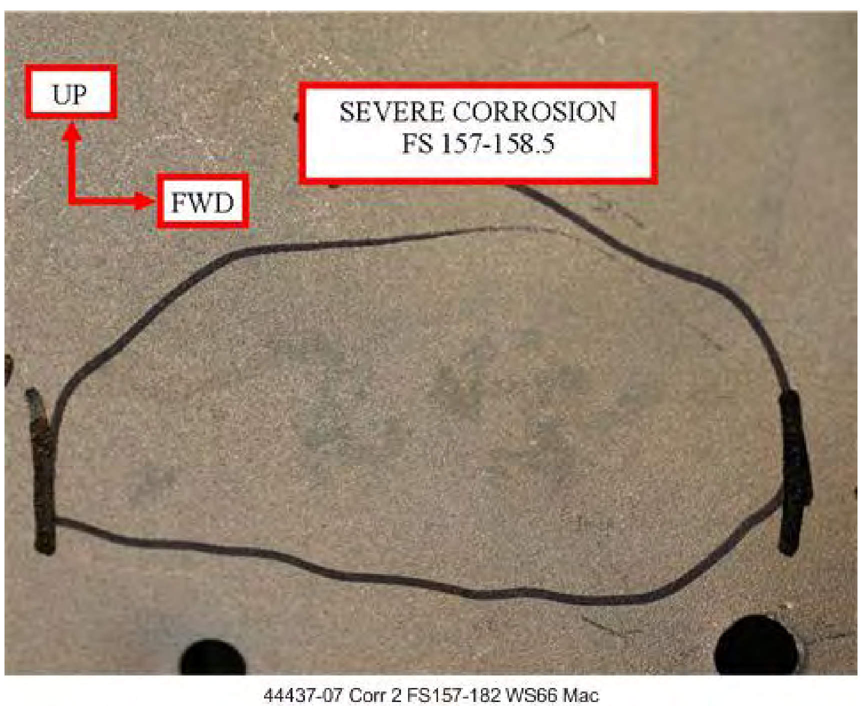


Figure 318. Macroscopic View of Severe Corrosion on the Right Nacelle Inboard Longitudinal Bulkhead WS 66 FS 157 Through FS 158.5

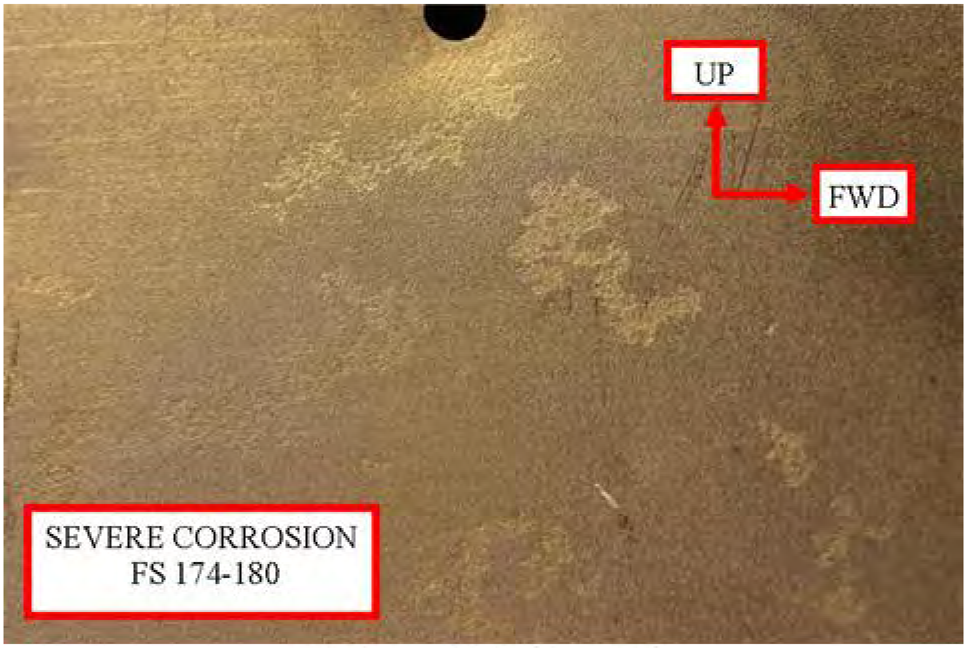


Figure 319. Macroscopic View of Severe Corrosion on the Right Nacelle Inboard Longitudinal Bulkhead WS 66 FS 174 Through FS 180

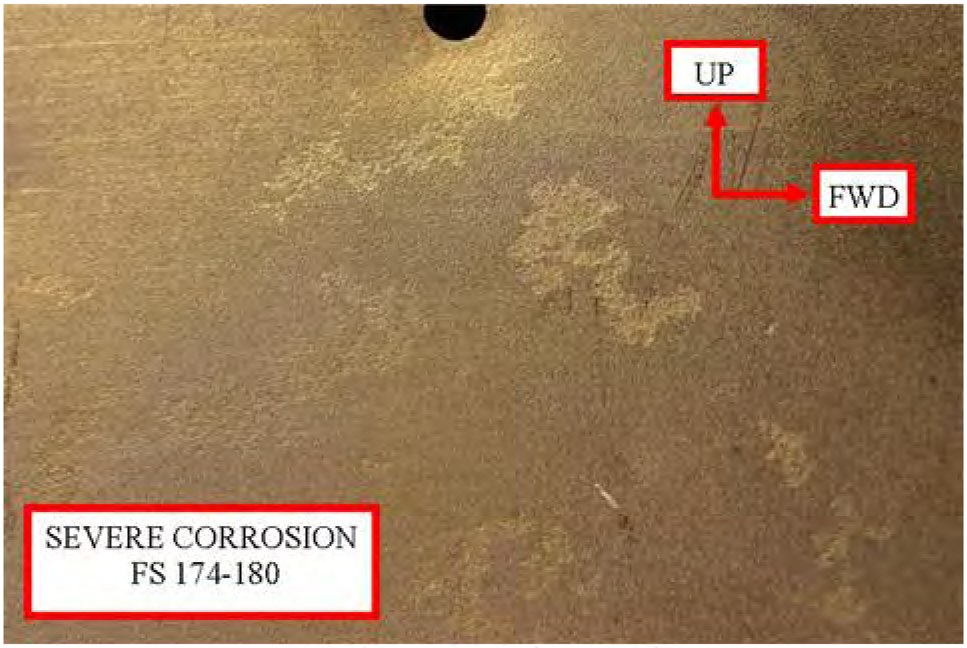


Figure 320. Macroscopic View of Crack on the Right Nacelle Inboard Longitudinal Bulkhead WS 66 FS 174

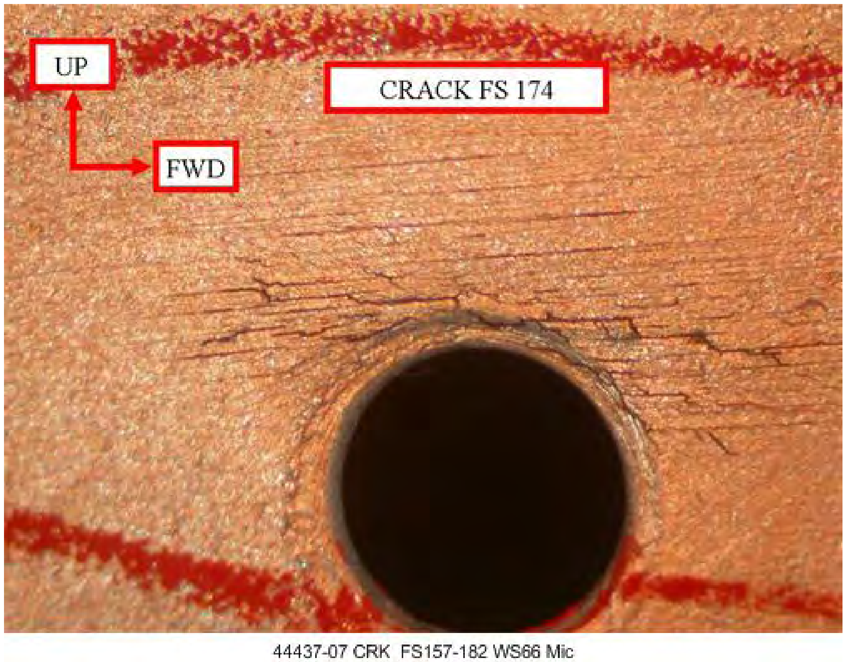


Figure 321. Microscopic View of Crack on the Right Nacelle Inboard Longitudinal Bulkhead WS 66 FS 174

Figure 322 shows the location of light-moderate corrosion on the right nacelle bulkhead longitudinal inboard, part number 44437-07. This corrosion caused a localized reduction in thickness of 4%. The location of light-moderate scattered corrosion on the right wing nacelle bulkhead assembly, part number 44474-06A, is shown in figure 323. This corrosion caused a maximum localized thickness loss of 3%. Figure 324 shows the location of a 0.21-inch-crack on the right wing nacelle torque tube bracket assembly, part number 41815, located from WS 66 to WS 89 FS 111.5. A macroscopic view of this crack is shown in figure 325, and the fluorescent liquid penetrant indication of this crack is shown in figure 326. A microscopic view of the crack on the torque tube bracket is shown in figure 327.

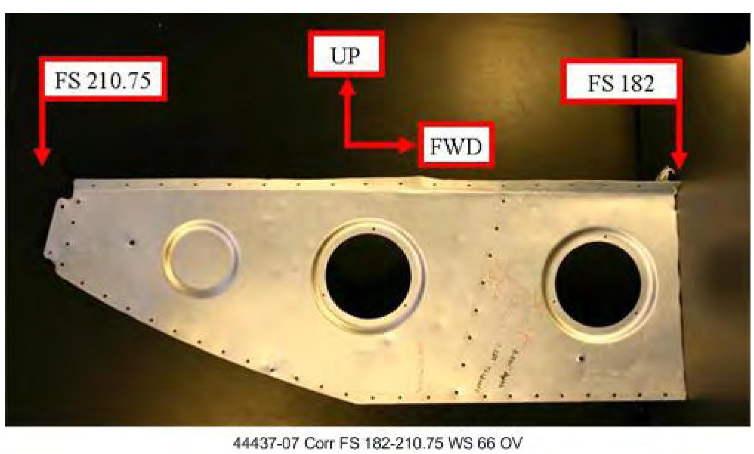
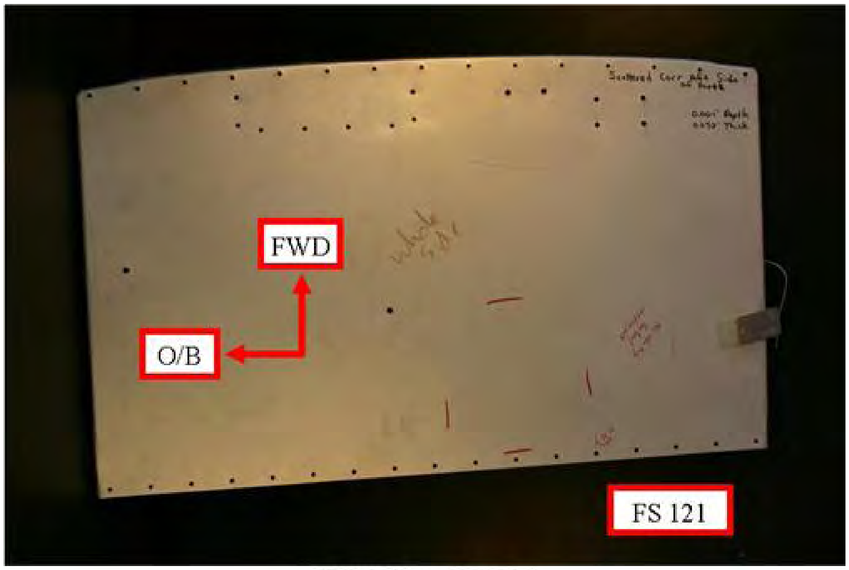
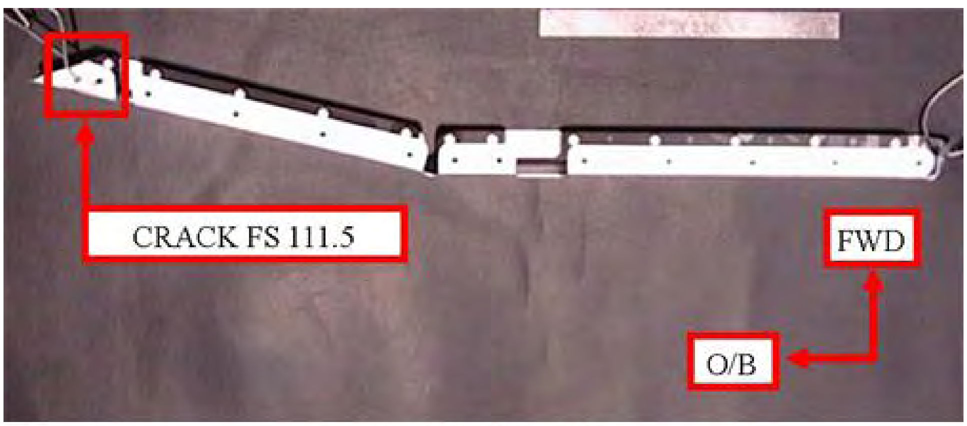


Figure 322. Location of Light-Moderate Corrosion on the Right Nacelle Bulkhead Longitudinal Inboard WS 66 FS 182 Through FS 210.75



44474-06A Corr FS 121 OV

Figure 323. Location of Light-Moderate Corrosion on the Right Wing Nacelle Bulkhead Assembly WS 66 Through WS 89 FS 121



41815 CRK FS 111.5 OV

Figure 324. Location of the Crack on the Right Wing Nacelle Torque Tube Bracket Assembly WS 66 Through WS 89 FS 111.5

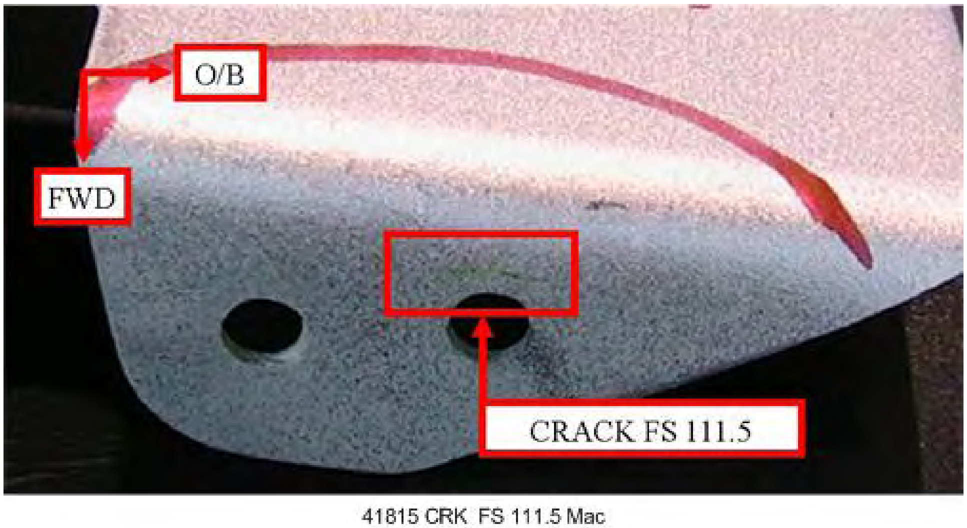


Figure 325. Macroscopic View of Crack on the Right Wing Nacelle Torque Tube Bracket Assembly FS 111.5

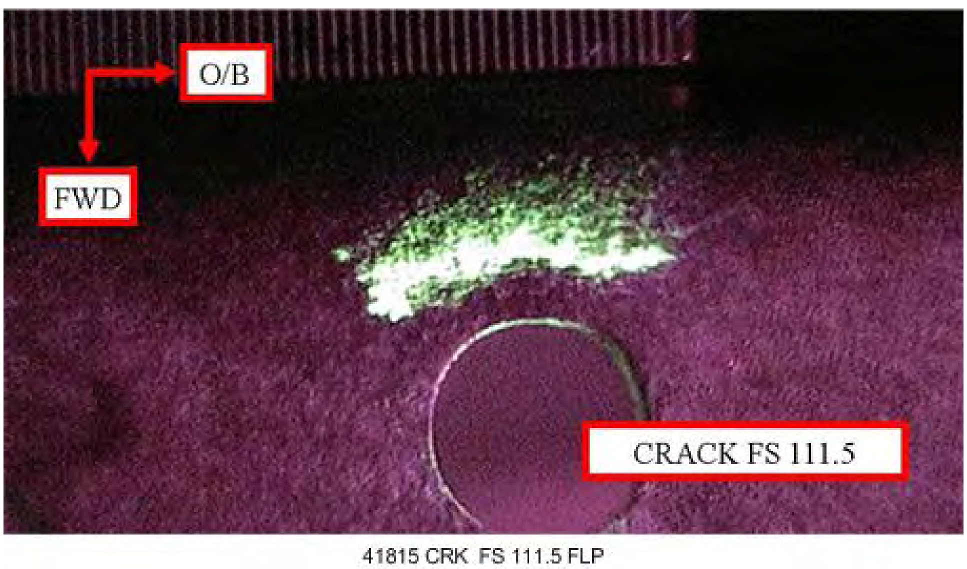


Figure 326. Fluorescent Liquid Penetrant Indication on the Right Wing Nacelle Torque Tube Bracket Assembly FS 111.5

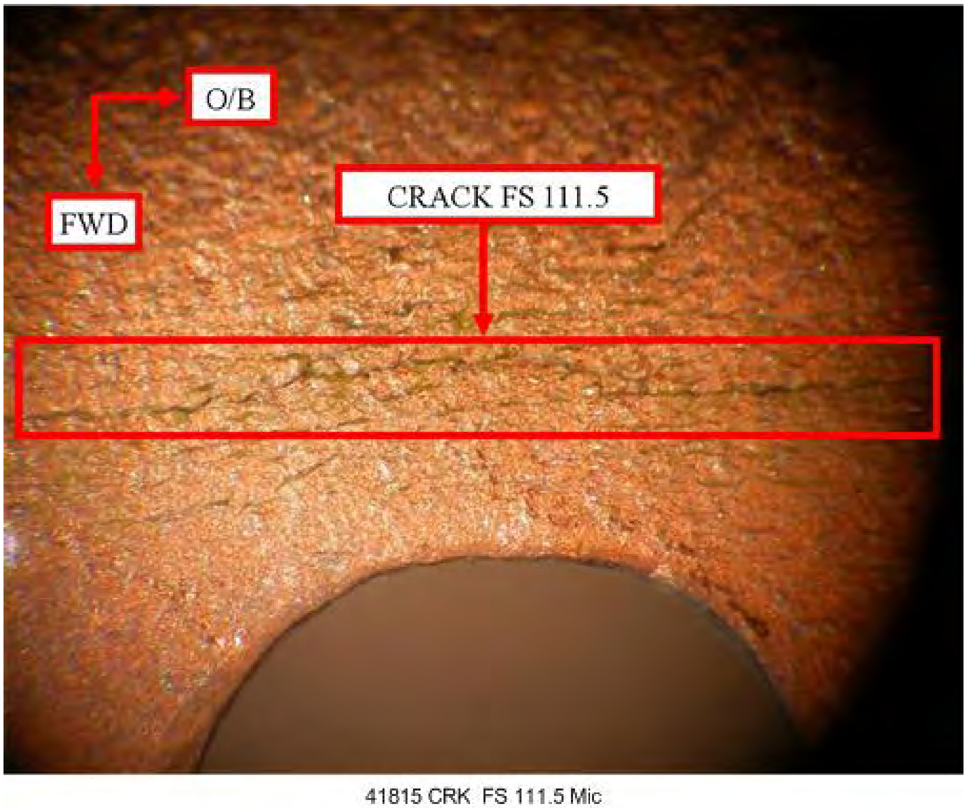


Figure 327. Microscopic View of Crack on the Right Wing Nacelle Torque Tube Bracket Assembly FS 111.5

The location of two areas of wear on the right wing nacelle plate assembly firewall shear, part number 40671-01A, is shown in figure 328. A macroscopic view of wear 1 is shown in figure 329. This 1.14-square-inch area of wear was located at WS 71 and caused a maximum reduction in thickness of 60%. Figure 330 shows a macroscopic view of wear 2 on the firewall shear at WS 82. This 0.44-square-inch area of corrosion caused a maximum thickness loss of 12.5%. Figure 331 shows the location of two cracks on the right wing trunnion landing gear aft, part number 40288-00A, at WS 77.5. A macroscopic view of crack 1, measuring 0.210 inch, is shown in figure 332, and the fluorescent liquid penetrant indication of this crack is shown in figure 333. Figure 334 shows a macroscopic view of crack 2, and figure 335 shows the fluorescent liquid penetrant indication for this 0.172-inch-long crack. A microscopic view of crack 2 is shown in figure 336.

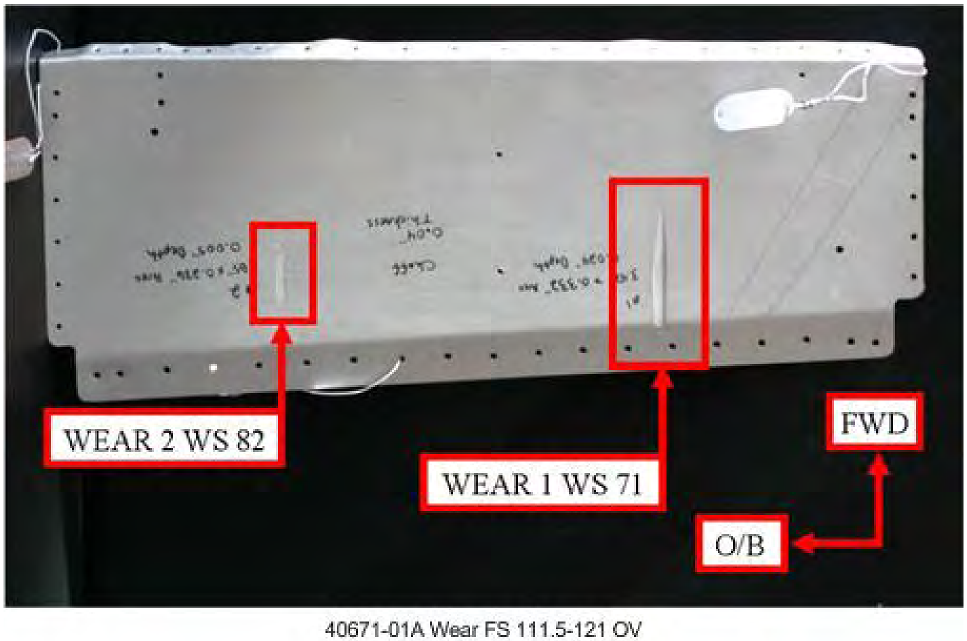


Figure 328. Location of Wear 1 and 2 on the Right Wing Nacelle Plate Assembly Firewall Shear WS 71 and WS 82

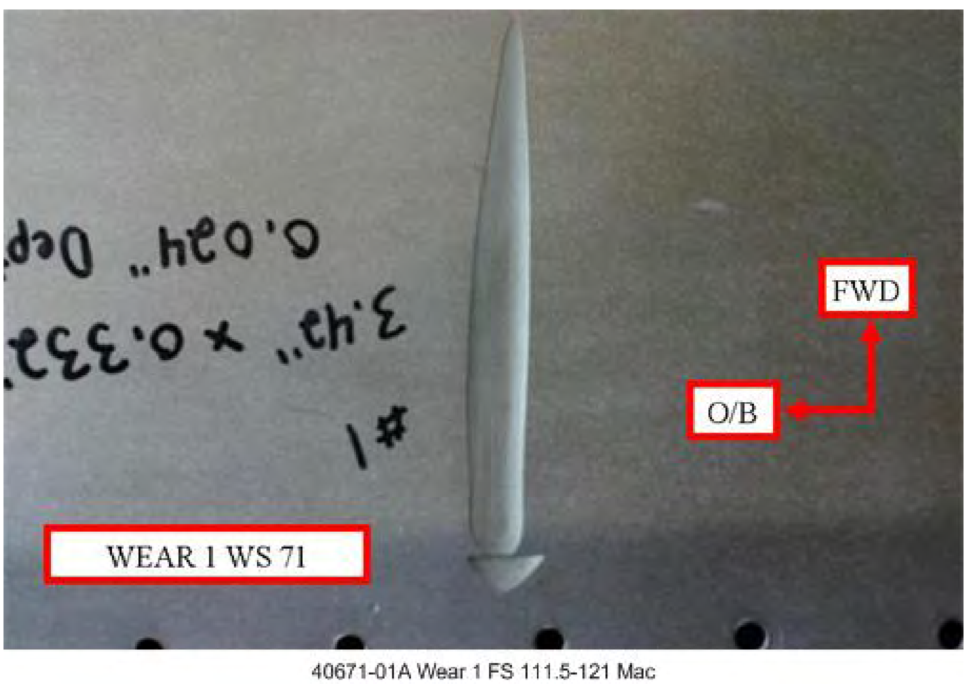
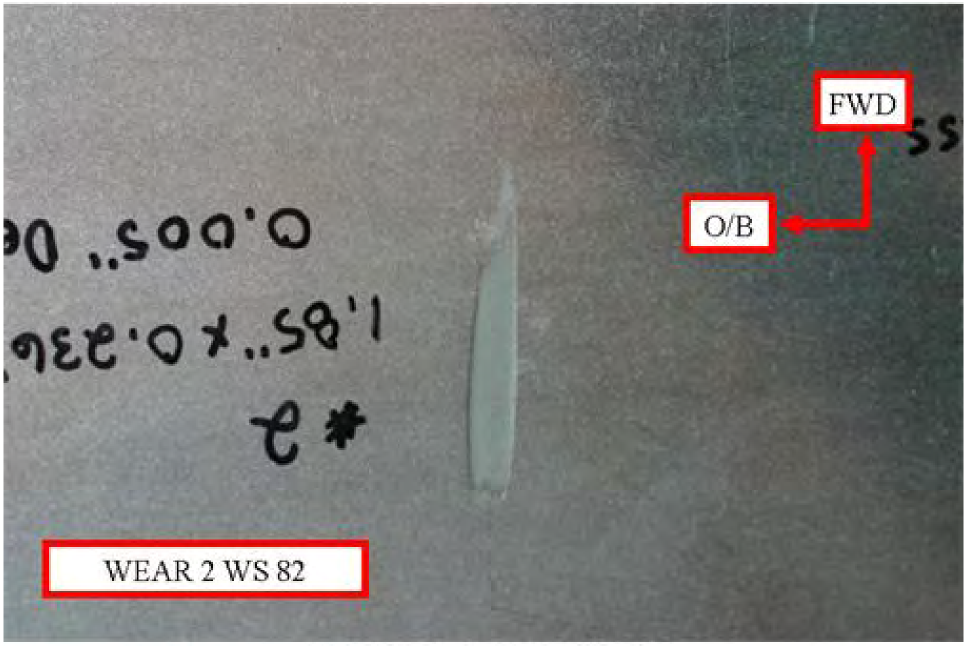
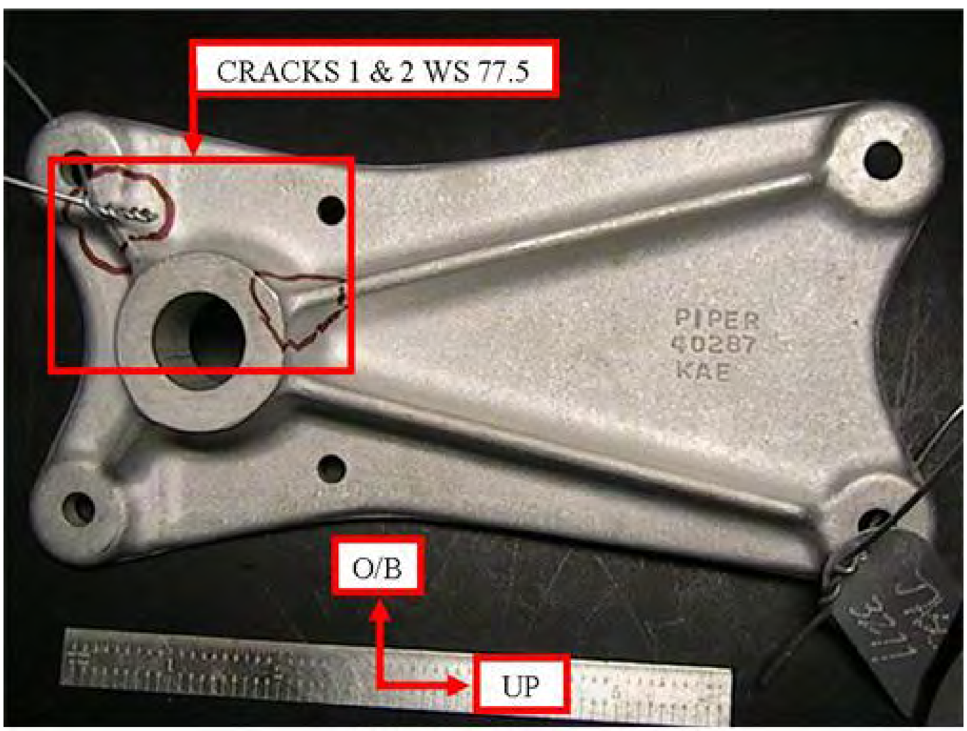


Figure 329. Macroscopic View of Wear 1 on the Right Wing Nacelle Plate Assembly Firewall Shear WS 71



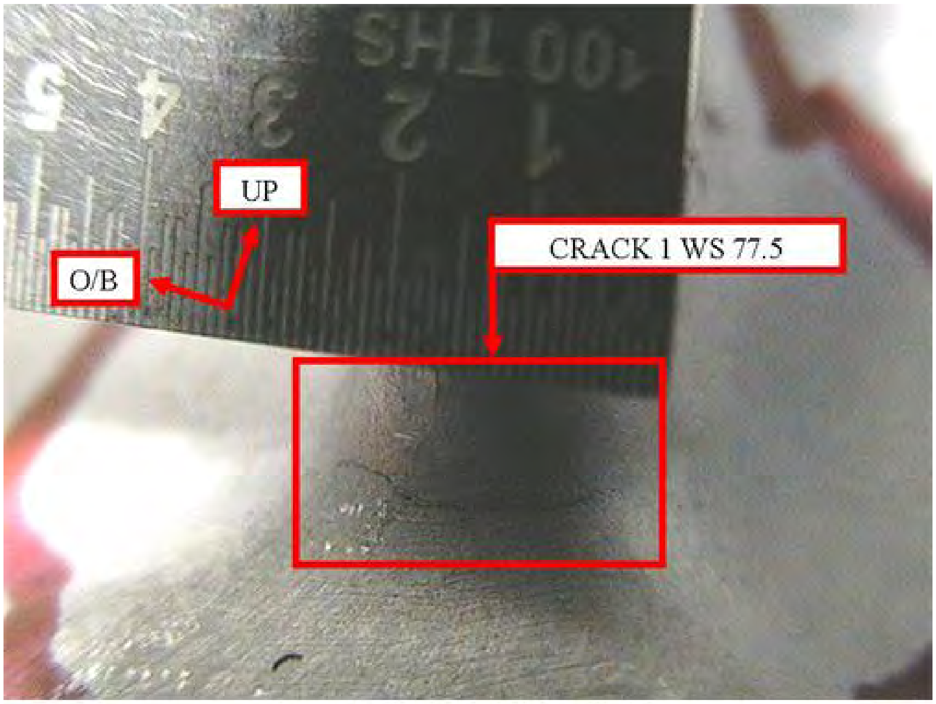
40671-01A Wear 2 FS 111.5-121 Mac

Figure 330. Macroscopic View of Wear 2 on the Right Wing Nacelle Plate Assembly Firewall Shear WS 82



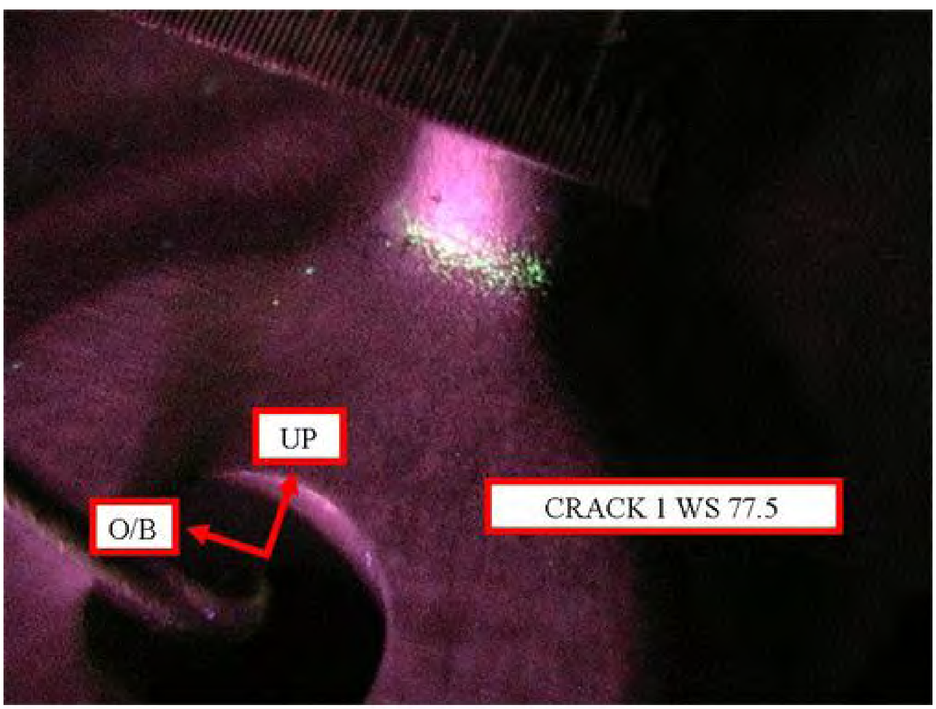
40288-00A CRK WS 77.5 OV

Figure 331. Location of Cracks 1 and 2 on the Right Wing Trunnion Landing Gear Aft WS 77.5



40288-00A CRK 1 WS 77.5 Mac

Figure 332. Macroscopic View of Crack 1 on the Right Wing Trunnion Landing Gear Aft WS 77.5



40288-00A CRK 1 WS 77.5 FLP

Figure 333. Fluorescent Liquid Penetrant Indication of Crack 1 on the Right Wing Trunnion Landing Gear Aft WS 77.5

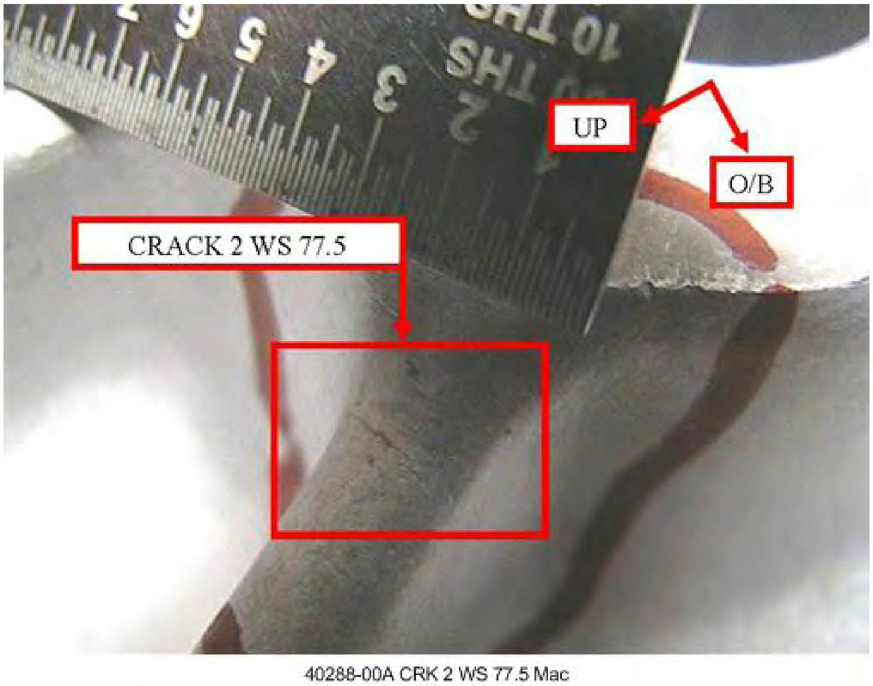


Figure 334. Macroscopic View of Crack 2 on the Right Wing Trunnion Landing Gear Aft WS 77.5

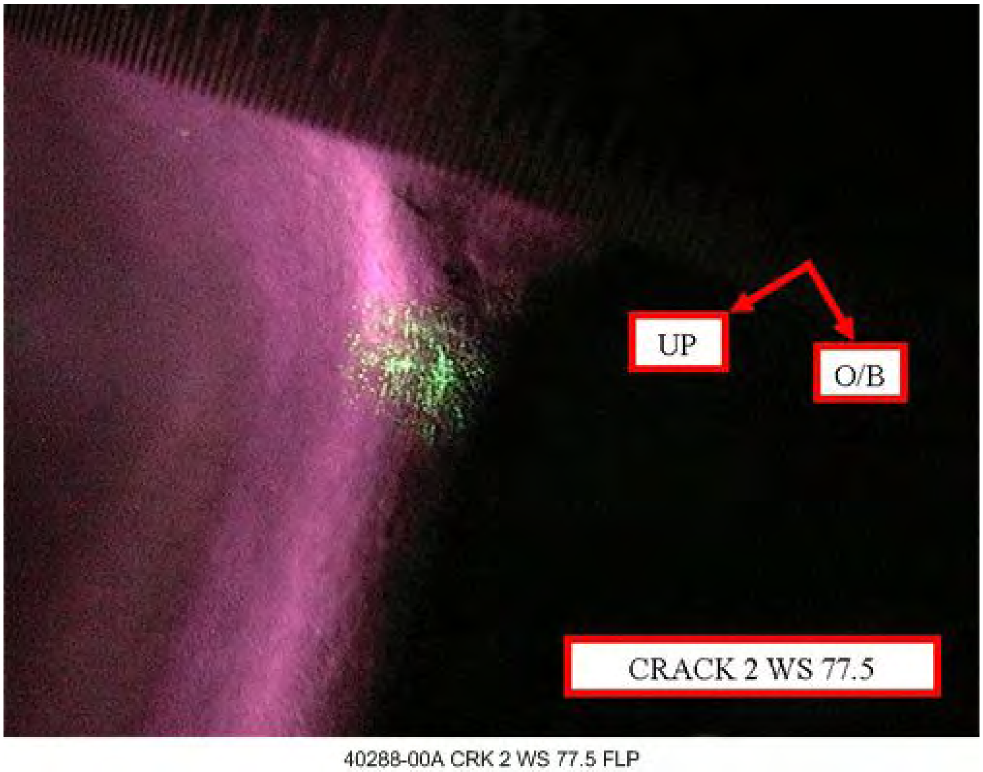
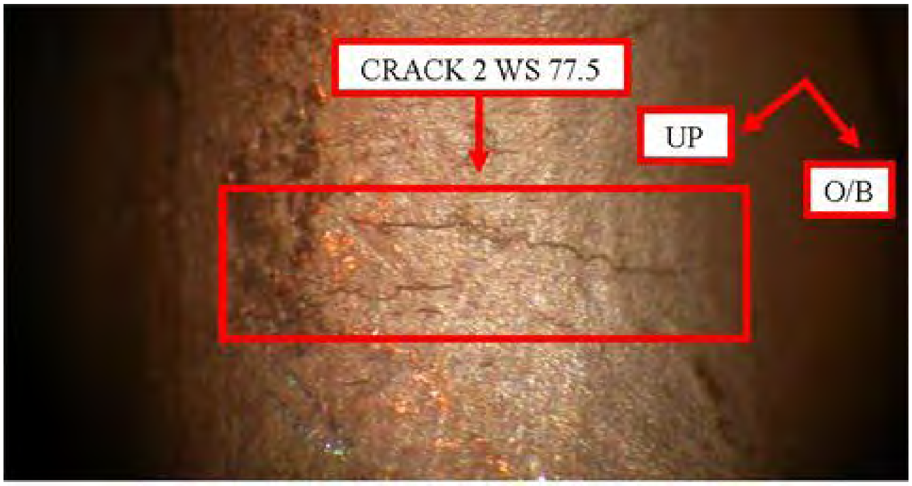


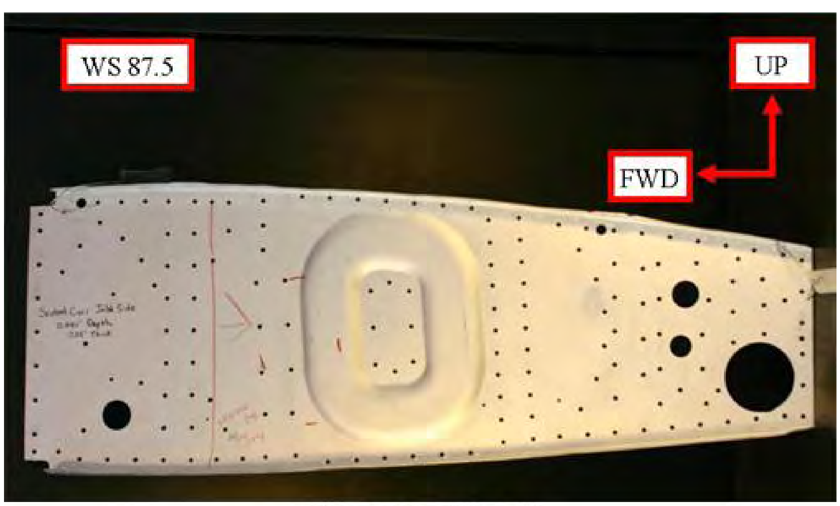
Figure 335. Fluorescent Liquid Penetrant Indication of Crack 2 on the Right Wing Trunnion Landing Gear Aft WS 77.5



40288-00A CRK 2 WS 77.5 Mic

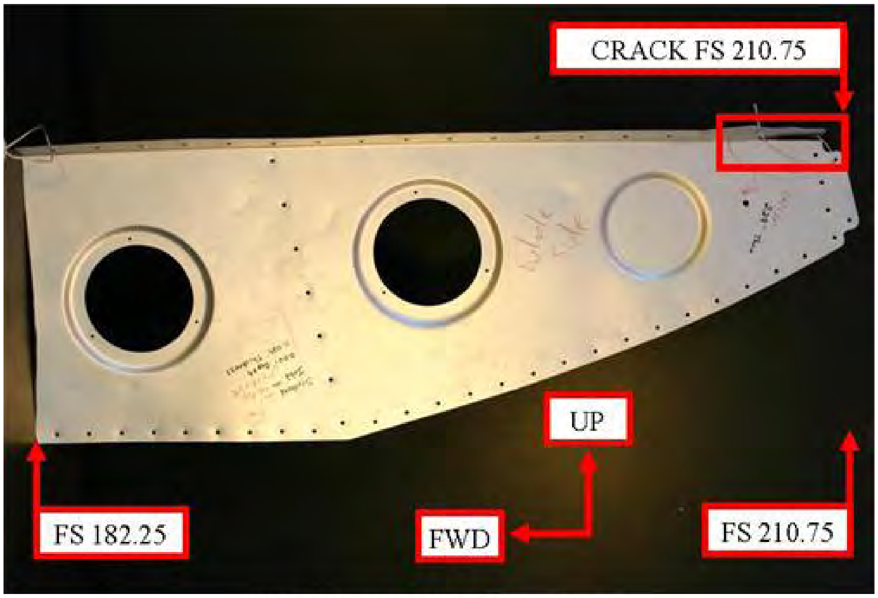
Figure 336. Microscopic View of Crack 2 on the Right Wing Trunnion Landing Gear Aft WS 77.5

The location of light scattered corrosion, causing a maximum thickness loss of 2%, on the right wing rear spar rib assembly, part number 40455-25B, is shown in figure 337 at WS 87.5. Figure 338 shows the location of a crack and light-moderate scattered corrosion on the right wing nacelle outboard longitudinal bulkhead, part number 44437-07. The light-moderate corrosion was scattered at WS 89 FS 182.25 to FS 210.75 and resulted in a maximum reduction in thickness of 4%. A macroscopic view of the 2.29-inch crack located at WS 89 FS 210.75 is shown in figure 339. The location of a 0.047-inch crack on the right wing flap track, part number 45381-15B, is shown in figure 340, and a macroscopic view of this crack is shown in figure 341. The fluorescent liquid penetrant indication of this crack, located at WS 147.5, is shown in figure 342. Figure 343 shows the location of light scattered corrosion on the right aileron hinge, part number 45384-00. This corrosion caused a localized thickness loss of less than 1%.



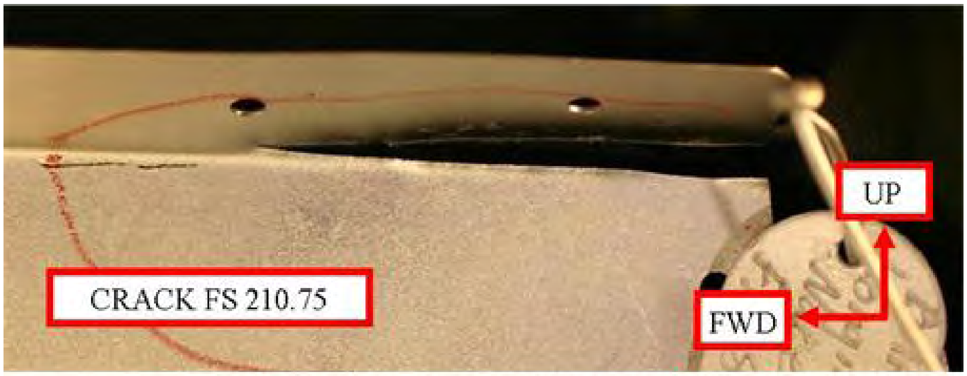
40455-25B Corr WS 87.5 OV

Figure 337. Location of Light Corrosion on the Right Wing Rear Spar Rib Assembly WS 87.5



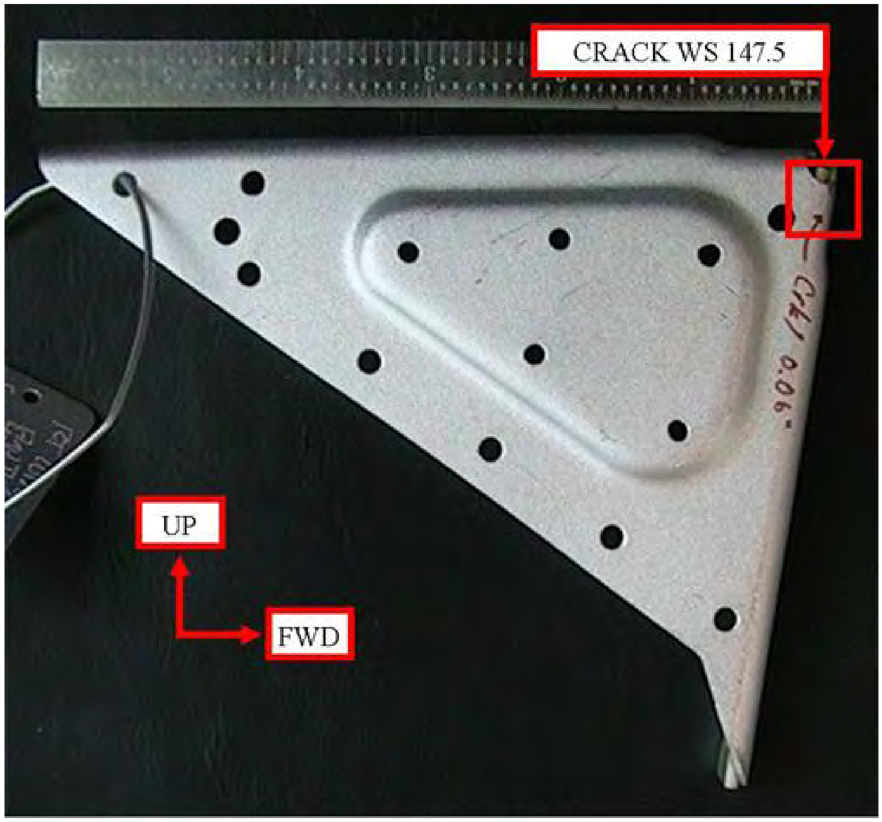
44437-07 Corr FS 182.25-210.75 WS 89 OV

Figure 338. Location of Crack and Overview of Right Wing Nacelle Outboard Longitudinal Bulkhead WS 89 FS 182.25 Through FS 210.75



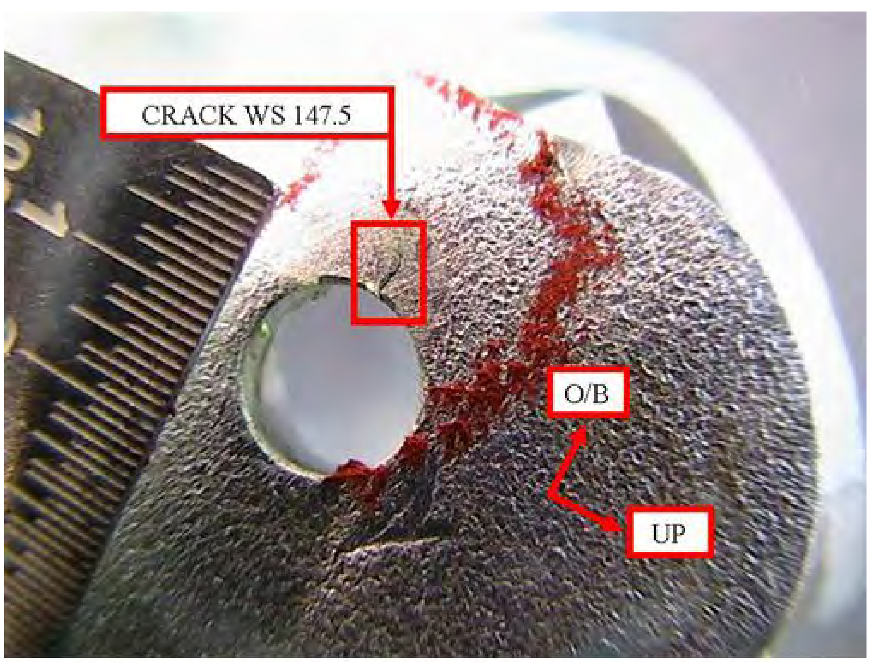
44437-07 CRK FS 182.25-210.75 WS 89 Mac

Figure 339. Macroscopic View of Crack on the Right Wing Nacelle Outboard Longitudinal Bulkhead WS 89 FS 210.75



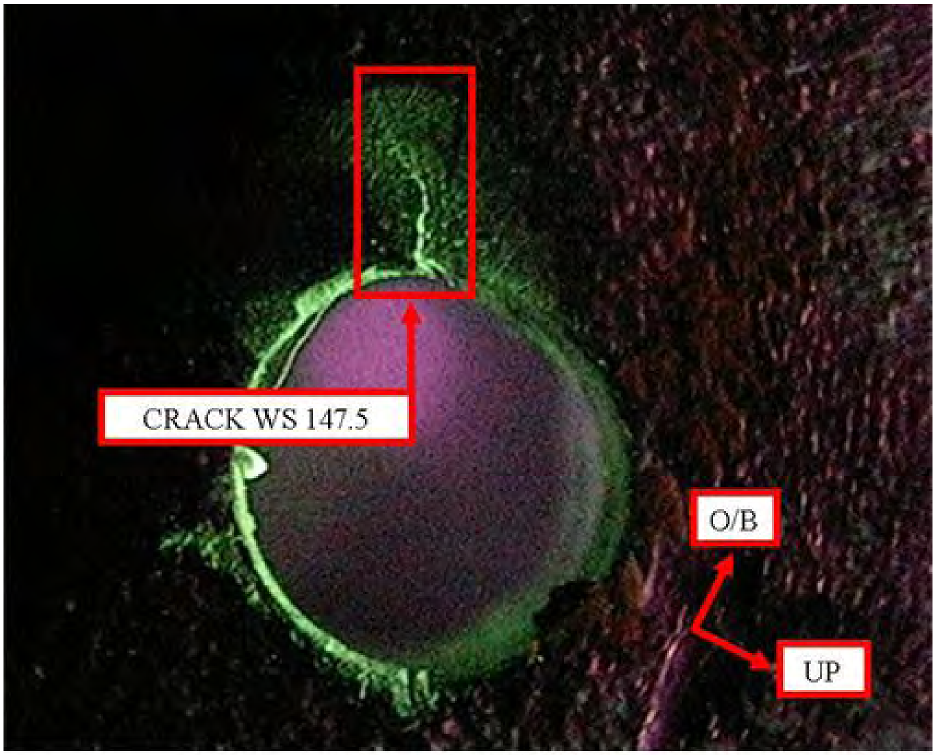
45351-15B CRK WS147.5 OV

Figure 340. Location of Crack on the Right Wing Flap Track WS 147.5



45381-15B CRK WS 147.5 Mac

Figure 341. Macroscopic View of Crack on the Right Wing Flap Track WS 147.5



45381-15B CRK WS 147.5 FLP

Figure 342. Fluorescent Liquid Penetrant Indication of the Crack on the Right Wing Flap Track WS 147.5

45384-00 Corr WS 174.5 OV

Figure 343. Location of Light Corrosion on the Right Aileron Hinge WS 174.5

<u>5.2.1.1 Fuselage</u>.

During the teardown evaluation of the fuselage, 136 parts were inspected and 121 defects were identified and characterized. There were 79 cracks recorded on the fuselage, along with 33 areas of corrosion, 5 instances of wear, and 4 occurrences of other damage. Multiple cracks were characterized in each of the following areas: windshield bracket, stiffener angle, various bulkhead assemblies, fuselage web-beam assembly lower forward frame right side, fuselage left lower forward web, fuselage lower plate assembly, floorboard support structure, left and right front spar attachments, and the door sill support aft body. Multiple areas of corrosion were found on the floorboard support structure and various bulkheads.

5.2.1.1.1 3.4.2.3.1 Fuselage FS 2 Through FS 34.

Numerous cracks and areas of corrosion were identified during the teardown evaluation of the fuselage. Table 34 provides detailed characterizations of all defects found on FS 2 to FS 34.

Table 34. Inspection Results From FS 2 Through FS 34

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Windshield bracket, figure 345	Crack	BL 0	1.685 inches	Not inspected[1]	Fatigue	346 347
	Crack	BL 0	0.461 inch	Not inspected[1]	Fatigue	348 349
Fuselage lower fwd skin, figure 350	Corrosion	FS 2 through FS 23	21 inches by 7 inches	Corrosion indication	Light-moderate corrosion 5% thickness loss	105
Stiffener angle, figure 351	Multiple cracks	FS 2	0.03 inch	No indication	10% through thickness cracks	352
Lower fwd fuselage right side skin angle attachment, figure 253	Corrosion	FS 2 through FS 20	19.25 inches by 0.5 inch	Corrosion indication	Severe corrosion 15.6% thickness loss	354
Battery box, figure 355	Crack	FS 2	0.341 inch	Crack indication	Surface crack	356 257
Bulkhead assembly, figure 258	Corrosion	FS 2	13.0 inches by 1.0 inch	Corrosion indication	Severe corrosion 22% thickness loss	259
	Crack	FS 2	0.836 inch	Crack indication	Surface crack	360 361 362
	Crack	FS 2	0.545 inch	Crack indication	Surface crack	363 364 365
Right fuselage upper angle longitudinal beam assembly, figure 366	Corrosion	FS 2 through FS 15.25	13.25 inches by 0.875 inch	No indication	Light-moderate corrosion 4.2% thickness loss	105
Fuselage web-beam assembly lower fwd frame right side, figure 367	Multiple cracks	FS 2	0.806 inch	Crack indication	Hole crack	368
	Multiple cracks	FS 2	0.933 inch	Crack indication	Surface crack bend radii	369
	Multiple cracks	FS 2	0.409 inch	Crack indication	Surface crack bend radii	370 371
	Multiple cracks	FS 2	0.211 inch	Crack indication	Hole crack	372 373

Table 34. Inspection Results From FS 2 Through FS 34 (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Fuselage left upper angle #1 longitudinal beam assembly, figure 374	Corrosion	FS 10 through FS 35	Scattered over entire part	Corrosion indication	Light corrosion 2% thickness loss	103
	Corrosion	FS 2 through FS 10	8 inches by 1.5 inches	Corrosion indication	Severe corrosion 12% thickness loss	375
Fuselage left lower fwd web, figure 376	Corrosion	FS 2 through FS 18	Scattered over entire part	Corrosion indication	Light-moderate corrosion 2.5% thickness loss	105
	Crack	FS 2	0.096 inch	Crack indication	Bend radii	377 378
	Multiple cracks	FS 2	0.76 inch	Crack indication	Hole crack surface crack	379 380 381
	Multiple cracks	FS 2	0.221 inch	Crack indication	Hole crack	382 383
Baggage compartment door lock bracket, figure 384	Corrosion	FS 2 through FS 7.5	5.5 inches by 2 inches	Corrosion indication	Moderate corrosion 6% thickness loss	385
Fuselage lower plate assembly, figure 386	Crack	FS 22	0.570 inch	Not inspected[1]	Fatigue	387 388
	Crack	FS 27	0.668 inch	Not inspected[1]	Fatigue	389 390

[1] Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 344 shows the location of two cracks on the windshield bracket, part number 40908, at BL 0. A macroscopic view of crack 1, which measured 1.685 inches, is shown in figure 345, and the fracture face is shown in figure 346. This fracture face shows crack face characteristics typical of fatigue. Figure 347 shows a macroscopic view of crack 2 on the windshield bracket, and the fracture face is shown in figure 348. Through analysis of the crack face, it was determined that this 0.461-inch crack was caused by fatigue. The location of light-moderate corrosion on the fuselage lower forward skin, part number 40887-6, is shown in figure 349. This 147-square-inch area of corrosion was located from FS 2 to FS 23 and caused a maximum reduction in thickness of 5%. The location of multiple cracks on the stiffener angle, part number 44717-5, is shown in figure 350. These cracks measured 0.03 inch each and had grown only about 10% through the thickness of the part. A microscopic view of these cracks, which occurred at FS 2, are shown in figure 351.

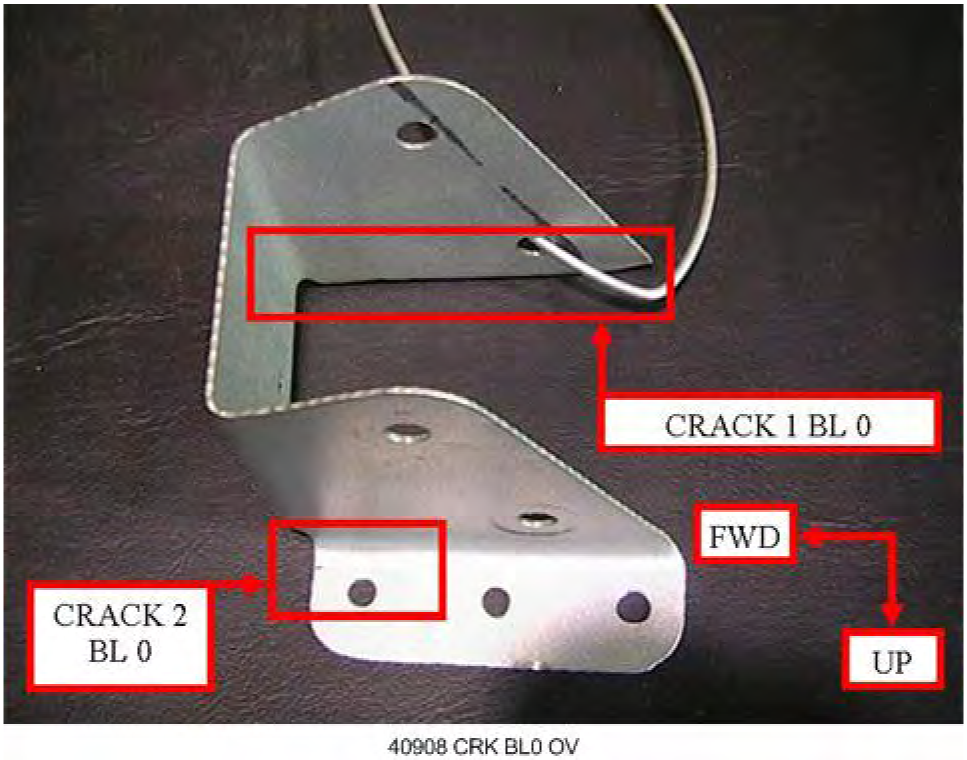


Figure 344. Location of Crack 1 and 2 on the Windshield Bracket BL 0

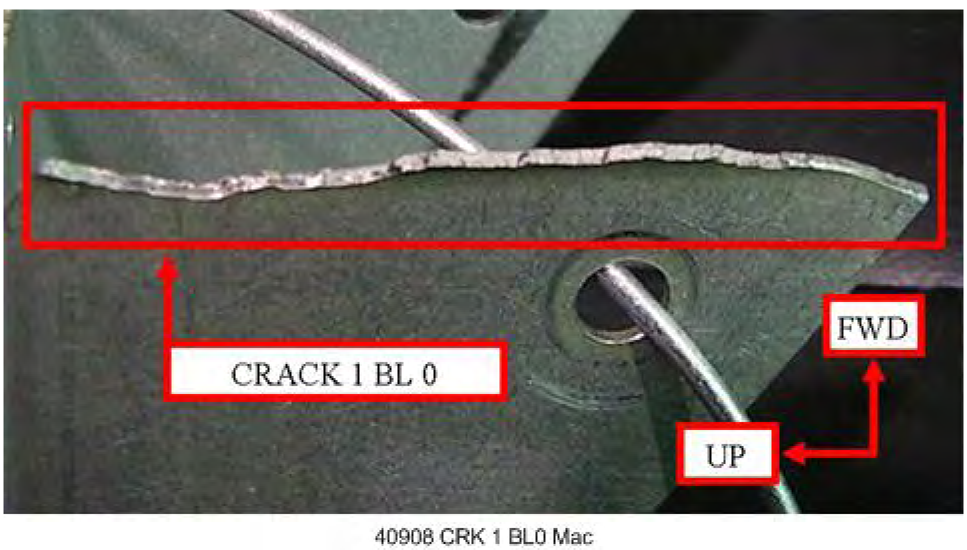


Figure 345. Macroscopic View of Crack 1 on the Windshield Bracket BL 0

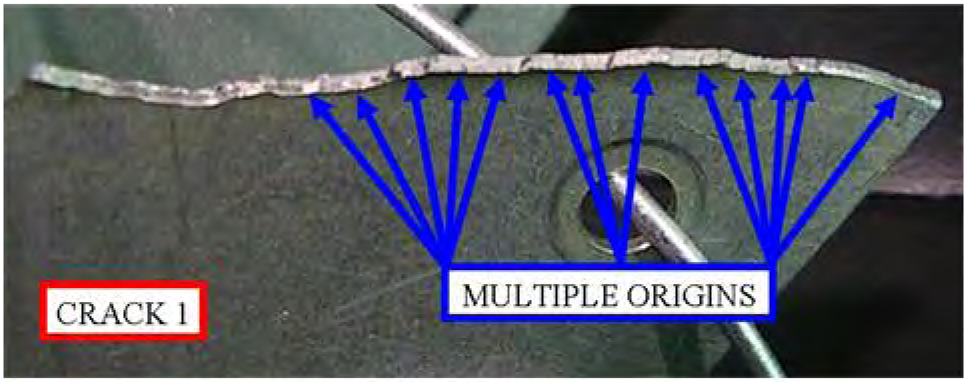


Figure 346. Fracture Face of Crack 1 on the Windshield Bracket BL 0

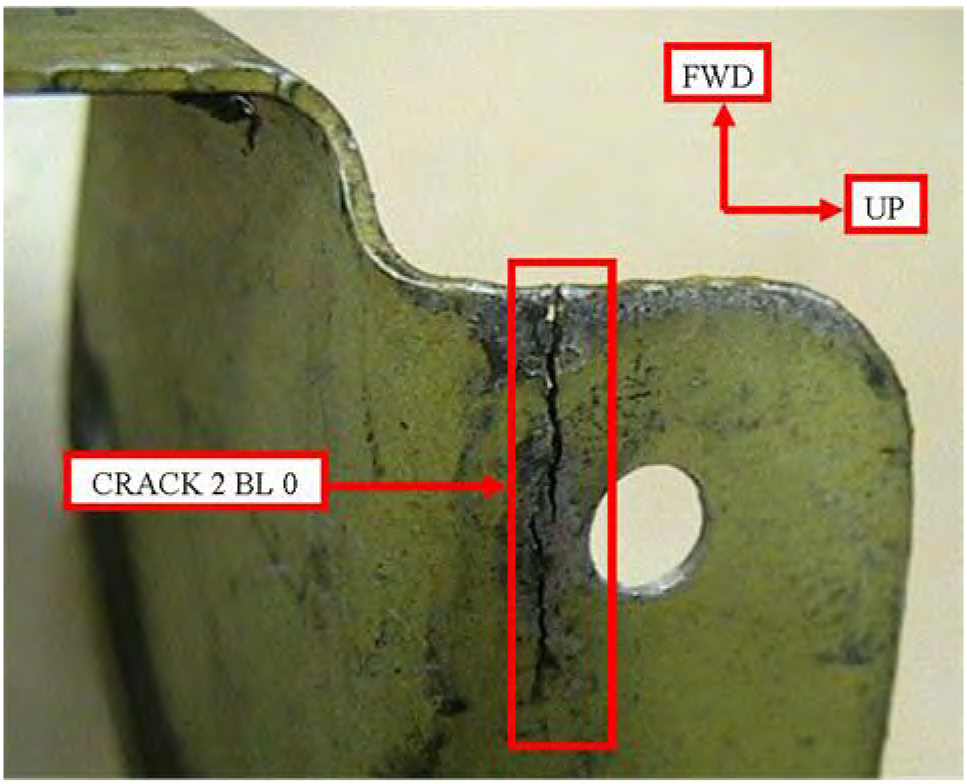


Figure 347. Macroscopic View of Crack 2 on the Windshield Bracket BL 0

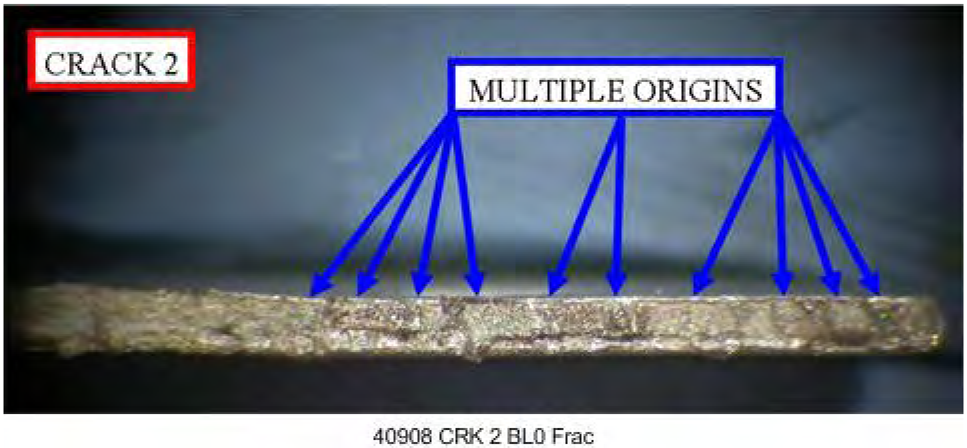


Figure 348. Fracture Face of Crack 2 on the Windshield Bracket BL 0

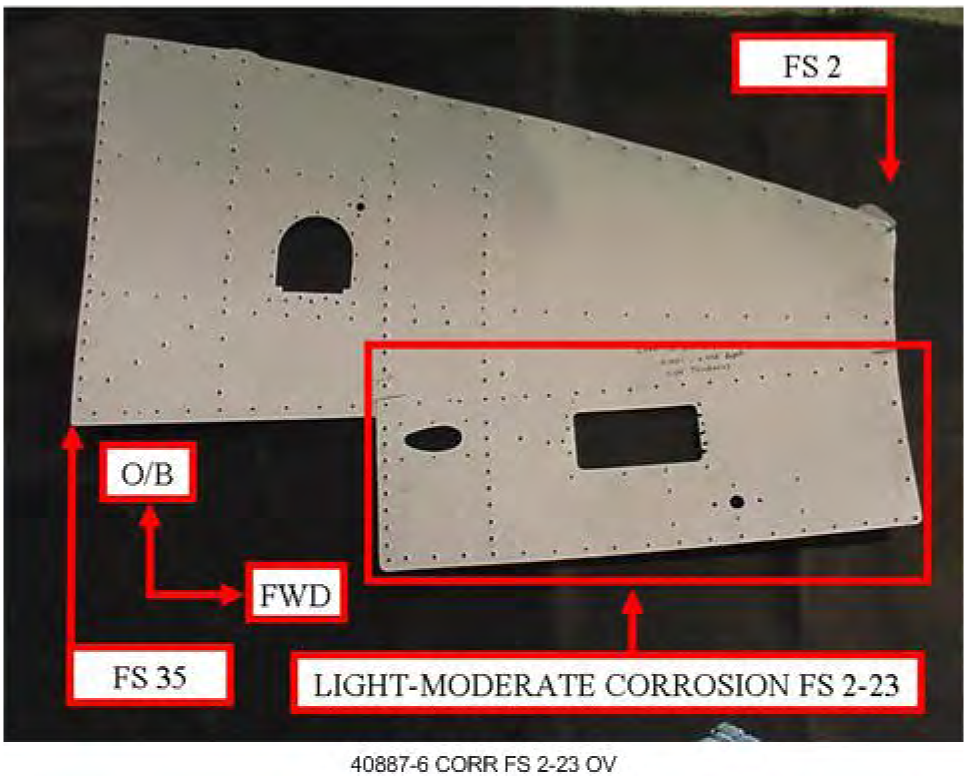
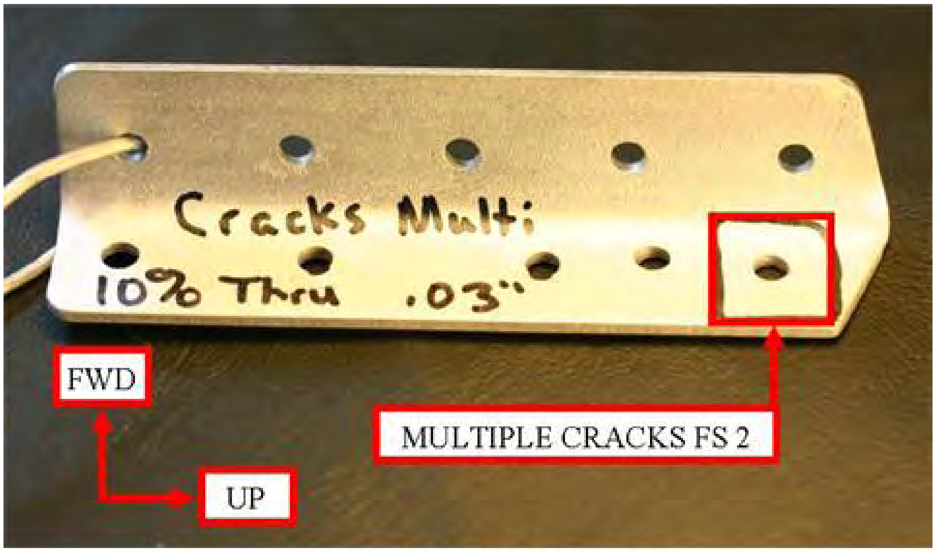
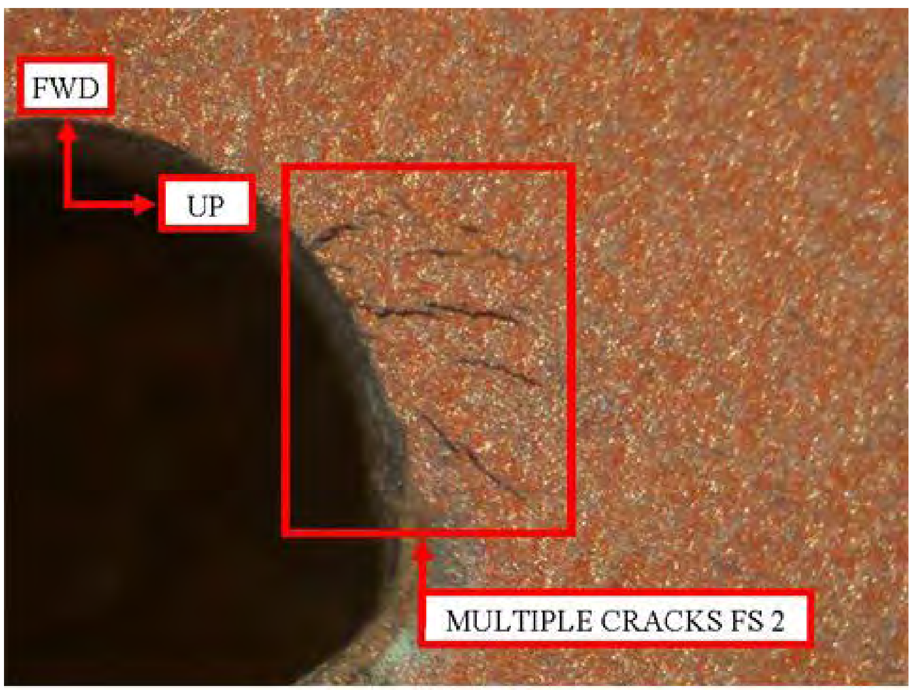


Figure 349. Location of Light-Moderate Corrosion on the Fuselage Lower Forward Skin FS 2 Through FS 23



44717-5 CRK FS2 OV

Figure 350. Location of Multiple Cracks on the Stiffener Angle FS 2



44717-5 MULTI CRKS FS2 Mic

Figure 351. Microscopic View of Multiple Cracks on the Stiffener Angle FS 2

Figure 352 shows the location of severe corrosion on the lower forward fuselage right side skin angle attachment, part number 40887-28, from FS 2 to FS 20. This 9.64-square-inch area of corrosion caused a maximum localized thickness loss of 15.6%. Figure 353 shows a macroscopic view of this area of corrosion. The location of a 0.341-inch-long crack on the battery box, part number 53959-000, is shown in figure 354 at FS 2. Figure 355 shows a macroscopic view of this crack, while a microscopic view is shown in figure 356.

Figure 352. Location of Severe Corrosion on the Lower Forward Fuselage Right Side Skin Angle Attachment FS 2 Through FS 20

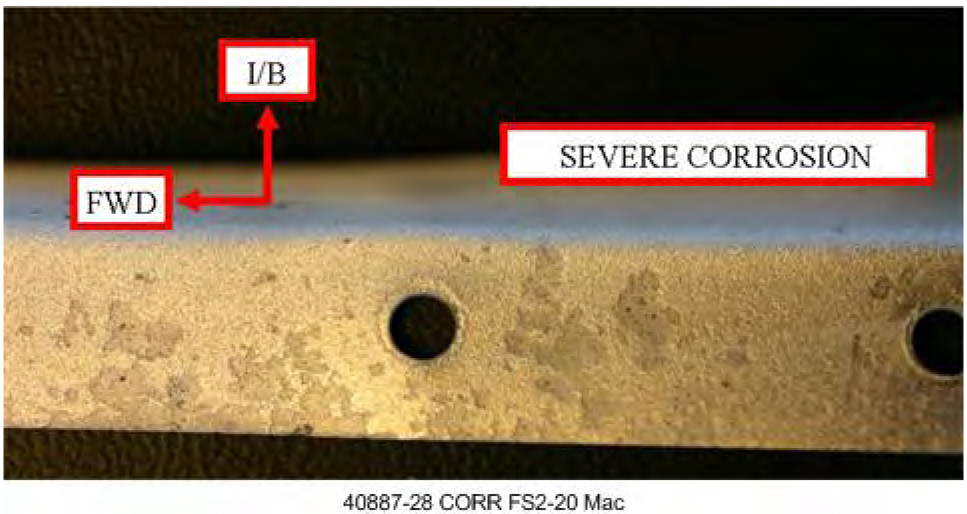
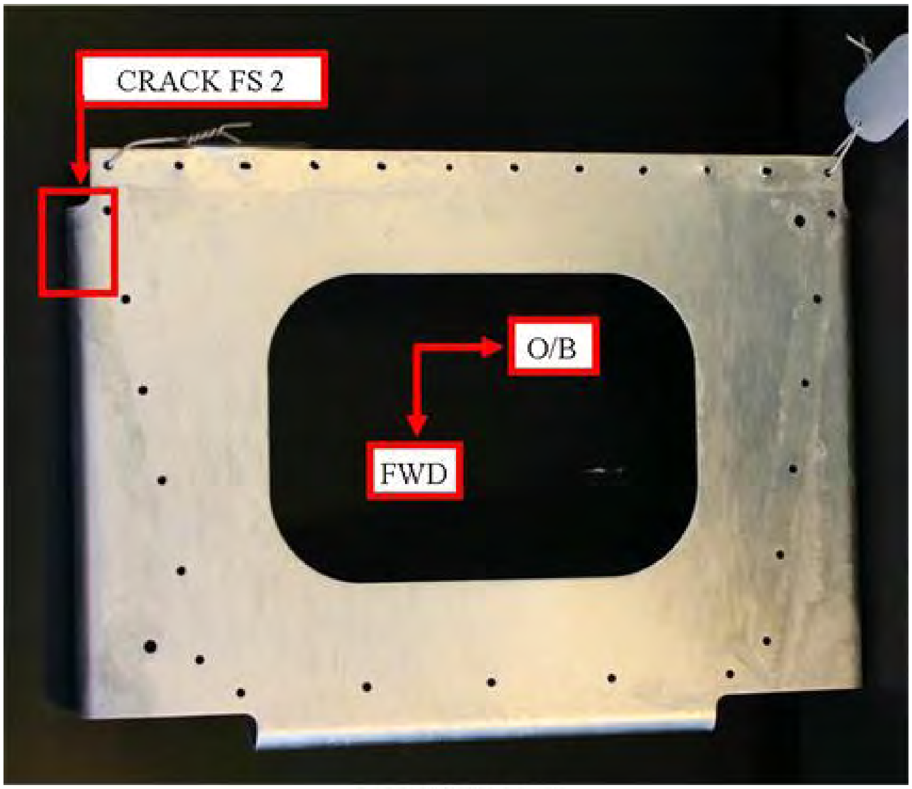
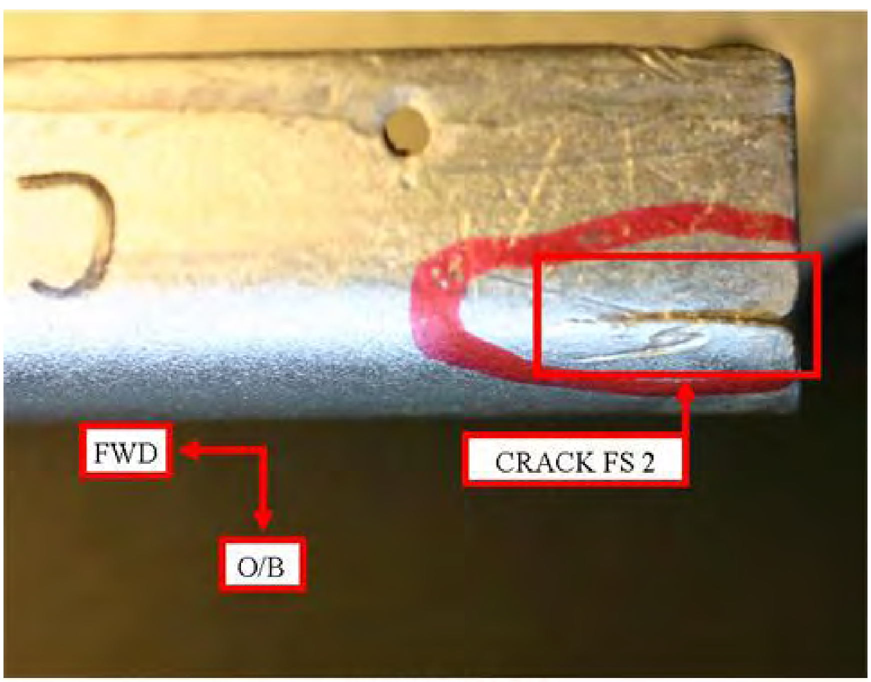


Figure 353. Macroscopic View of Severe Corrosion on the Lower Forward Fuselage Right Side Skin Angle Attachment FS 2 Through FS 20



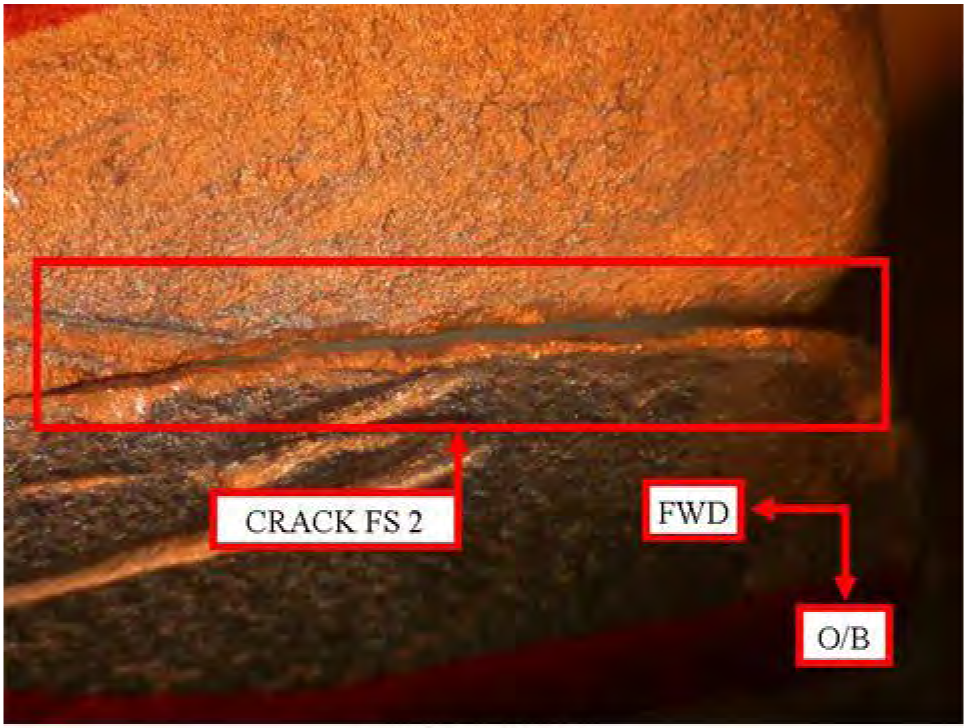
53959-00 CRK FS2 OV

Figure 354. Location of Crack on the Battery Box FS 2



53959-00 CRK FS2 Mac

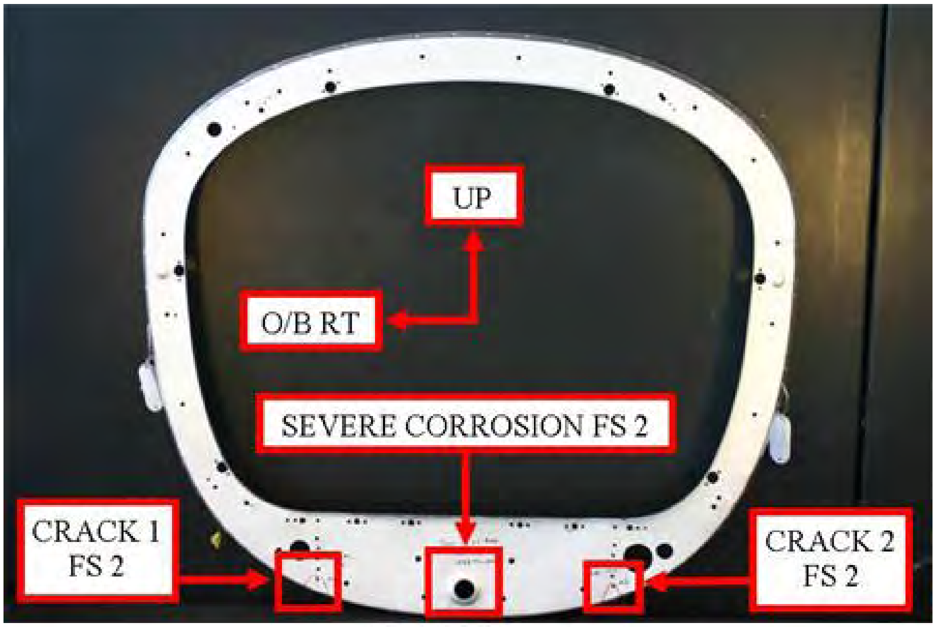
Figure 355. Macroscopic View of the Crack on the Battery Box FS 2



53959-00 CRK FS2 Mic

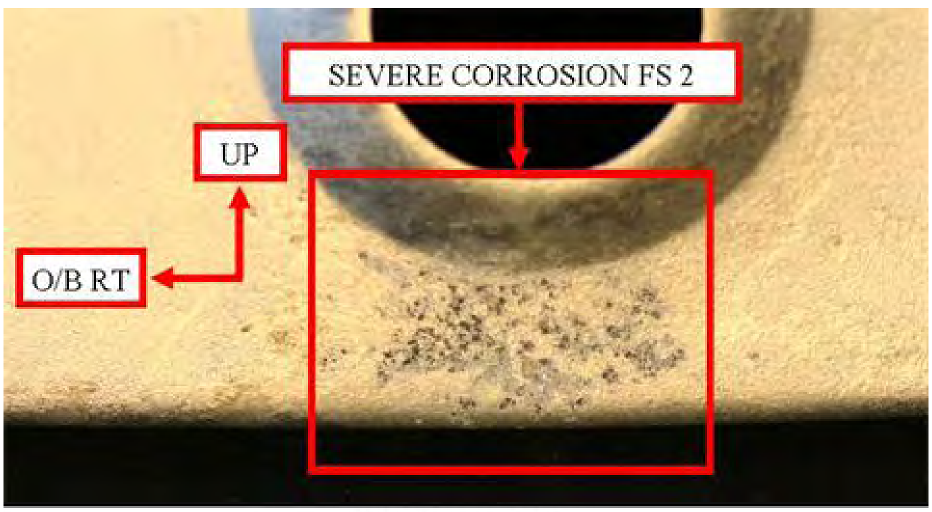
Figure 356. Microscopic View of Crack on the Battery Box FS 2

Figure 357 shows the location of two cracks and one 13-square-inch area of severe corrosion on the bulkhead assembly, part number 71790-2. A macroscopic view of the severe corrosion, located at FS 2, is shown in figure 358. This area of corrosion caused a localized reduction in thickness of 22%, which is categorized as severe corrosion. Figure 359 shows a macroscopic view of crack 1, which measured 0.836 inch and was located at FS 2. The fluorescent liquid penetrant indication is shown in figure 360, and a microscopic view of crack 1 is shown in figure 361. Figure 362 shows a macroscopic view of crack 2 on the bulkhead assembly, located at FS 2. The fluorescent liquid penetrant indication is shown in figure 363. A microscopic view of crack 2, which measures 0.545 inch in length and also occurs at FS 2, is shown in figure 364. Figure 365 shows the location of an 11.69-square-inch area of light-moderate corrosion, located from FS 2 to FS 15.25 on the right fuselage upper angle longitudinal beam assembly, part number 44877-01A. This area of corrosion caused a localized reduction in thickness of 4.2%.



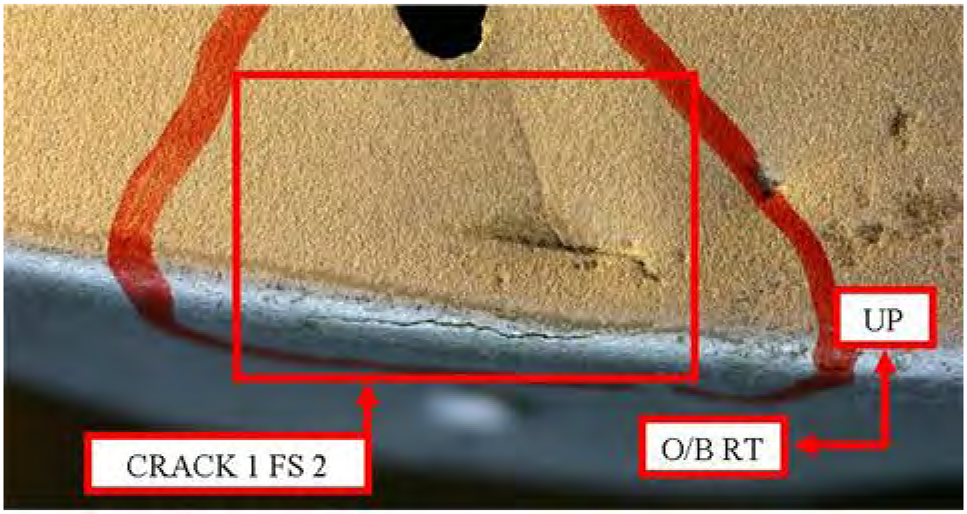
71790-02 CRK FS2 OV

Figure 357. Location of Two Cracks and Severe Corrosion on the Bulkhead Assembly FS 2



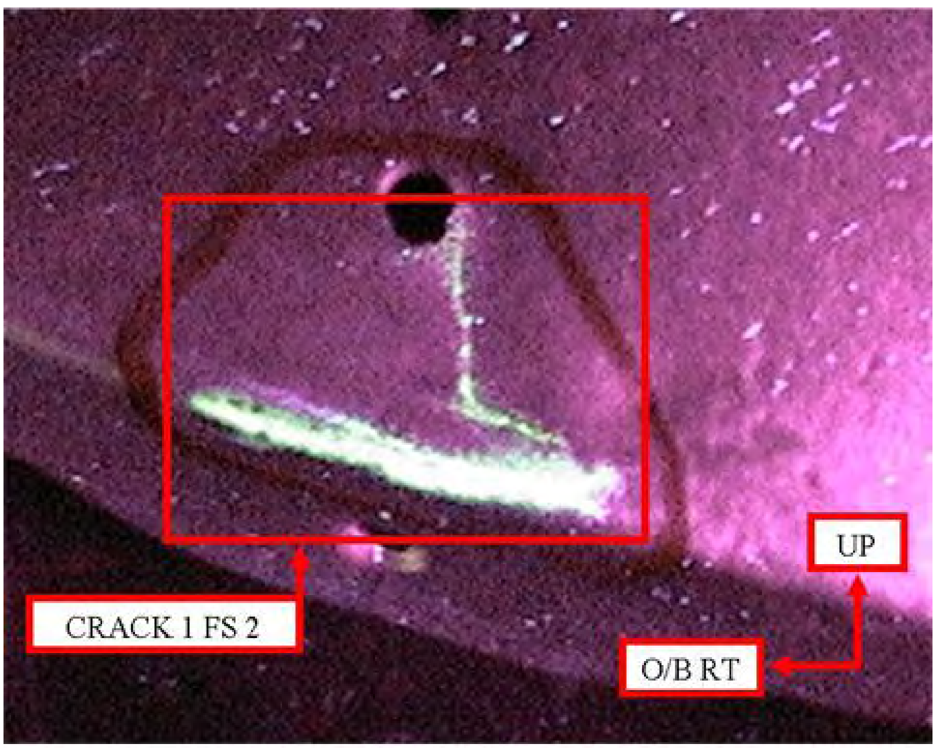
71790-02 CORR FS2 Mac

Figure 358. Macroscopic View of Severe Corrosion on the Bulkhead Assembly FS 2



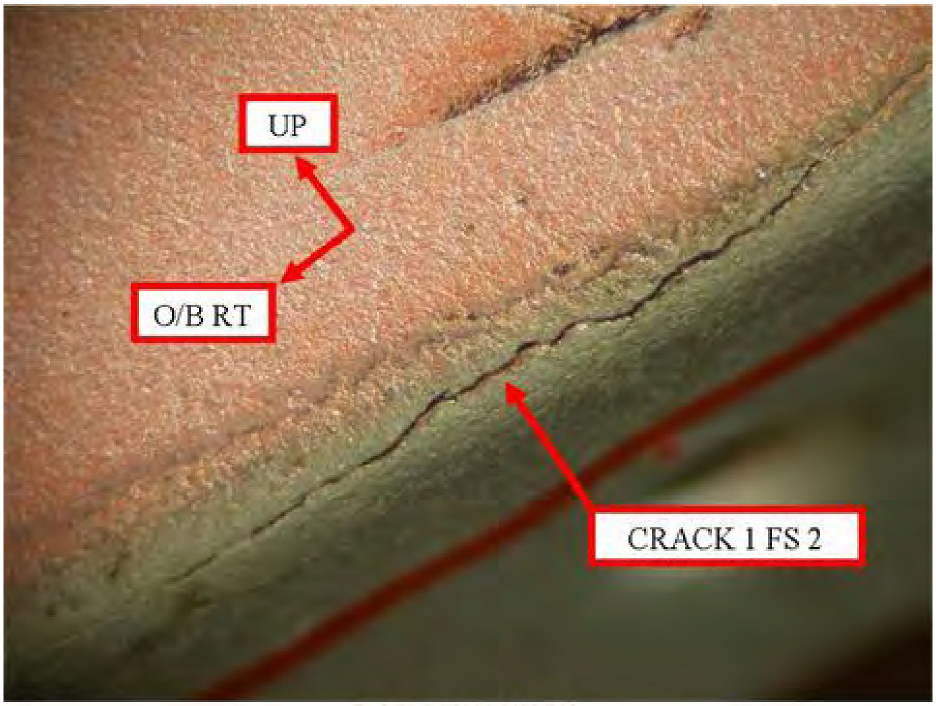
71790-02 CRK 1 FS2 Mac

Figure 359. Macroscopic View of Crack 1 on the Bulkhead Assembly FS 2



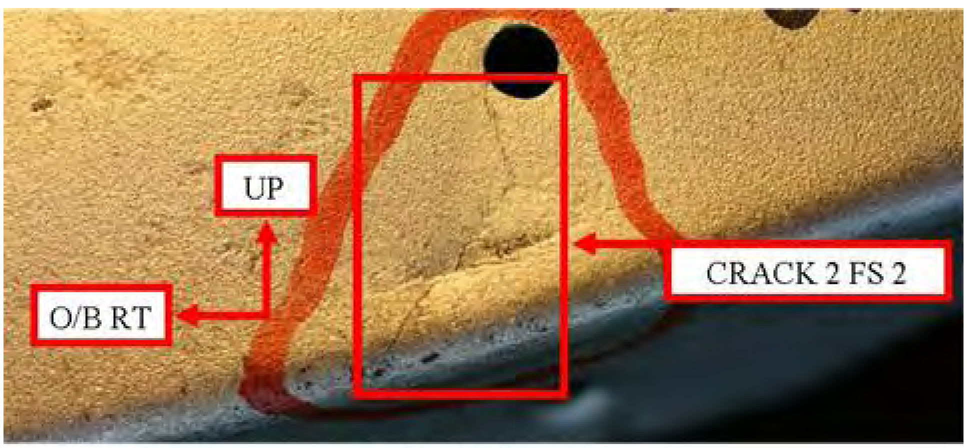
71790-02 CRK 1 FS2 FLP

Figure 360. Fluorescent Liquid Penetrant Indication of Crack 1 on the Bulkhead Assembly FS 2



71790-02 CRK 1 FS2 Mic

Figure 361. Microscopic View of Crack 1 on the Bulkhead Assembly FS 2



71790-02 CRK 2 FS2 Mac

Figure 362. Macroscopic View of Crack 2 on the Bulkhead Assembly FS 2

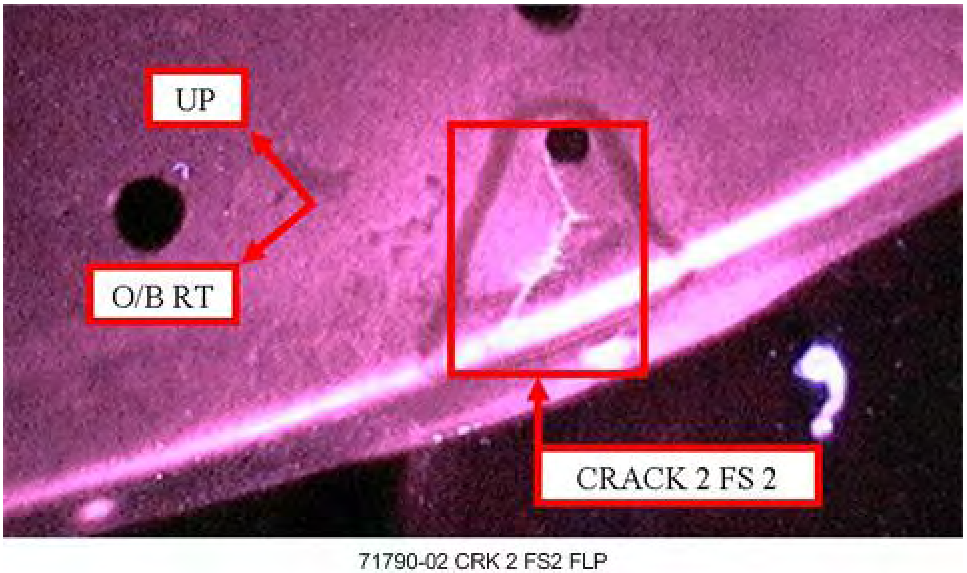


Figure 363. Fluorescent Liquid Penetrant Indication of Crack 2 on the Bulkhead Assembly FS 2

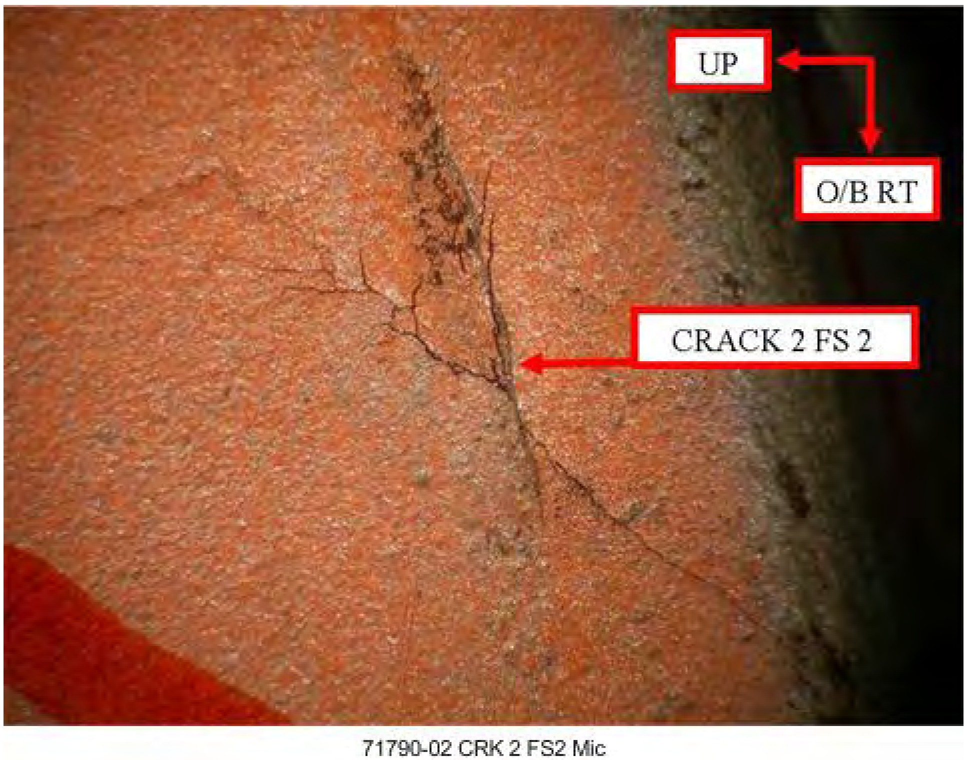
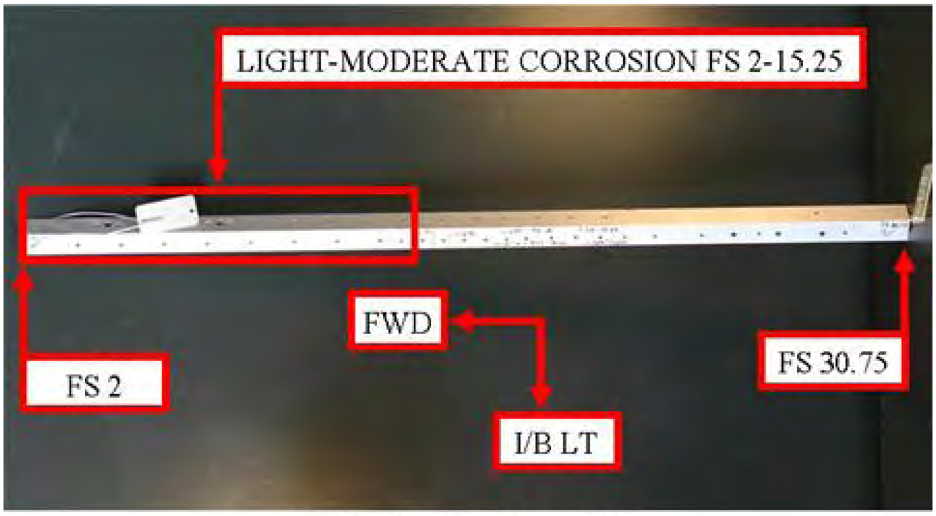


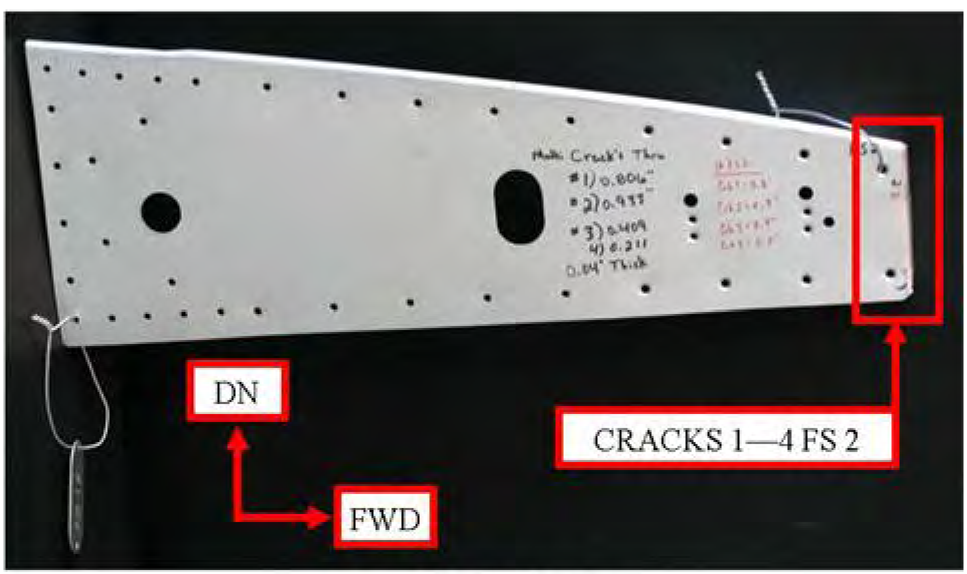
Figure 364. Microscopic View of Crack 2 on the Bulkhead Assembly FS 2



44877-01A CORR FS 2-15.25 OV

Figure 365. Location of Light-Moderate Corrosion on the Right Fuselage Upper Angle Longitudinal Beam Assembly FS 2 Through FS 15.25

The location of four cracks on the fuselage web-beam assembly lower forward frame right side, part number 41090-03, is shown in figure 366 at FS 2. A macroscopic view of crack 1, which measured 0.806 inch, is shown in figure 367. A macroscopic view of crack 2, which measured 0.933 inch and occurred at FS 2, is shown in figure 368. Figure 369 shows a macroscopic view of crack 3 on the fuselage web-beam, and figure 370 shows the fluorescent liquid penetrant indication of this 0.409-inch crack. A macroscopic view of crack 4, which measures 0.211 inch, is shown in figure 371, and the fluorescent liquid penetrant indication for crack 4 is shown in figure 372.



41090-03 CRK FS2-18 OV

Figure 366. Location of Cracks on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

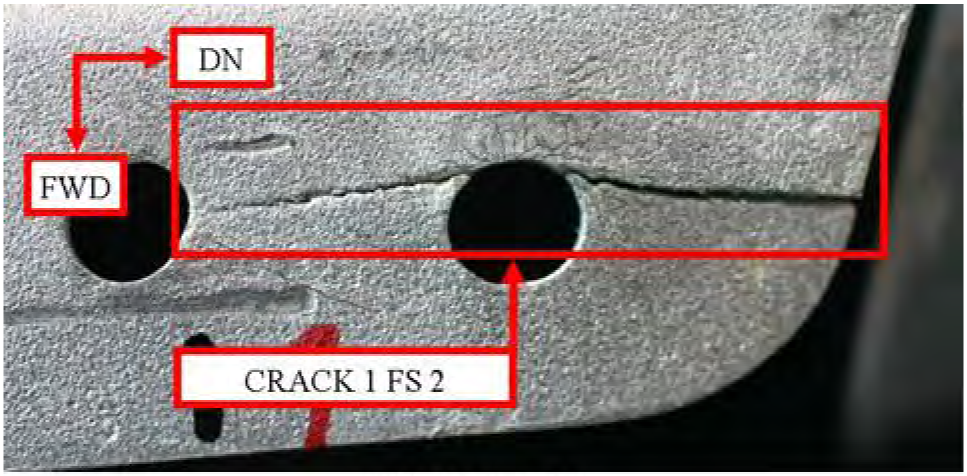


Figure 367. Macroscopic View of Crack 1 on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

Figure 368. Macroscopic View of Crack 2 on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

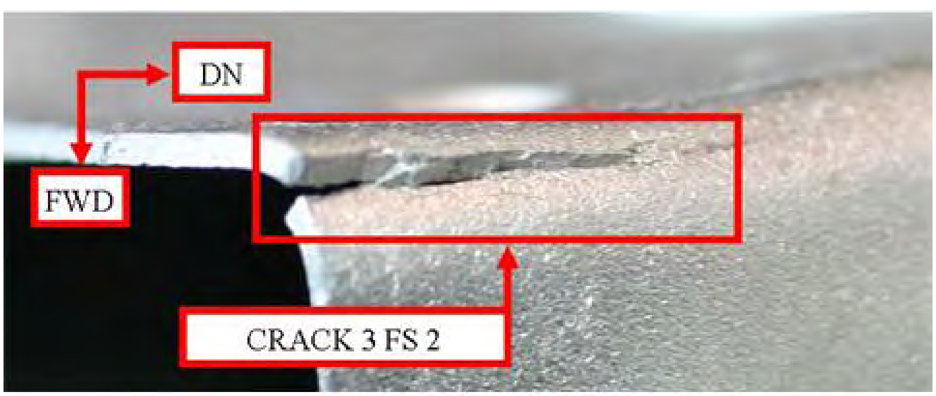


Figure 369. Macroscopic View of Crack 3 on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

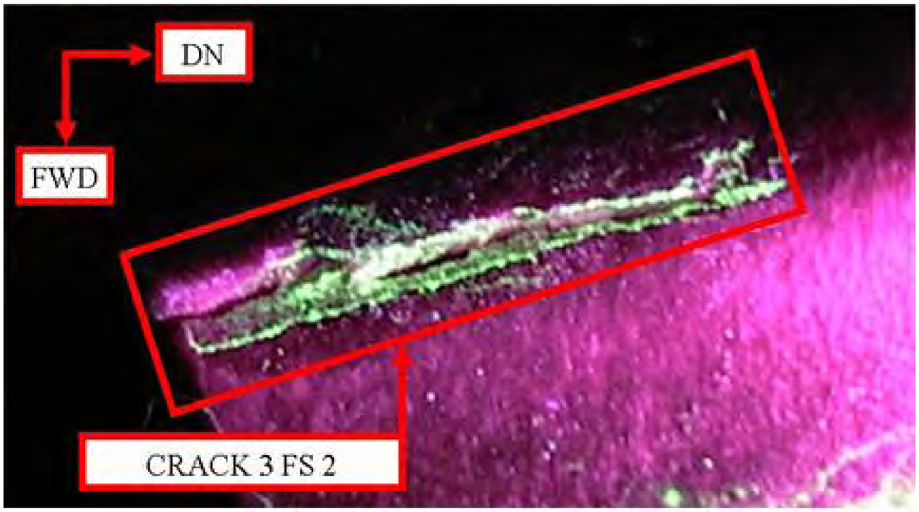


Figure 370. Fluorescent Liquid Penetrant Indication of Crack 3 on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

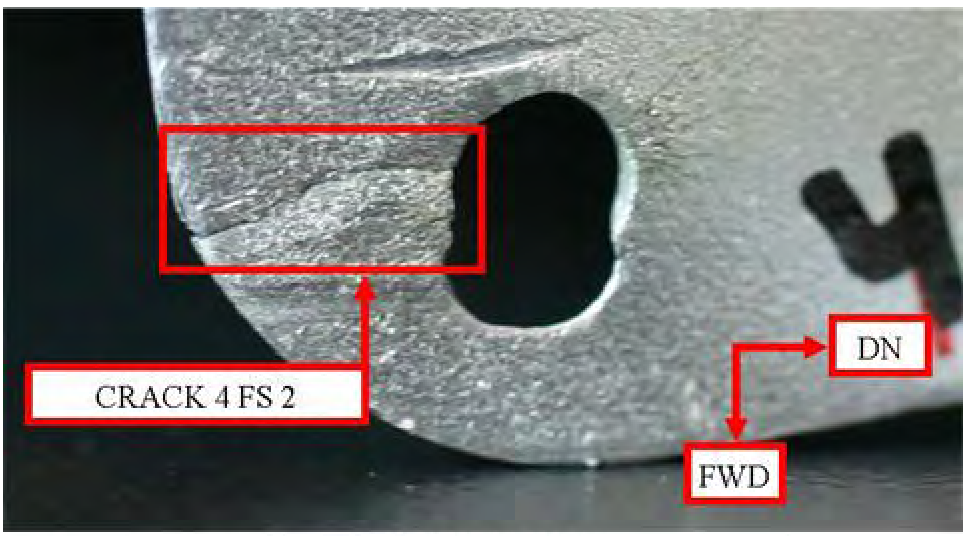


Figure 371. Macroscopic View of Crack 4 on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

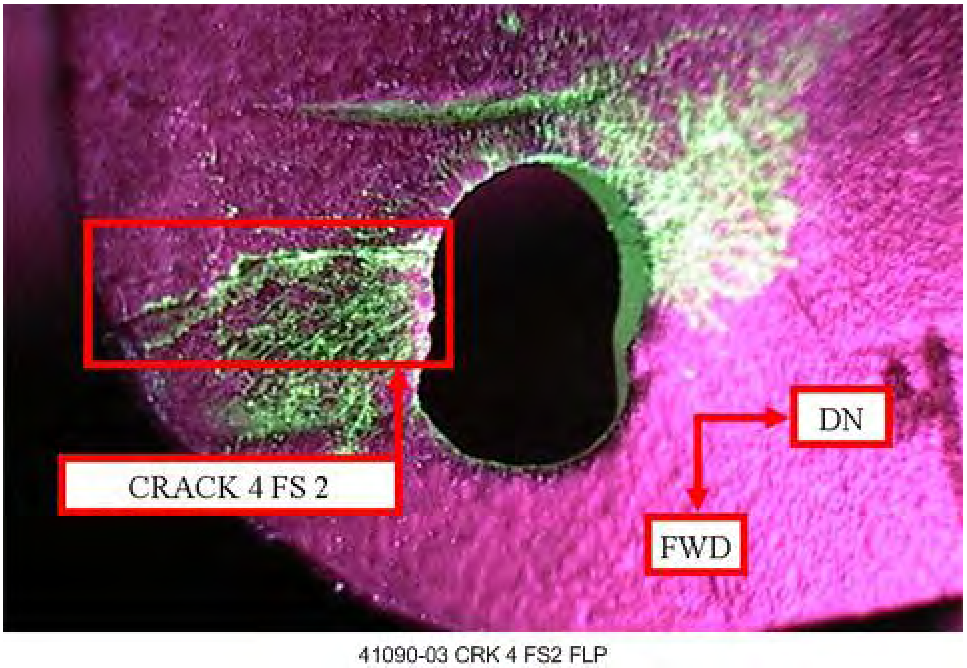
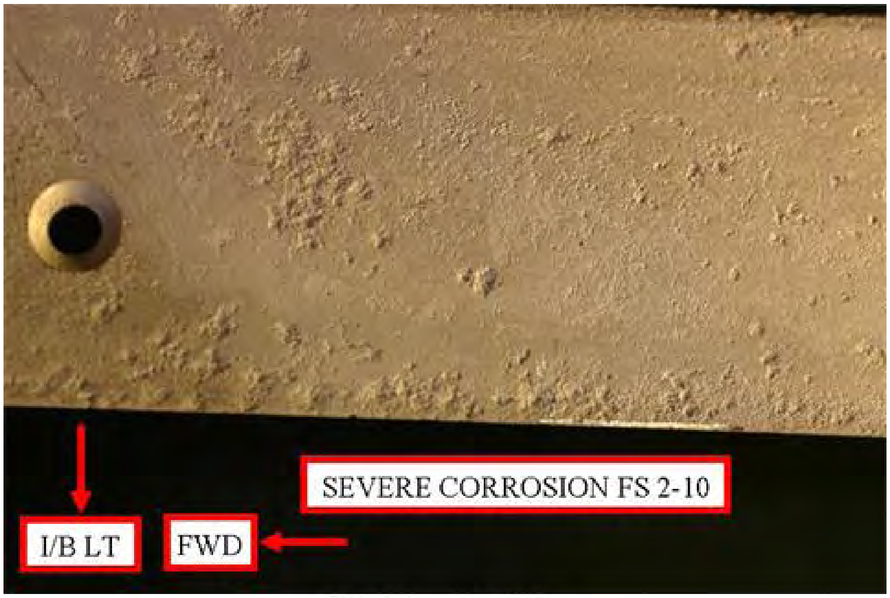


Figure 372. Fluorescent Liquid Penetrant Indication of Crack 4 on the Fuselage Web-Beam Assembly Lower Forward Frame Right Side FS 2

Figure 373 shows the locations of a 12-square-inch area of severe corrosion and light scattered corrosion on the fuselage left upper angle 1 longitudinal beam assembly, part number 44877-00. The light scattered corrosion occurred from FS 10 to FS 35 and caused a maximum localized thickness loss of 2%. A macroscopic view of the severe corrosion that occurred from FS 2 to FS 10 is shown in figure 374. This area of corrosion caused a localized reduction in thickness of 12%. An overview of light-moderate scattered corrosion and the location of three cracks on the fuselage left lower forward web, part number 41090-02, are shown in figure 375. Light-moderate corrosion was found scattered across the web from FS 2 to FS 18 and caused a localized reduction in thickness of 2.5%. A macroscopic view of crack 1, which measured 0.096 inch and occurred at FS 2, is shown in figure 376, and a microscopic view of crack 1 is shown in figure 377. Figure 378 shows a macroscopic view of crack 2, which measured 0.76 inch and was located at FS 2 on the fuselage left lower forward web. Figure 379 shows the fluorescent liquid penetrant indication of crack 2, and figure 380 shows a microscopic view of this crack. A macroscopic view of crack 3 is shown in figure 381, and a microscopic view of this 0.221-inch-crack is shown in figure 382.

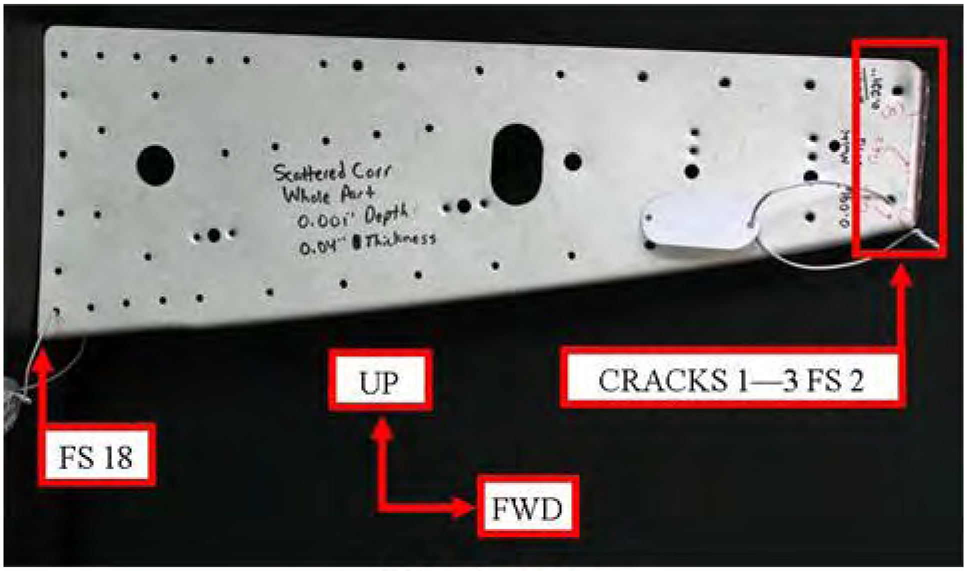
44877-00 Corr FS 2-10 OV

Figure 373. Location of Severe Corrosion on the Fuselage Left Upper Angle 1 Longitudinal Beam Assembly FS 2 Through FS 35



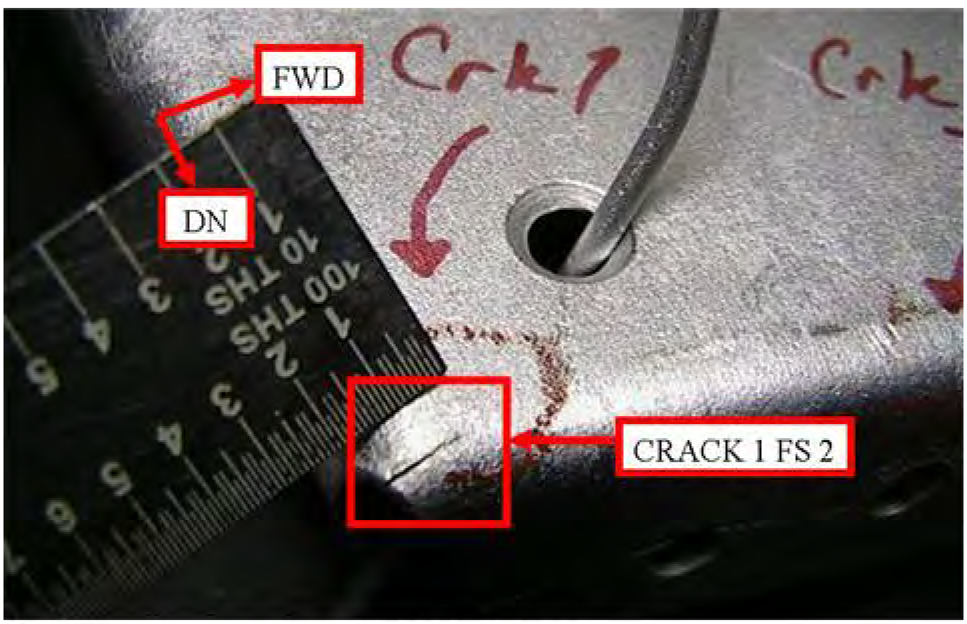
44877-00 Corr FS 2-10 Mac

Figure 374. Macroscopic View of Severe Corrosion on the Fuselage Left Upper Angle 1 Longitudinal Beam Assembly FS 2 Through FS 10



41090-02 CRK FS2 OV

Figure 375. Location of Three Cracks and Overview of Fuselage Left Lower Forward Web FS 2 Through FS 18



41090-02 CRK 1 FS2 Mac

Figure 376. Macroscopic View of Crack 1 on the Fuselage Left Lower Forward Web FS 2

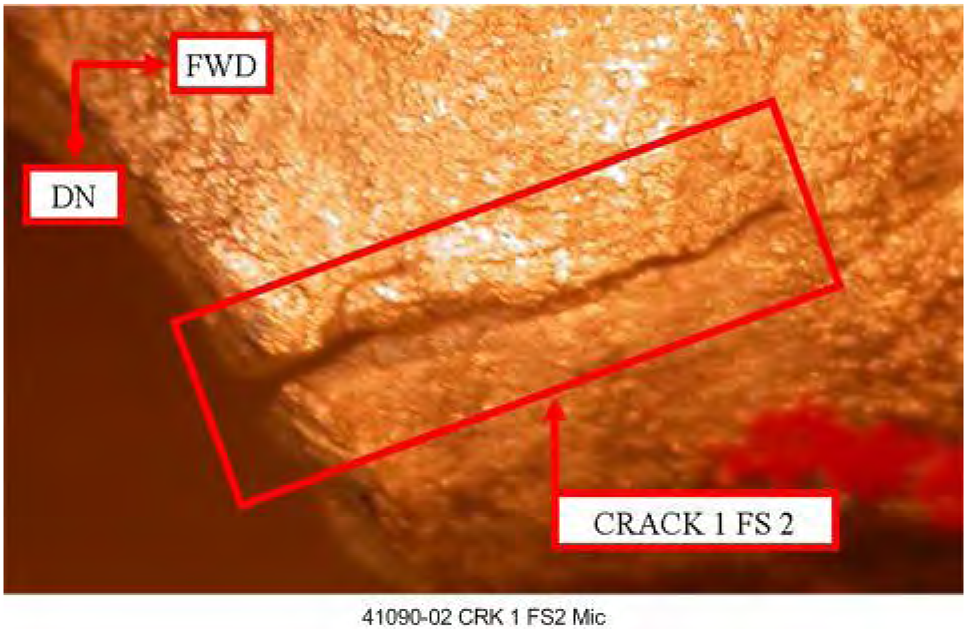


Figure 377. Microscopic View of Crack 1 on the Fuselage Left Lower Forward Web FS 2

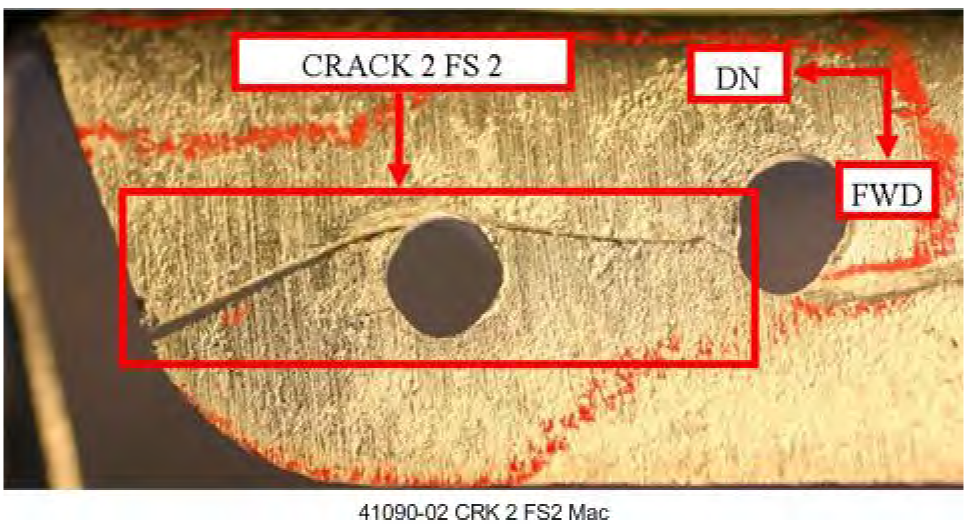


Figure 378. Macroscopic View of Crack 2 on the Fuselage Left Lower Forward Web FS 2

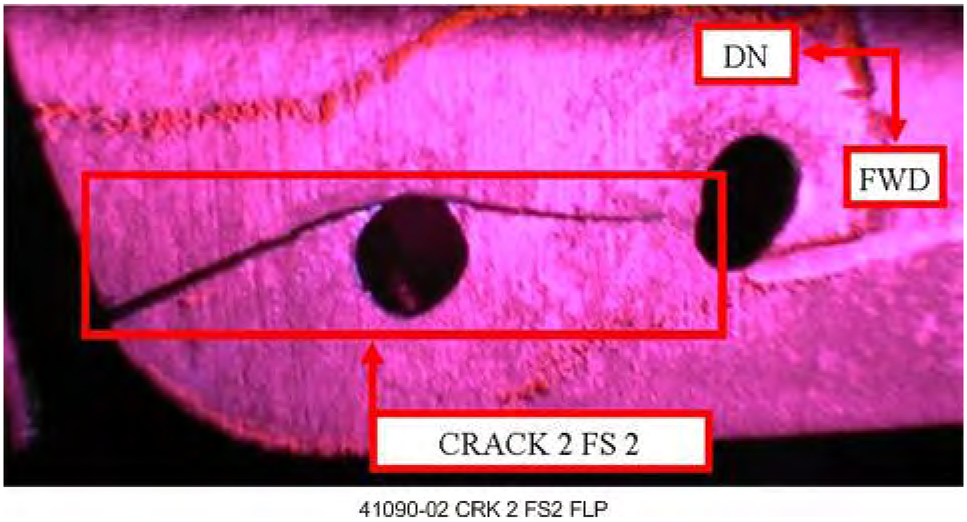


Figure 379. Fluorescent Liquid Penetrant Indication of Crack 2 on the Fuselage Left Forward Web FS 2

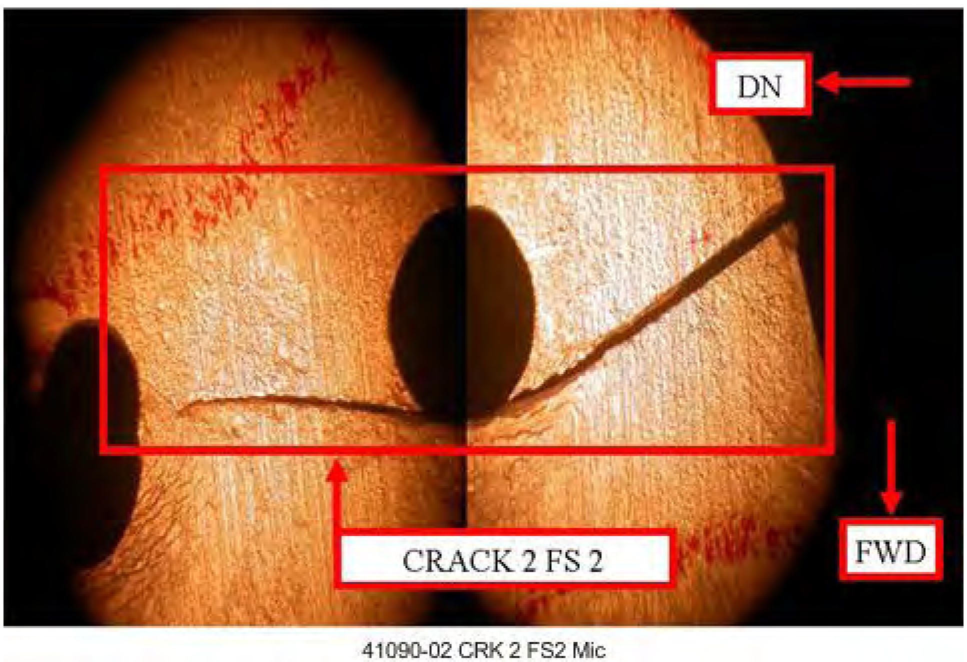


Figure 380. Microscopic View of Crack 2 on the Fuselage Left Lower Forward Web FS 2

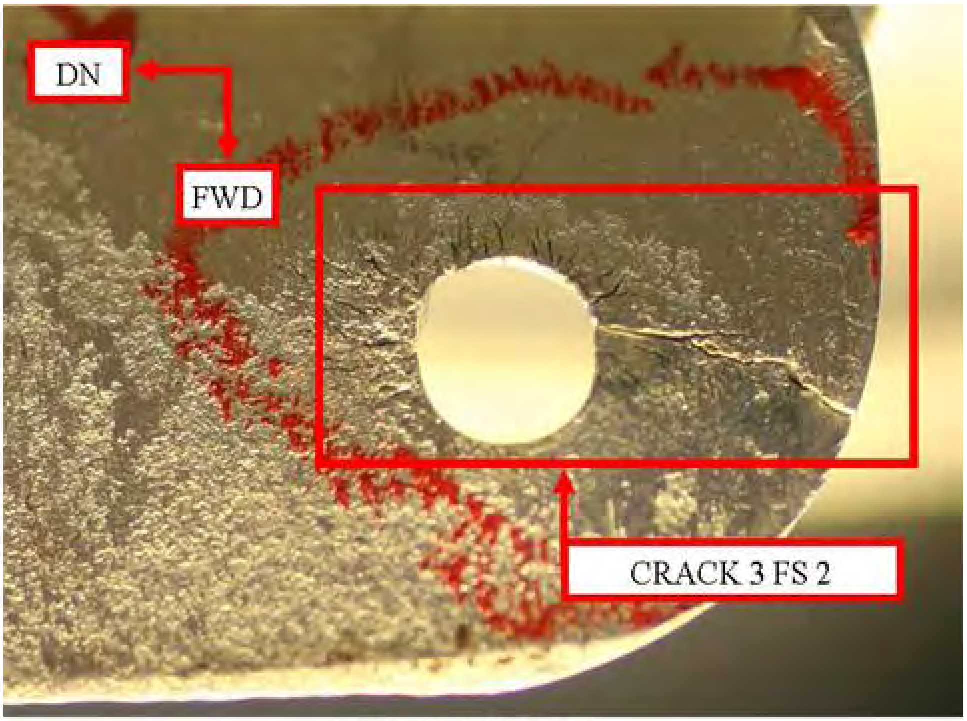


Figure 381. Macroscopic View of Crack 3 on the Fuselage Left Lower Forward Web FS 2

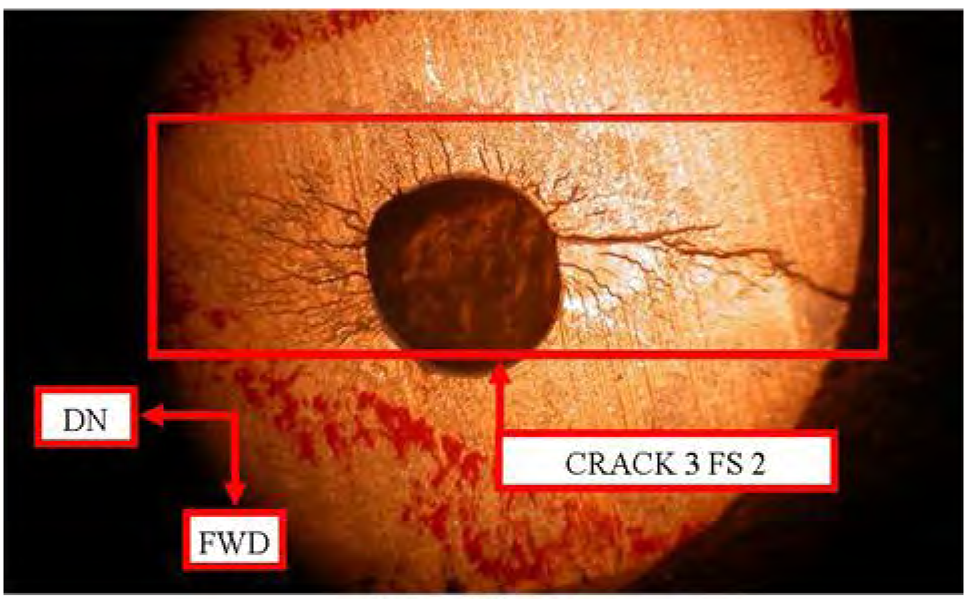


Figure 382. Microscopic View of Crack 3 on the Fuselage Left Lower Forward Web FS 2

Figure 383 shows the location of an 11-square-inch area of moderate corrosion on the baggage compartment door lock bracket, part number 40208-00. This area of corrosion caused a maximum localized reduction in thickness of 6%. A macroscopic view of a segment of this area of corrosion is shown in figure 384. Figure 385 shows the location of cracks 1 and 2 on the fuselage lower plate assembly, part number 44814-00 from FS 18 to FS 28.62. A macroscopic view of crack 1, which measured 0.570 inch and is located at FS 22, is shown in figure 386, and a picture of the fracture face is shown in figure 387. Figure 388 shows a macroscopic view of crack 2, which measured 0.668 inch and was located at FS 27, and the fracture face is shown in figure 389 for crack 2. The failure mode for both cracks was determined to be fatigue.

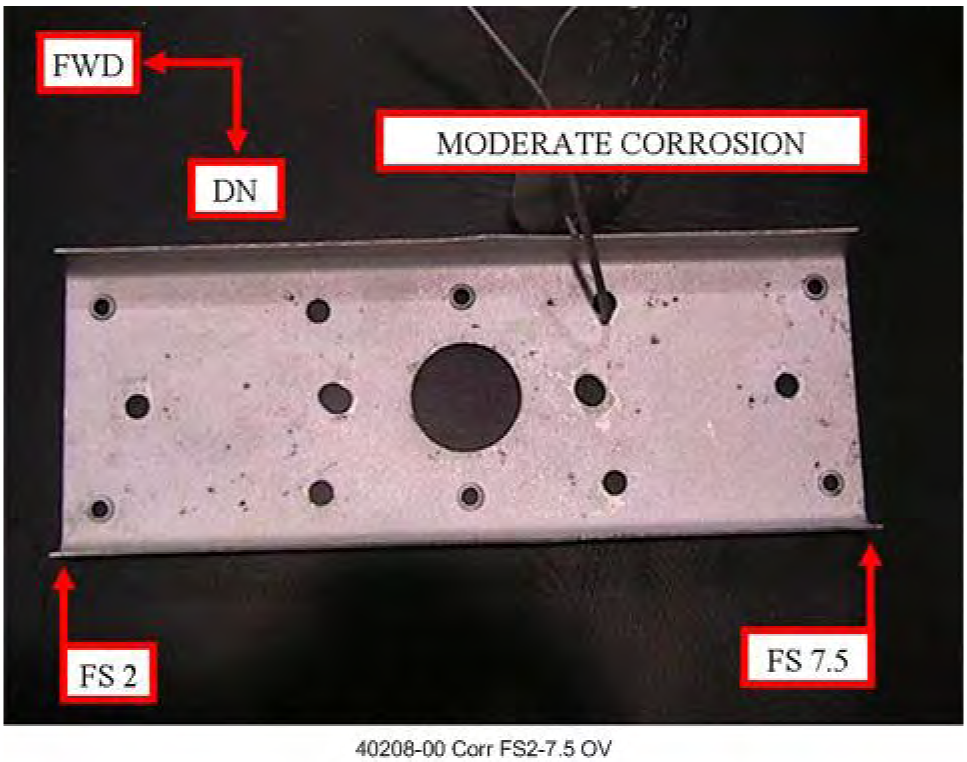


Figure 383. Location of Moderate Corrosion on the Baggage Compartment Door Lock Bracket FS 2 Through FS 7.5

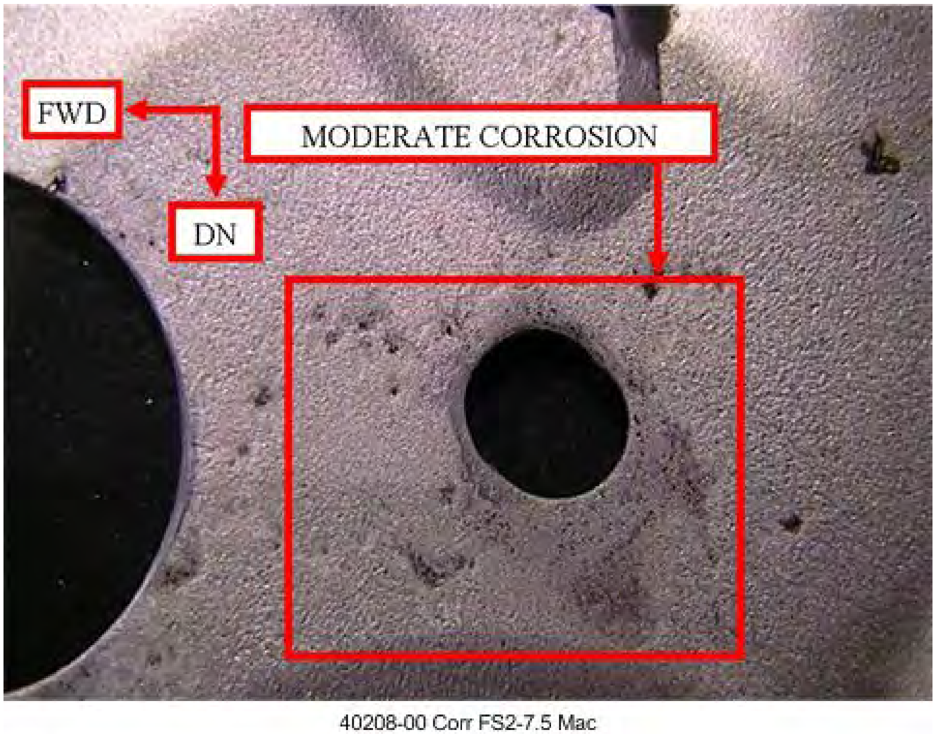


Figure 384. Macroscopic View of Moderate Corrosion on the Baggage Compartment Door Lock Bracket

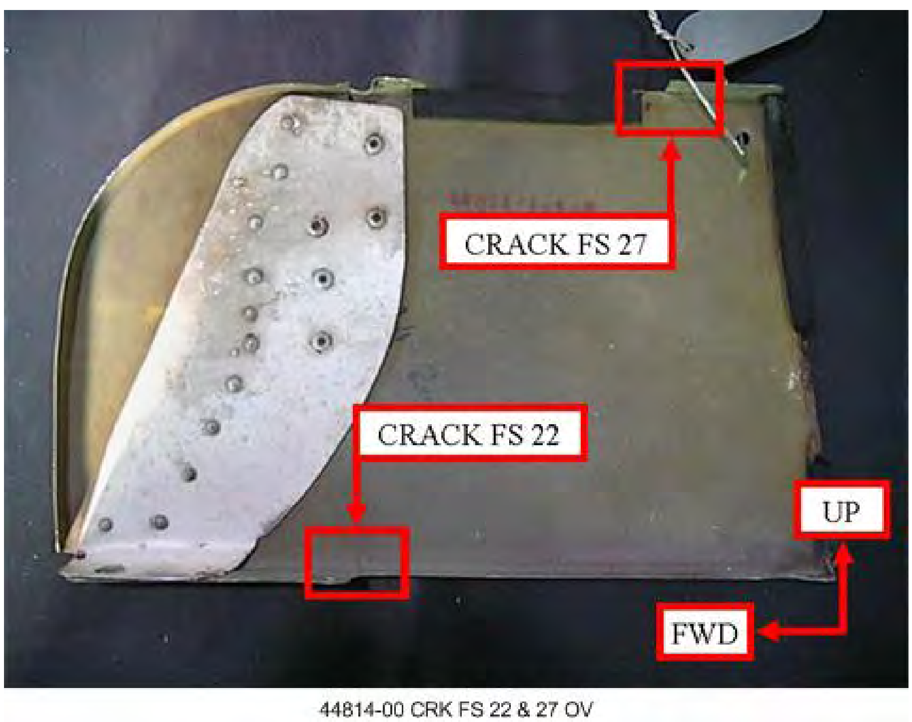


Figure 385. Location of Cracks 1 and 2 on the Fuselage Lower Plate Assembly FS 18 Through FS 28.62

Figure 386. Macroscopic View of Crack 1 on the Fuselage Lower Plate Assembly FS 22

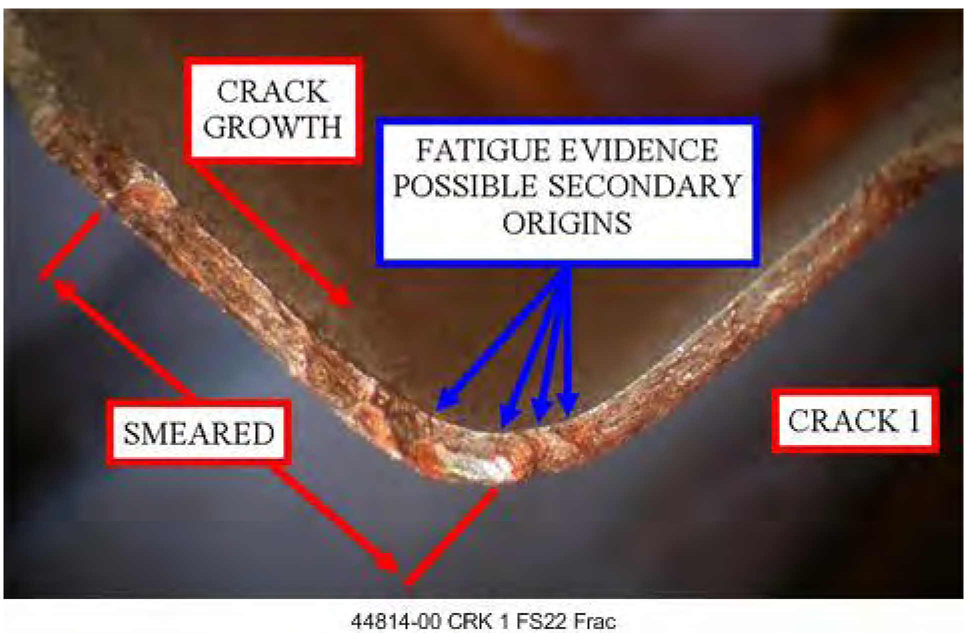


Figure 387. Fracture Face of Crack 1 on the Fuselage Lower Plate Assembly FS 22

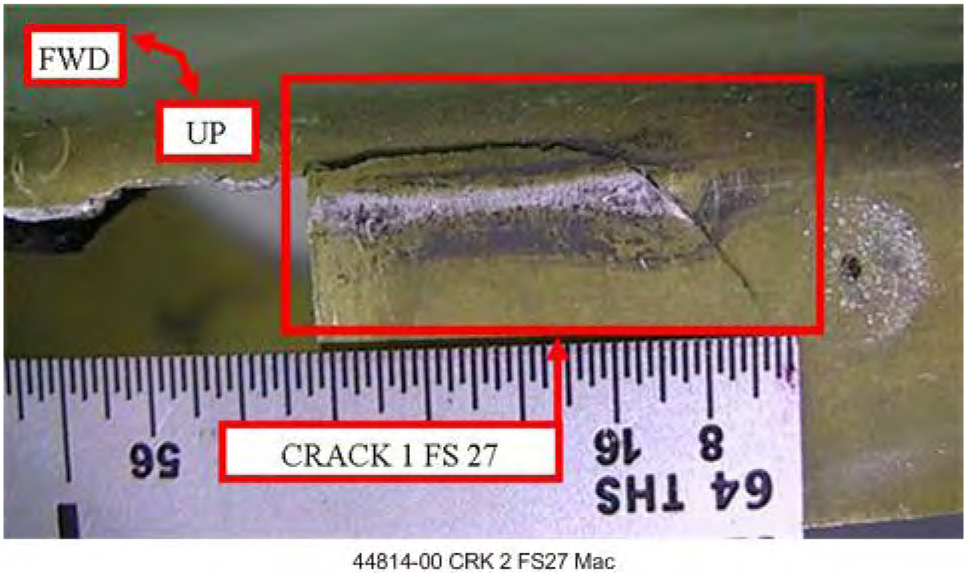


Figure 388. Macroscopic View of Crack 2 on the Fuselage Lower Plate Assembly FS 27

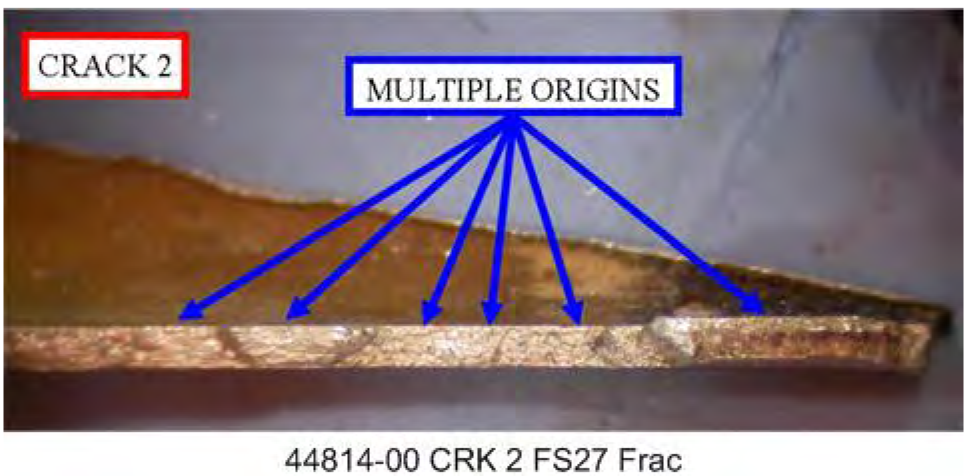


Figure 389. Fracture Face of Crack 2 on the Fuselage Lower Plate Assembly FS 27

3.4.2.3.2 Fuselage FS 35 Through FS 99.

Table 35 provides a detailed characterization of all defects found on the fuselage from FS 35 to FS 99 during the teardown evaluation.

Table 35. Inspection Results From FS 35 Through FS 99

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Fuselage upper left angle #2 beam assembly, figure 391	Corrosion	FS 35–68	Scattered over entire part	Corrosion indication	Light corrosion 2% thickness loss	103
Fuselage upper right angle B longitudinal beam assembly, figure 392	Corrosion	FS 57 through FS 61.75	1.5 inches by 5.5 inches	Corrosion indication	Light corrosion 1% thickness loss	103
Left floorboard support channel, figure 393	Crack	FS 57	0.09 inch	Crack indication	Hole crack	394 395
	Crack	FS 57	0.156 inch	Crack indication	Bend radii surface crack	396 397
	Crack	FS 57	1.25 inches	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	398
Fuselage upper fwd skin, figure 399	Crack	FS 63.25	1.691 inches	Crack indication	Surface crack	400
Fuselage bulkhead assembly, figure 401	Crack	FS 66.5	0.262 inch	Crack indication	Hole crack	402
Fuselage bulkhead assembly, figure 403	Damage	FS 87.5	0.693 inch by 0.34 inch	Indication	Damage 100% thickness loss	404
	Damage	FS 87.5	0.408 inch by 0.25 inch	Indication	Damage 100% thickness loss	404
	Damage	FS 87.5	0.354 inch by 0.21 inch	Indication	Damage 100% thickness loss	405 406

Table 35. Inspection Results From FS 35 Through FS 99 (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Bulkhead assembly, figure 407	Crack	FS 99	0.436 inch	Crack indication	Bend radii	408 409
	Crack	FS 99	0.196 inch	Crack indication	Bend radii surface crack	410 411
Center fuselage right stringer, figure 412	Crack	FS 102.25	0.12 inch	Crack indication	Bend radii surface crack	413 414 415

[1] Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 390 shows and overview of light scattered corrosion on the fuselage upper left angle 2 beam assembly, part number 44877-00, from FS 35 to FS 68. The corrosion was categorized as light corrosion because it caused a maximum thickness loss of only 2%. The location of an 8.25-square-inch area of light corrosion is shown in figure 391 on the fuselage upper right angle B longitudinal beam assembly, part number 44877-01B. This corrosion was located from FS 57 to FS 61.75 and caused a maximum localized reduction in thickness of 1%.

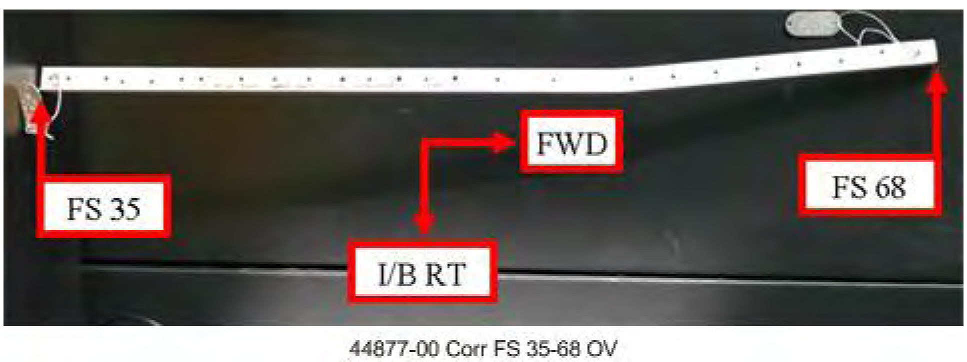


Figure 390. Overview of Fuselage Upper Left Angle 2 Beam Assembly FS 35 Through FS 68

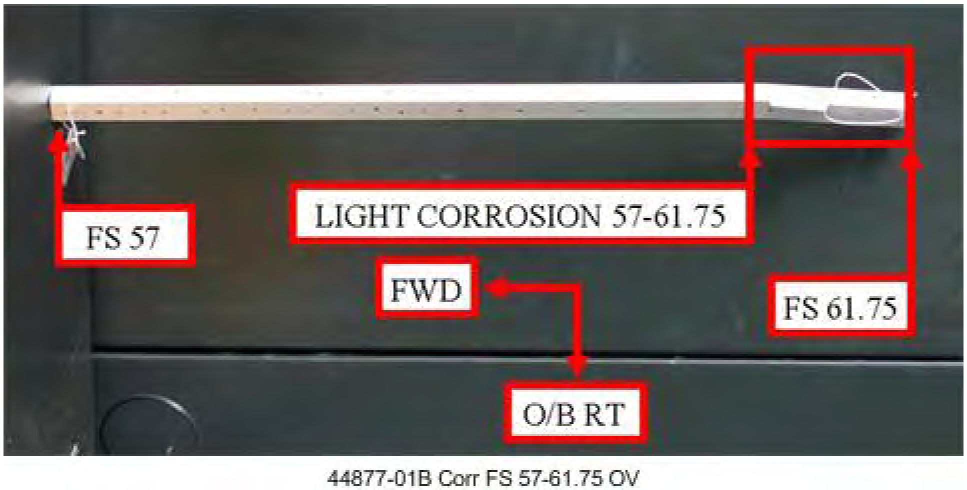


Figure 391. Location of Light Corrosion on the Fuselage Upper Right Angle B Longitudinal Beam Assembly FS 57 Through FS 61.75

The location of three cracks on the left floorboard support channel, part number 44804-00, is shown in figure 392. A macroscopic view of crack 1, which measured 0.09 inch in length, is shown in figure 393, and a microscopic view of this crack is shown in figure 394. Crack 2, which measured 0.156 inch, is shown macroscopically in figure 395 and microscopically in figure 396. A macroscopic view of crack 3 is shown in figure 397. This 1.25-inch crack was analyzed, but due to extensive crack face smearing, the failure mode could not be determined.

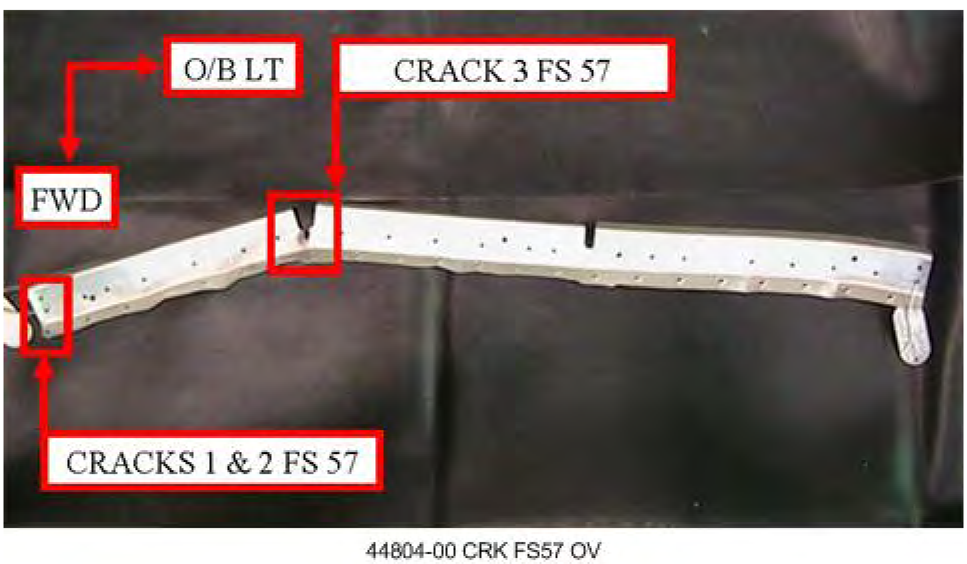


Figure 392. Location of Three Cracks on the Left Floorboard Support Channel FS 57

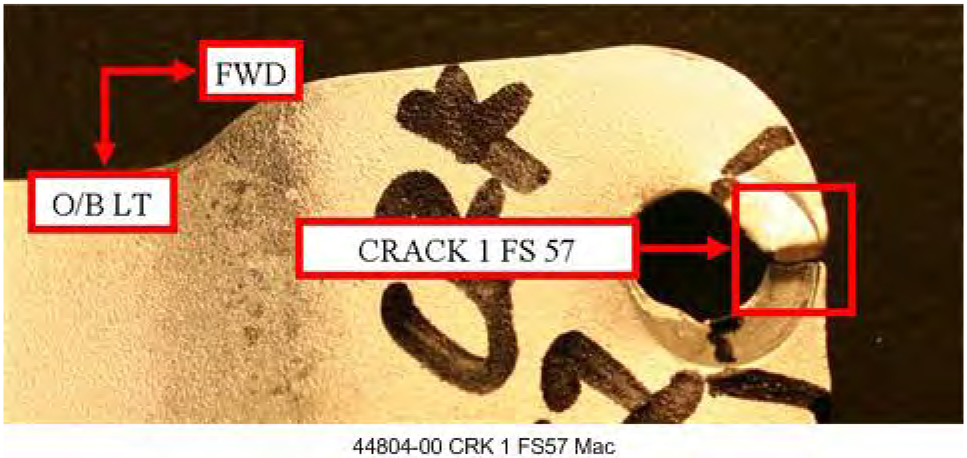


Figure 393. Macroscopic View of Crack 1 on the Left Floorboard Support Channel FS 57

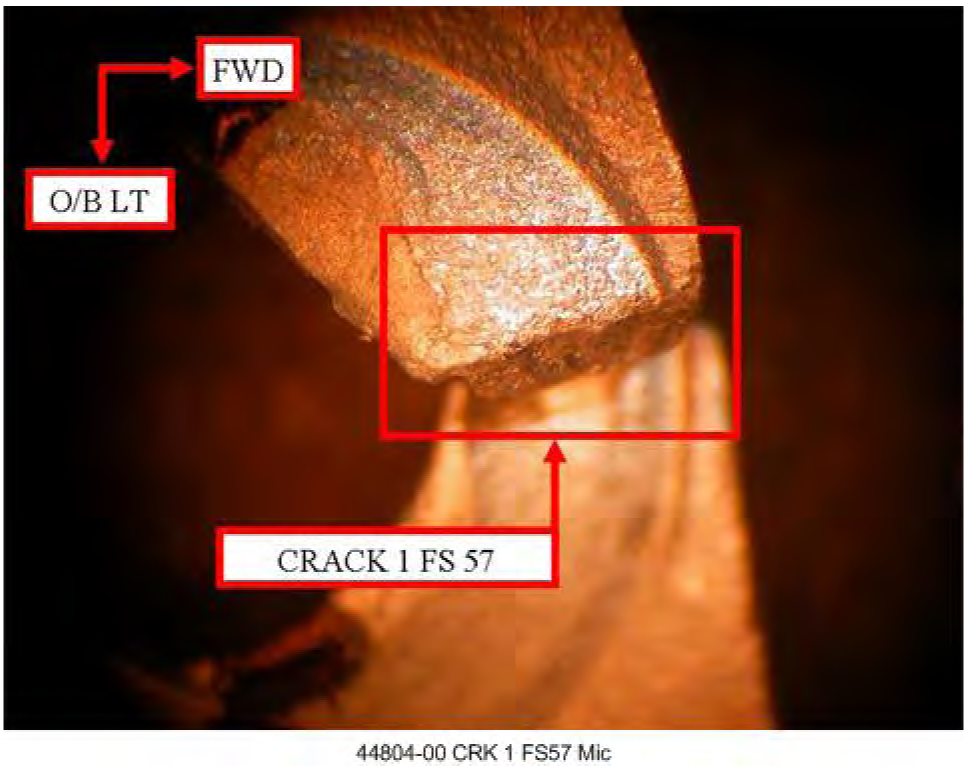


Figure 394. Microscopic View of Crack 1 on the Left Floorboard Support Channel FS 57

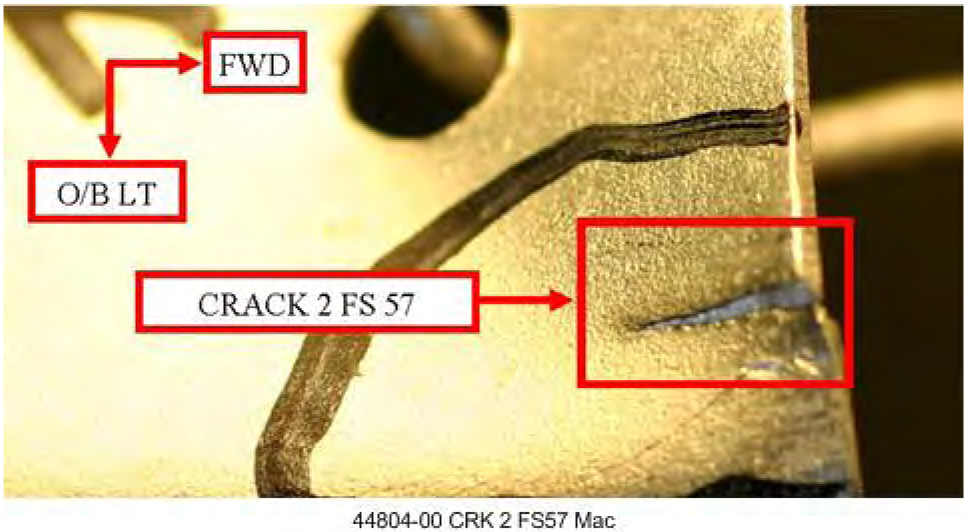


Figure 395. Macroscopic View of Crack 2 on the Left Floorboard Support Channel FS 57

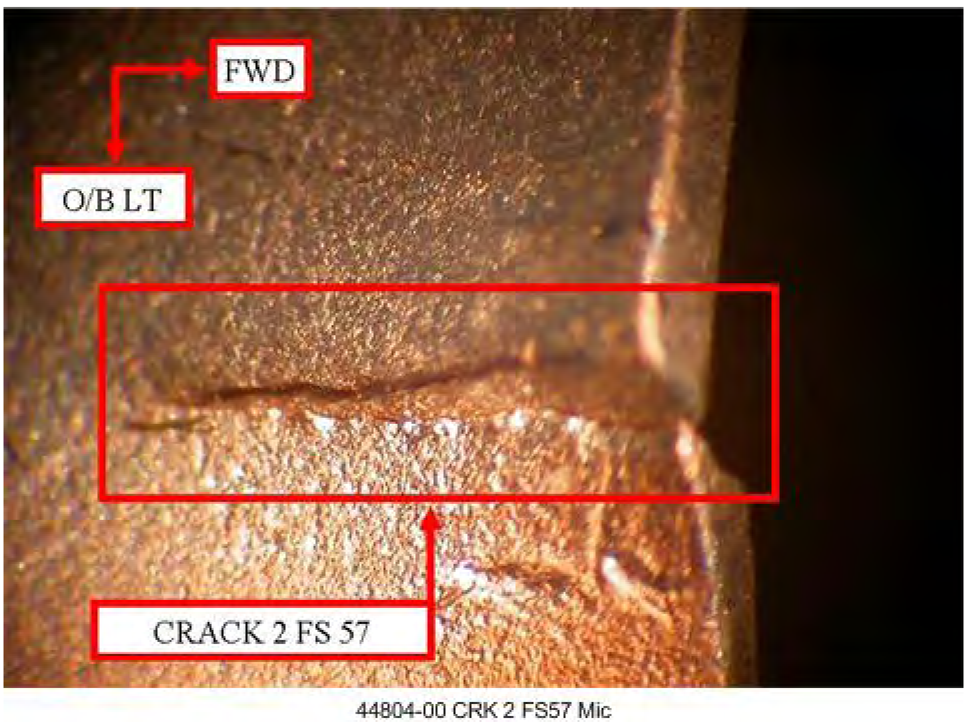
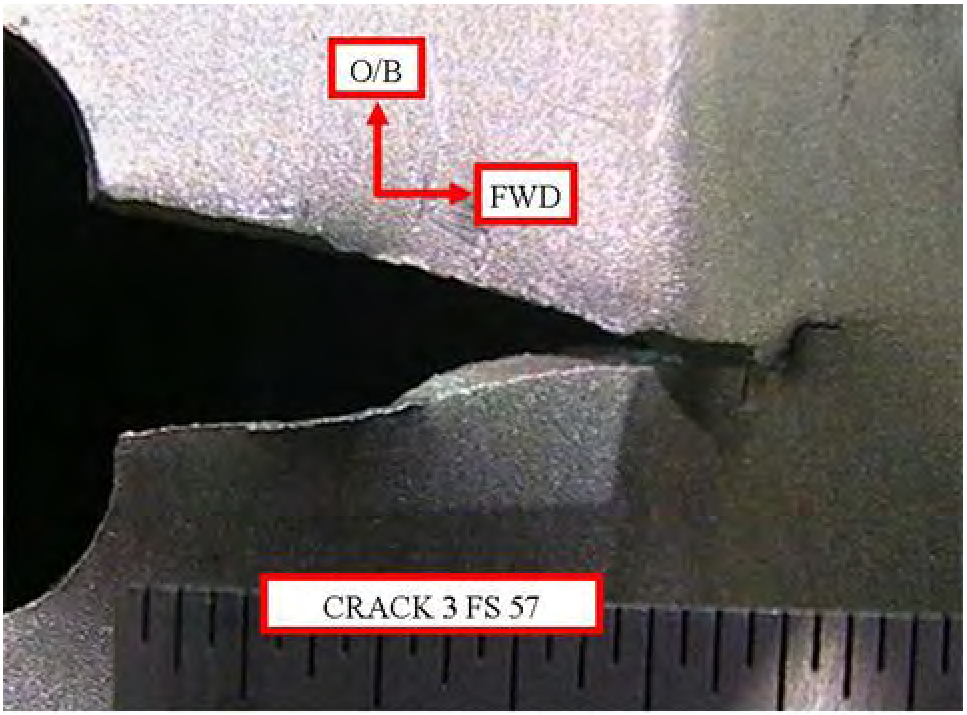


Figure 396. Microscopic View of Crack 2 on the Left Floorboard Support Channel FS 57



44804-00 CRK 3 FS57 Mac

Figure 397. Macroscopic View of Crack 3 on the Left Floorboard Support Channel FS 57

The location of a 1.691-inch crack on the fuselage upper forward skin, part number 52118-00, is shown in figure 398, and a macroscopic view of this crack is shown in figure 399. Figure 400 shows the location of a 0.262-inch crack on the fuselage bulkhead assembly, part number 40962-3. A macroscopic view of this crack, which occurs at FS 66.5, is shown in figure 401. The location of three areas of damage on the fuselage bulkhead assembly, located at FS 87.5, is shown in figure 402. Figure 403 shows a macroscopic view of damage 1 and 2 on the bulkhead assembly, part number 52157-6. Damage 1, which covered a surface area of 0.24 square inch, and damage 2, which covered 0.1 square inch, both resulted in localized areas of complete thickness loss. A macroscopic picture of damage 3 is shown in figure 404. This damage covered a surface area of 0.07 square inch and caused a maximum localized reduction in thickness of 100%. The fluorescent liquid penetrant indication for this damage is shown in figure 405.

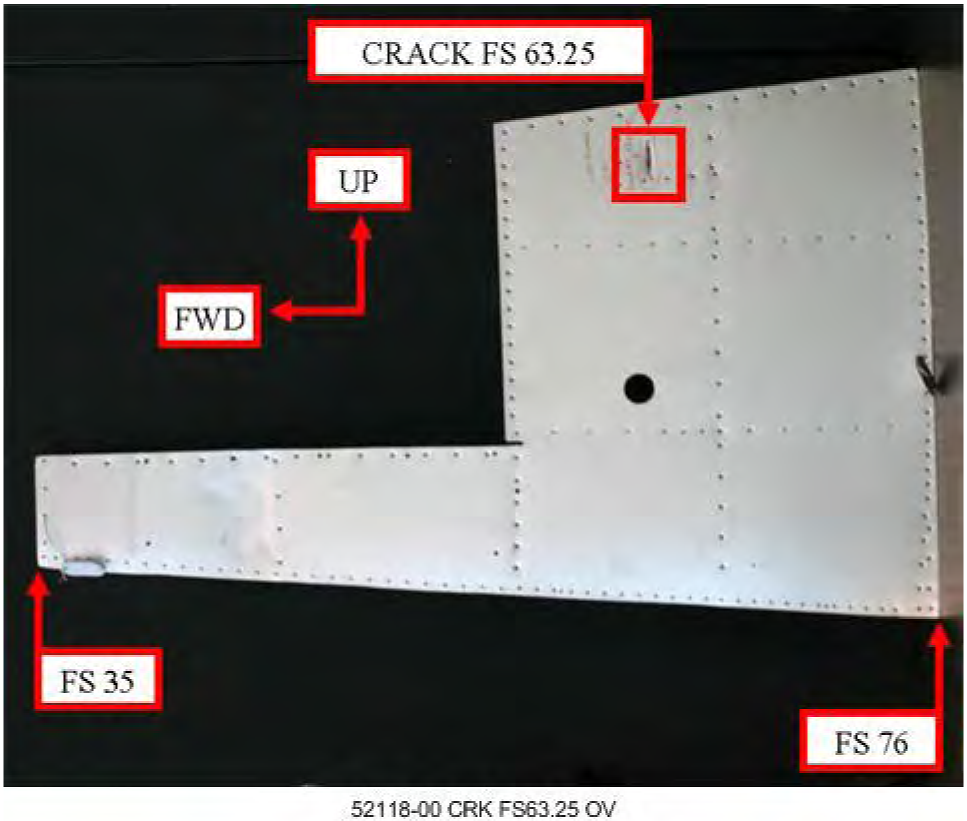


Figure 398. Location of Crack on the Fuselage Upper Forward Skin FS 63.25

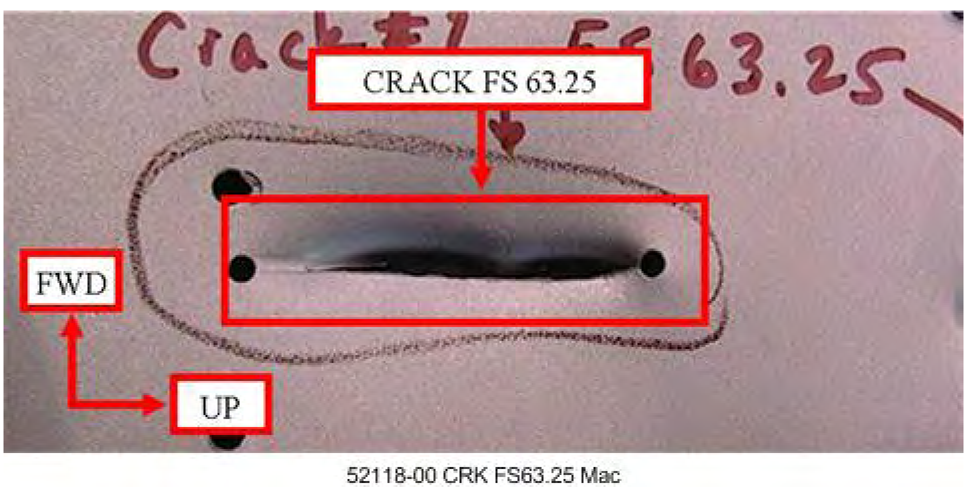


Figure 399. Macroscopic View of the Crack in the Fuselage Upper Forward Skin FS 63.25

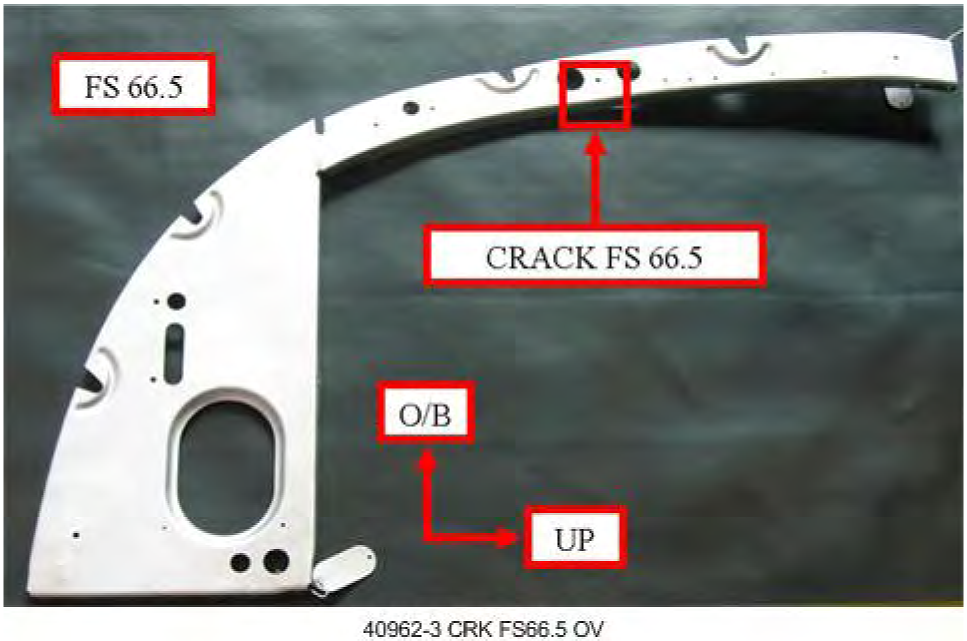


Figure 400. Location of Crack on the Fuselage Bulkhead Assembly FS 66.5

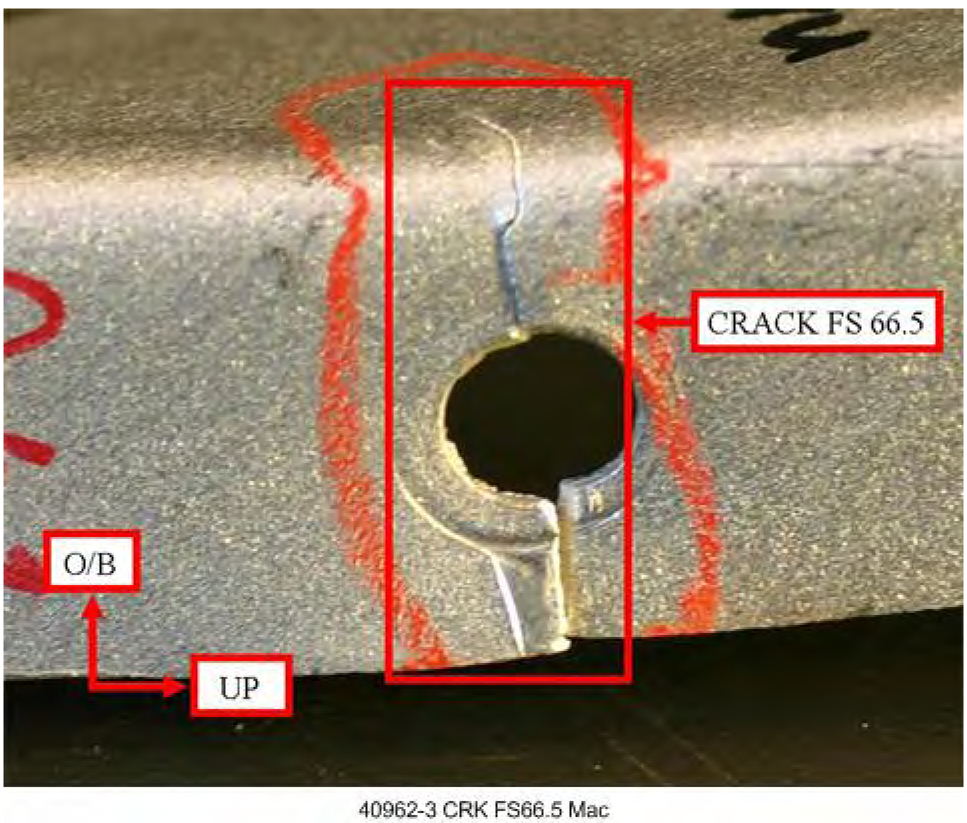


Figure 401. Macroscopic View of Crack on the Fuselage Bulkhead Assembly FS 66.5

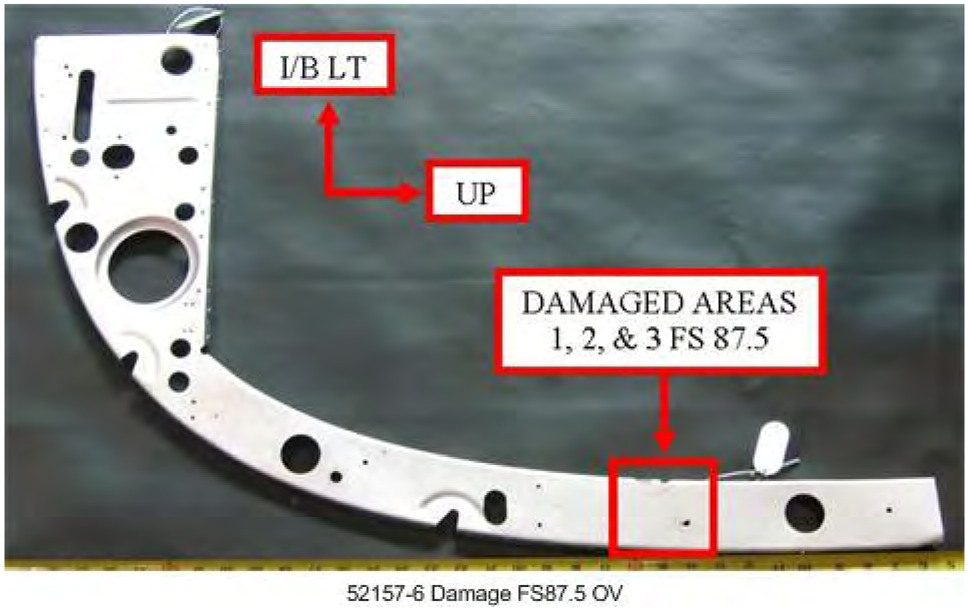


Figure 402. Location of Three Areas of Damage on the Fuselage Bulkhead Assembly FS 87.5

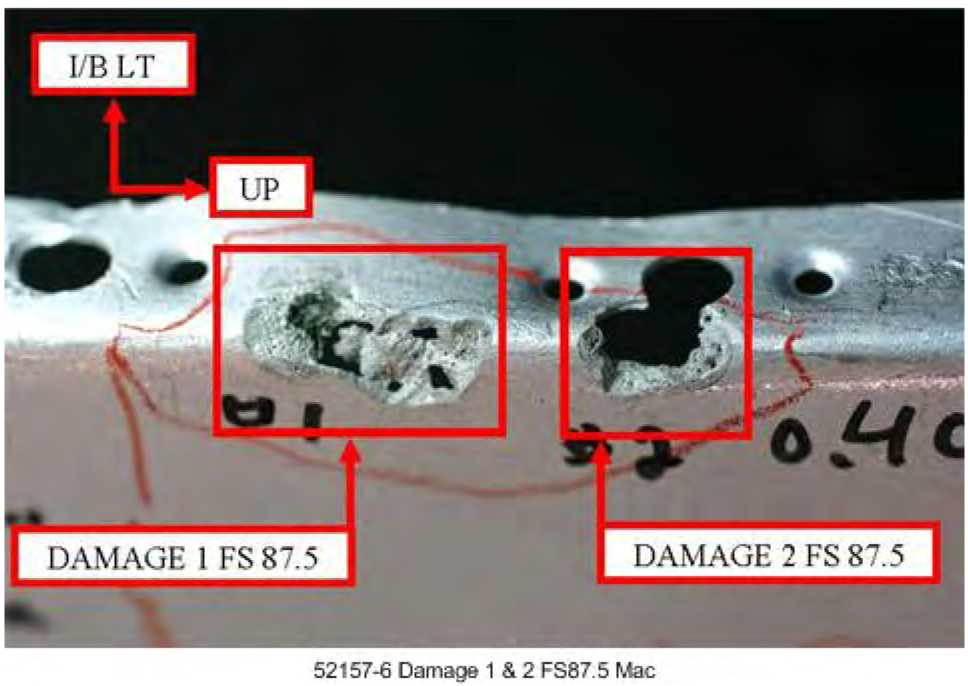
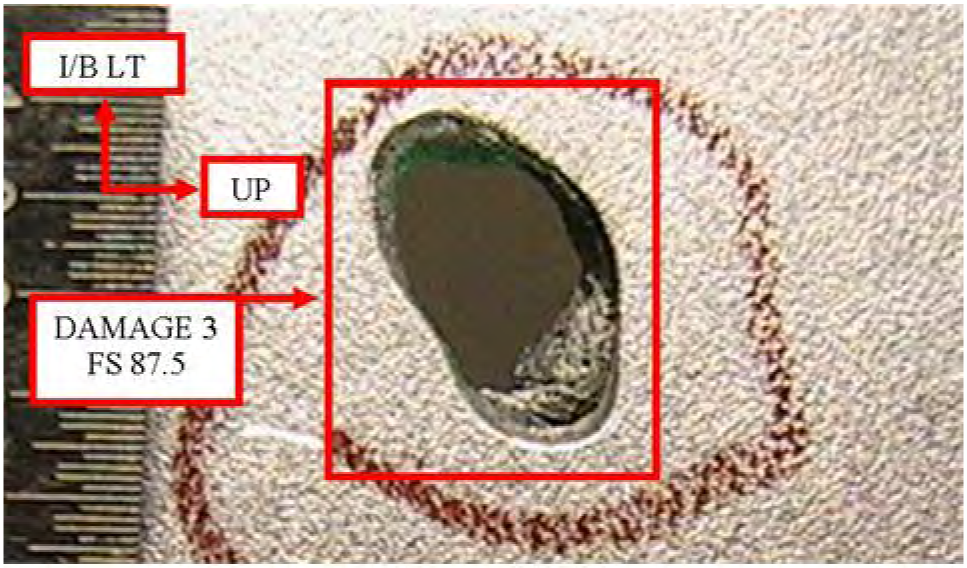
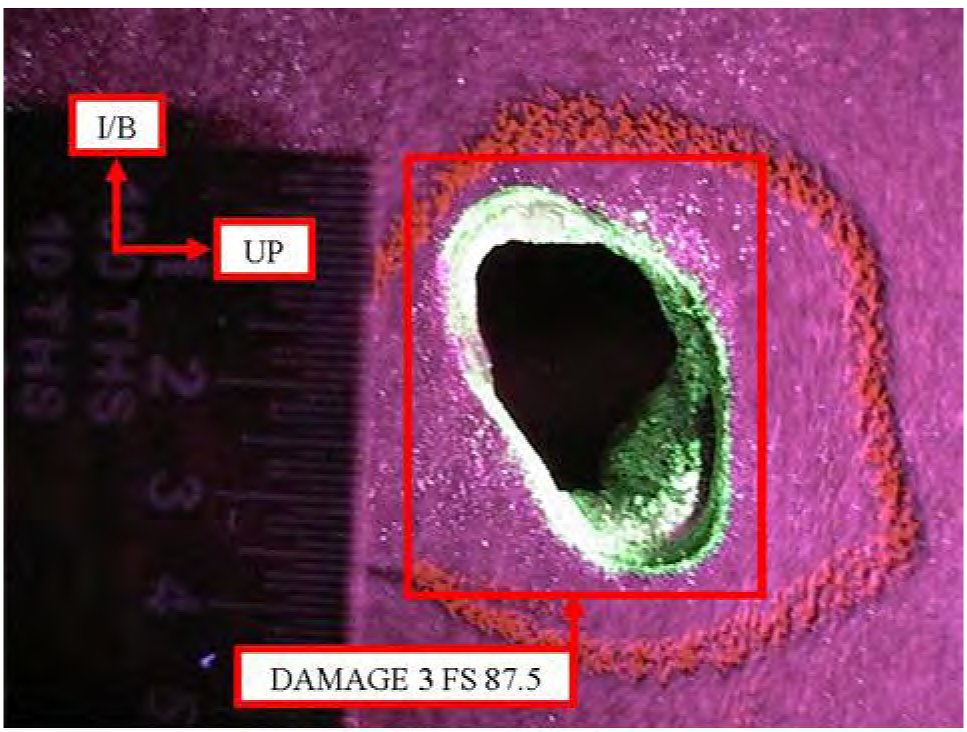


Figure 403. Macroscopic View of Damage 1 and 2 on the Fuselage Bulkhead Assembly FS 87.5



52157-6 Damage 3 FS87.5 Mac

Figure 404. Macroscopic View of Damage 3 on the Fuselage Bulkhead Assembly FS 87.5



52157-6 Damage 3 FS87.5 FLP

Figure 405. Fluorescent Liquid Penetrant Indication of Damage 3 on the Fuselage Bulkhead Assembly FS 87.5

The location of two cracks on the bulkhead assembly, part number 52084-002, located at FS 99 is shown in figure 406. Crack 1, which measured 0.436 inch in length, is shown macroscopically in figure 407 and microscopically in figure 408. A macroscopic view of crack 2 is shown in figure 409. This crack, which measured 0.196 inch, is shown microscopically in figure 410. The location of a 0.12-inch crack on the center fuselage right stringer, part number 52080-05, is shown in figure 411. This crack, located at FS 102.25 is shown macroscopically in figure 412. The fluorescent liquid penetrant indication for this crack is shown in figure 413, and a microscopic view of the crack is shown in figure 414.

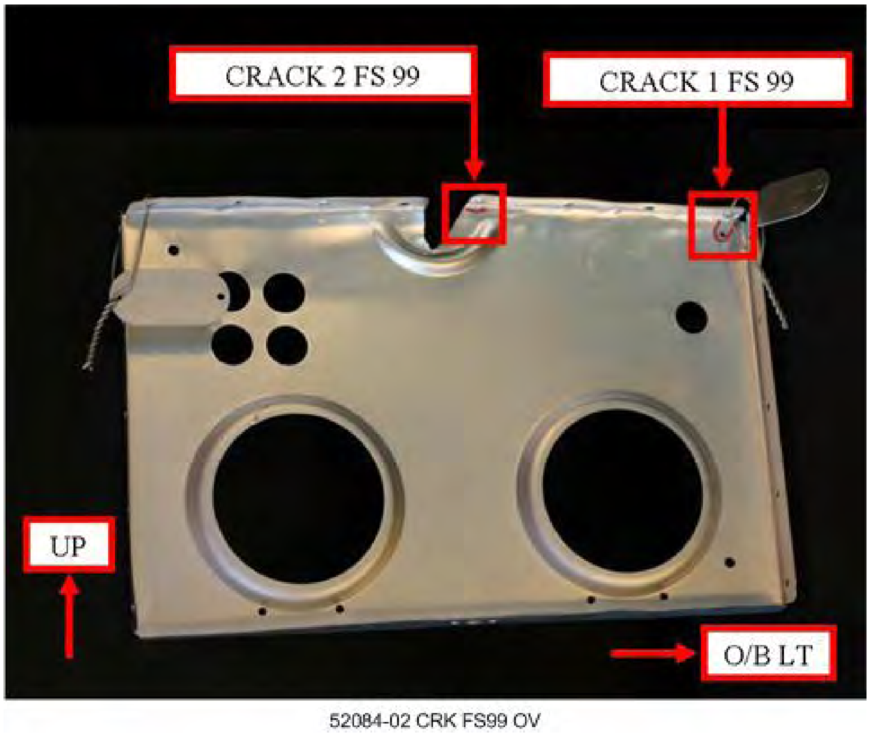


Figure 406. Location of Cracks 1 and 2 on the Bulkhead Assembly FS 99

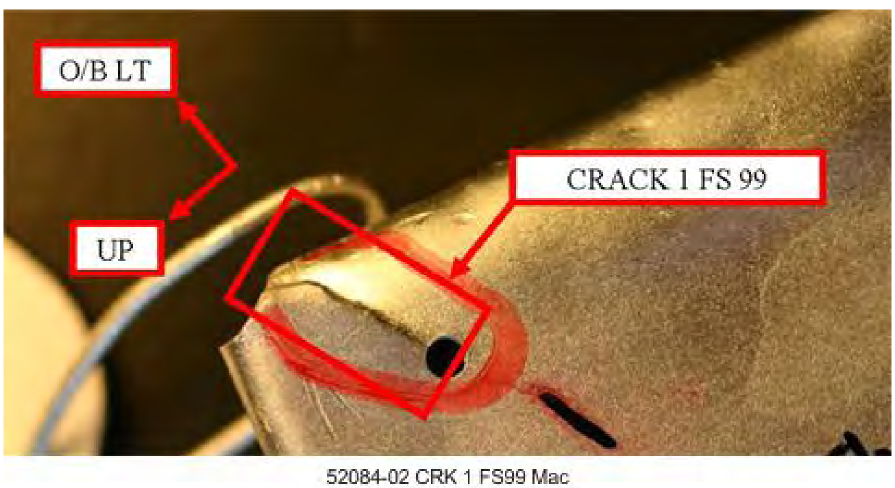
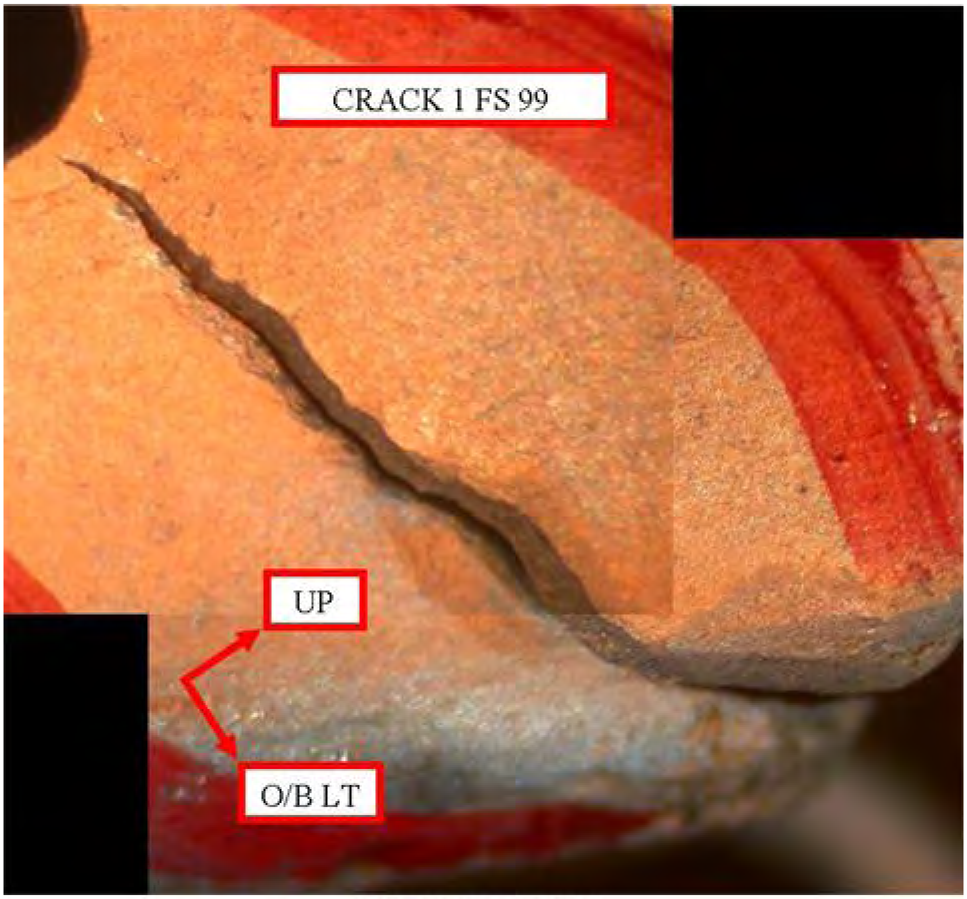
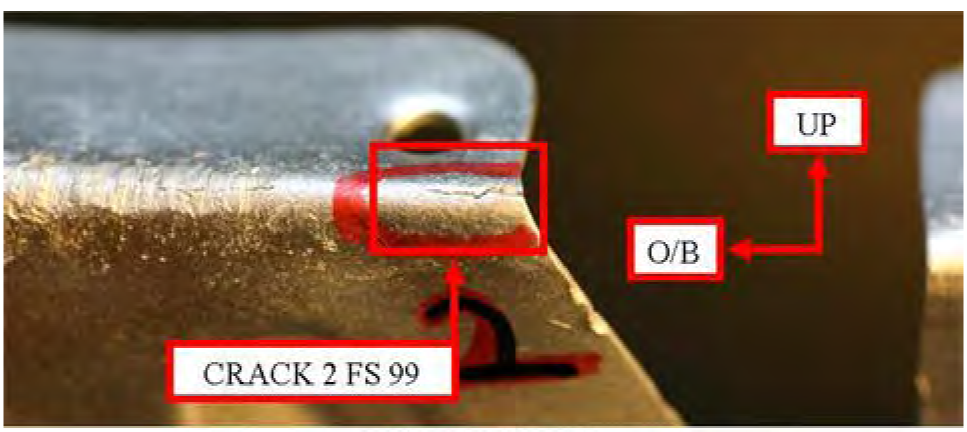


Figure 407. Macroscopic View of Crack 1 on the Bulkhead Assembly FS 99



52084-02 CRK 1 FS99 Mic

Figure 408. Microscopic View of Crack 1 on the Bulkhead Assembly FS 99

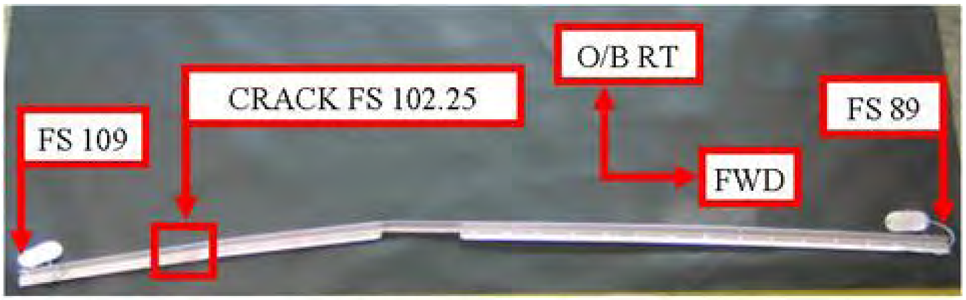


52084-02 CRK 2 FS99 Mac

Figure 409. Macroscopic View of Crack 2 on the Bulkhead Assembly FS 99

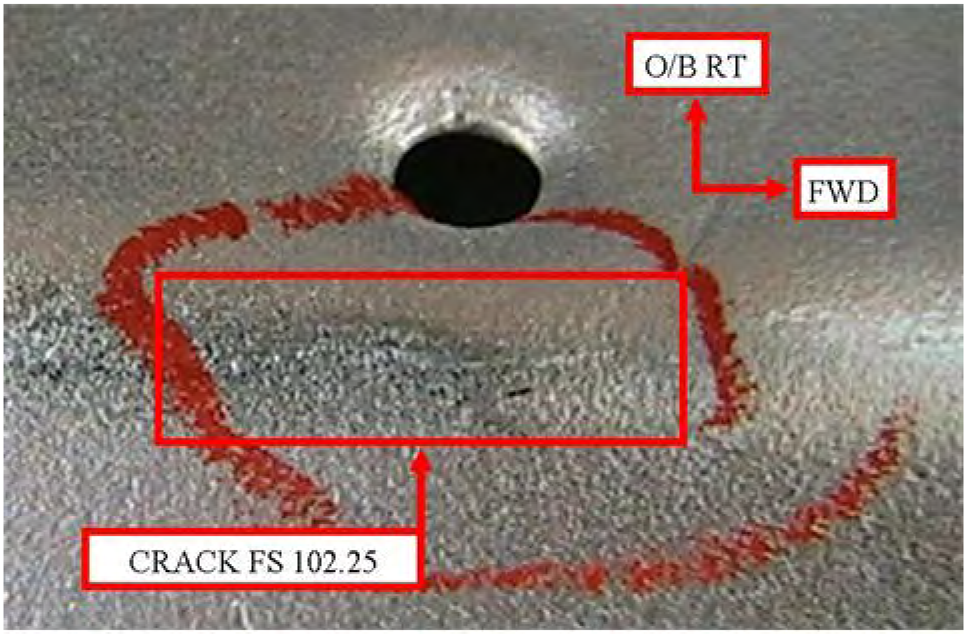
52084-02 CRK 2 FS99 Mic

Figure 410. Microscopic View of Crack 2 on the Bulkhead Assembly FS 99



52080-05 CRK FS102.25 OV

Figure 411. Location of Crack on the Center Fuselage Right Stringer FS 89 Through FS 109



52080-05 CRK FS102.25 Mac

Figure 412. Macroscopic View of Crack on the Center Fuselage Right Stringer at FS 102.25

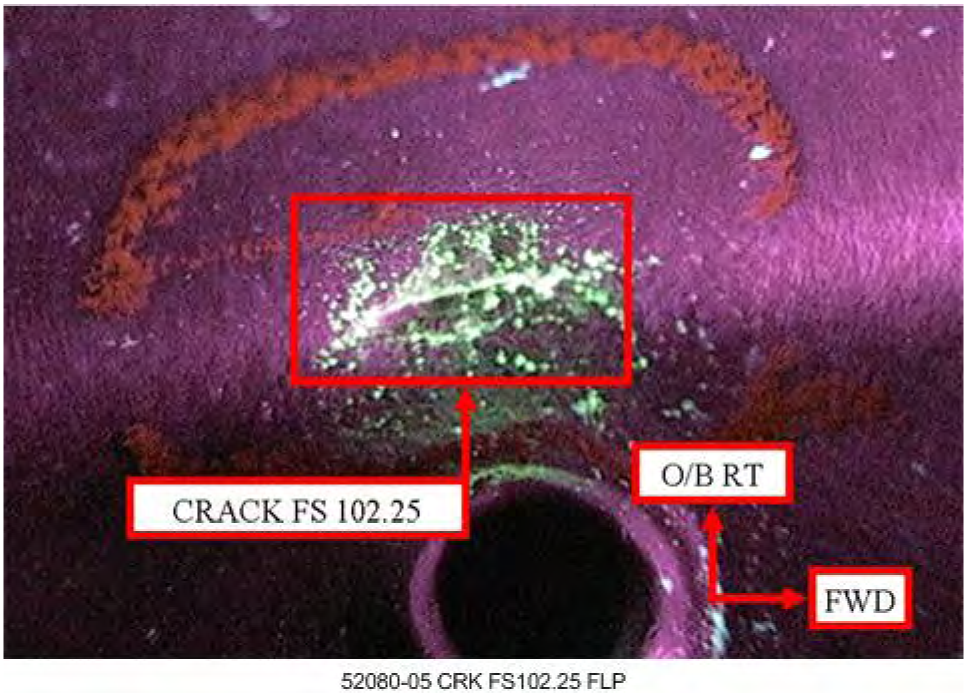


Figure 413. Fluorescent Liquid Penetrant Indication of the Crack on the Center Fuselage Right Stringer FS 102.25

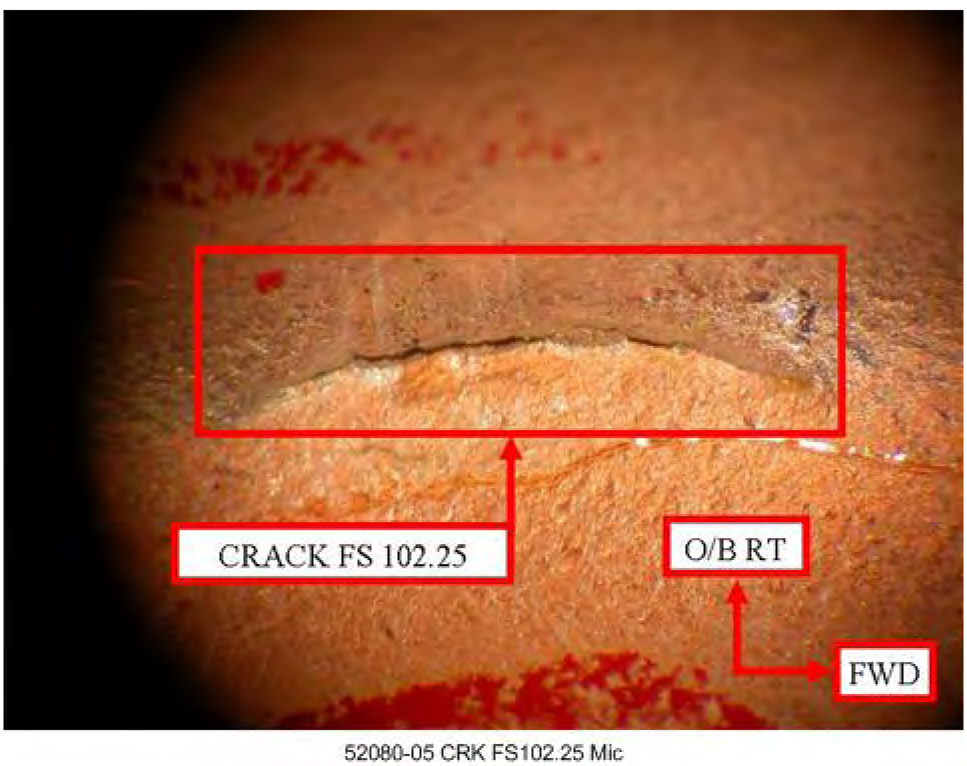


Figure 414. Microscopic View of the Crack on the Center Fuselage Right Stringer FS 102.25

3.4.2.3.3 Fuselage FS 100 Through FS 139.

A detailed characterization of all defects found on the fuselage from FS 100 to FS 139 is shown in table 36.

Table 36. Inspection Results From FS 100 Through FS 139

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right floorboard channel, figure 416	Crack	FS 91.5	0.045 inch	Crack indication	Hole crack	417
	Crack	FS 112	0.097 inch	Crack indication	Hole crack	418
	Crack	FS 123	1.1605 inches	Crack indication	Surface crack	419
Right front spar attachment, figure 428	Crack	FS 104.31	0.353 inch	Not inspected[1]	Fatigue	421 422
	Crack	FS 104.31	0.310 inch	Not inspected[1]	Fatigue	421 423
	Crack	FS 104.31	0.112 inch	Not inspected[1]	Fatigue	424
Right front spar attachment 2 of 3 (B), figure 425	Wear	FS 104.31	2.59 inches by 1 inch	No indication	Wear 28% thickness loss	426
Left front spar attachment 2 of 3 (B), figure 427	Wear	FS 104.31	2.58 inches by 1.15 inches	No indication	Wear 28% thickness loss	428
Left front spar attachment, figure 429	Crack	FS 104.68	1 inch	Not inspected[1]	Fatigue	430 431 432
	Crack	FS 104.68	0.334 inch	Not inspected[1]	Fatigue	433
	Crack	FS 104.68	0.250 inch	Not inspected[1]	Fatigue	434
Left lower forward longitudinal frame beam assembly, figure 436	Wear	FS 106 through FS 120	14 inches by 9 inches	No indication	Wear 100% thickness loss	436
	Crack	FS 112	0.223 inch	Crack indication	Surface crack	437

Table 36. Inspection Results From FS 100 Through FS 139 (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Bulkhead assembly lower forward left frame, figure 438	Crack	FS 111	0.289 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	439
	Crack	FS 111	0.375 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	440
	Crack	FS 111	0.336 inch	Crack indication	Bend radii	441 442
Left inboard floorboard support, figure 443	Corrosion	FS 111	0.44 inch by 0.39 inch	Corrosion indication	Moderate-severe corrosion 9.3% thickness loss	443
	Corrosion	FS 118	0.525 inch by 0.6 inch	Corrosion indication	Moderate corrosion 6.2% thickness loss	445
	Corrosion	FS 123	2.1 inches by 0.89 inch	Corrosion indication	Severe corrosion 25% thickness loss	446
Fuselage upper left angle 4 longitudinal beam assembly, figure 447	Corrosion	FS 115.5	0.75 inch by 0.75 inch	Corrosion indication	Light corrosion 1% thickness loss	103
	Corrosion	FS 129.5	2.25 inches by 0.5 inch	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
Fuselage lower forward stringer, figure 448	Crack	FS 122	0.274 inch	Not inspected[1]	Overload bend radii	448 450
Fuselage right floorboard channel support, figure 450	Crack	FS 123	0.738 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	452
	Crack	FS 135	1.297 inches	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	453
Fuselage longitudinal beam lower right frame, figure 454	Corrosion	FS 130.75 through FS 133	2.25 inches by 0.95 inch	Corrosion indication	Light-moderate corrosion 2.5% thickness loss	105

[1] Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 415 shows the location of three cracks on the right floorboard channel, part number 44801-01. A macroscopic view of crack 1 on the right floorboard channel is shown in figure 416. This crack, which was located at FS 91.5, measured 0.045 inch in length. A macroscopic view of crack 2 is shown in figure 417, which measured 0.097 inch and was located at FS 112. Figure 418 shows a macroscopic view of crack 3, which was located at FS 123 and measured 1.1605 inch in length.

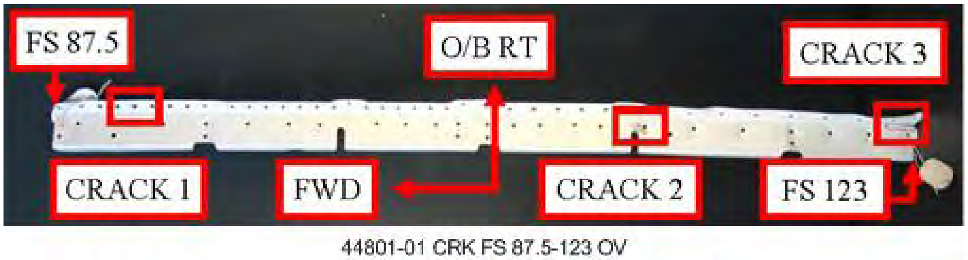


Figure 415. Location of Three Cracks on the Right Floorboard Channel FS 87.5 Through FS 123

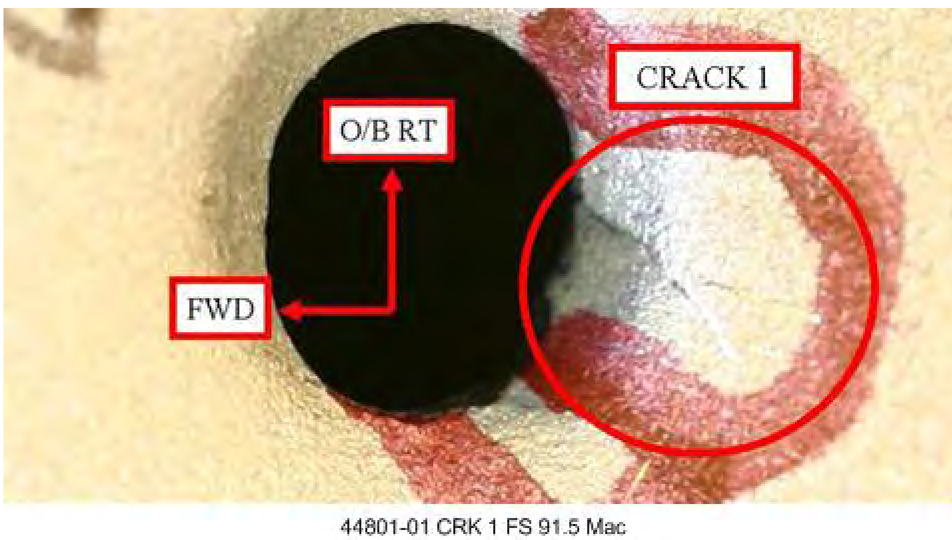


Figure 416. Macroscopic View of Crack 1 on the Right Floorboard Channel FS 91.5

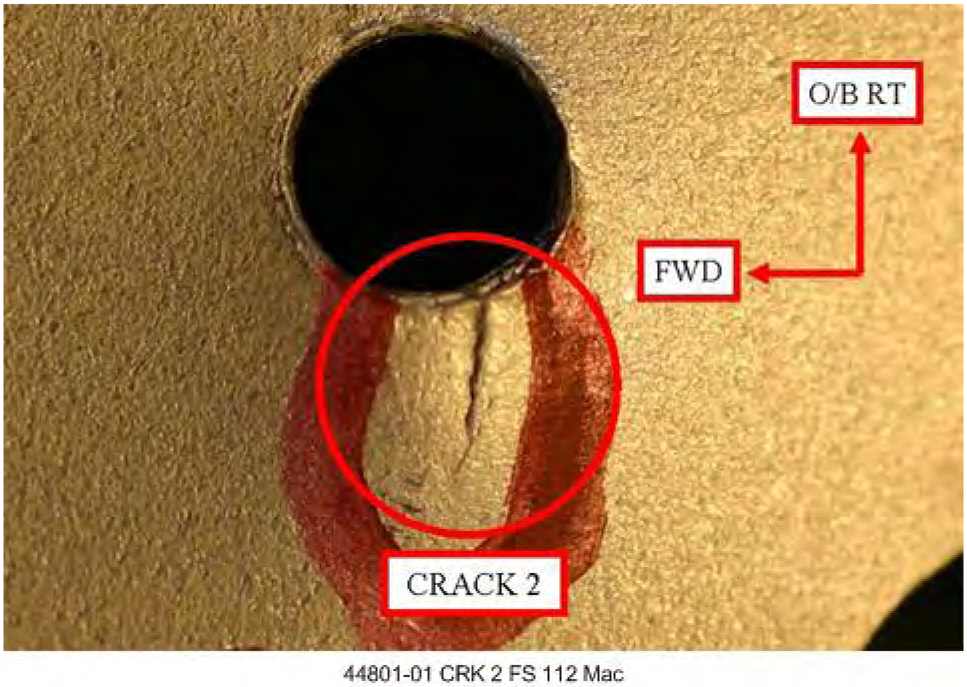


Figure 417. Macroscopic View of Crack 2 on the Right Floorboard Channel FS 112

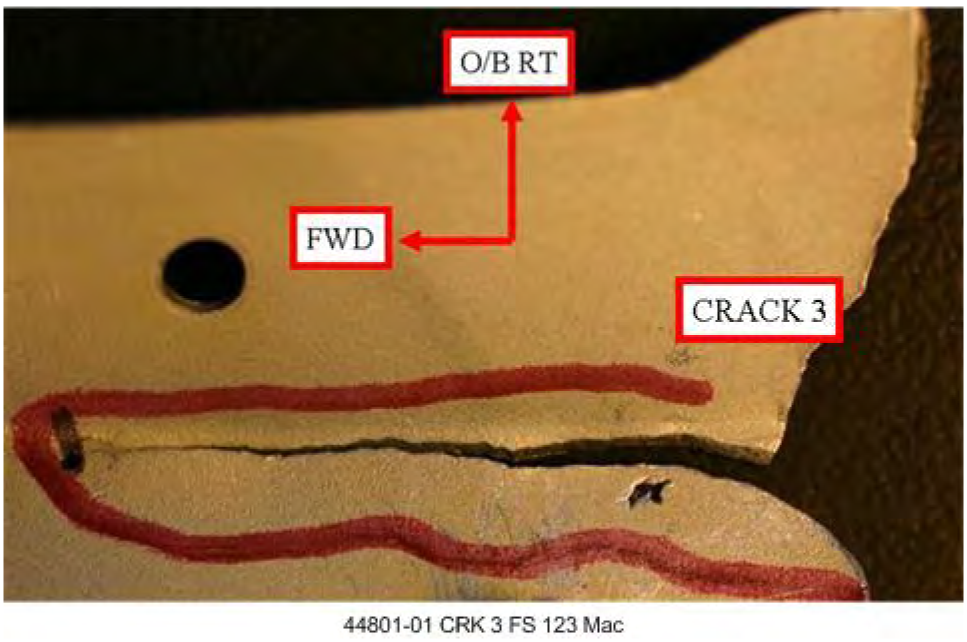
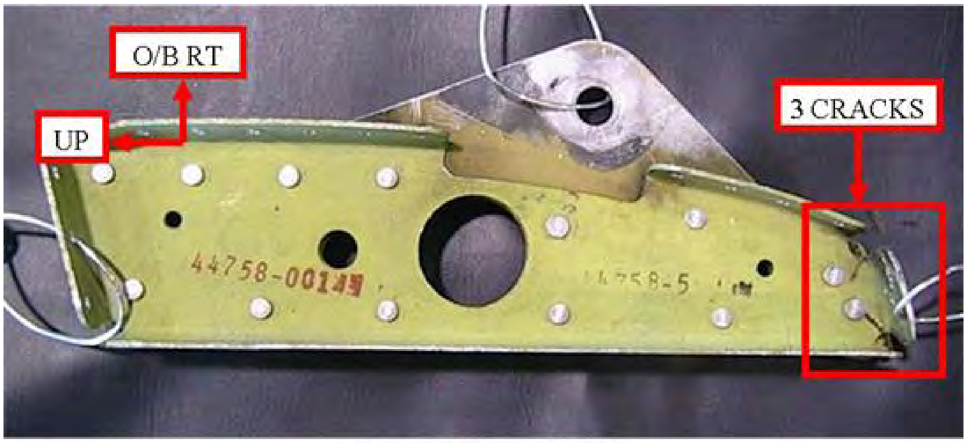


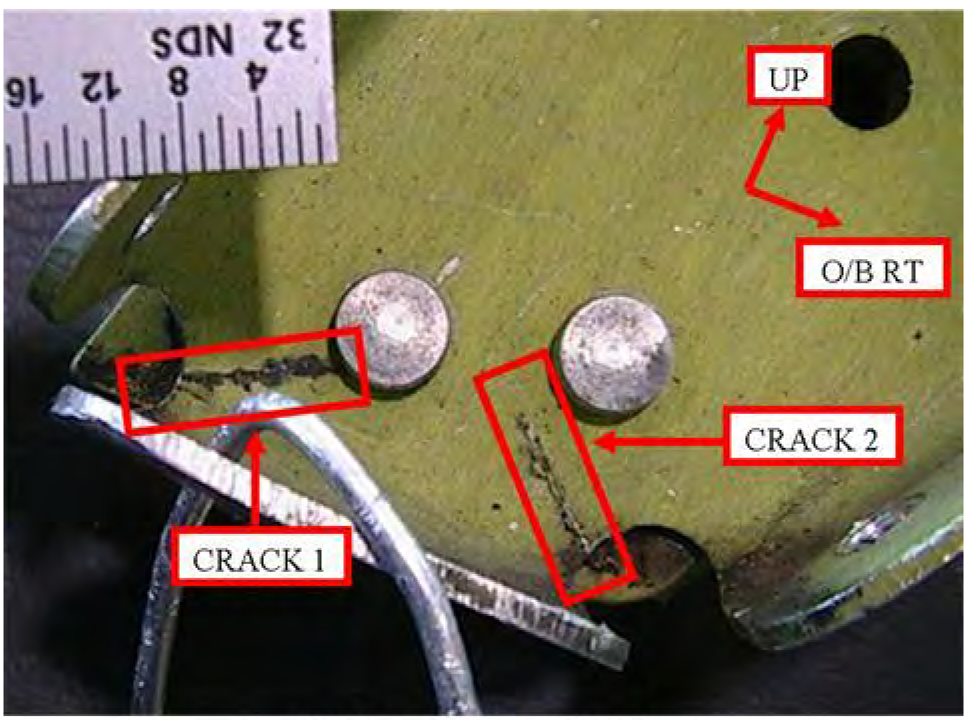
Figure 418. Macroscopic View of Crack 3 on the Right Floorboard Support FS 123

The location of three cracks on the right front spar attachment, part number 44758-01, is shown in figure 419. Figure 420 shows a macroscopic view of cracks 1 and 2 at FS 104.31. Figure 421 shows a fracture face picture of crack 1, which measured 0.353 inch, on the right front spar attachment, and figure 422 shows a fracture face picture for the 0.310-inch crack, identified as crack 2. Figure 423 shows a macroscopic view of crack 3, which measured 0.112 inch. From the fatigue striations evident on the fracture faces of cracks 1 and 2, it was determined that the failure mode for all three cracks was fatigue.



44758-00(RT) CRK FS 104.68 OV

Figure 419. Location of Three Cracks on the Right Front Spar Attachment FS 104.31



44758-00(RT) CRK 1 & 2 FS 104.68 Mac

Figure 420. Macroscopic View of Cracks 1 and 2 on the Right Front Spar Attachment FS 104.31

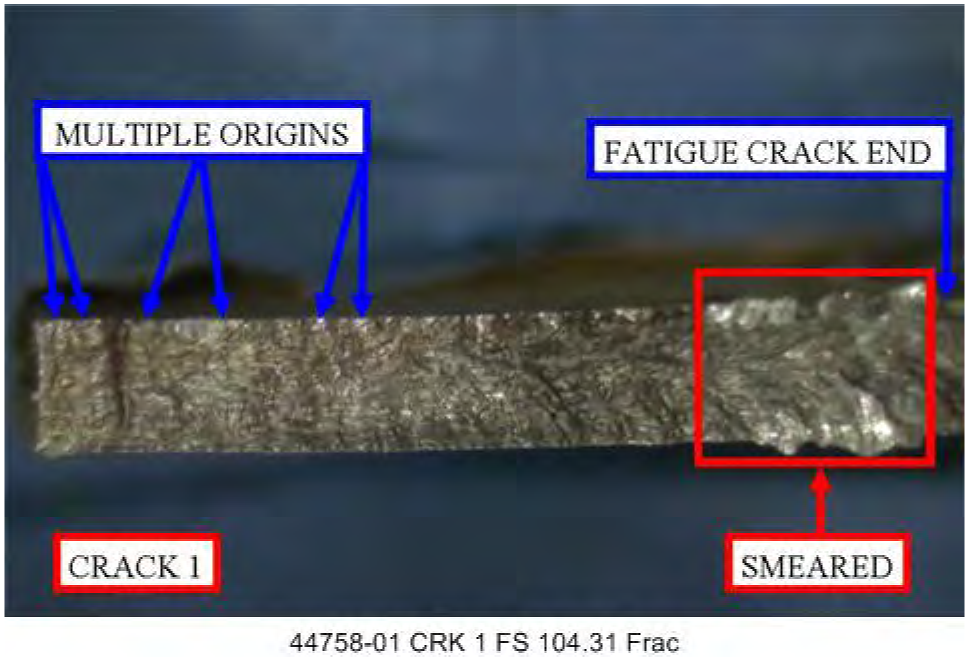


Figure 421. Fracture Face of Crack 1 on the Right Front Spar Attachment FS 104.31

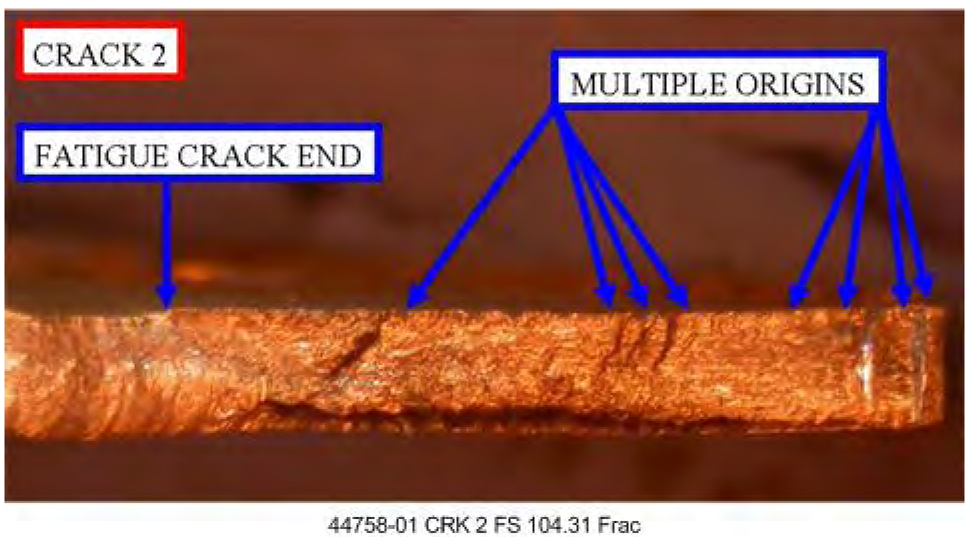
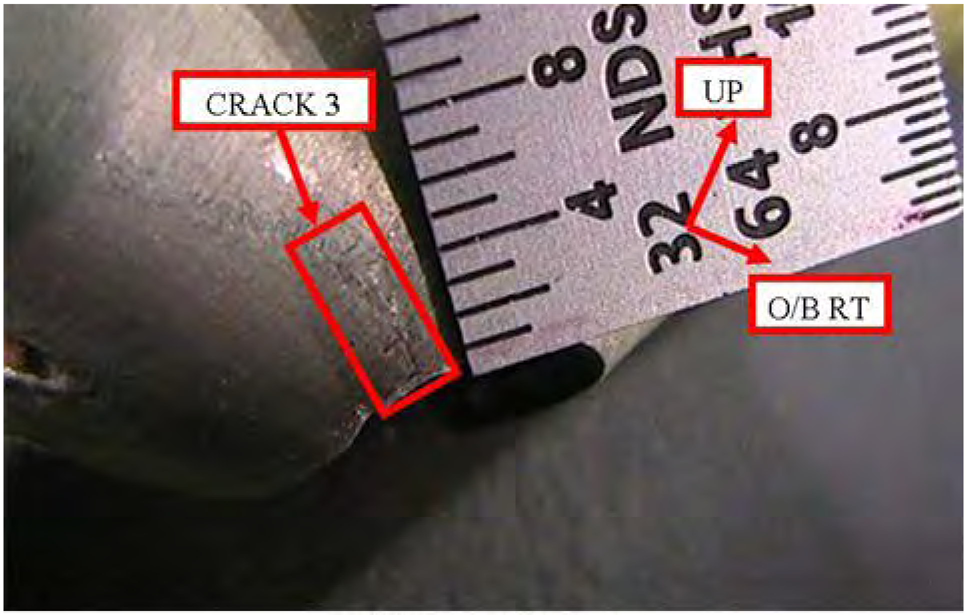


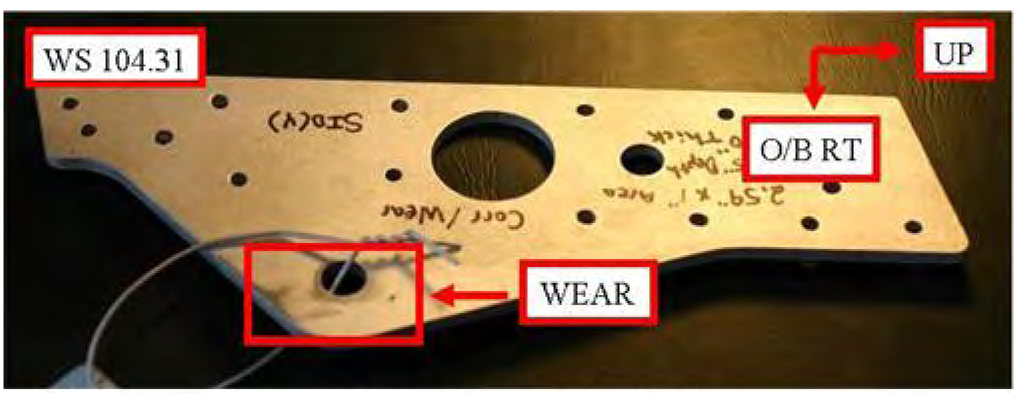
Figure 422. Fracture Face of Crack 2 on the Right Front Spar Attachment FS 104.31



44758-00 CRK 3 FS 104.68 Mac

Figure 423. Macroscopic View of Crack 3 on the Right Front Spar Attachment FS 104.31

Figure 424 shows the location of wear on the right front spar attachment 2 of 3 (B), part number 44758-01. A macroscopic view of the wear is shown in figure 425. This area of wear covered a surface area of 2.59 square inches and caused a localized reduction in thickness of 28%. The location of a similar area of wear on the left front spar attachment 2 of 3 (B) is shown in figure 426. A macroscopic view of this 2.97-square-inch area of corrosion, which caused a maximum localized thickness loss of 28%, is shown in figure 427.



44758-1(RT) Wear FS 104.31 2 of 3 OV

Figure 424. Location of Wear on the Right Front Spar Attachment 2 of 3 (B) FS 104.31

Figure 425. Macroscopic View of Wear on the Right Front Spar Attachment 2 of 3 (B) FS 104.31

Figure 426. Location of Wear on the Left Front Spar Attachment 2 of 3 (B) FS 104.31

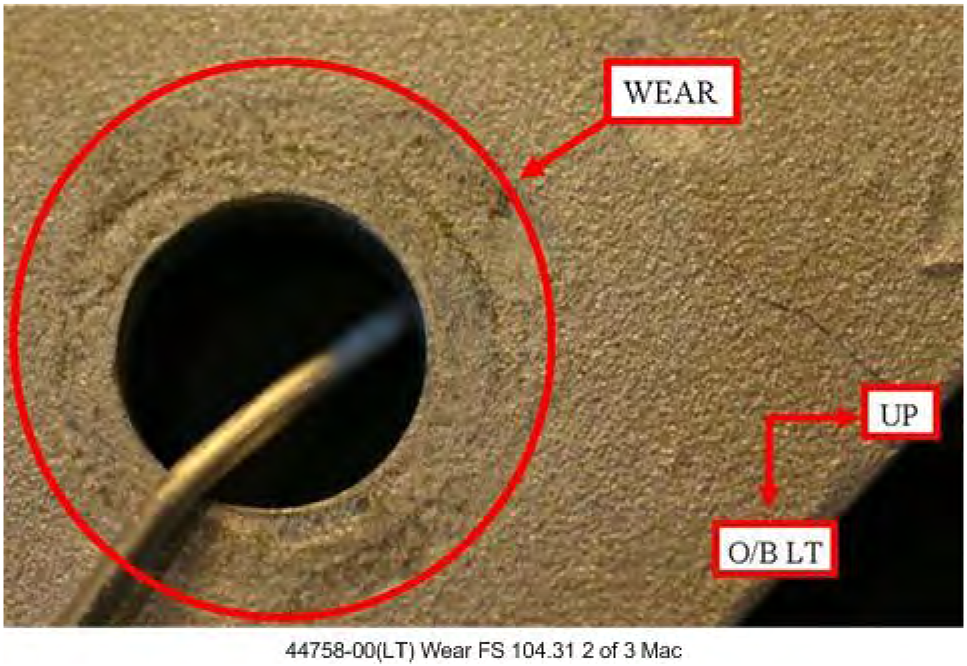


Figure 427. Macroscopic View of Wear on the Left Front Spar Attachment 2 of 3 (B) FS 104.31

The location of three cracks on the left front spar attachment is shown in figure 428. A macroscopic view of crack 1, which measured 1 inch, is shown in figure 429, and the fracture face of crack 1 is shown in figure 430. A fractograph of this crack, located at FS 104.68, is shown in figure 431, which shows crack face characteristics typical of fatigue. A macroscopic view of crack 2, which measured 0.334 inch, on the left front spar attachment, part number 44758-00, is shown in figure 432. A macroscopic view of crack 3, which measured 0.25 inch, is shown in figure 433. All three cracks were determined to be caused by fatigue.

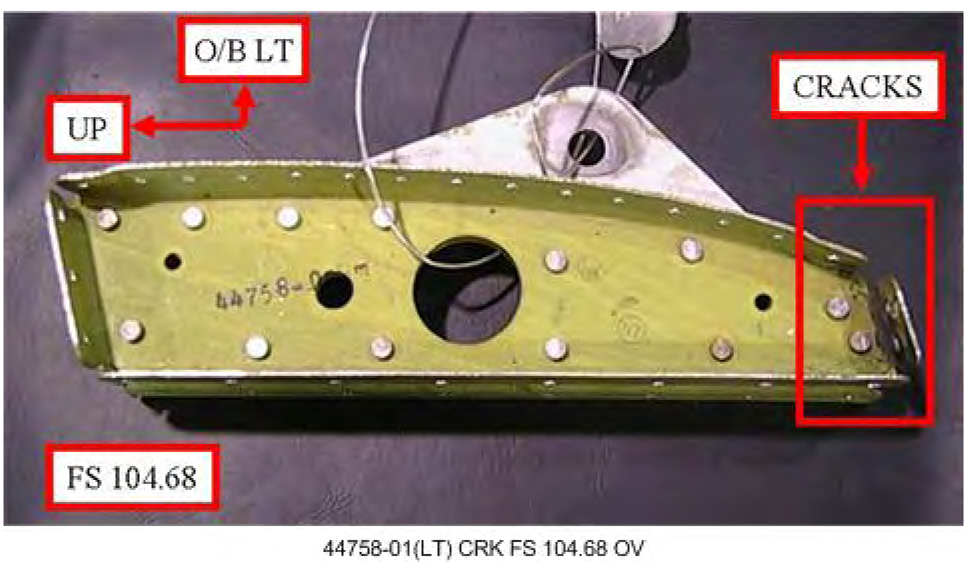


Figure 428. Location of Three Cracks on the Left Front Spar Attachment FS 104.68

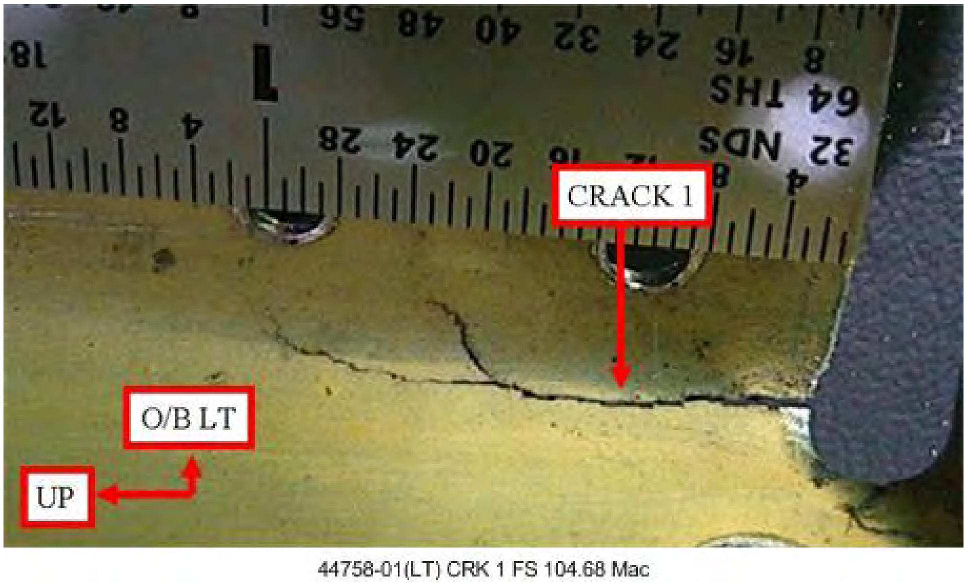


Figure 429. Macroscopic View of Crack 1 on the Left Front Spar Attachment FS 104.68

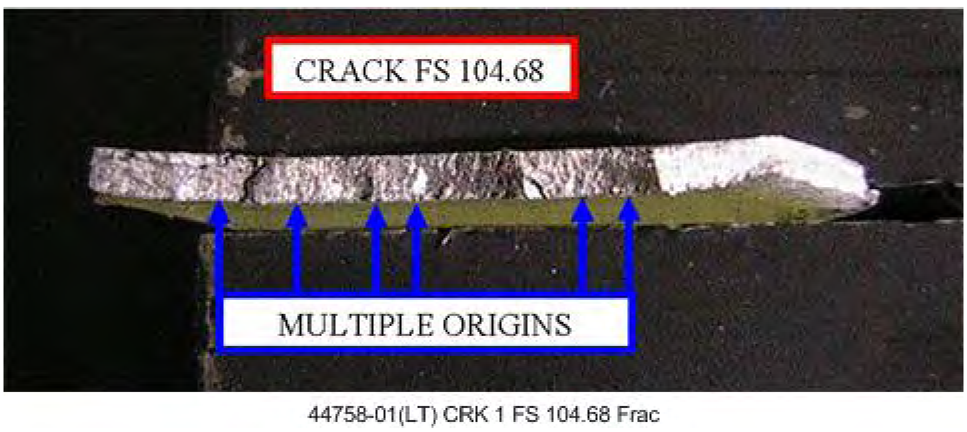


Figure 430. Fracture Face of Crack 1 on the Left Front Spar Attachment FS 104.68

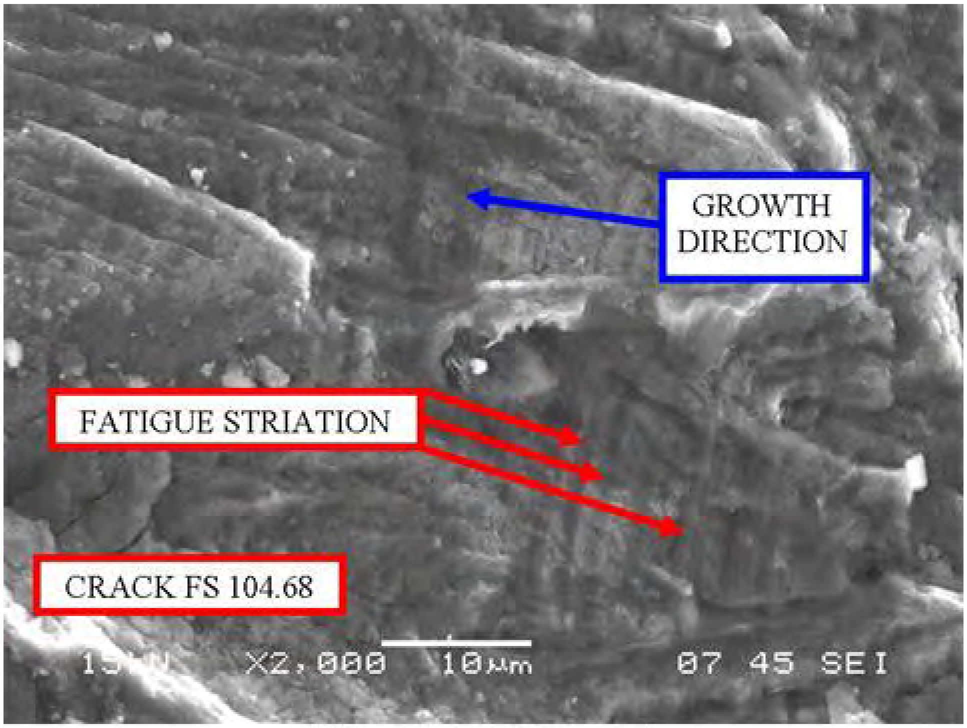


Figure 431. Fractograph of Crack 1 on the Left Front Spar Attachment FS 104.68

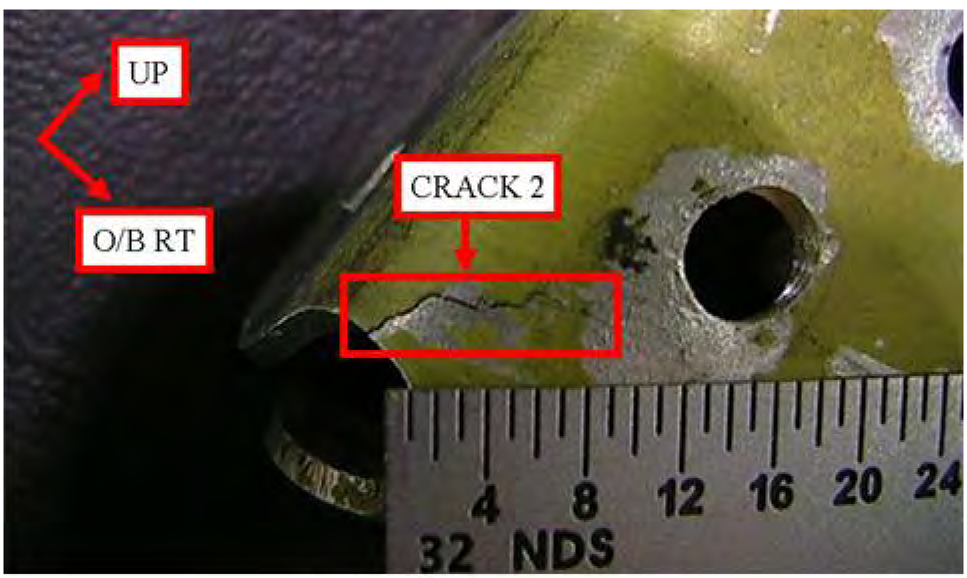


Figure 432. Macroscopic View of Crack 2 on the Left Front Spar Attachment FS 104.68

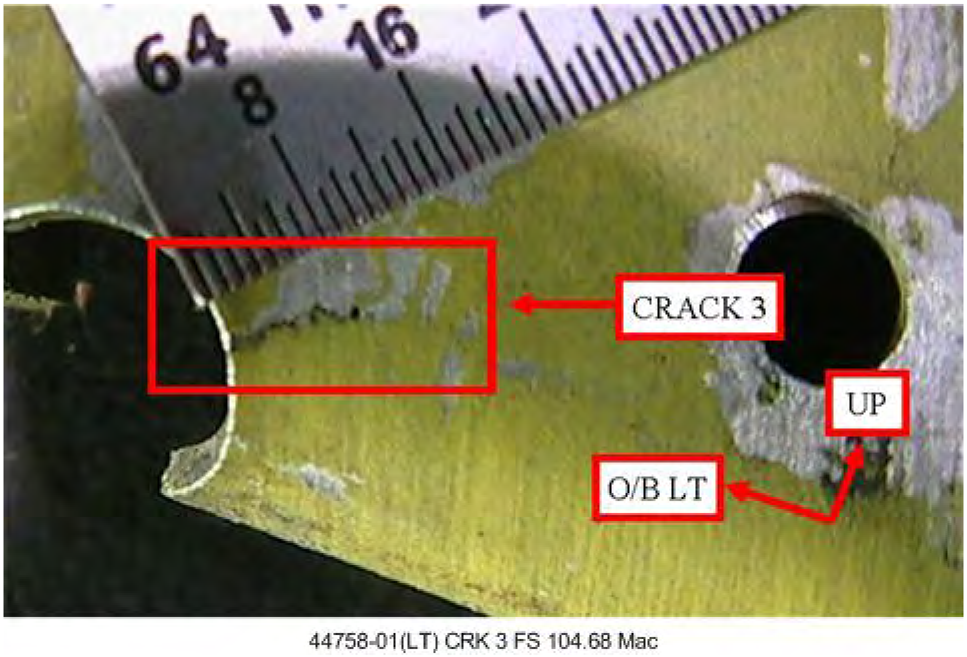


Figure 433. Macroscopic View of Crack 3 on the Left Front Spar Attachment FS 104.68

Figure 434 shows the location of a 0.223-inch crack and a 126-square-inch area of wear, which resulted in a localized maximum thickness loss of 100% on the left lower forward longitudinal frame beam assembly, part number 44877-2. The wear, which stretched from FS 106 to FS 120, and the crack, which was located at FS 112, are shown macroscopically in figure 435, and microscopically in figure 436.

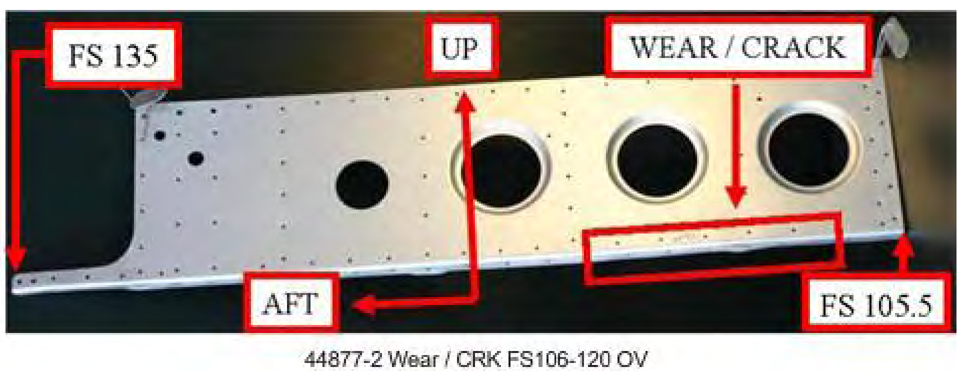


Figure 434. Location of Crack and Wear on the Left Lower Forward Longitudinal Frame Beam Assembly FS 105.5 Through FS 135

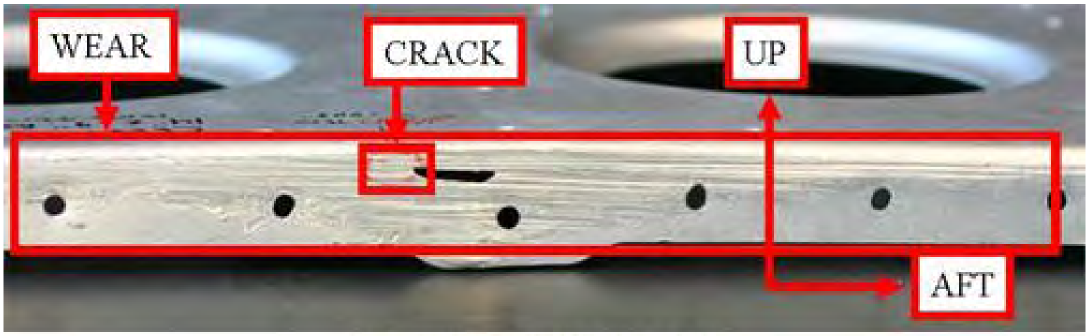


Figure 435. Macroscopic View of Crack and Wear on the Left Lower Forward Longitudinal Frame Beam Assembly FS 106 Through FS 120

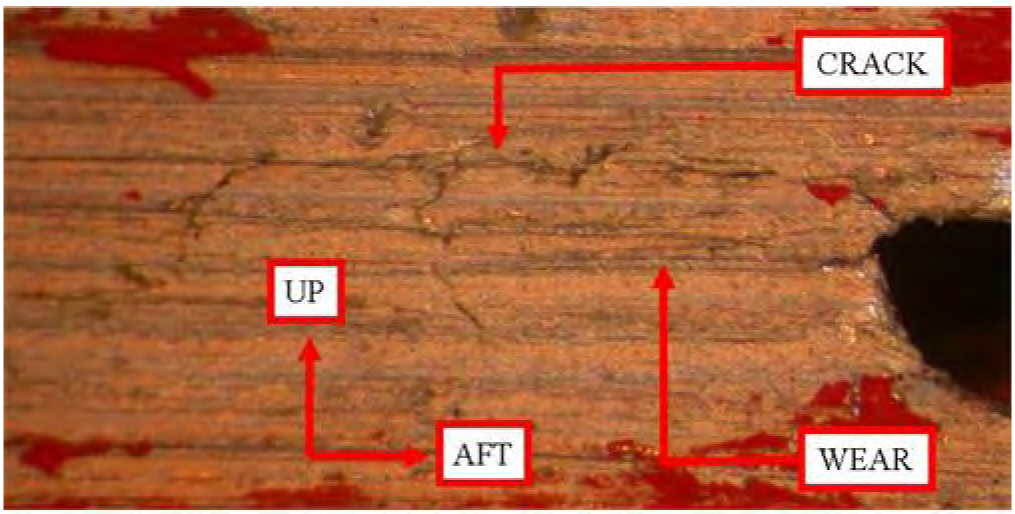
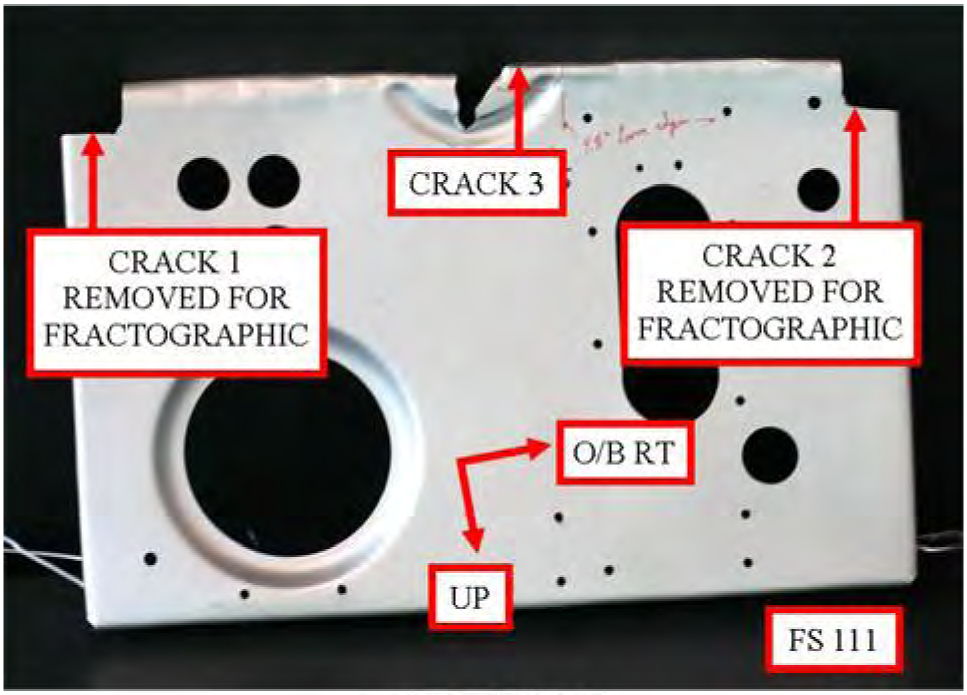


Figure 436. Microscopic View of Crack and Wear on the Left Lower Forward Longitudinal Frame Beam Assembly FS 112

The location of three cracks on the bulkhead assembly lower forward left frame, part number 52190-05 is shown in figure 437. A macroscopic view of crack 1, which was located at FS 111 and measured 0.289 inches in length, is shown in figure 438. A macroscopic view of crack 2, which measured 0.375 inch, is shown in figure 439. Crack 3, which measured 0.336 inches and was located at FS 111, is shown macroscopically in figure 440 and microscopically in figure 441. Cracks 1 and 2 were opened for fractographic analysis but, due to extensive crack face smearing, a failure mode could not be obtained.

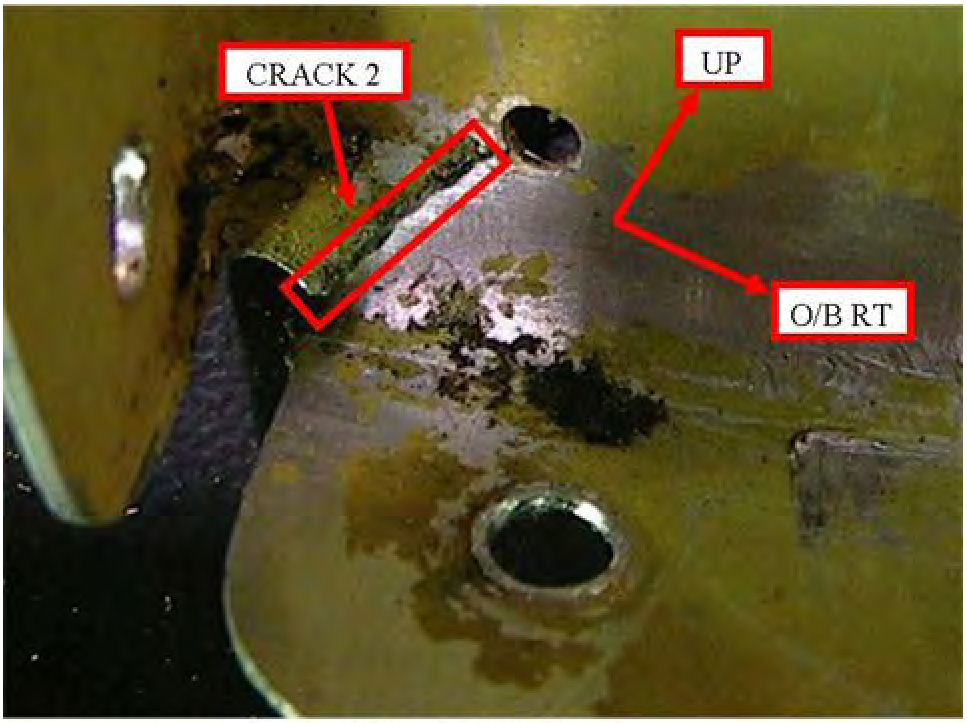


52190-05 CRK FS 111 OV

Figure 437. Location of Three Cracks in the Bulkhead Assembly Lower Forward Left Frame FS 111

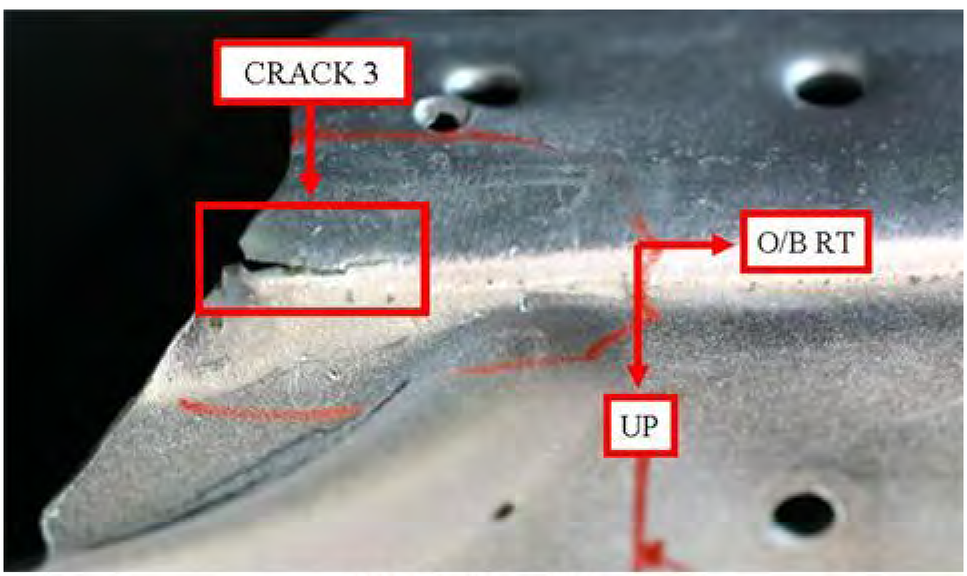
52190-05 CRK 1 FS 111 Mac

Figure 438. Macroscopic View of Crack 1 on the Bulkhead Assembly Lower Forward Left Frame FS 111



52190-05 CRK 2 FS 111 Mac

Figure 439. Macroscopic View of Crack 2 on the Bulkhead Assembly Lower Forward Left Frame FS 111



52190-05 CRK 3 FS 111 Mac

Figure 440. Macroscopic View of Crack 3 on the Bulkhead Assembly Lower Forward Left Frame FS 111

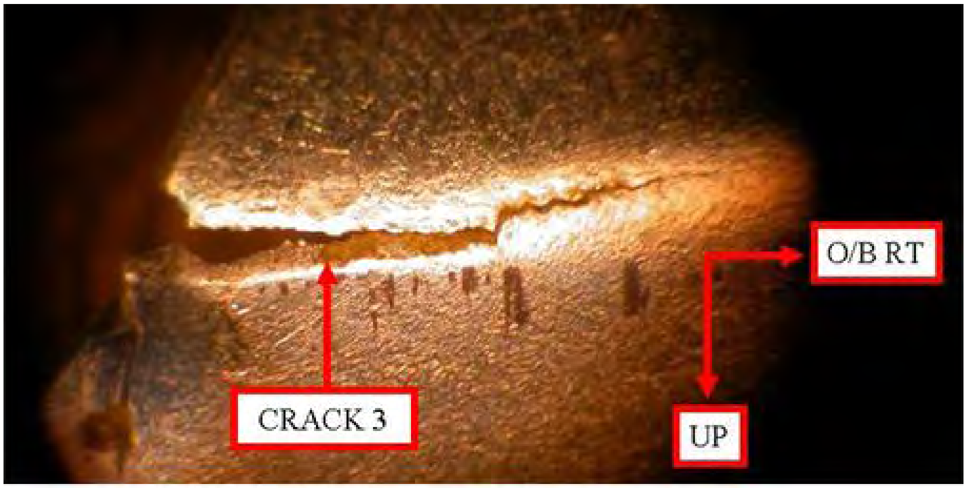


Figure 441. Microscopic View of Crack 3 on the Bulkhead Assembly Lower Forward Left Frame FS 111

The location of areas of moderate, moderate-severe, and severe corrosion on the left inboard floorboard support, part number 44885-00, is shown in figure 442. A macroscopic view of a 0.17-square-inch area of moderate-severe corrosion, located at FS 111, is shown in figure 443. This area of corrosion resulted in a maximum reduction in thickness of 9.3%. Figure 444 shows a macroscopic view of corrosion on the floorboard support at FS 118. This 0.32-square-inch area of corrosion caused a maximum localized thickness loss of 6.2%, which is classified as moderate corrosion. Figure 445 shows a macroscopic view of 1.87-square-inch area of severe corrosion, which caused a maximum localized thickness loss of 25%. This area of corrosion was located at FS 123.

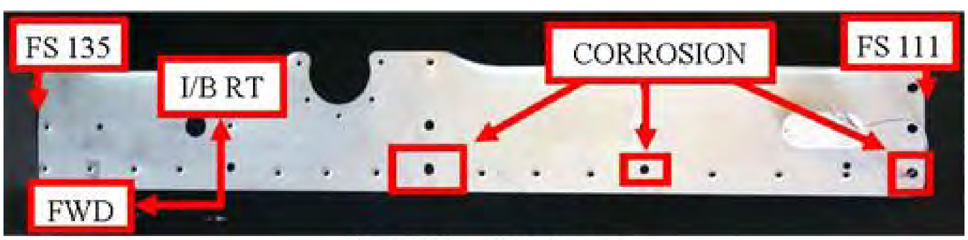
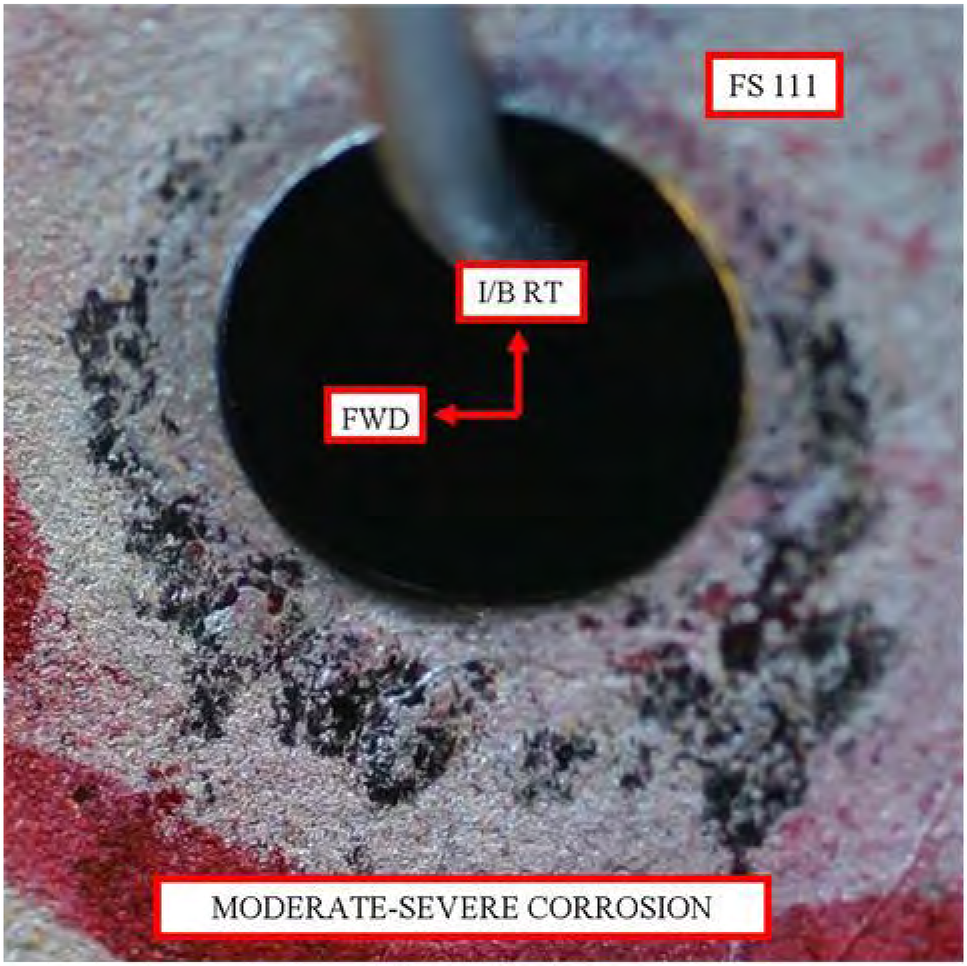


Figure 442. Location of Moderate, Moderate-Severe, and Severe Corrosion on the Left Inboard Floorboard Support FS 111 Through FS 135



44885-00 Corr FS111 Mac

Figure 443. Macroscopic View of Moderate-Severe Corrosion on the Left Inboard Floorboard Support FS 111

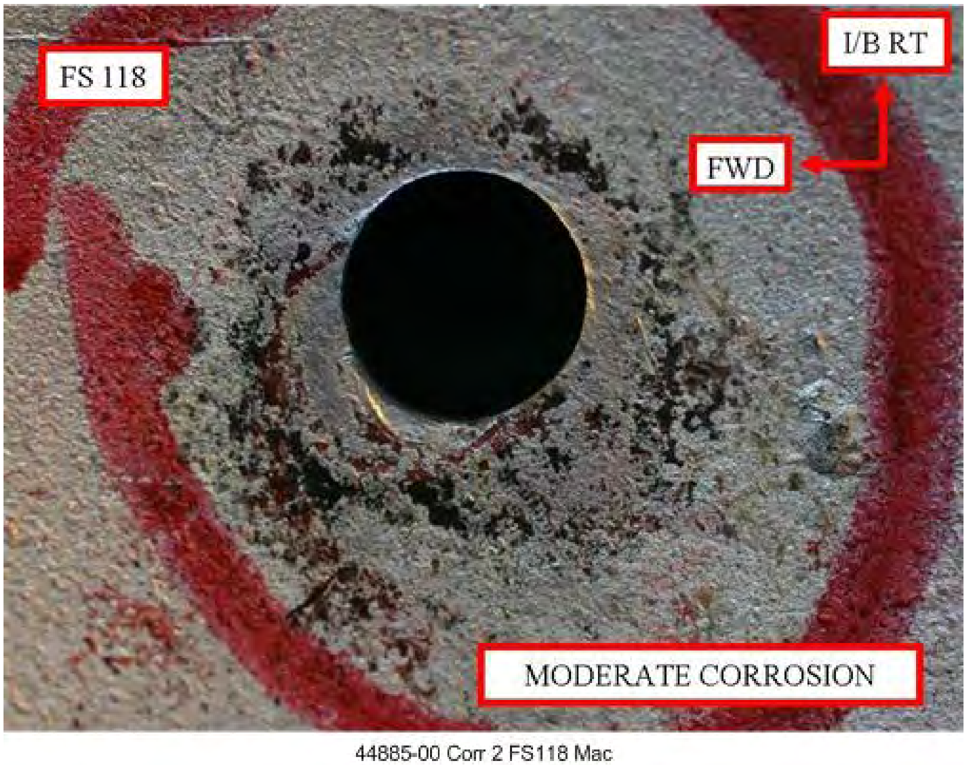


Figure 444. Macroscopic View of Moderate Corrosion on the Left Inboard Floorboard Support FS 118

Figure 445. Macroscopic View of Severe Corrosion on the Left Inboard Floorboard Support FS 123

Figure 446 shows the location of light and light-moderate corrosion on the fuselage upper left angle 4 longitudinal beam assembly, part number 44877-00, at FS 115.5 and FS 129.5. The light corrosion covered a surface area of 0.56 square inch and resulted in a localized thickness loss of 1%, and the light-moderate corrosion covered a surface area of 1.125 square inches and resulted in a reduction in localized thickness of 4%. The location of a 0.274-inch crack on the fuselage lower forward stringer, part number 52080-09, is shown in figure 447, and a macroscopic view of this crack is shown in figure 448. The fracture face of this crack, located at FS 122, is shown in figure 449. The failure mode for this crack was determined to be overload from analysis of this fracture face.

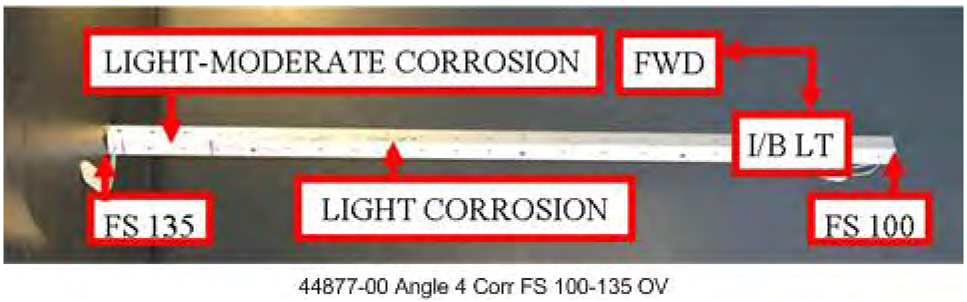


Figure 446. Location of Light and Light-Moderate Corrosion on the Fuselage Upper Left Angle 4 Longitudinal Beam Assembly FS 115.5 and FS 129.5

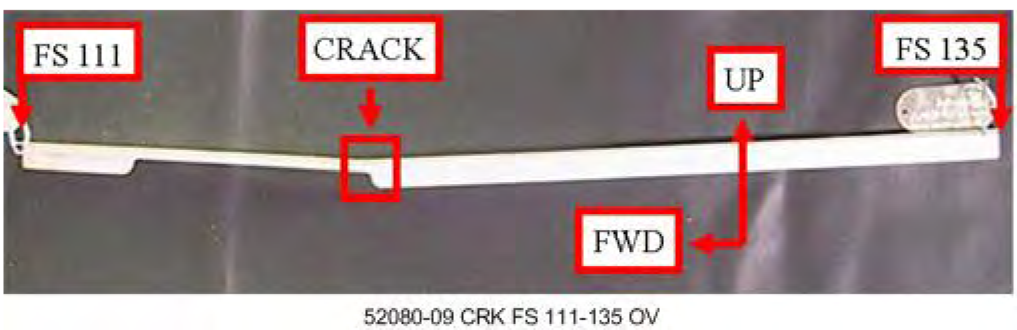


Figure 447. Location of Crack on the Fuselage Lower Forward Stringer FS 122

Figure 448. Macroscopic View of Crack on the Fuselage Lower Forward Stringer FS 122

Figure 449. Fracture Face of Crack on the Fuselage Lower Forward Stringer FS 122

Figure 450 shows the location of two cracks on the fuselage right floorboard channel support, part number 44801-01. A macroscopic view of crack 1, which is located at FS 123 and measured 0.738 inch, is shown in figure 451. Figure 452 shows a macroscopic view of a 1.297-inch crack, crack 2, at FS 135. Both crack faces were analyzed; however, due to extensive crack face smearing, the failure mode of both cracks could not be determined. Figure 453 shows the location of a 2.14-square-inch area of light-moderate corrosion on the fuselage longitudinal beam lower right frame, part number 44877-03. This corrosion caused a maximum reduction in thickness of 2.5%.

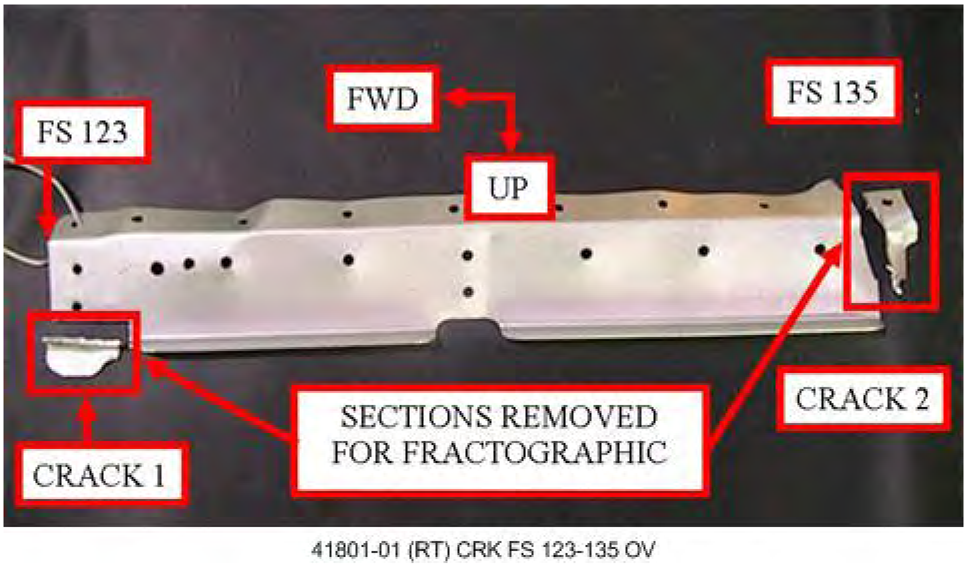


Figure 450. Location of Cracks 1 and 2 on the Fuselage Right Floorboard Channel Support FS 123 Through FS 135

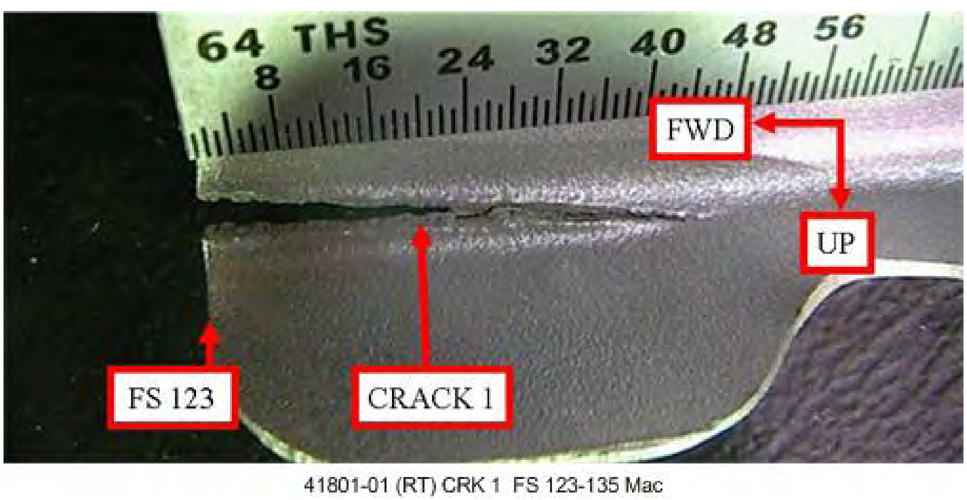


Figure 451. Macroscopic View of Crack 1 on the Fuselage Right Floorboard Channel Support FS 123

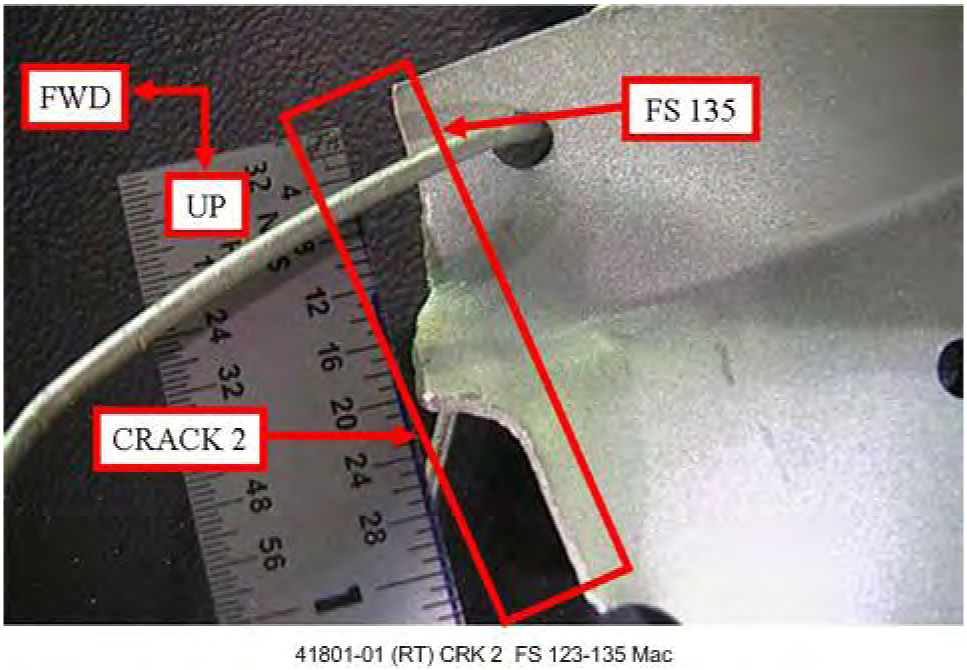


Figure 452. Macroscopic View of Crack 2 on the Fuselage Right Floorboard Channel Support FS 135

Figure 453. Location of Light-Moderate Corrosion on the Fuselage Longitudinal Beam Lower Right Frame FS 130.75 Through FS 133

3.4.2.3.3 Fuselage FS 140 Through FS 177.

Each defect found during the teardown evaluation from FS 140 to FS 177 has been completely characterized in table 37.

Table 37. Inspection Results From FS 140 Through FS 177

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left center body bulkhead, figure 455	Crack	FS 140	0.072 inch	Crack indication	Hole crack	456
Right center body bulkhead, figure 457	Corrosion	FS 140	0.43 inch by 0.26 inch	No indication	Severe corrosion 12.5% thickness loss	458
	Wear	FS 140	0.495 inch by 0.275 inch	No indication	Wear 7.5% thickness loss	459
Left floorboard support, figure 460	Corrosion	FS 140	0.415 inch by 0.55 inch	Corrosion indication	Severe corrosion 18% thickness loss	461
	Corrosion	FS 146	0.35 inch by 0.1 inch	Corrosion indication	Moderate-severe corrosion 9.4% thickness loss	462
	Corrosion	FS 151	0.275 inch by 0.26 inch	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
Fuselage bottom bulkhead assembly, figure 463	Crack	FS 162.6	1.514 inches	Not inspected[1]	Fatigue	464 465
	Crack	FS 162.6	0.642 inch	Not inspected[1]	Fatigue	466 467
	Crack	FS 162.6	1.804 inches	Not inspected[1]	Fatigue	468 469
Center body right, figure 470	Corrosion	FS 174	1.27 inches by 0.4875 inch	Corrosion indication	Light-moderate corrosion 3% thickness loss	105
Center body left, figure 471	Corrosion	FS 174	0.773 inch by 1 inch	Corrosion indication	Light corrosion 2% thickness loss	103
Fuselage right longitudinal beam assembly, figure 472	Corrosion	FS 174 through FS 203	Scattered over entire part	Corrosion indication	Light corrosion 1% thickness loss	103

[1]Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

The location of a crack on the left center body bulkhead at FS 140, part number 40995-18, is shown in figure 454. A macroscopic view of this 0.072-inch crack is shown in figure 455. Figure 456 shows the location of an area of severe corrosion and an area of wear on the right center body bulkhead FS 140, part number 40995-17. A macroscopic view of the area of severe corrosion, located at FS 140, is shown in figure 457. This corrosion covered a surface area of 0.11 square inch and caused a maximum reduction in thickness of 12.5%. A macroscopic view of a 0.14-square-inch area of wear is shown in figure 458. This wear occurred at FS 140 and caused a localized reduction in thickness of 7.5%.

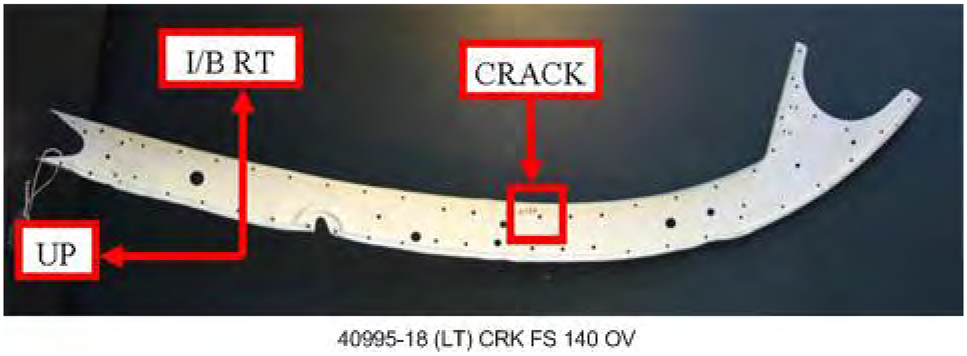


Figure 454. Location of Crack on the Left Center Body Bulkhead FS 140

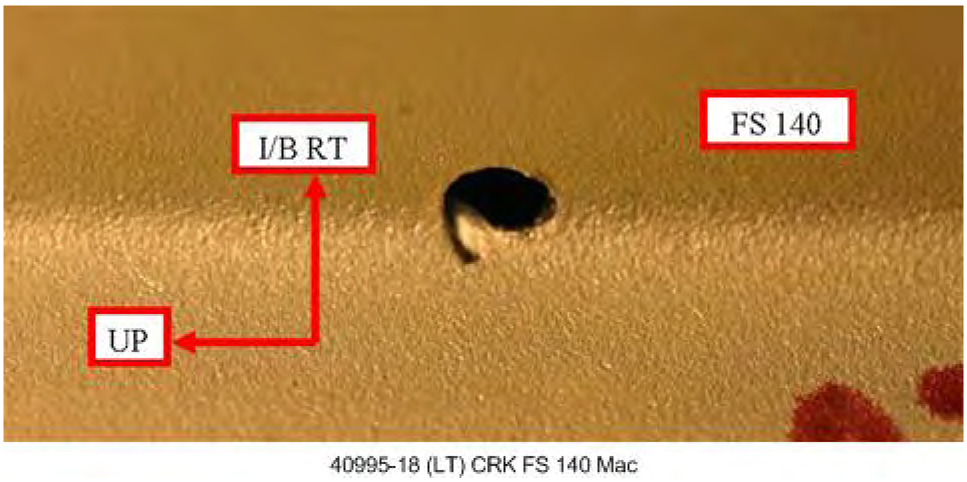
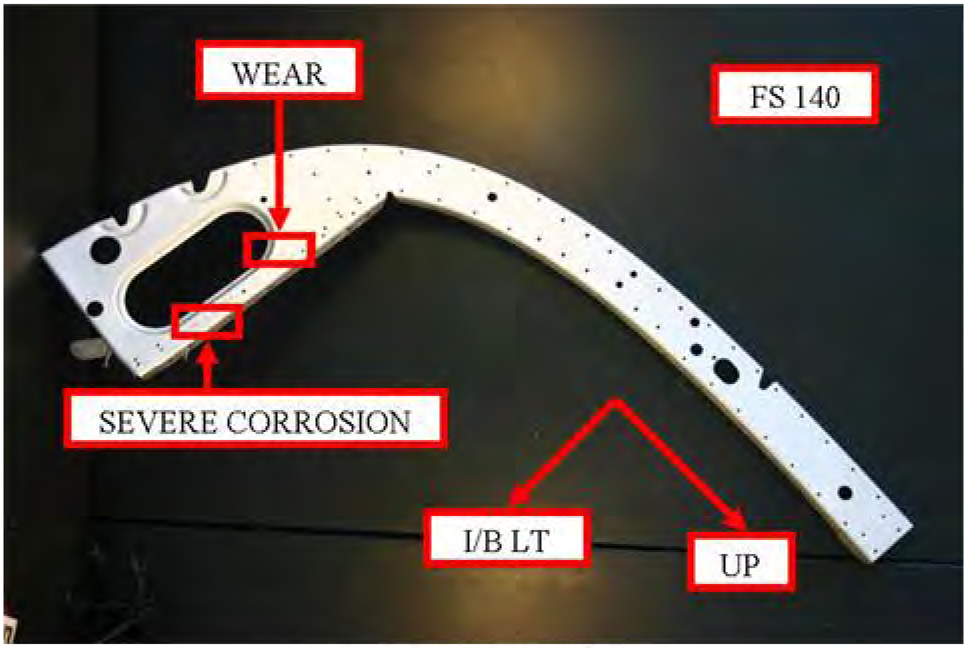
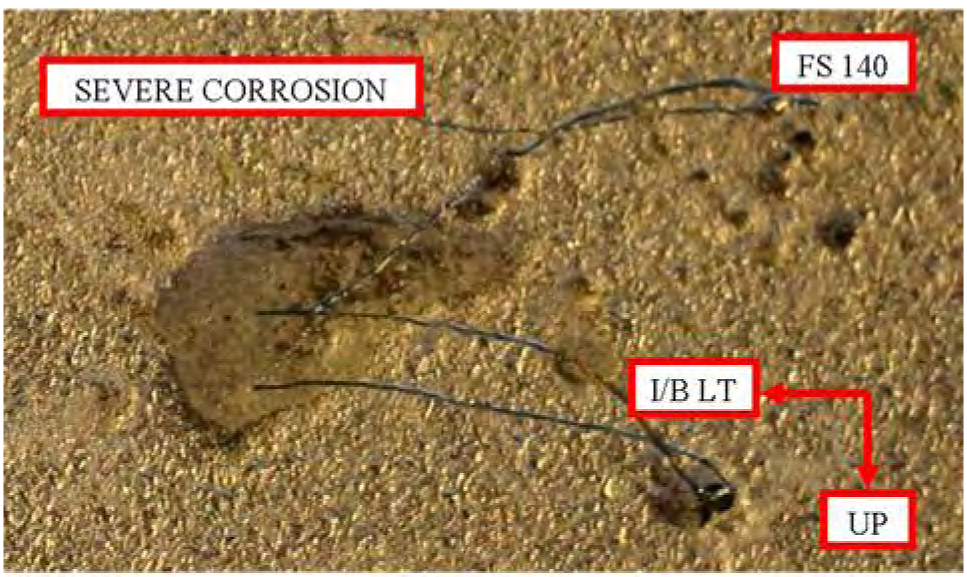


Figure 455. Macroscopic View of Crack on the Left Center Body Bulkhead FS 140



40995-17 (RT) Corr FS140 OV

Figure 456. Location of Severe Corrosion and Wear on the Right Center Body Bulkhead FS 140



40995-17 (RT) Corr FS140 Mac

Figure 457. Macroscopic View of Severe Corrosion on the Right Center Body Bulkhead FS 140

Figure 458. Macroscopic View of Wear on the Right Center Body Bulkhead FS 140

The location of light-moderate, moderate-severe, and severe corrosion identified on the left floorboard support, part number 41614-00, is shown in figure 459. Figure 460 shows a macroscopic view of a 0.23-square-inch area of corrosion located at FS 140. The maximum localized reduction in thickness caused by this area of corrosion was 18%. A macroscopic view of a 0.04-square-inch area of moderate-severe corrosion, located at FS 146, is shown in figure 461. This area of corrosion caused a localized thickness loss of 9.4%. A 0.07-square-inch area of light-moderate corrosion that caused a localized thickness loss of 3% was noted on the floorboard support at FS 151.

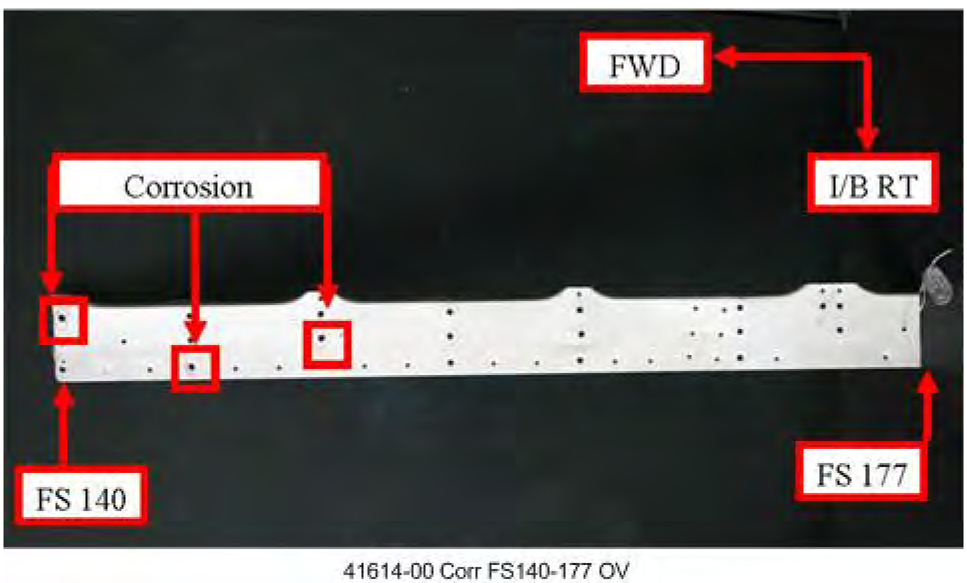


Figure 459. Location of Light-Moderate, Moderate-Severe, and Severe Corrosion on the Left Floorboard Support FS 140 Through FS 177

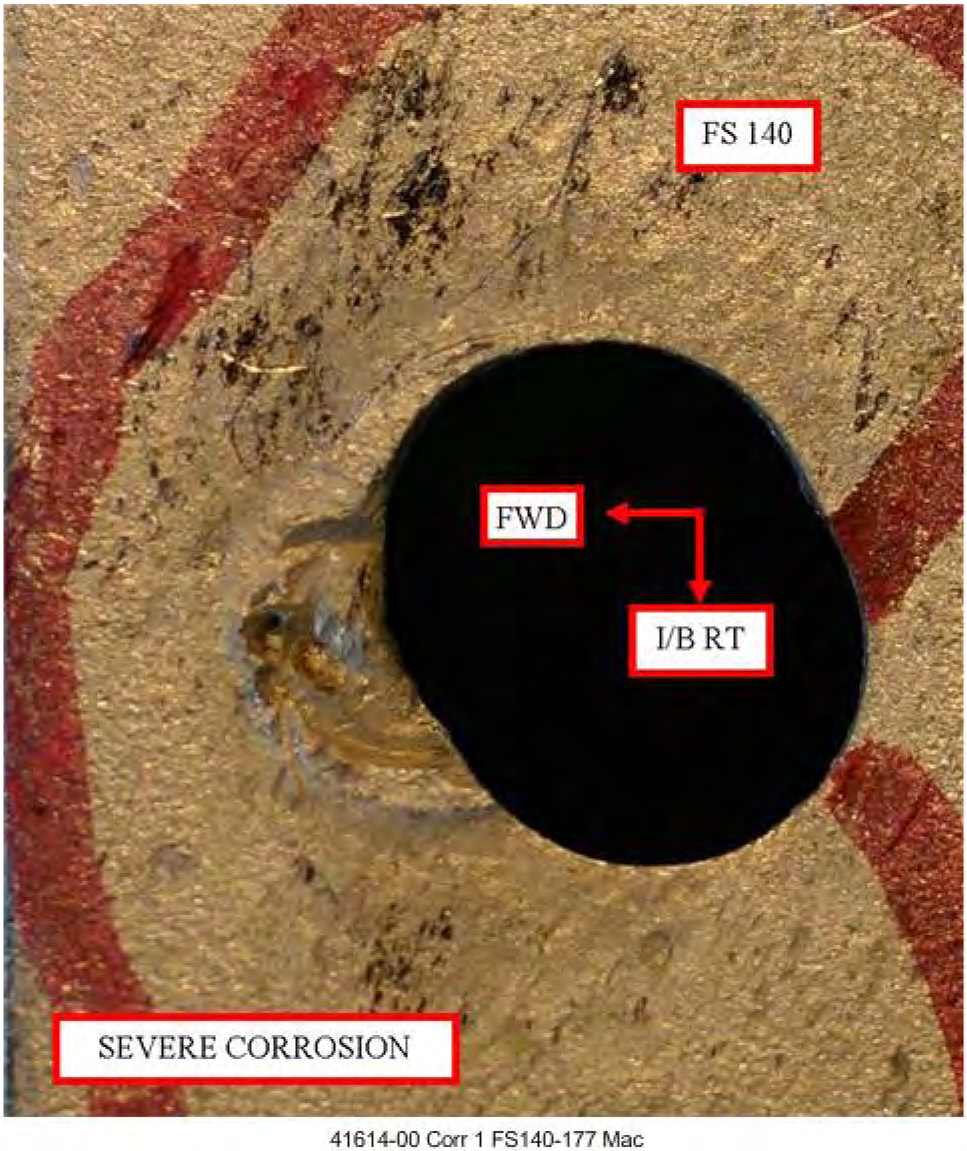


Figure 460. Macroscopic View of Severe Corrosion on the Left Floorboard Support FS 140

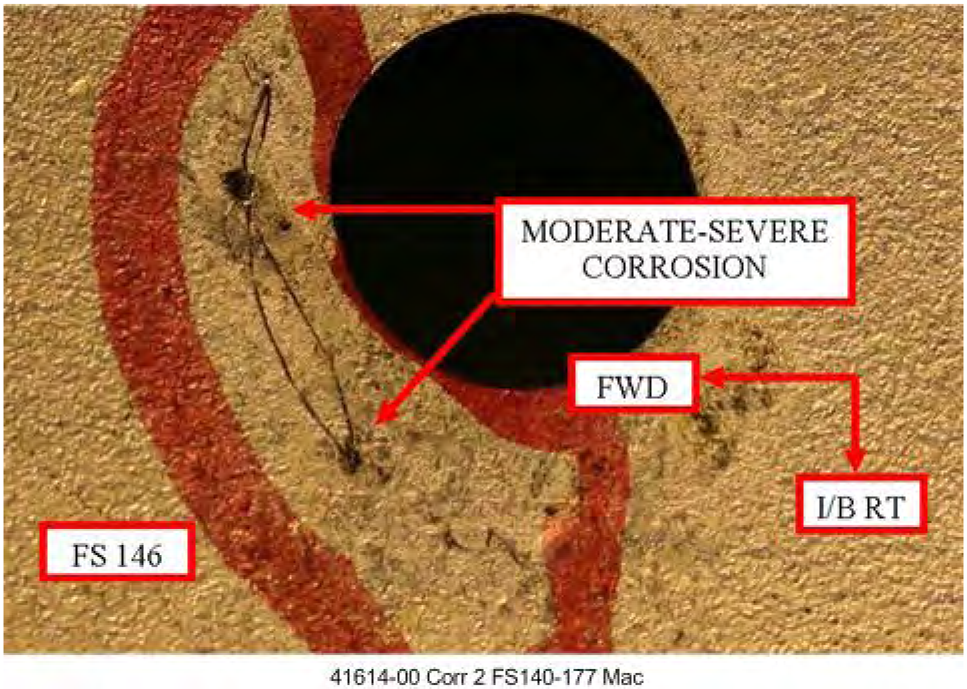


Figure 461. Macroscopic View of Moderate-Severe Corrosion on the Left Floorboard Support FS 146

The location of three cracks on the fuselage bottom bulkhead assembly, part number 41233-05, is shown in figure 462. A macroscopic view of crack 1, located at FS 162.6, is shown in figure 463. The fracture face of this 1.514-inch crack is shown in figure 464. A macroscopic view of crack 2, which measured 0.642 inch in length, is shown in figure 465. A fracture face of this crack is shown in figure 466. A macroscopic view of crack 3, which measured 1.804 inches in length and occurred at FS 162.6, is shown in figure 467, and the fracture face is shown in figure 468. From the fracture face characteristics, it was determined that all three cracks were caused by fatigue; however, significant crack face smearing was also observed.

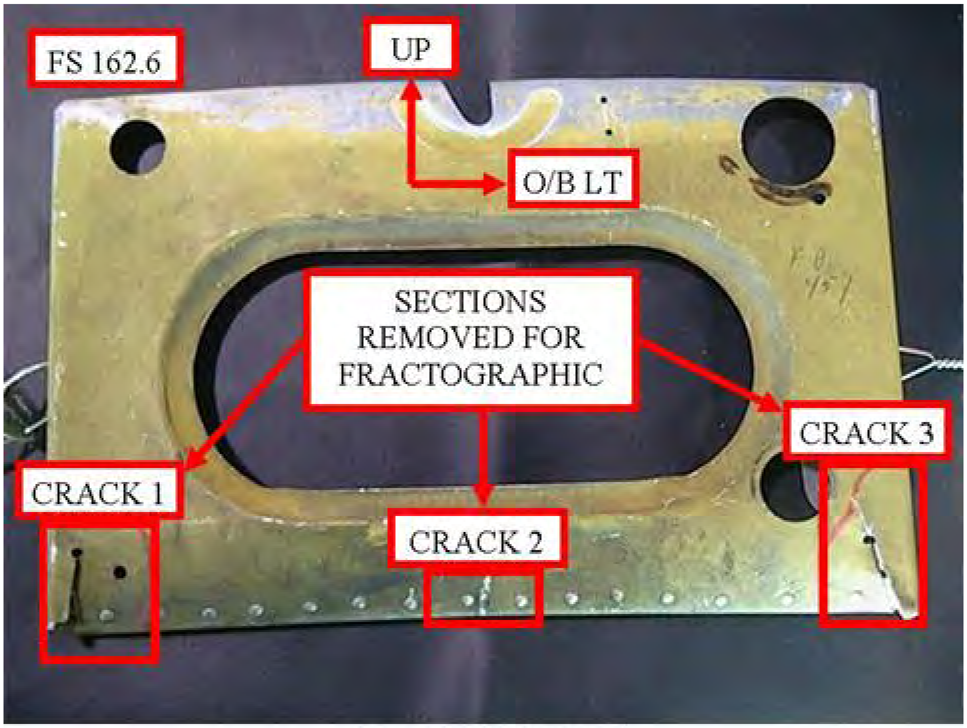


Figure 462. Location of Three Cracks on the Fuselage Bottom Bulkhead Assembly FS 162.6

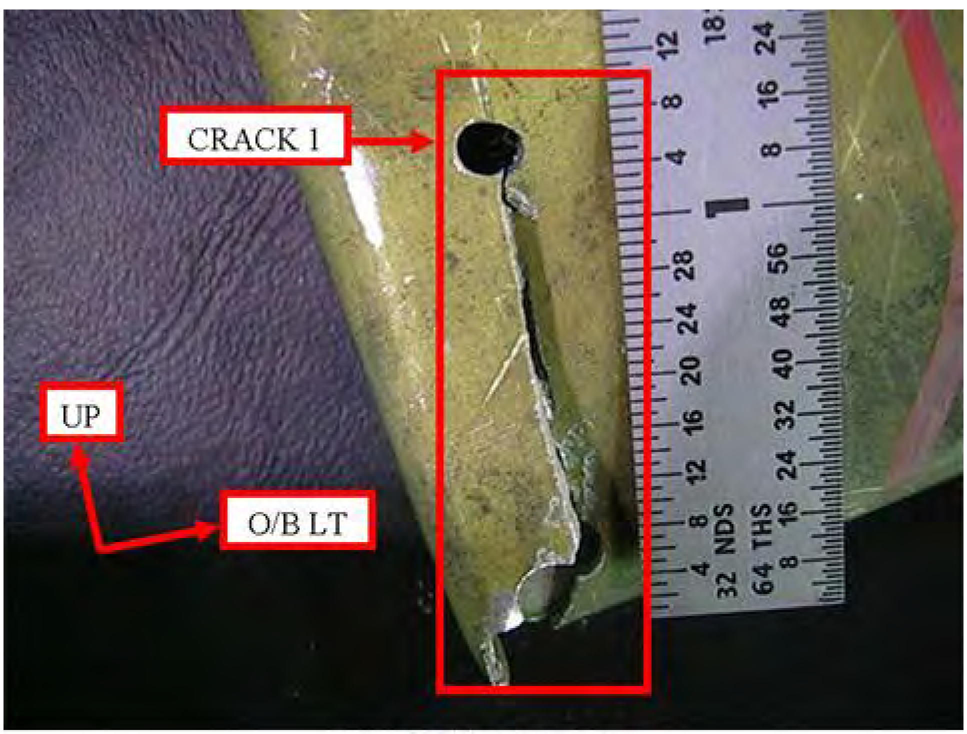
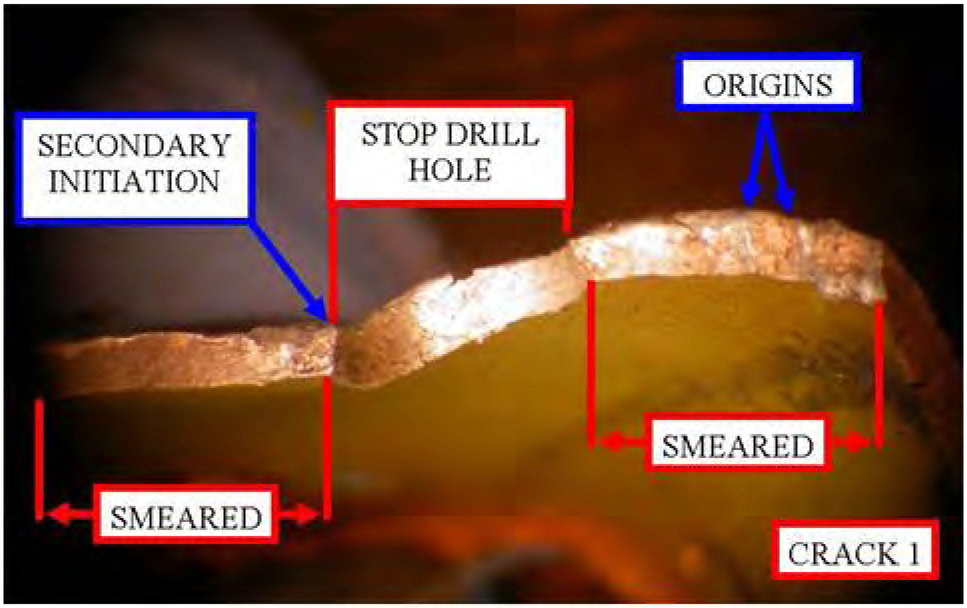
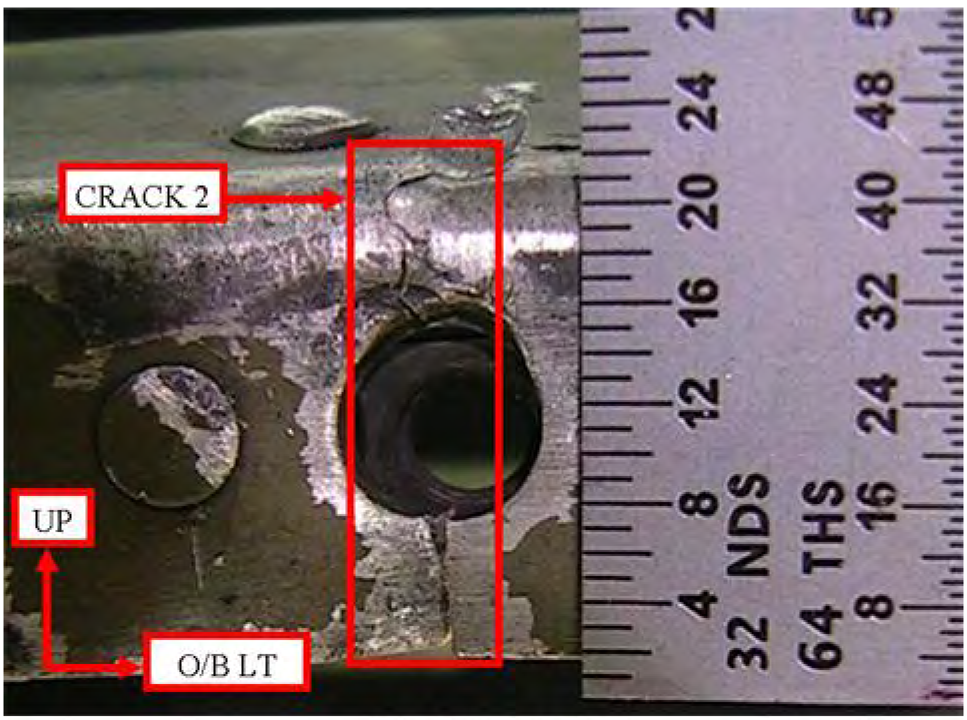


Figure 463. Macroscopic View of Crack 1 on the Fuselage Bottom Bulkhead Assembly FS 162.6



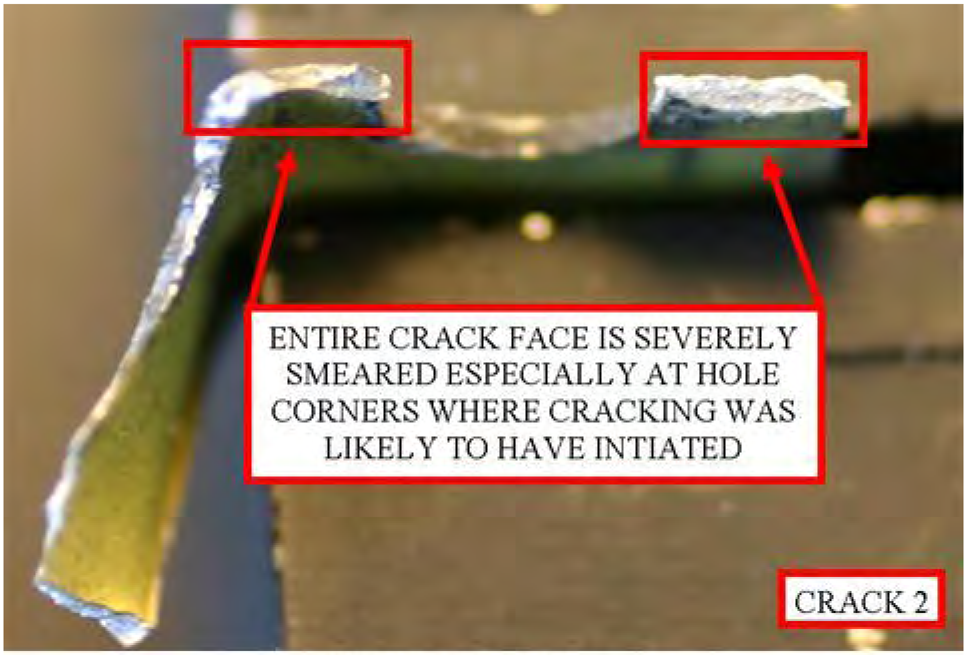
41233-05 CRK 1 FS 162.6 Frac

Figure 464. Fracture Face of Crack 1 on the Fuselage Bottom Bulkhead Assembly FS 162.6



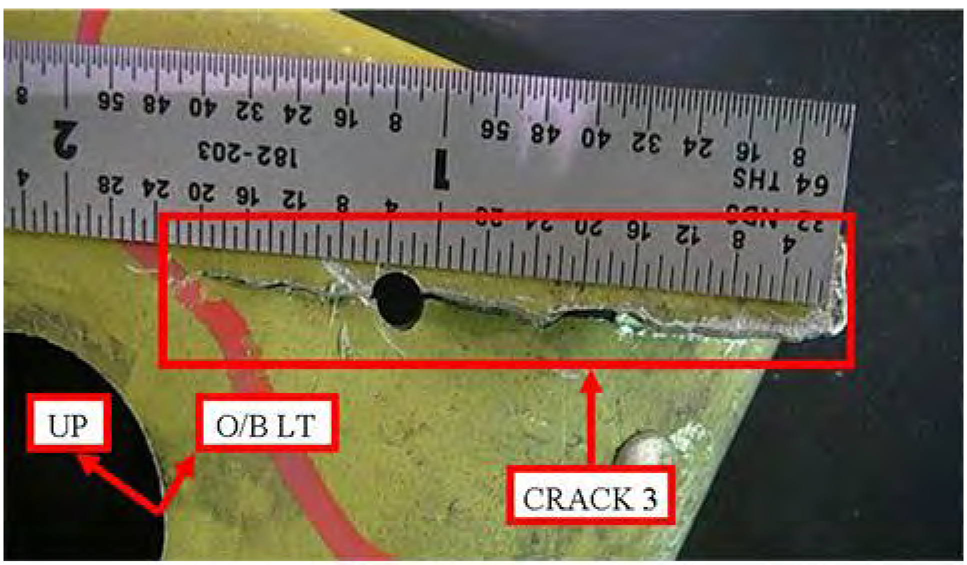
41233-05 CRK 2 FS 162.6 Mac

Figure 465. Macroscopic View of Crack 2 on the Fuselage Bottom Bulkhead Assembly FS 162.6



41233-05 CRK 2 FS 162.6 Frac

Figure 466. Fracture Face of Crack 2 on the Fuselage Bottom Bulkhead Assembly FS 162.6



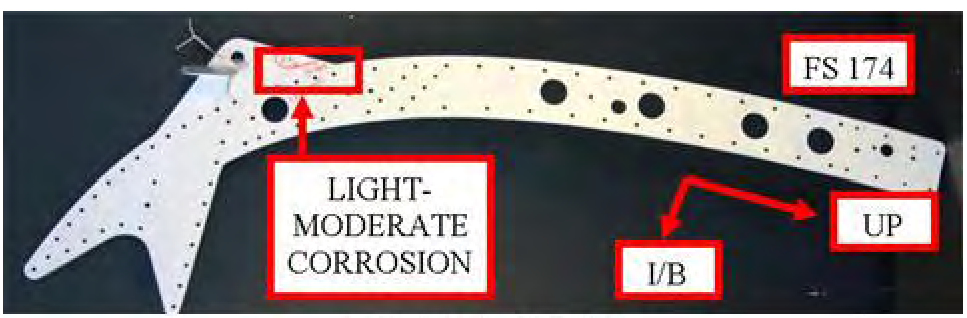
41233-05 CRK 3 FS 162.6 Mac

Figure 467. Macroscopic View of Crack 3 on the Fuselage Bottom Bulkhead Assembly FS 162.6

41233-05 CRK 3 FS 162.6 Frac

Figure 468. Fracture Face of Crack 3 on the Fuselage Bottom Bulkhead Assembly FS 162.6

Figure 469 shows the location of a 0.62-square-inch area of light-moderate corrosion on the center body right, part number 41178-6. This area of corrosion caused a localized thickness loss of 3% and occurred at FS 174. The location of a 0.77-square-inch area of light corrosion, resulting in a maximum reduction in thickness of 2%, is shown in figure 470 on the center body left, part number 41178-6. Light scattered corrosion, shown in figure 471, was found scattered across the fuselage right longitudinal beam assembly, part number 41119, from FS 174 to FS 203. This corrosion caused a maximum reduction in thickness of 1%.



41178-6(RT) Corr FS174 OV

Figure 469. Location of Light-Moderate Corrosion on the Center Body Right FS 174

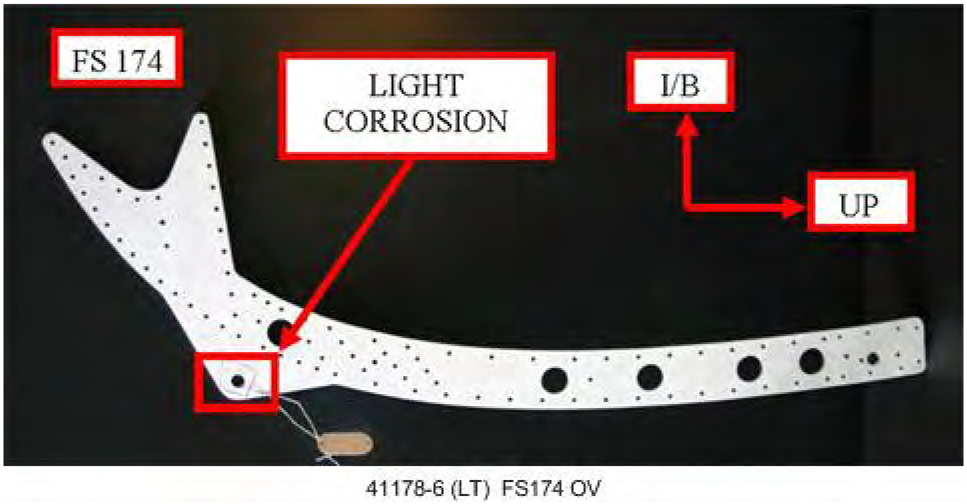


Figure 470. Location of Light Corrosion on the Center Body Left FS 174

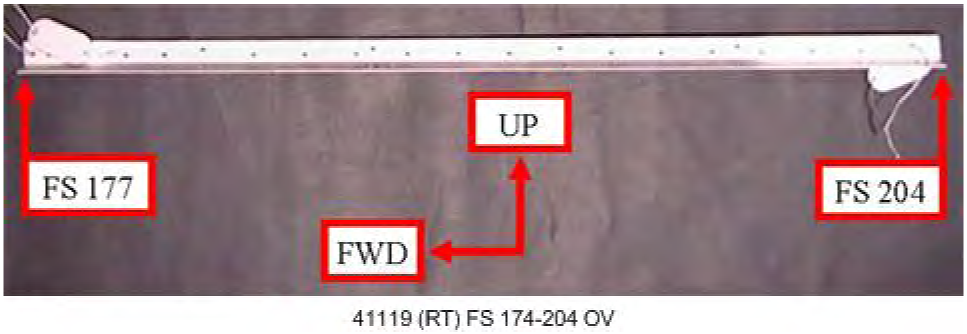


Figure 471. Location of Light Scattered Corrosion on the Fuselage Right Longitudinal Beam Assembly FS 174 Through FS 204

3.4.2.3.4 Fuselage FS 178 Through FS 244.

Numerous areas of corrosion and cracks were observed on the fuselage from FS 178 to FS 244. Detailed characterizations of each of these defects are shown in table 38.

Table 38. Inspection Results From FS 178 Through FS 244

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Right floorboard support, figure 473	Corrosion	FS 179 through FS 215	37 inches by 3.5 inches	Corrosion indication	Moderate corrosion 6.3% thickness loss	474
	Crack	FS 184.5	0.054 inch	Crack indication	Hole crack	475 476

Table 38. Inspection Results From FS 178 Through FS 244 (Continued)

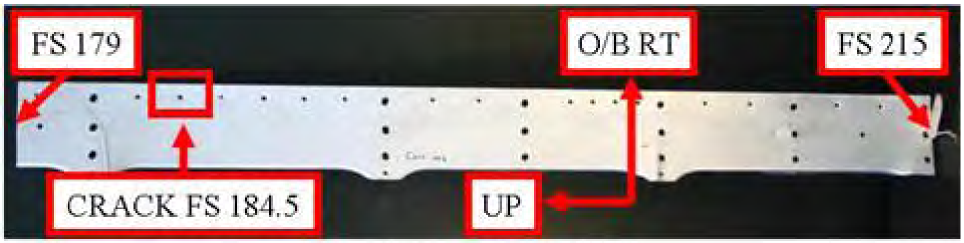
Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Fuselage right longitudinal bulkhead outboard angle, figure 477	Corrosion	FS 182.25 through FS 210.75	Scattered over entire part	No indication	Light-moderate corrosion 4% thickness loss	105
Left floorboard support, figure 478	Corrosion	FS 204.5	0.54 inch by 0.5 inch	Corrosion indication	Severe corrosion 22% thickness loss	479
	Corrosion	FS 210	0.36 inch by 0.15 inch	Corrosion indication	Moderate corrosion 6.3% thickness loss	480
	Corrosion	FS 215	0.15 inch by 0.07 inch	Corrosion indication	Moderate-severe corrosion 9.4% thickness loss	481
Left longitudinal beam assembly, figure 482	Wear	FS 208	1.046 inches by 0.66 inch	No indication	Wear 12.5% thickness loss	483
Fuselage right beam assembly longitudinal, figure 484	Corrosion	FS 204 through FS 244	Scattered over entire part	Corrosion indication	Light corrosion 1% thickness loss	103
Fuselage left beam assembly lower rear frame, figure 485	Corrosion	FS 213.5 through FS 228.5	15 inches by 7 inches	No indication	Light corrosion 2% thickness loss	103
	Crack	FS 220.5	1.59 inches	Crack indication	Bend radii Surface crack	486
Left inboard floorboard support assembly, figure 487	Crack	FS 216.5	0.014 inch	No indication	Hole crack	488 489
	Crack	FS 220.25	0.215 inch	Crack indication	Hole crack	490 491
	Crack	FS 222	0.508 inch	Crack indication	Surface crack	492 493
	Crack	FS 222	0.157 inch	Crack indication	Hole crack	494 495
	Crack	FS 223	0.03 inch	Crack indication	Hole crack	496 497
	Crack	FS 225	0.036 inch	Crack indication	Hole crack	3498 499
	Crack	FS 227.5	0.166 inch	Crack indication	Surface crack	500 501
	Multiple cracks	FS 229.5	0.26 inch	Crack indication	Hole crack	502 503
	Crack	FS 230	0.249 inch	Crack indication	Surface crack	504 505

Table 38. Inspection Results From FS 178 Through FS 244 (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left inboard floorboard support assembly, figure 487 (Continued)	Crack	FS 236.25	0.142 inch	Crack indication	Surface crack	506 507
	Crack	FS 238	0.31 inch	Crack indication	Hole crack	508 509
	Multiple Cracks	FS 244	0.038 inch	Crack indication	Hole crack	510 511
Door sill support aft body, figure 512	Crack	FS 236	0.409 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	513
	Crack	FS 236	0.437 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	514
	Crack	FS 237.5	0.428 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	515
	Crack	FS 237.5	0.267 inch	Not inspected[1]	Failure mode undetermined due to extensive crack face smearing	516
Fuselage center body angle, figure 517	Corrosion	FS 204 through FS 244	Scattered over entire part	No indication	Light-moderate corrosion 3.2% thickness loss	105

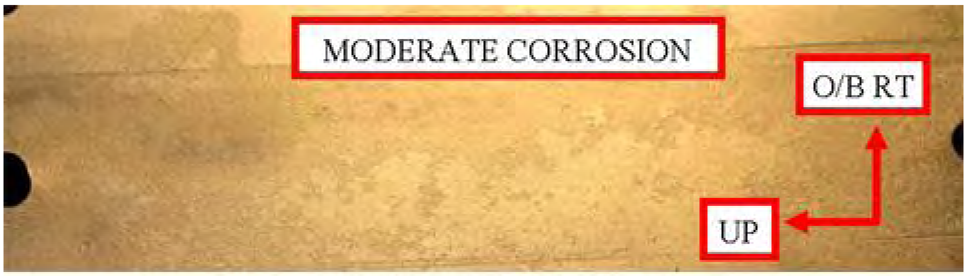
[1]Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 472 shows the location of a crack and moderate corrosion on the right floorboard support, part number 41615-00. A macroscopic view of the moderate corrosion, which covered 129.5 square inches from FS 179 to FS 215 and caused a localized reduction in thickness of 6.3%, is shown in figure 473. A macroscopic view of the 0.054-inch crack, located at FS 184.5, is shown in figure 474, while a microscopic view of this crack is shown in figure 475. The location of scattered light-moderate corrosion on the fuselage right longitudinal bulkhead outboard angle, part number 44437-07, is shown in figure 476. This area of scattered corrosion occurred from FS 182.25 to FS 210.75 and caused a maximum reduction in thickness of 4%.



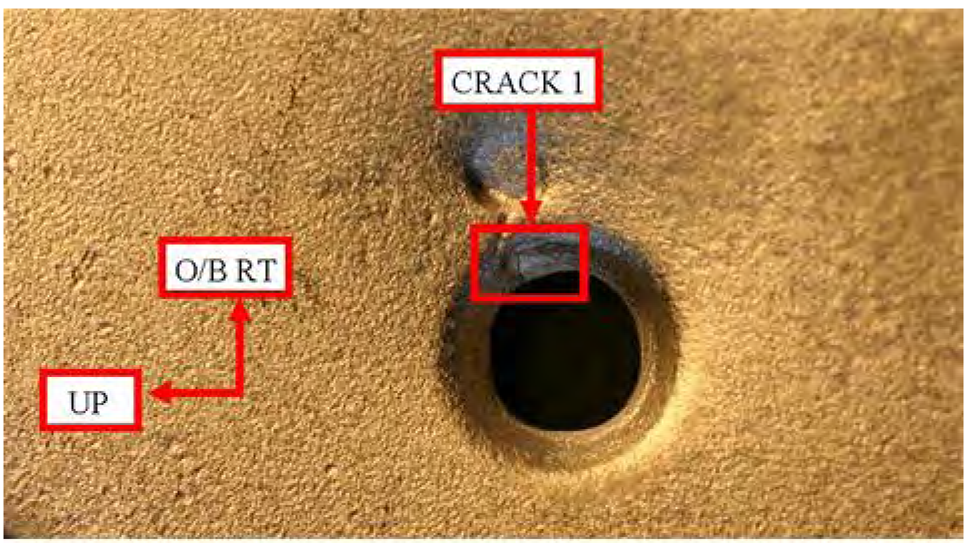
41615-00 (RT) FS179-215 OV

Figure 472. Location of Crack and Moderate Corrosion on the Right Floorboard Support FS 179 Through FS 215



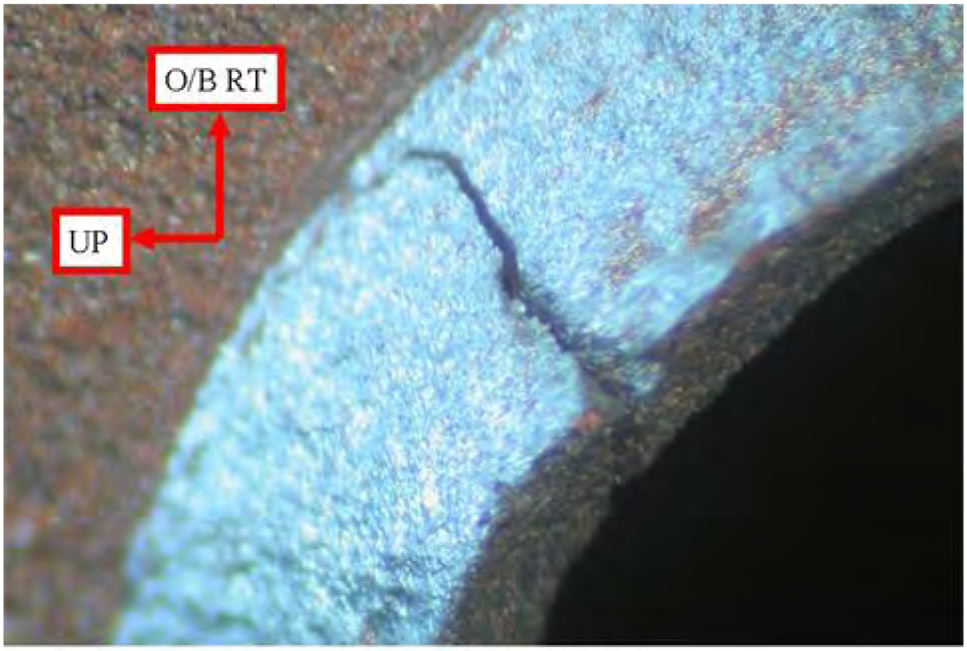
41615-00 (RT)Corr FS179-215 Mac

Figure 473. Macroscopic View of Moderate Corrosion on the Right Floorboard Support FS 179 Through FS 215



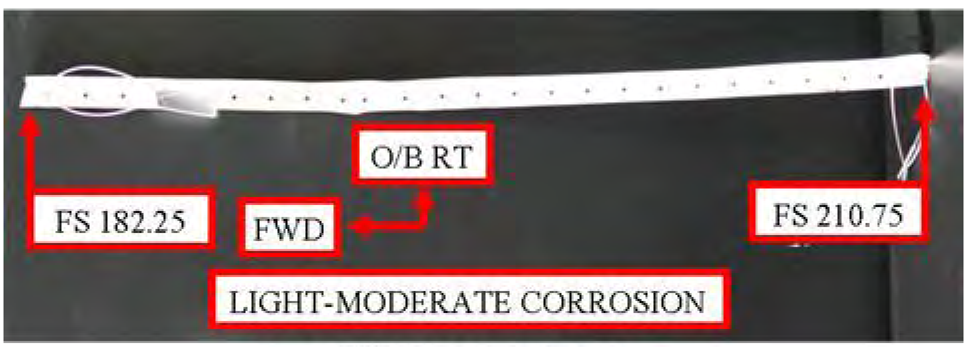
41615-00 (RT) CRK FS 184.5 Mac

Figure 474. Macroscopic View of Crack on the Right Floorboard Support FS 184.5



41615-00 (RT) CRK FS179-215 Mic

Figure 475. Microscopic View of Crack on the Right Floorboard Support FS 184.5



44437-07 FS182.25-210.75 Ov

Figure 476. Location of Light-Moderate Scattered Corrosion on the Fuselage Right Longitudinal Bulkhead Outboard Angle FS 182.25 Through FS 210.75

Figure 477 shows the location of moderate, moderate-severe, and severe corrosion on the left floorboard support, part number 41614-00. A macroscopic view of a 0.27-square-inch area of severe corrosion, located at FS 204.5, is shown in figure 478. This area of corrosion caused a maximum reduction in thickness of 22%. Figure 479 shows a 0.05-square-inch area of moderate corrosion that caused a localized thickness loss of 6.3% and was located at FS 210. A macroscopic view of a 0.01-square-inch area of moderate-severe corrosion is shown in figure 480. This area of corrosion caused a maximum localized reduction in thickness of 9.4%.

Figure 477. Location of Moderate, Moderate-Severe, and Severe Corrosion on the Left Floorboard Support FS 179 Through FS 215

Figure 478. Macroscopic View of Severe Corrosion on the Left Floorboard Support FS 204.5

Figure 479. Macroscopic View of Moderate Corrosion on the Left Floorboard Support FS 210

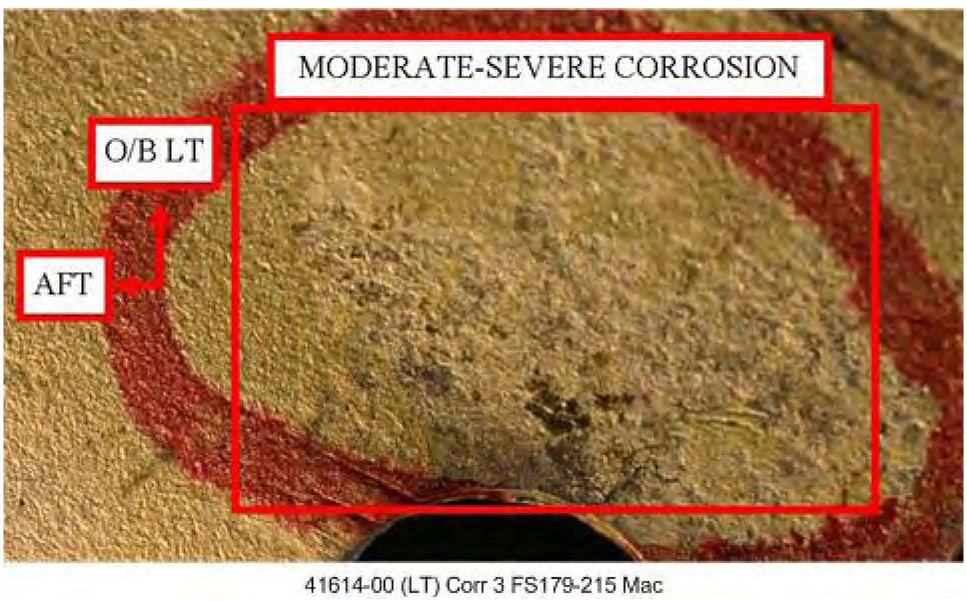
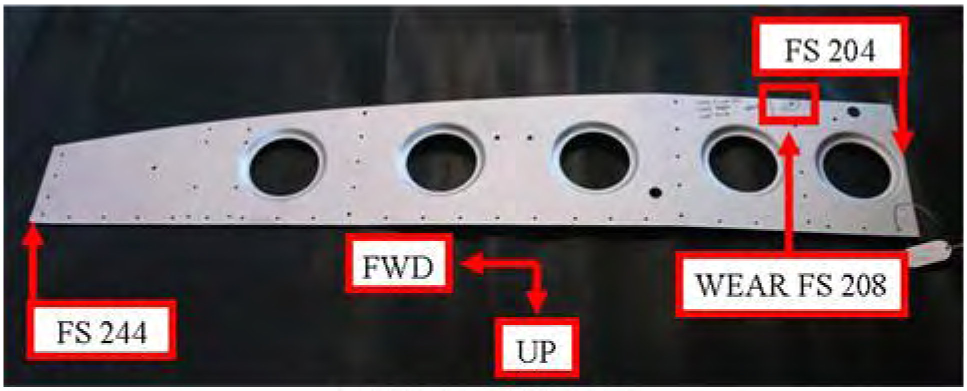


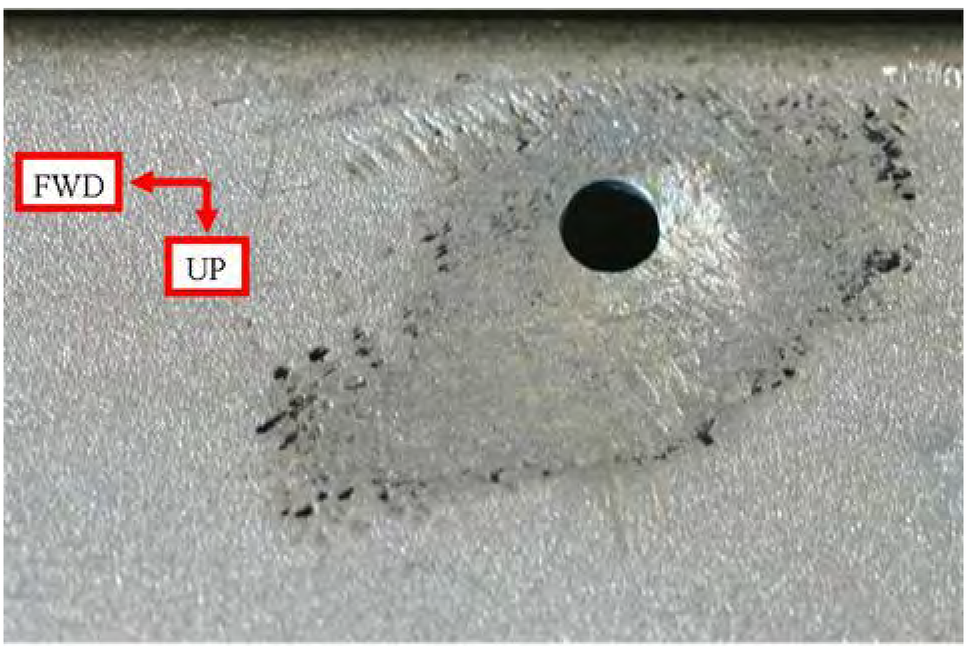
Figure 480. Macroscopic View of Moderate-Severe Corrosion on the Left Floorboard Support FS 215

The location of a 0.69-square-inch area of wear on the left longitudinal beam assembly, part number 41119, is shown in figure 481. This area of wear, located at FS 208, caused a reduction in thickness of 12.5% locally, as shown in figure 482. Figure 483 shows the location of scattered light corrosion on the fuselage right beam assembly longitudinal, part number 41119. This location was scattered across the surface of the part from FS 204 to FS 244 and caused a localized reduction in thickness of 1%. The location of a crack and light corrosion on the fuselage left beam assembly lower rear frame, part number 41119-08, is shown in figure 484. The light corrosion covered a surface area of 105 square inches from FS 213.5 to FS 228.5 and resulted in a thickness loss of 2%. A macroscopic view of the 1.59-inch crack, located at FS 220.5, is shown in figure 485.



41119 (LT) Wear FS 204-244 OV

Figure 481. Location of Wear on the Left Longitudinal Beam Assembly FS 204 Through FS 244



41119 (LT) Wear FS 208 Mac

Figure 482. Macroscopic View of Wear on the Left Longitudinal Beam Assembly FS 208

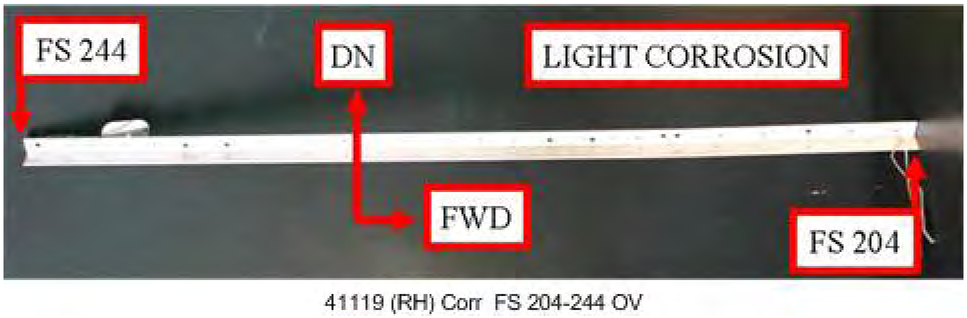


Figure 483. Location of Light Scattered Corrosion on the Fuselage Right Beam Assembly Longitudinal FS 204 Through FS 244

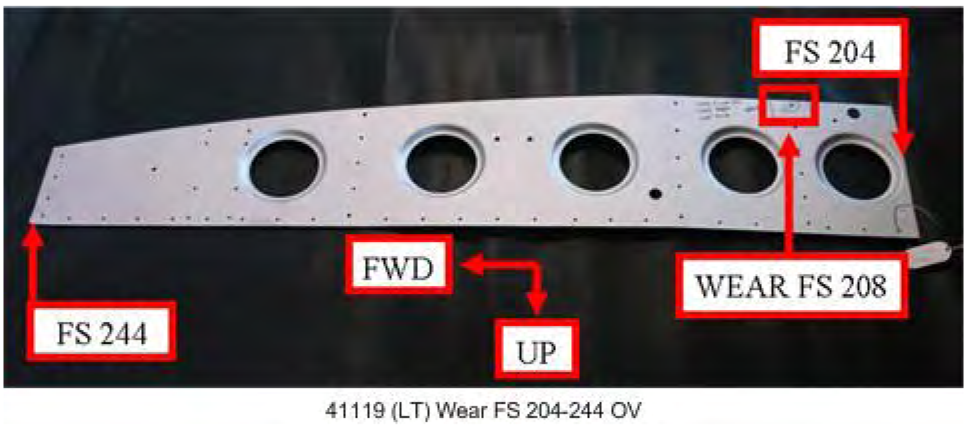


Figure 484. Location of Crack and Light Corrosion on the Fuselage Left Beam Assembly Lower Rear Frame FS 198.5 Through FS 228.5

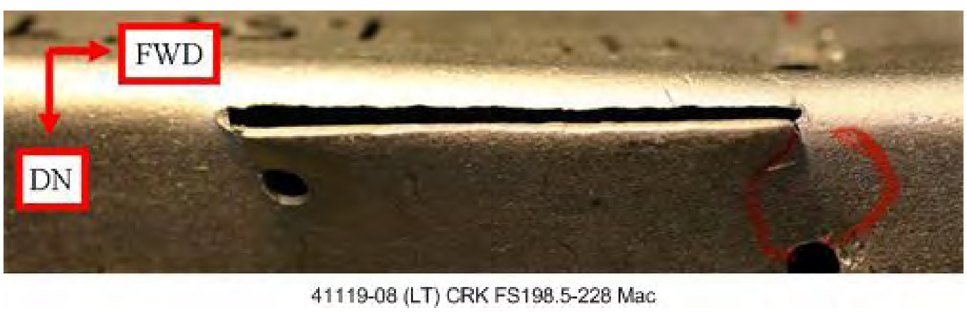
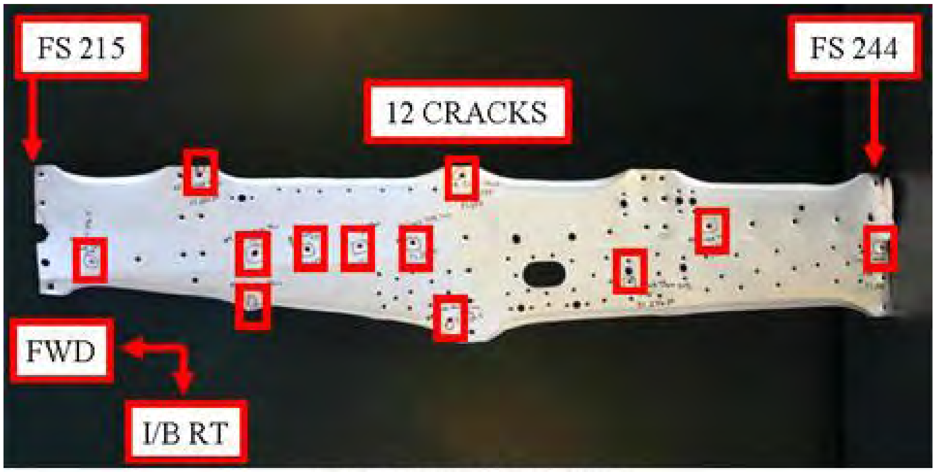


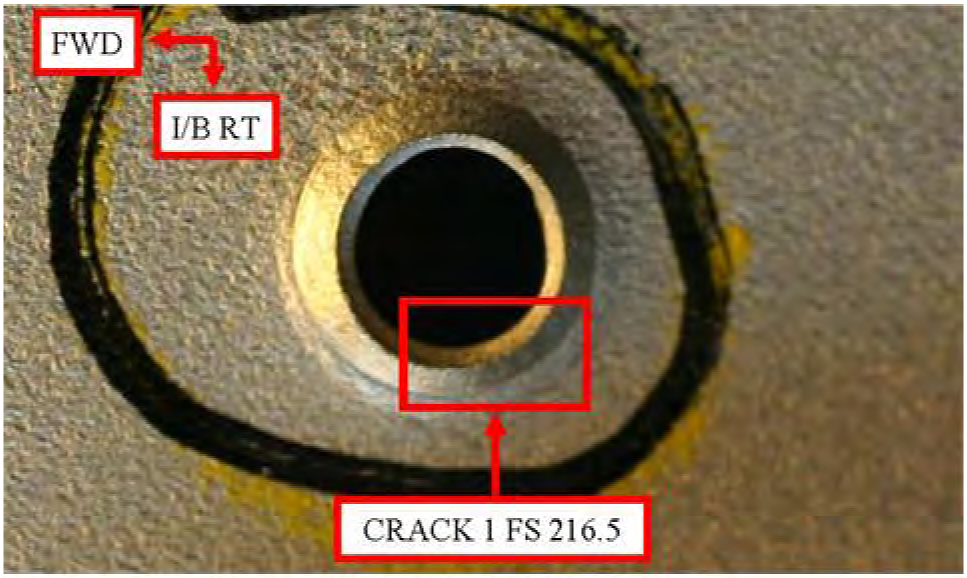
Figure 485. Macroscopic View of Crack on the Fuselage Left Beam Assembly Lower Rear Frame FS 220.5

The location of 12 cracks on the left inboard floorboard support assembly, part number 41022-0, is shown in figure 486. A macroscopic view of crack 1, which measures 0.014 inch and is located at FS 216.5, is shown in figure 487, and a microscopic view of this crack is shown in figure 488. Figure 489 shows a macroscopic view of crack 2, which measured 0.215 inch and was located at FS 220.5. A microscopic view of crack 2 is shown in figure 490. Figure 491 shows a macroscopic view of crack 3, which measured 0.508 inch and was located at FS 222, and a microscopic view of this crack is shown in figure 492. A macroscopic view of crack 4 was located at FS 222, and is shown in figure 493. A microscopic view of this 0.157-inch crack is shown in figure 494.



41022-0 (LT)CRK FS 215-244 OV

Figure 486. Location of 12 Cracks on the Left Inboard Floorboard Assembly
FS 215 Through FS 244



41022-0 (LT) CRK 1 FS 216.5 Mac

Figure 487. Macroscopic View of Crack 1 on the Left Inboard Floorboard Support Assembly FS 216.5

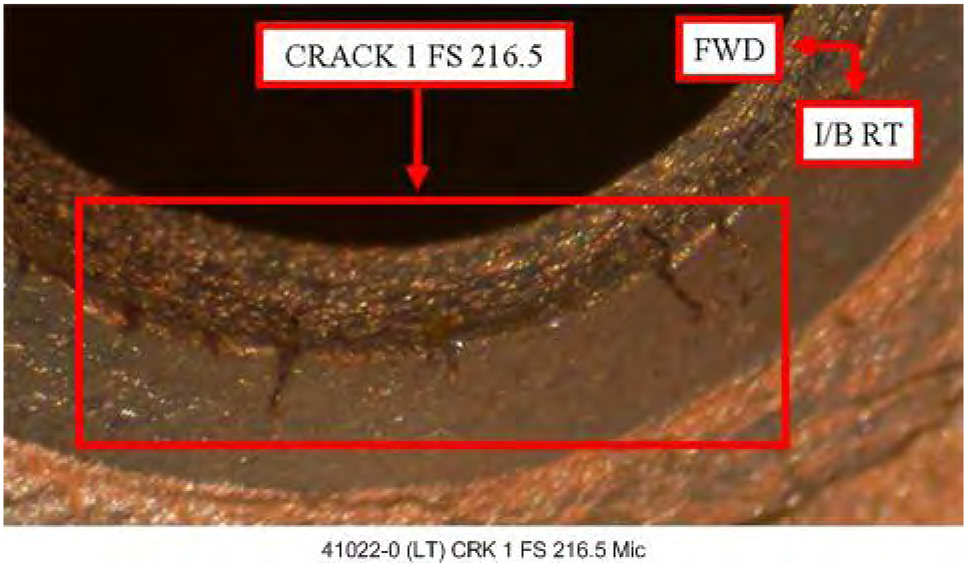


Figure 488. Microscopic View of Crack 1 on the Left Inboard Floorboard Support Assembly FS 216.5

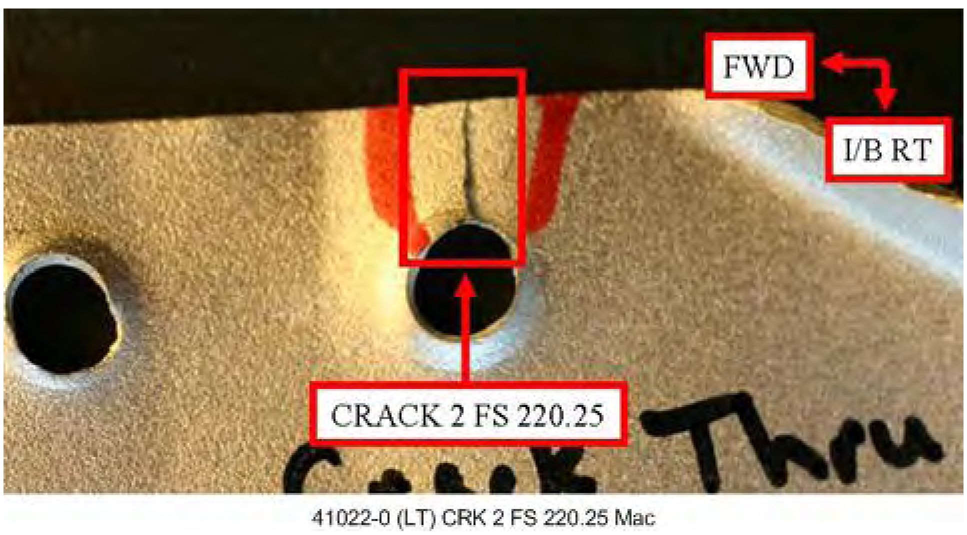


Figure 489. Macroscopic View of Crack 2 on the Left Inboard Floorboard Support Assembly FS 220.25

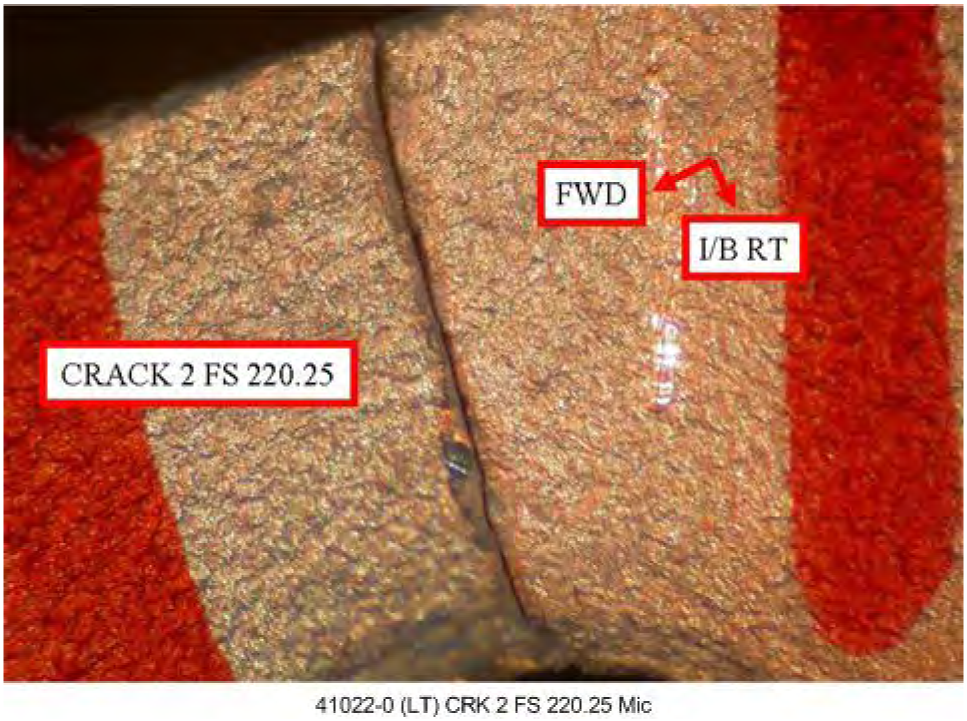


Figure 490. Microscopic View of Crack 2 on the Left Inboard Floorboard Support Assembly FS 220.25

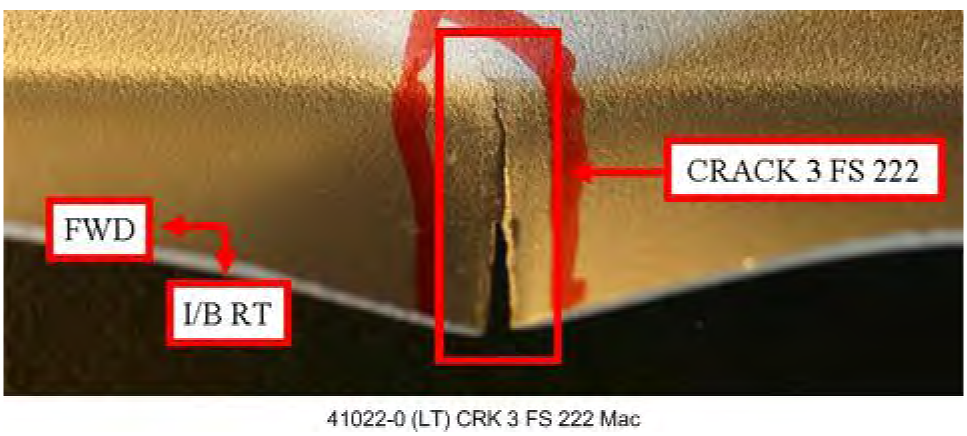
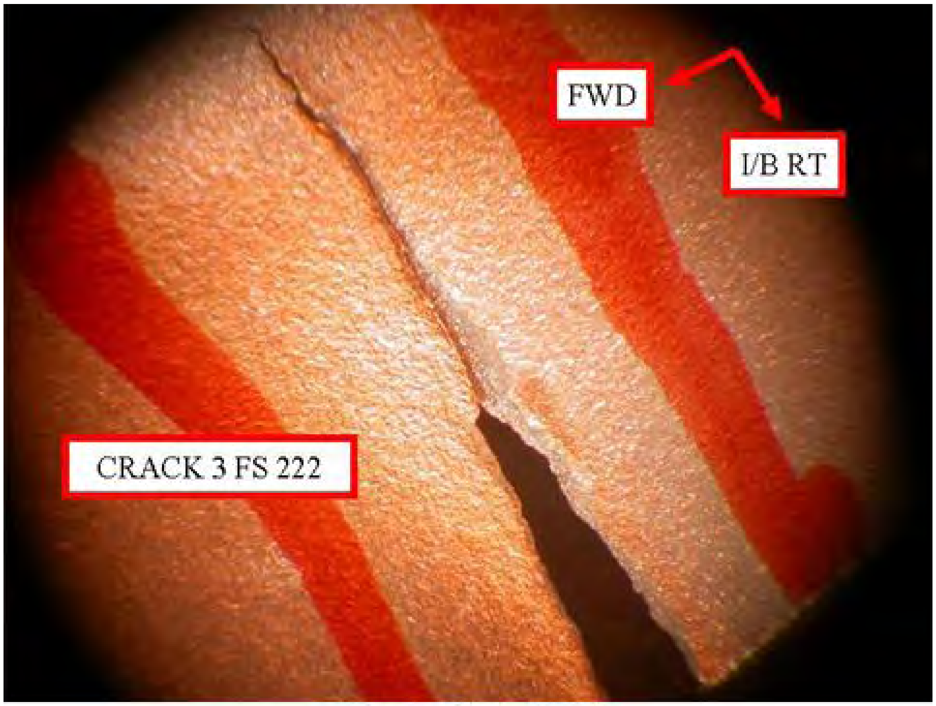
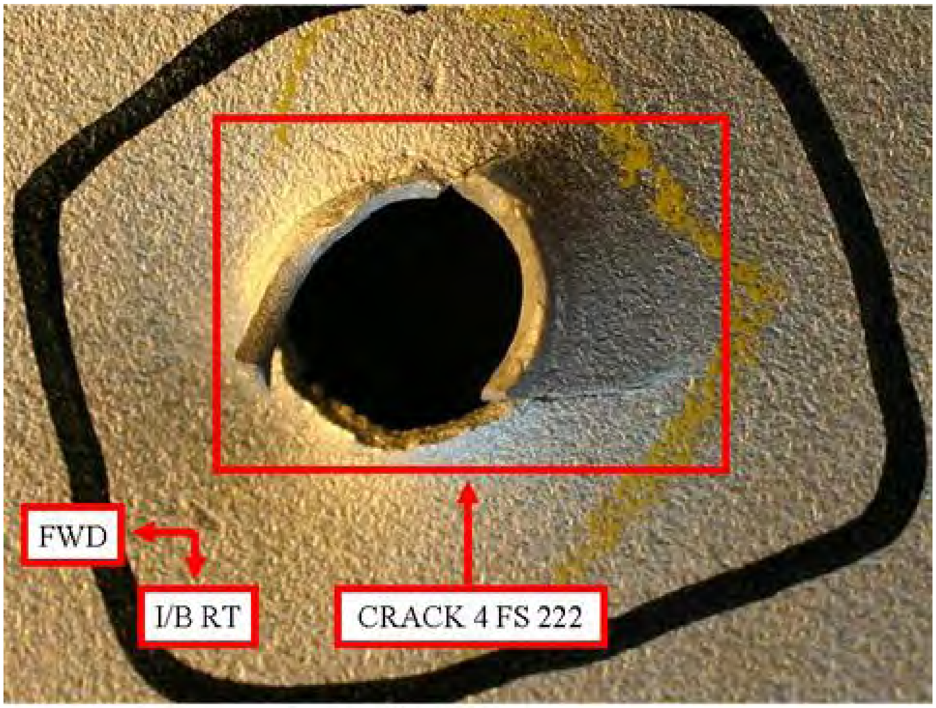


Figure 491. Macroscopic View of Crack 3 on the Left Inboard Floorboard Support Assembly FS 222



41022-0 (LT) CRK 3 FS 222 Mic

Figure 492. Microscopic View of Crack 3 on the Left Inboard Floorboard Support Assembly FS 222



41022-0 (LT) CRK 4 FS 222 Mac

Figure 493. Macroscopic View of Crack 4 on the Left Inboard Floorboard Support Assembly FS 222

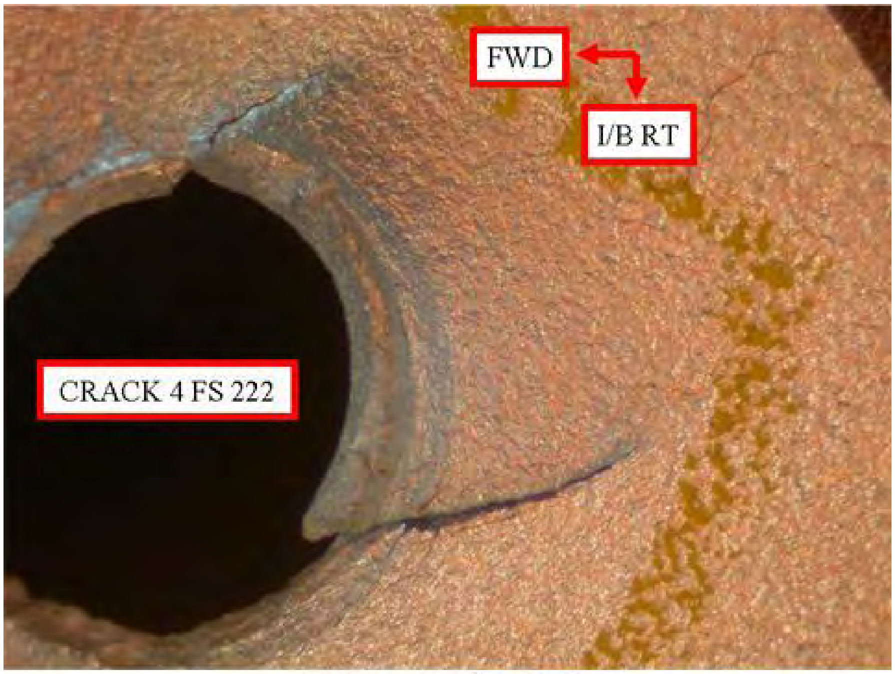


Figure 494. Microscopic View of Crack 4 on the Left Inboard Floorboard Support Assembly FS 222

A macroscopic view of crack 5 on the left inboard floorboard assembly, which measured 0.03 inch and was located at FS 223, is shown in figure 495, and figure 496 shows a microscopic view of crack 5. Crack 6, which measures 0.036 inch and was located at FS 225, is shown macroscopically in figure 497 and microscopically in figure 498. Figure 499 shows a macroscopic view of crack 7. This crack measured 0.166 inch and was located at FS 227.5. A microscopic view of crack 7 is shown in figure 500. Multiple cracks grouped together and identified as crack 8 measured 0.26 inch in length and occurred at FS 229.5. A macroscopic view of crack 8 is shown in figure 501, and a microscopic view is shown in figure 502.

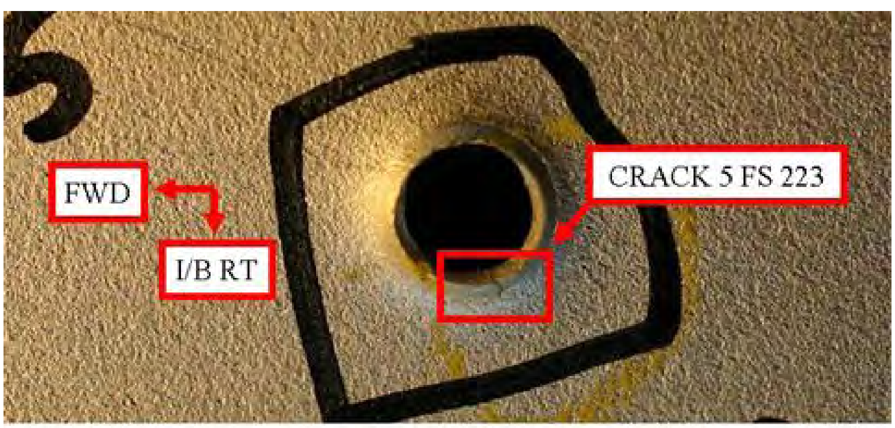
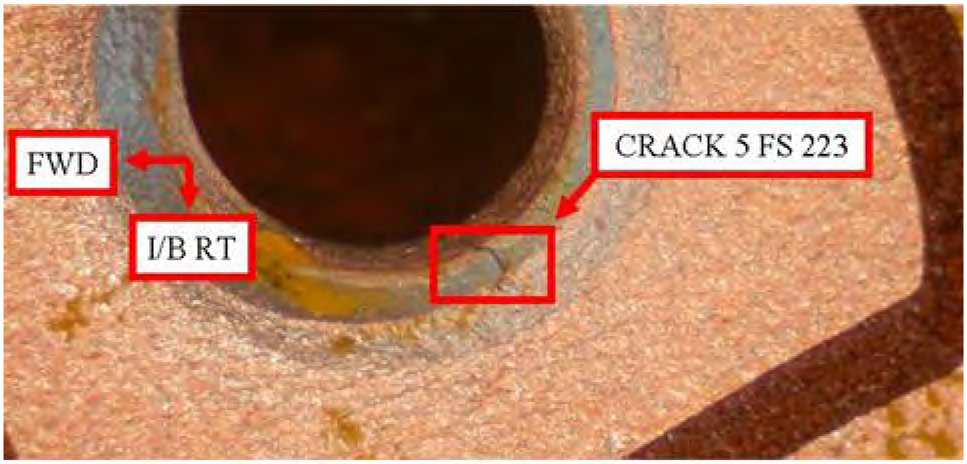
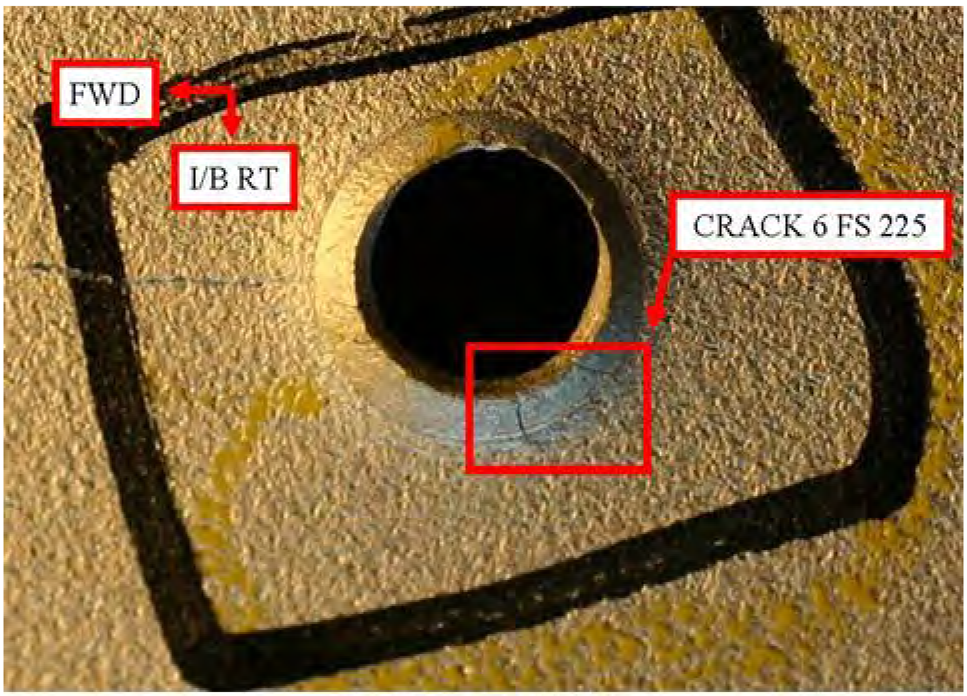


Figure 495. Macroscopic View of Crack 5 on the Left Inboard Floorboard Support Assembly FS 223



41022-0 (LT) CRK 5 FS 223 Mic

Figure 496. Microscopic View of Crack 5 on the Left Inboard Floorboard Support Assembly FS 223



41022-0 (LT) CRK 6 FS 225 Mac

Figure 497. Macroscopic View of Crack 6 on the Left Inboard Floorboard Support Assembly FS 225

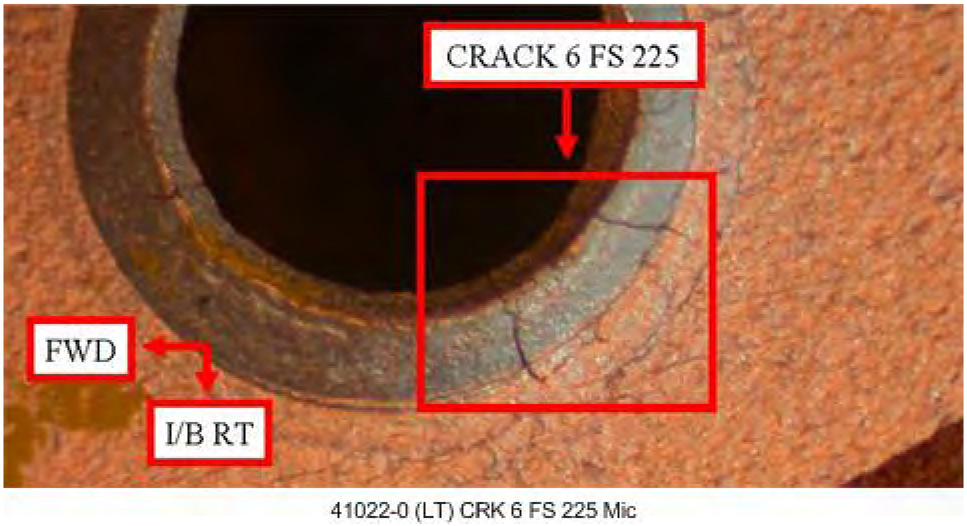


Figure 498. Microscopic View of Crack 6 on the Left Inboard Floorboard Support Assembly FS 225

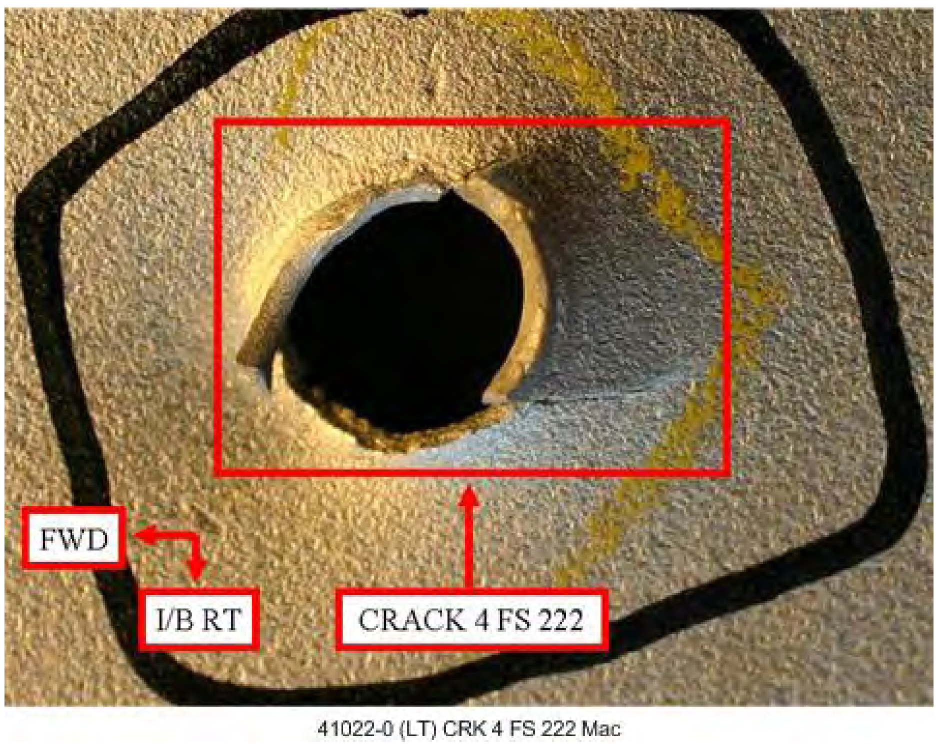


Figure 499. Macroscopic View of Crack 7 on the Left Inboard Floorboard Support Assembly FS 227.5

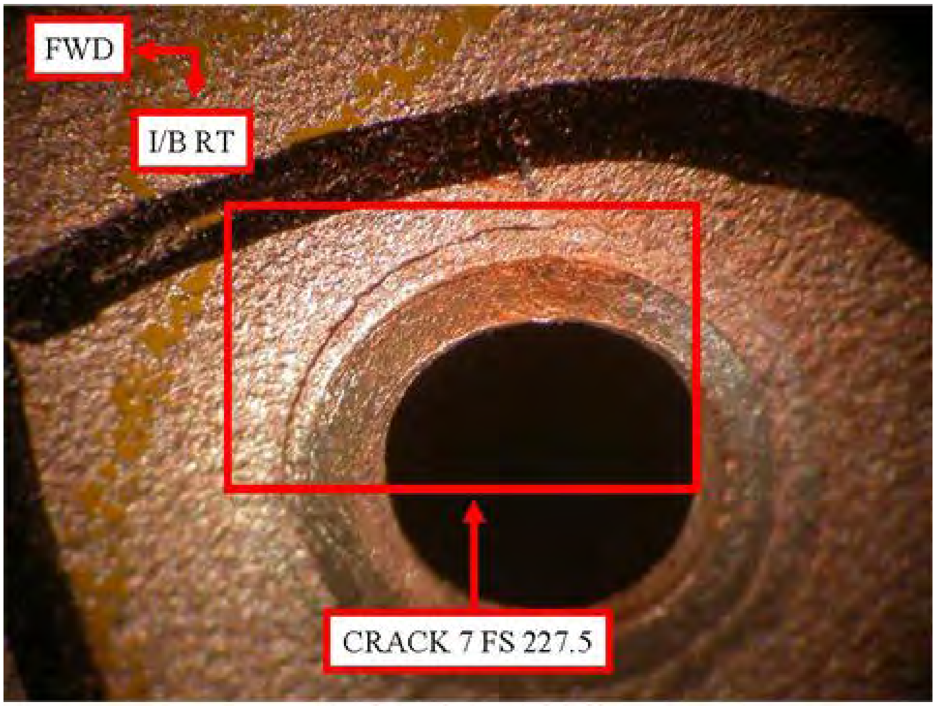


Figure 500. Microscopic View of Crack 7 on the Left Inboard Floorboard Support Assembly FS 227.5

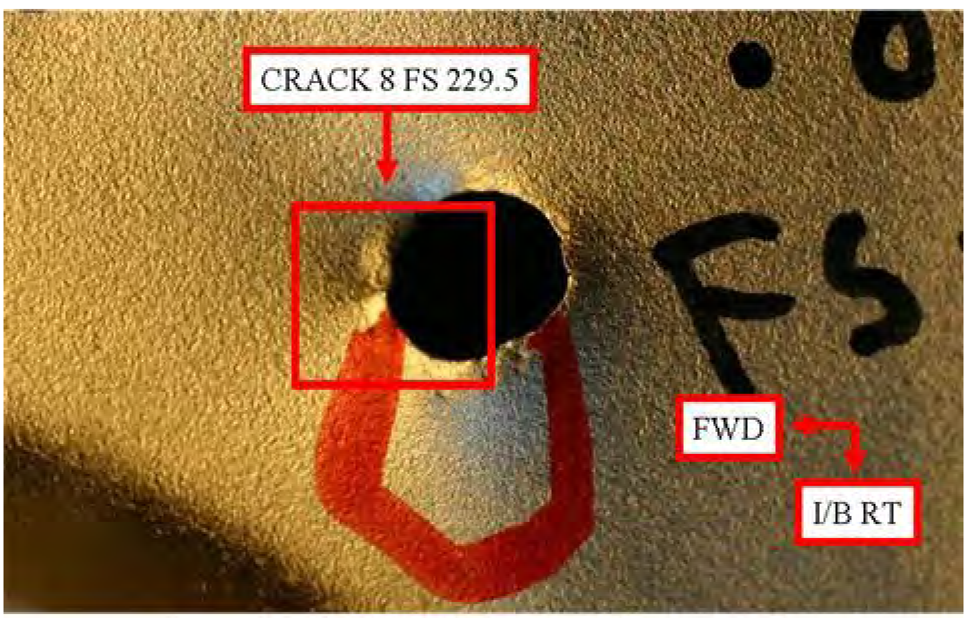


Figure 501. Macroscopic View of Crack 8 on the Left Inboard Floorboard Support Assembly FS 229.5

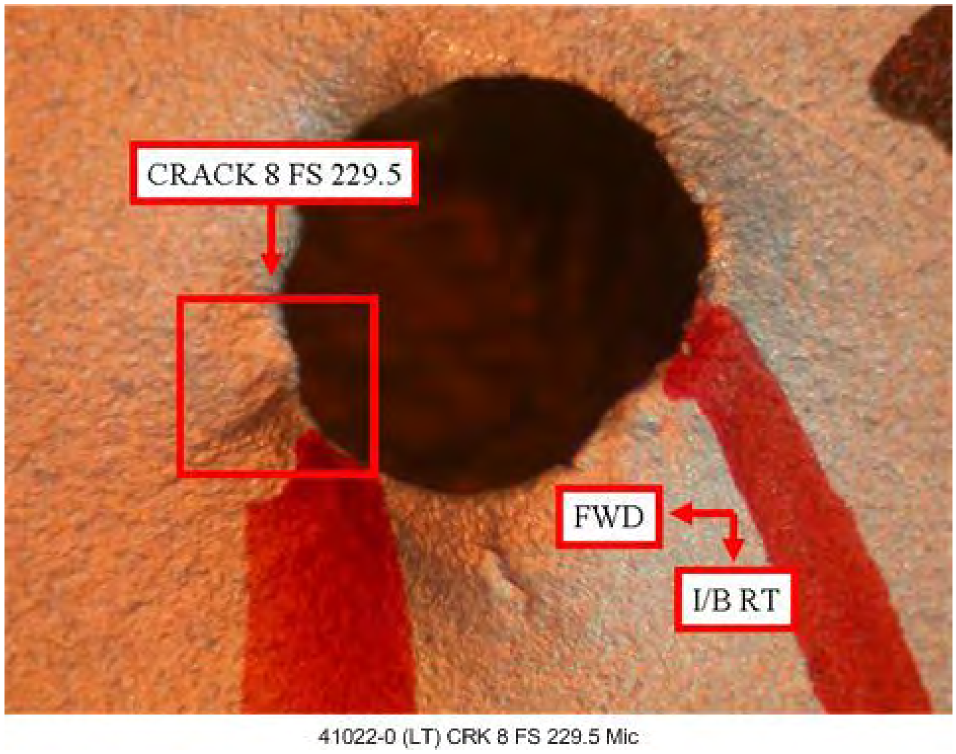
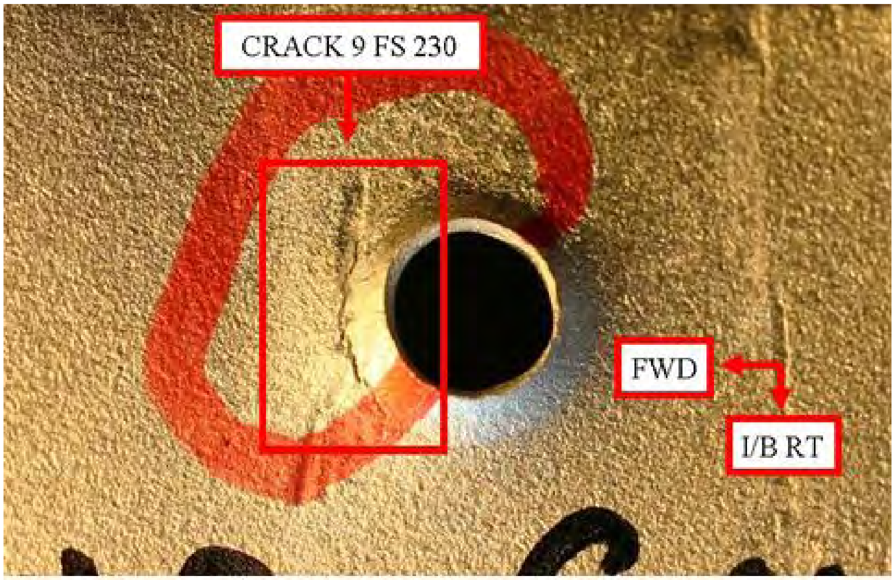


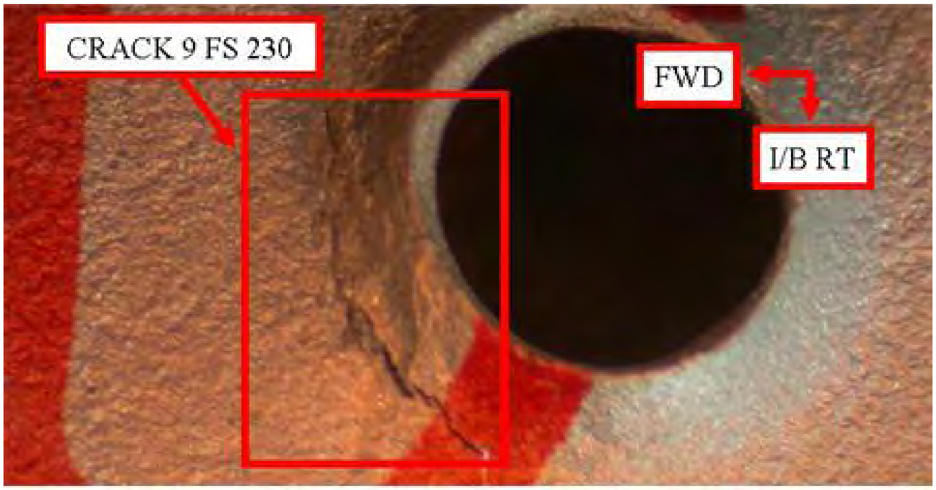
Figure 502. Microscopic View of Crack 8 on the Left Inboard Floorboard Support Assembly FS 229.5

Figure 503 shows a macroscopic view of crack 9 on the left inboard floorboard support assembly, and figure 504 shows a microscopic view of this crack. Crack 9 measured 0.249 inch and occurred at FS 230. Crack 10, which is shown macroscopically in figure 505, measured 0.142 inch in length and occurred at FS 236.25. A microscopic view of crack 10 is shown in figure 506. Figure 507 shows a macroscopic view of crack 11, and figure 508 provides a microscopic view of this 0.31-inch crack located at FS 238. Multiple cracks grouped together and identified as crack 12 are shown macroscopically in figure 509 and microscopically in figure 510. The longest of these cracks measured 0.038 inch and occurred at FS 244.



41022-0 (LT) CRK 9 FS 230 Mac

Figure 503. Macroscopic View of Crack 9 on the Left Inboard Floorboard Support Assembly FS 230



41022-0 (LT) CRK 9 FS 230 Mic

Figure 504. Microscopic View of Crack 9 on the Left Inboard Floorboard Support Assembly FS 230

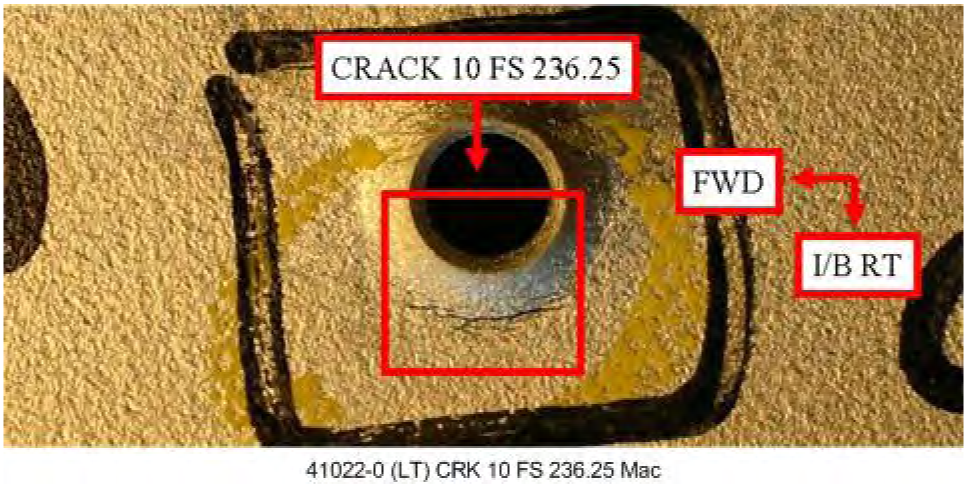


Figure 505. Macroscopic View of Crack 10 on the Left Inboard Floorboard Support Assembly FS 236.25

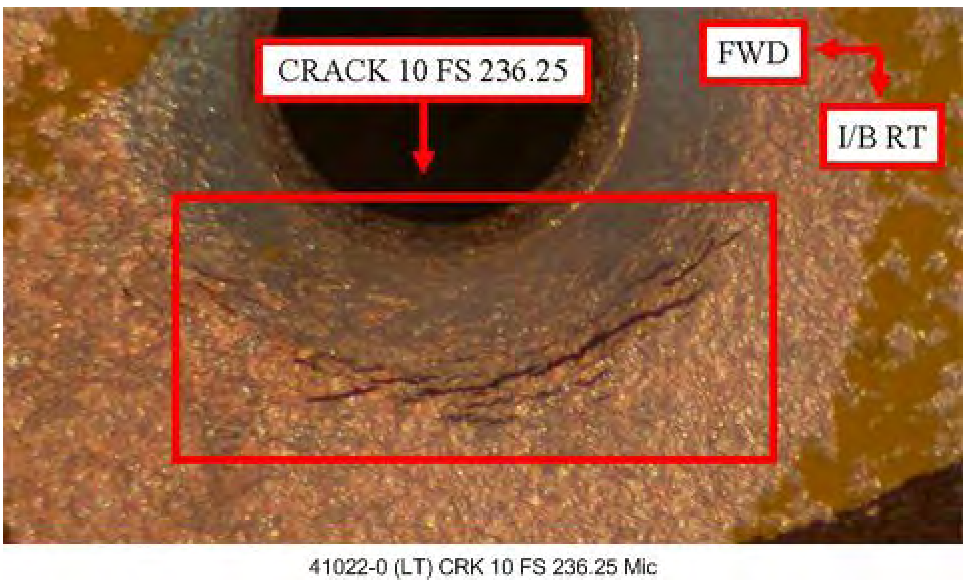
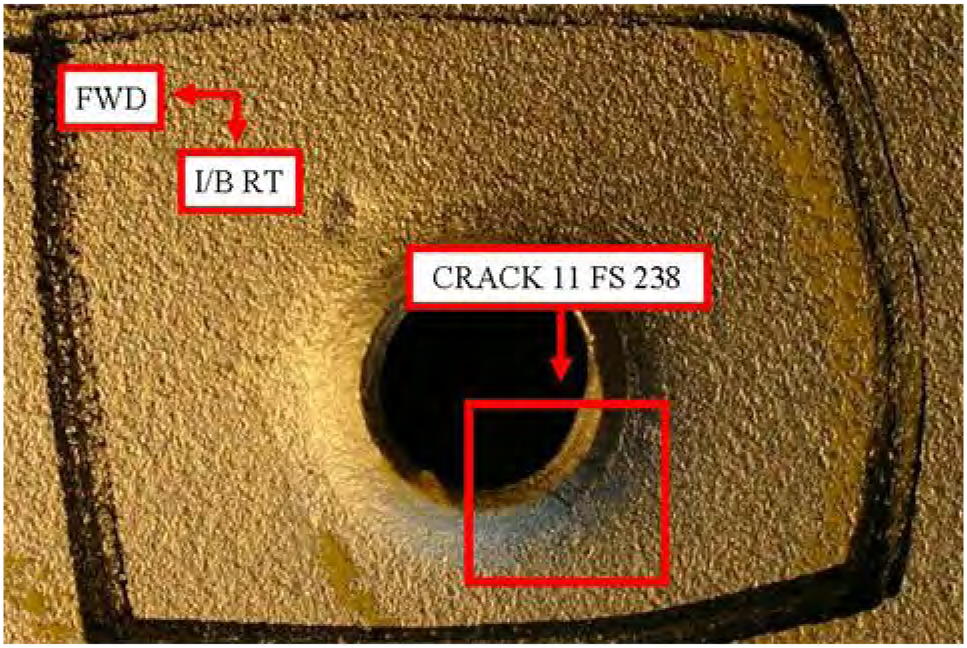
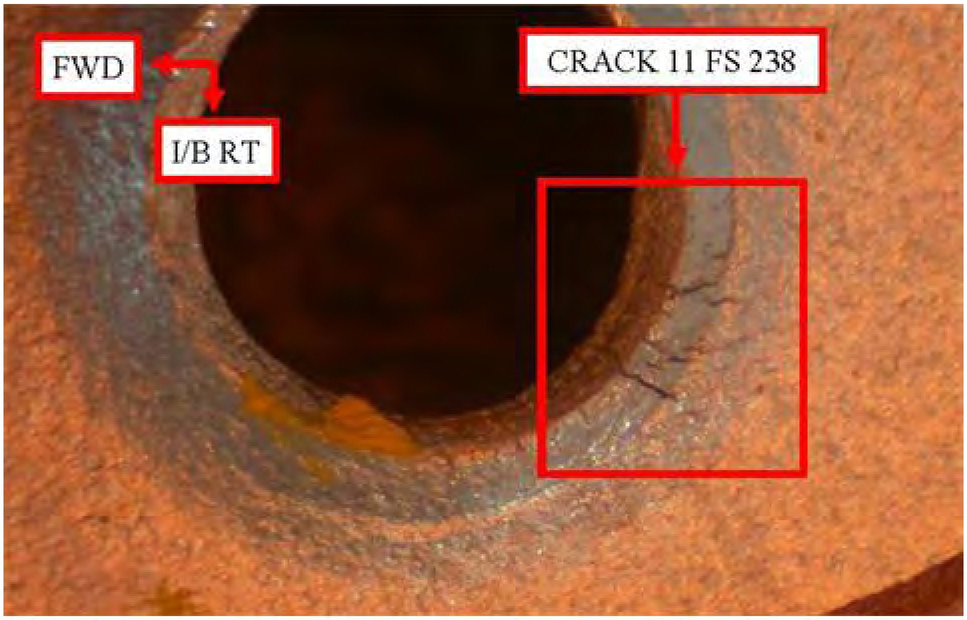


Figure 506. Microscopic View of Crack 10 on the Left Inboard Floorboard Support Assembly FS 236.25



41022-0 (LT) CRK 11 FS 238 Mac

Figure 507. Macroscopic View of Crack 11 on the Left Inboard Floorboard Support Assembly FS 238



41022-0 (LT) CRK 11 FS 238 Mic

Figure 508. Microscopic View of Crack 11 on the Left Inboard Floorboard Support Assembly FS 238

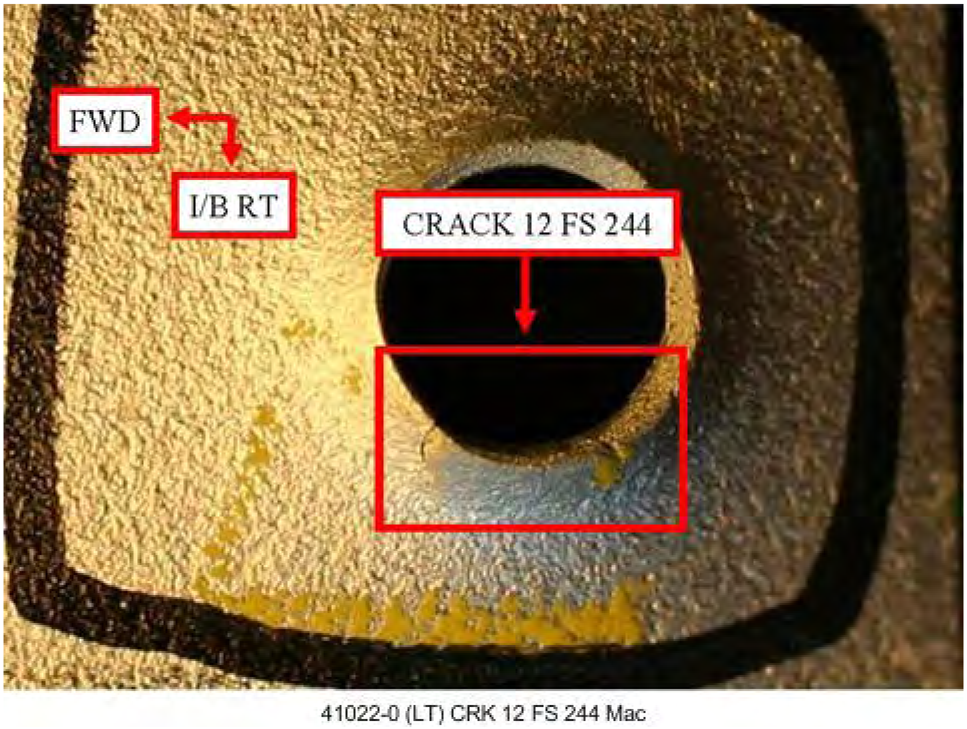


Figure 509. Macroscopic View of Crack 12 on the Left Inboard Floorboard Support Assembly FS 244

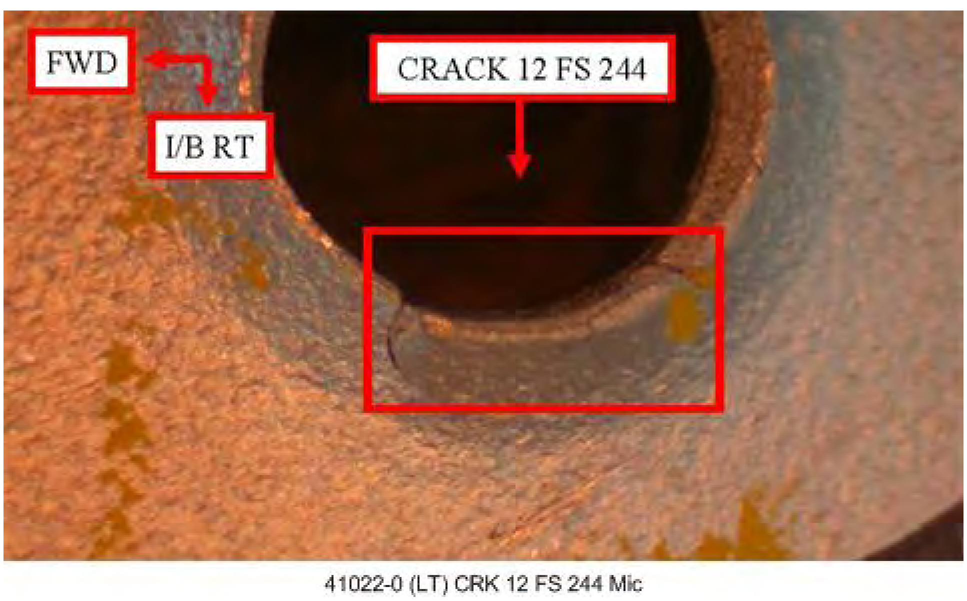
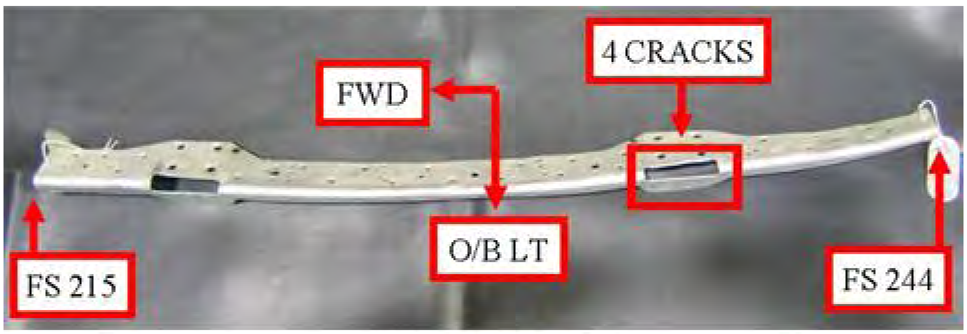


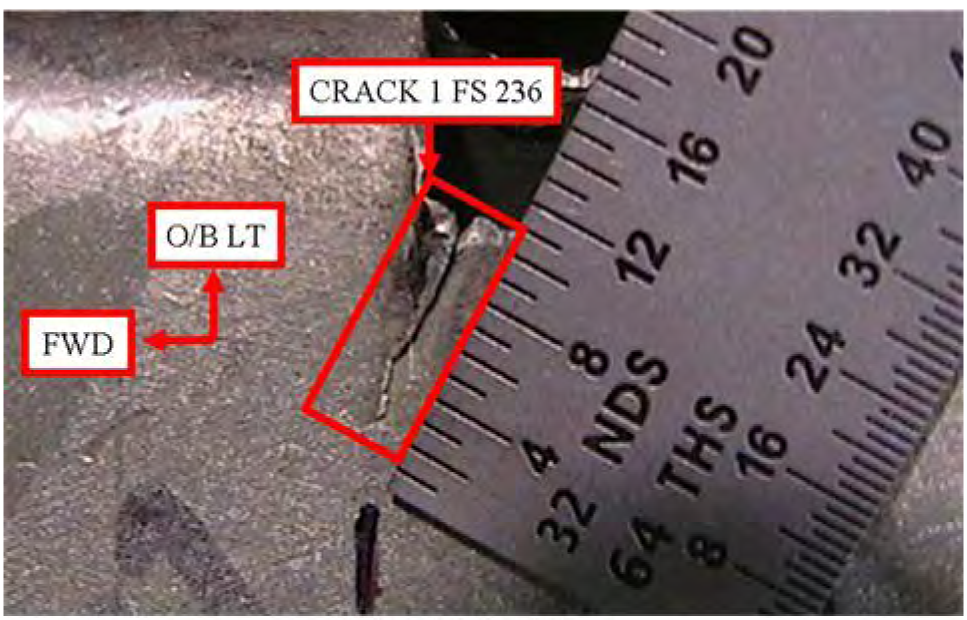
Figure 510. Microscopic View of Crack 12 on the Left Inboard Floorboard Support Assembly FS 244

During the teardown evaluation, four cracks were identified on the door sill support aft body, part number 41189-00. The location of these cracks is shown in figure 511. A macroscopic view of crack 1, which measured 0.409 inch and was located at FS 236, is shown in figure 512. Figure 513 shows a macroscopic view of crack 2. Crack 2 was also located at FS 236 and measured 0.437 inch in length. A macroscopic view of crack 3 is shown in figure 514. Crack 3 measured 0.428 inch in length and occurred at FS 237.5. Crack 4 measured 0.267 inch and was located at FS 237.5. A macroscopic view of crack 4 is shown in figure 515. All four cracks were examined in an effort to determine failure mode, but due to extensive crack face smearing, the failure mode could not be determined. Figure 516 shows the location of light-moderate scattered corrosion on the fuselage center body angle, part number 41119-010. This corrosion was scattered across the entire surface of the part and caused a maximum localized thickness of 3.2%.



41189-00 CRK FS215-244 OV

Figure 511. Location of Four Cracks on the Door Sill Support Aft Body FS 215 Through FS 244



41189-00 CRK 1 FS236 Mac

Figure 512. Macroscopic View of Crack 1 on the Door Sill Support Aft Body FS 236

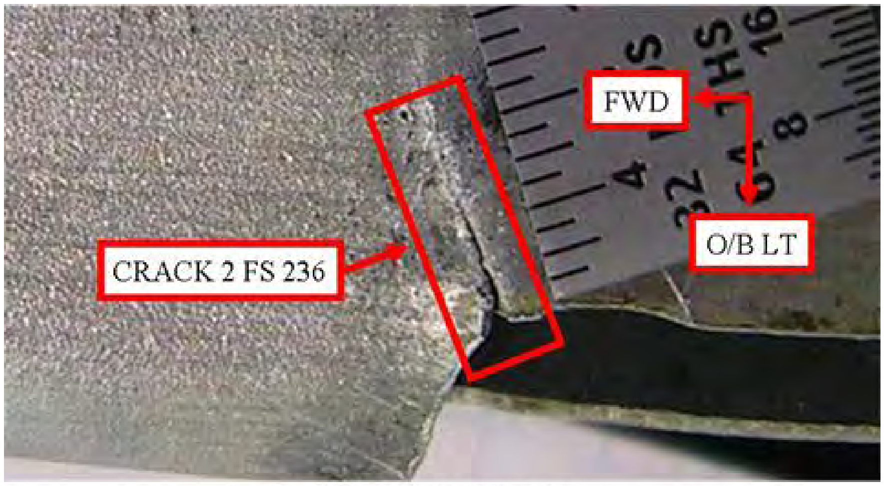


Figure 513. Macroscopic View of Crack 2 on the Door Sill Support Aft Body FS 236

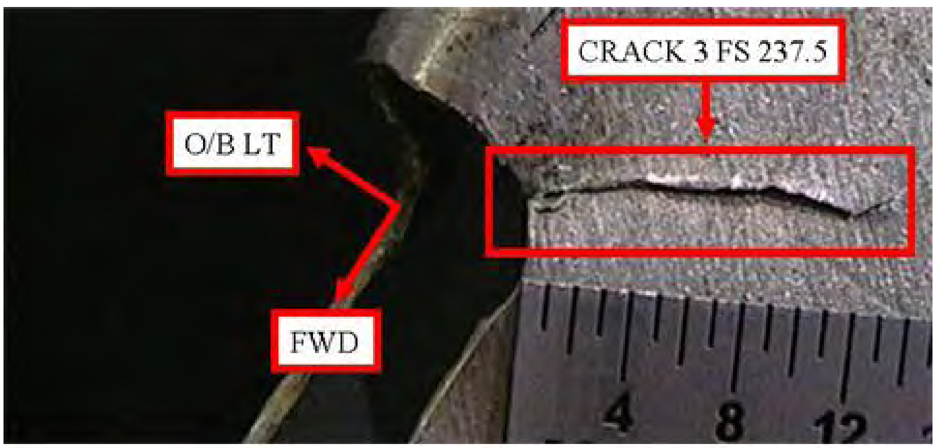


Figure 514. Macroscopic View of Crack 3 on the Door Sill Support Aft Body FS 237.5

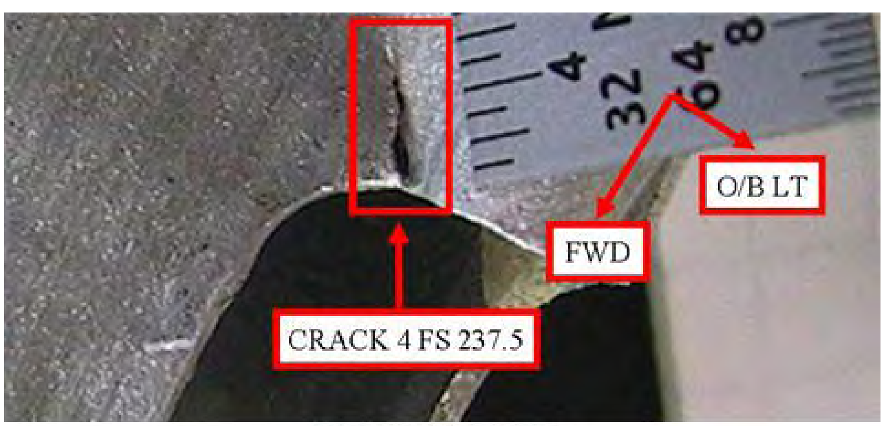


Figure 515. Macroscopic View of Crack 4 on the Door Sill Support Aft Body FS 237.5

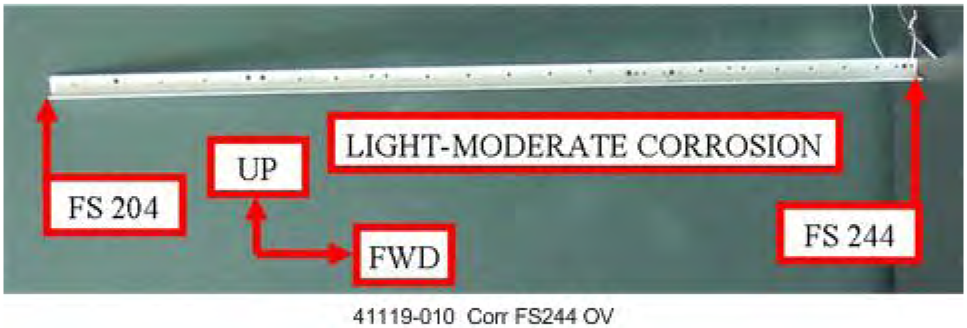


Figure 516. Location of Light-Moderate Scattered Corrosion on the Fuselage Center Body Angle FS 204 Through FS 244

3.4.2.3.5 Fuselage FS 245 Through FS 317.75.

Table 39 shows a detailed characterization of all defects found on the fuselage from FS 245 to FS 317.75.

Table 39. Inspection Results From FS 245 Through FS 317.75

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Bulkhead right aft fuselage, figure 518	Crack	FS 249	0.435 inch	Crack indication	Bend radii	519 520
Oxygen bottle bracket assembly, figure 521	Crack	FS 296	1.66 inches	Not inspected[1]	Fatigue	522 523 524
Aft fuselage bulkhead assembly (1 of 2), figure 525	Crack	FS 317.75	1.165 inches	Crack indication	Bend radii surface crack	526
	Crack	FS 317.75	0.75 inch	Crack indication	Bend radii surface crack	527
	Crack	FS 317.75	0.55 inch	Crack indication	Bend radii surface crack	528
Aft fuselage bulkhead assembly (2 of 2), figure 529	Crack	FS 317.75	1.138 inches	Crack indication	Bend radii surface crack	530
	Crack	FS 317.75	1.192 inches	Crack indication	Bend radii surface crack	531
Aft fuselage bulkhead assembly, figure 532	Crack	FS 317.75	0.48 inch	Crack indication	Surface crack	533 534

[1]Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

The location of a crack on the bulkhead right aft fuselage, part number 40732-01, is shown in figure 517. This crack measured 0.435 inch and was located at FS 249. A macroscopic view of this crack is shown in figure 518, and a microscopic view of this crack is shown in figure 519. The location of a 1.66-inch crack on the oxygen bottle bracket assembly, part number 52820-008, is shown in figure 520. A macroscopic view of this crack that was located at FS 296 is shown in figure 521. The fracture face of this crack is shown in figure 522, and a fractograph of the crack face, showing defined fatigue striations and crack growth direction, is shown in figure 523.

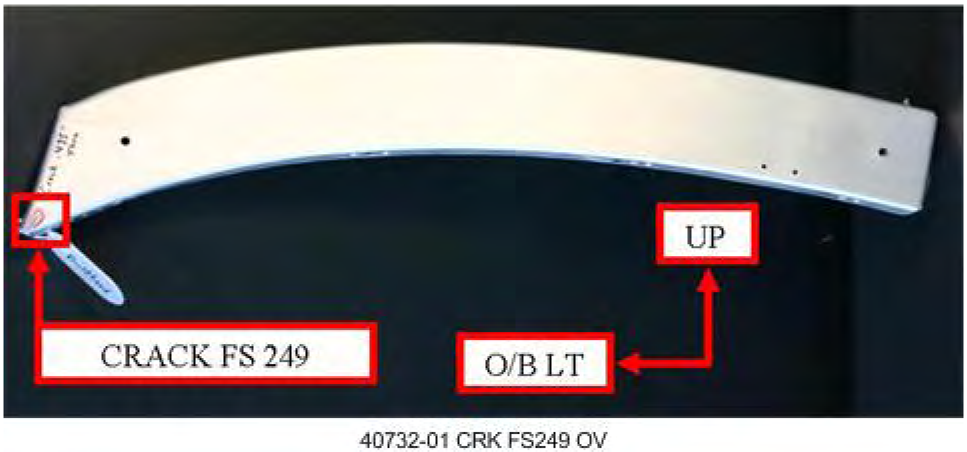


Figure 517. Location of Crack on the Bulkhead Right Aft Fuselage FS 249

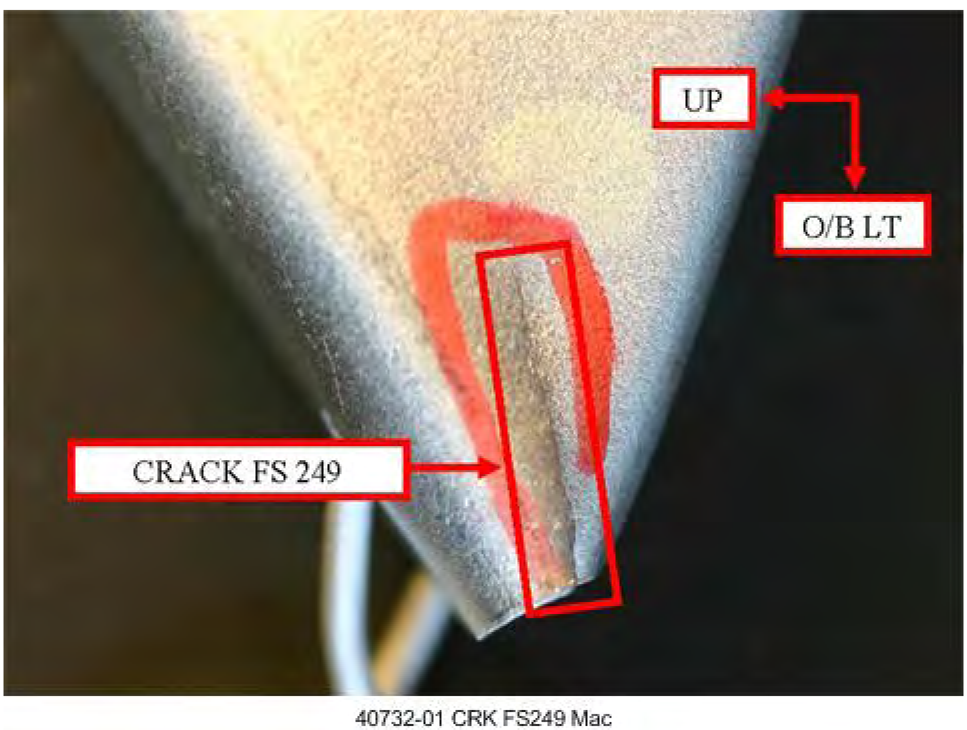


Figure 518. Macroscopic View of Crack on the Bulkhead Right Aft Fuselage FS 249

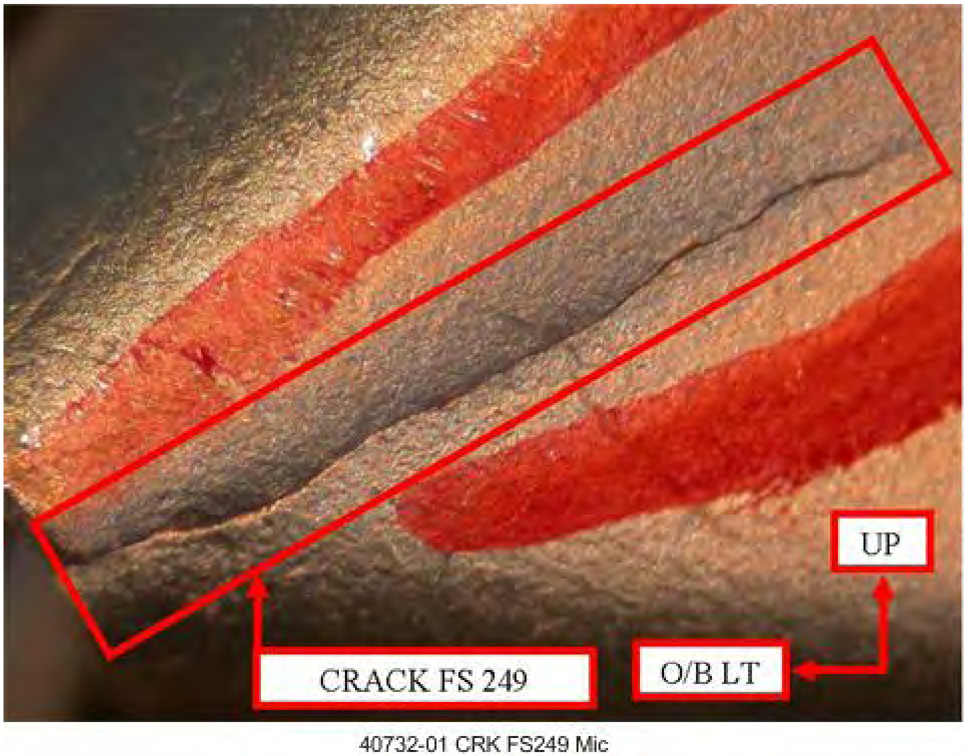


Figure 519. Microscopic View of Crack on the Bulkhead Right Aft Fuselage FS 249

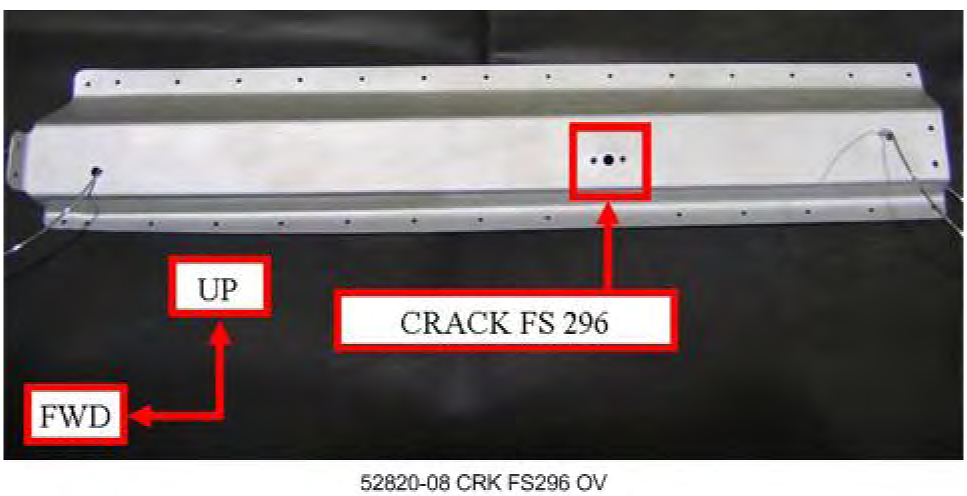
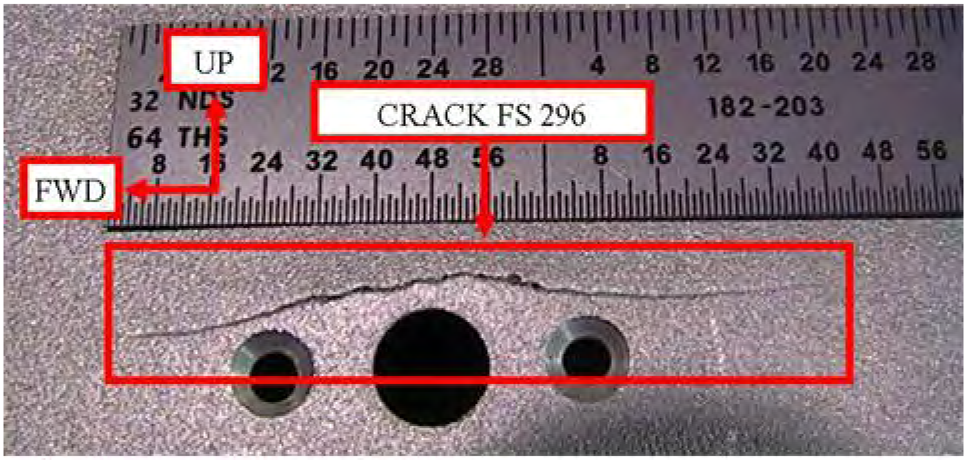
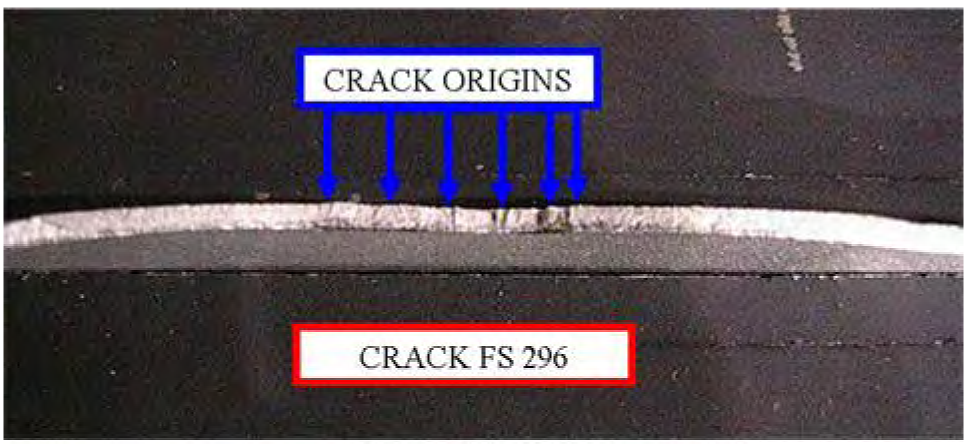


Figure 520. Location of Crack on the Oxygen Bottle Bracket Assembly FS 296



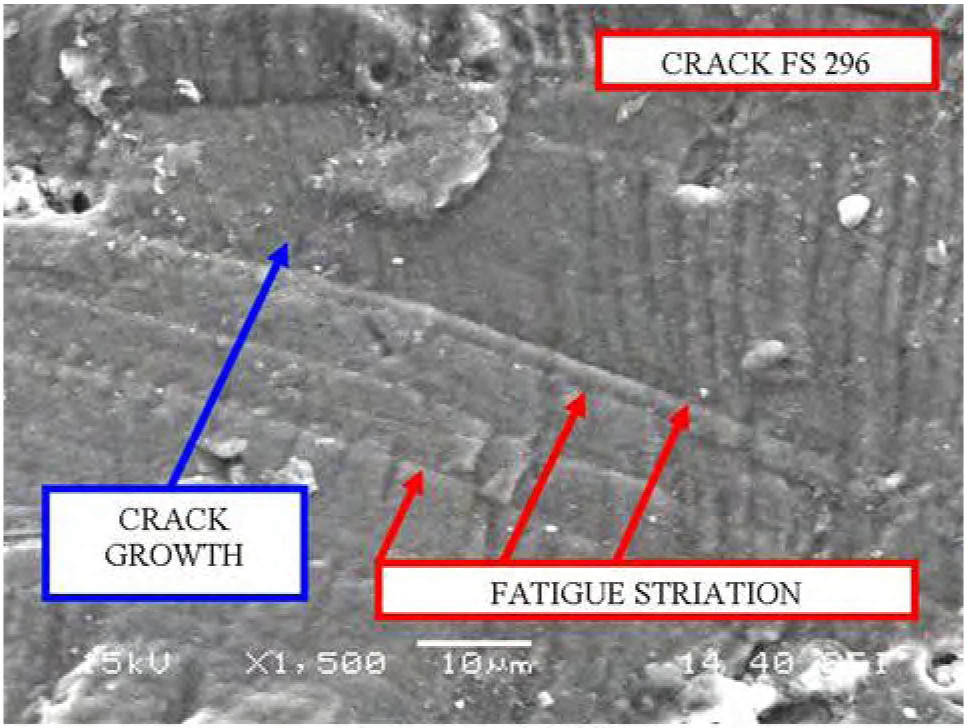
52820-08 CRK FS296 Mac

Figure 521. Macroscopic View of Crack on the Oxygen Bottle Bracket Assembly FS 296



52820-08 CRK FS296 Frac

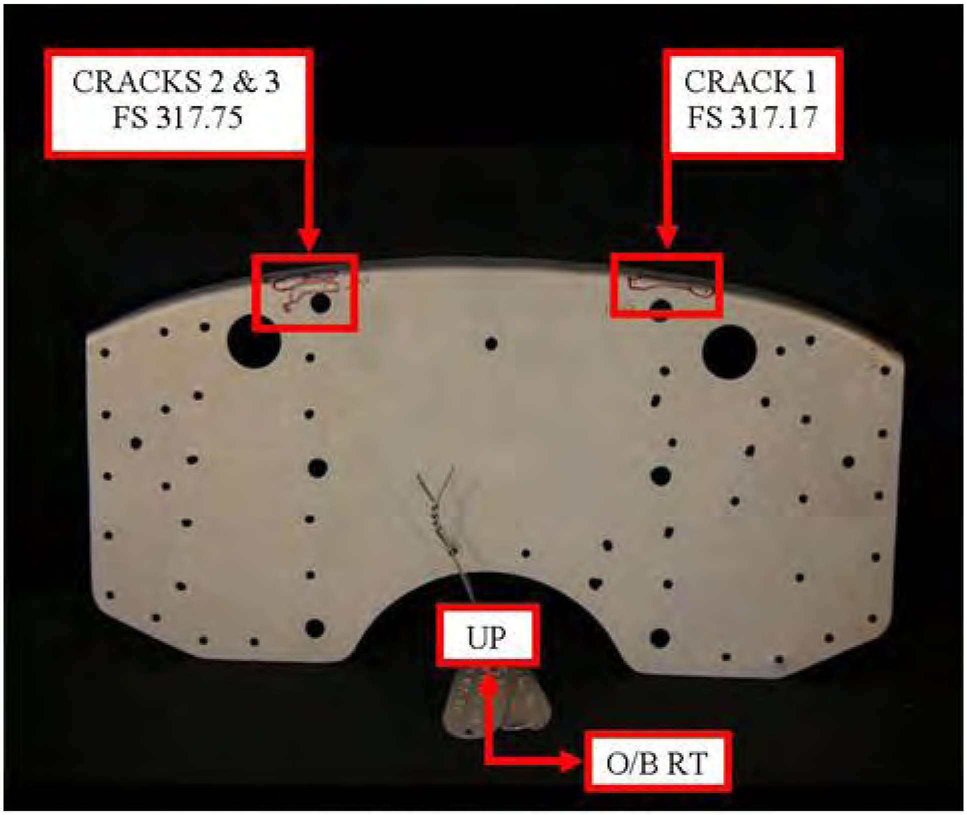
Figure 522. Fracture Face of Crack on the Oxygen Bottle Bracket Assembly FS 296



52820-08 CRK FS296 SEM

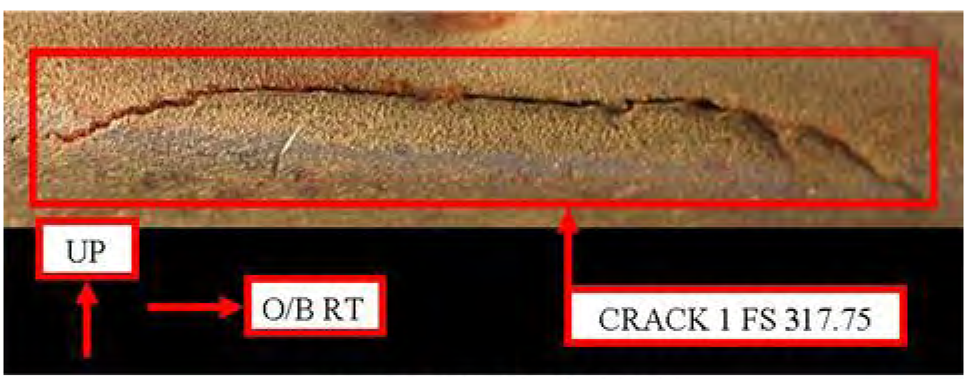
Figure 523. Fractograph of the Crack on the Oxygen Bottle Bracket Assembly FS 296

The location of three cracks on the aft fuselage bulkhead assembly (1 of 2), part number 40682-08, is shown in figure 524. Figure 525 shows a macroscopic view of crack 1, which measured 1.165 inches and was located at FS 317.75. A macroscopic view of crack 2, which measured 0.75 inch and was located at FS 317.75, is shown in figure 526, and figure 527 shows a macroscopic view of crack 3. Crack 3 measured 0.55 inch and was also located at FS 317.75.



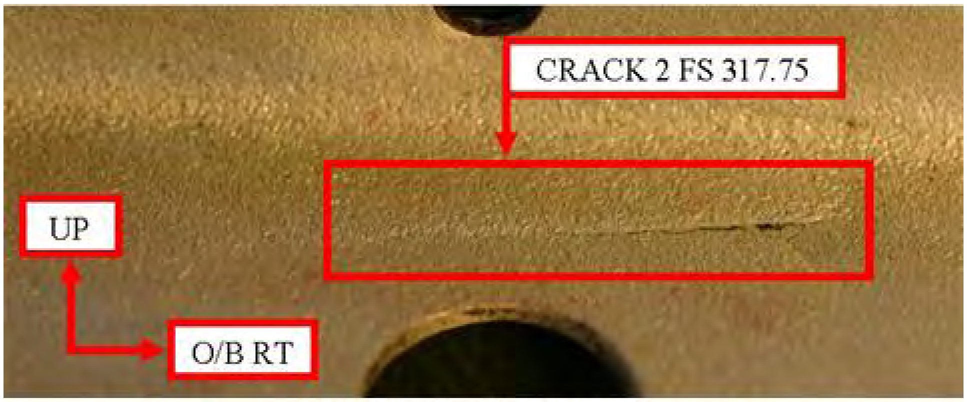
40682-08(1 of 2) CRK FS317.75 OV

Figure 524. Location of Three Cracks on the Aft Fuselage Bulkhead Assembly (1 of 2) FS 317.75



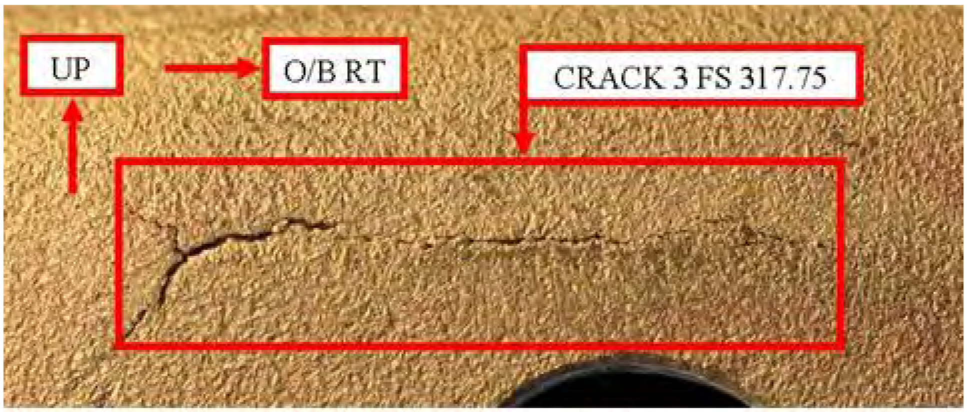
40682-08(1 of 2) CRK 1 FS317.75 Mac

Figure 525. Macroscopic View of Crack 1 on the Aft Fuselage Bulkhead Assembly (1 of 2) FS 317.75



40682-08(1 of 2) CRK 2 FS317.75 Mac

Figure 526. Macroscopic View of Crack 2 on the Aft Fuselage Bulkhead Assembly (1 of 2) FS 317.75



40682-08(1 of 2) CRK 3 FS317.75 Mac

Figure 527. Macroscopic View of Crack 3 on the Aft Fuselage Bulkhead Assembly (1 of 2) FS 317.75

Figure 528 shows the location of two cracks on the aft fuselage bulkhead assembly (2 of 2) at FS 317.75, part number 40682-08. A macroscopic view of crack 1, which measures 1.138 inches, is shown in figure 529. Figure 530 shows a macroscopic view of crack 2, which measures 1.192 inches. Figure 531 shows the location of a 0.48-inch crack on the aft fuselage bulkhead assembly, part number 40682-4. A macroscopic view of this crack is shown in figure 532, and a microscopic view of the crack is shown in figure 533.

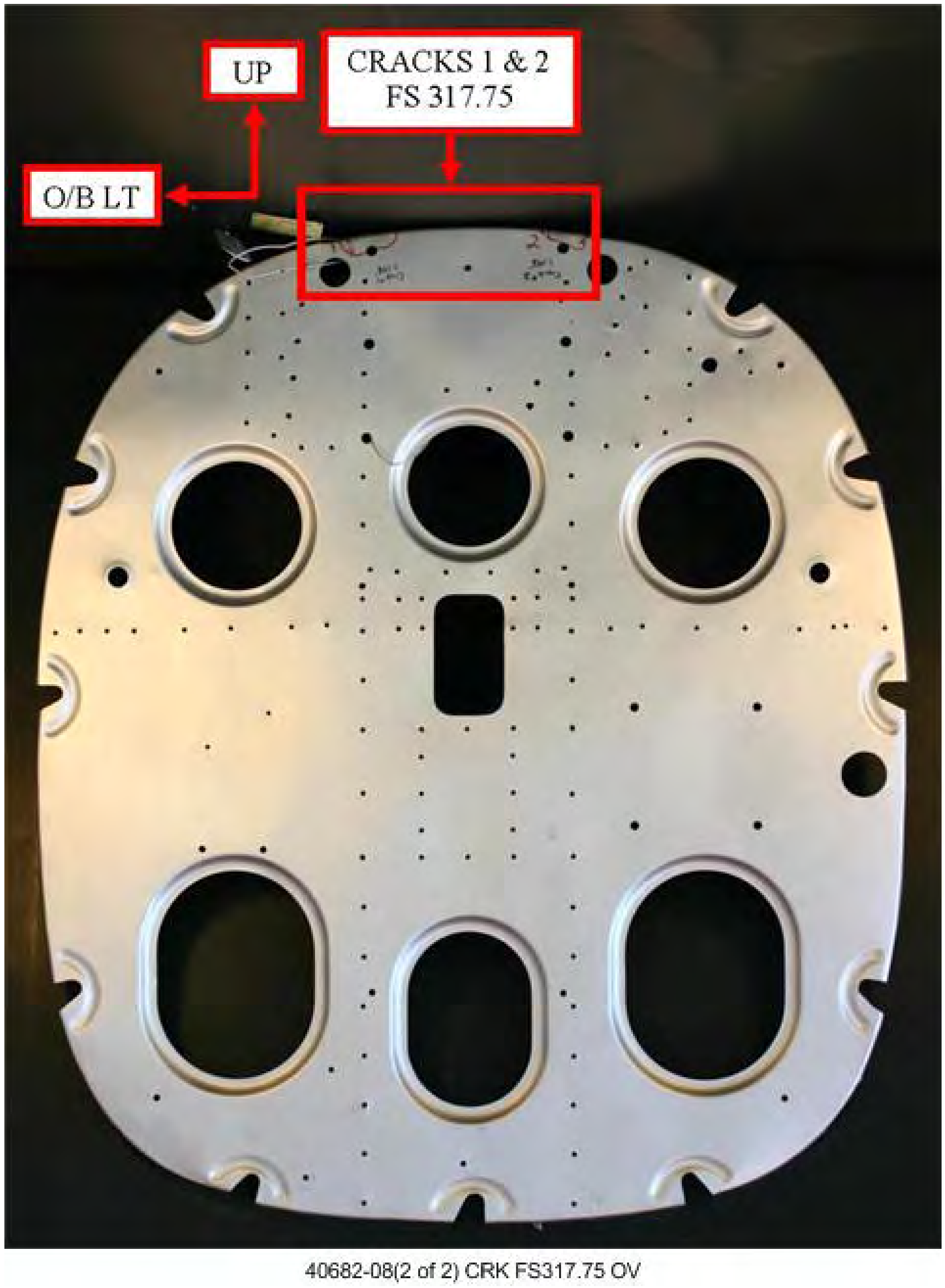


Figure 528. Location of Two Cracks on the Aft Fuselage Bulkhead Assembly (2 of 2) FS 317.75

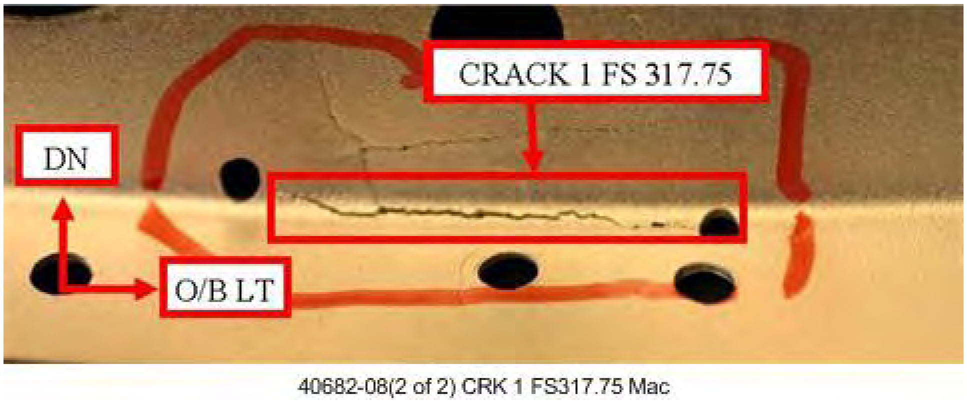


Figure 529. Macroscopic View of Crack 1 on the Aft Fuselage Bulkhead Assembly (2 of 2) FS 317.75

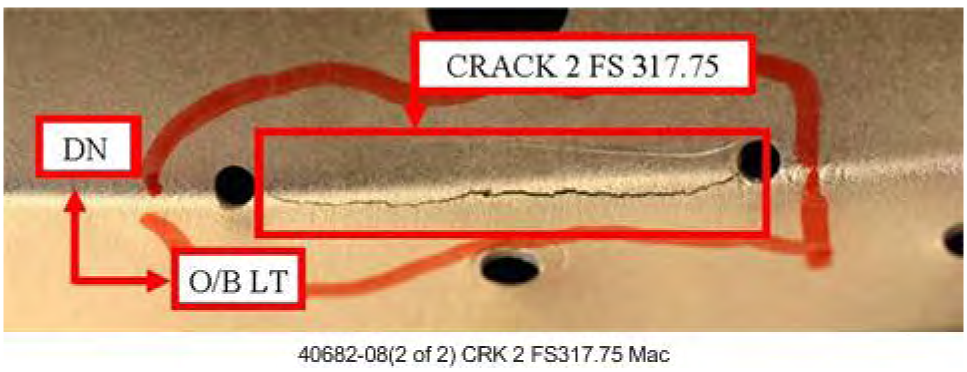


Figure 530. Macroscopic View of Crack 2 on the Aft Fuselage Bulkhead Assembly (2 of 2) FS 317.75

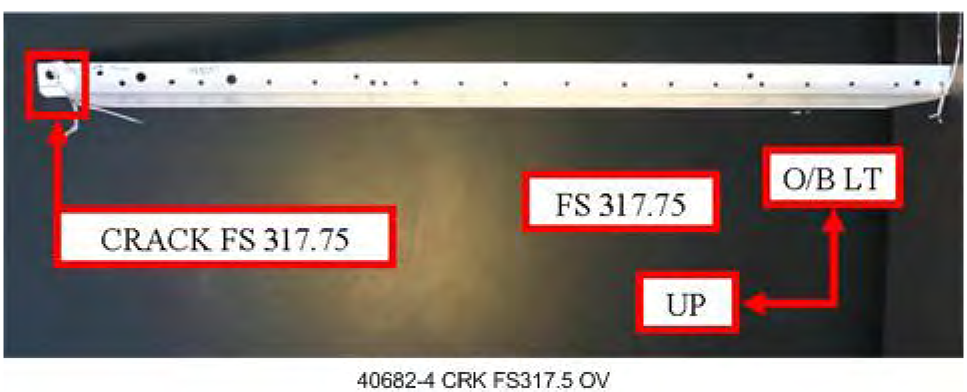


Figure 531. Location of a Crack on the Aft Fuselage Bulkhead Assembly FS 317.75

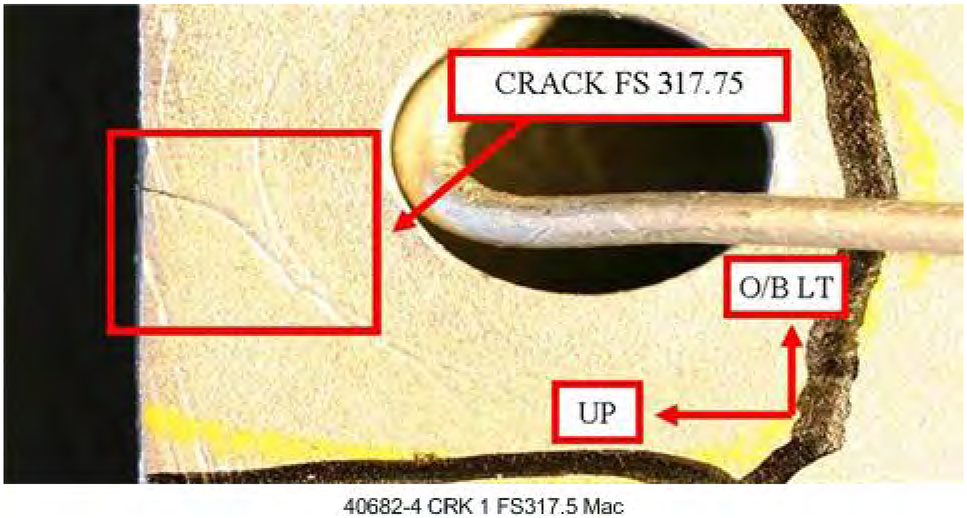


Figure 532. Macroscopic View of a Crack on the Aft Fuselage Bulkhead Assembly FS 317.75

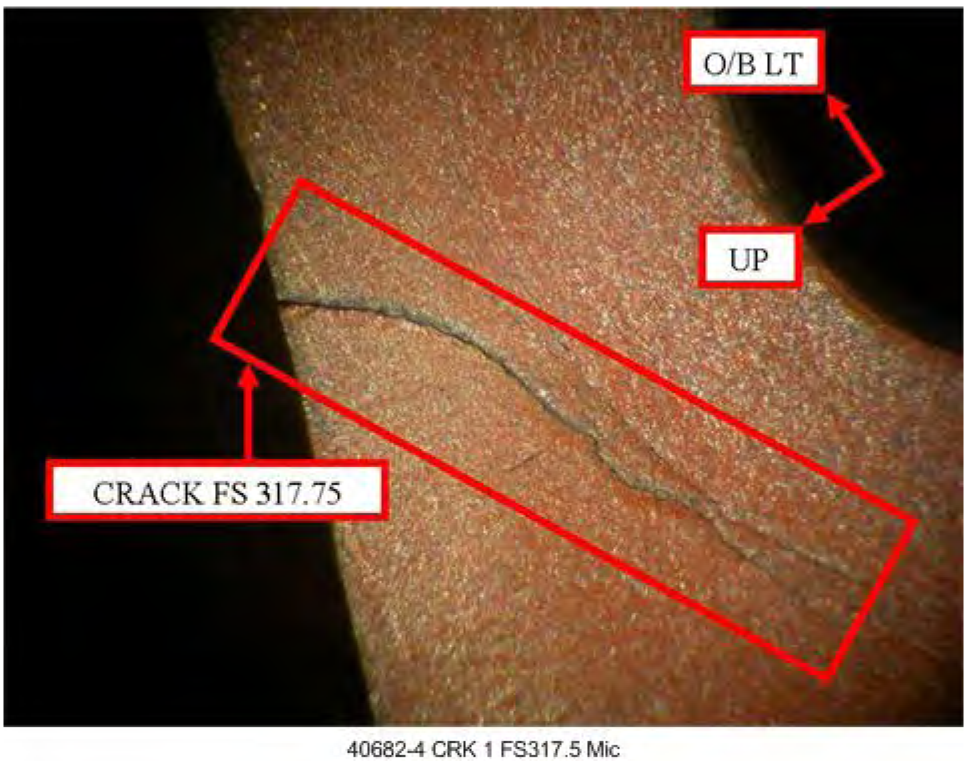


Figure 533. Microscopic View of a Crack Aft Fuselage Bulkhead Assembly FS 317.75

3.4.2.3.6 Fuselage FS 318 Through FS 379.

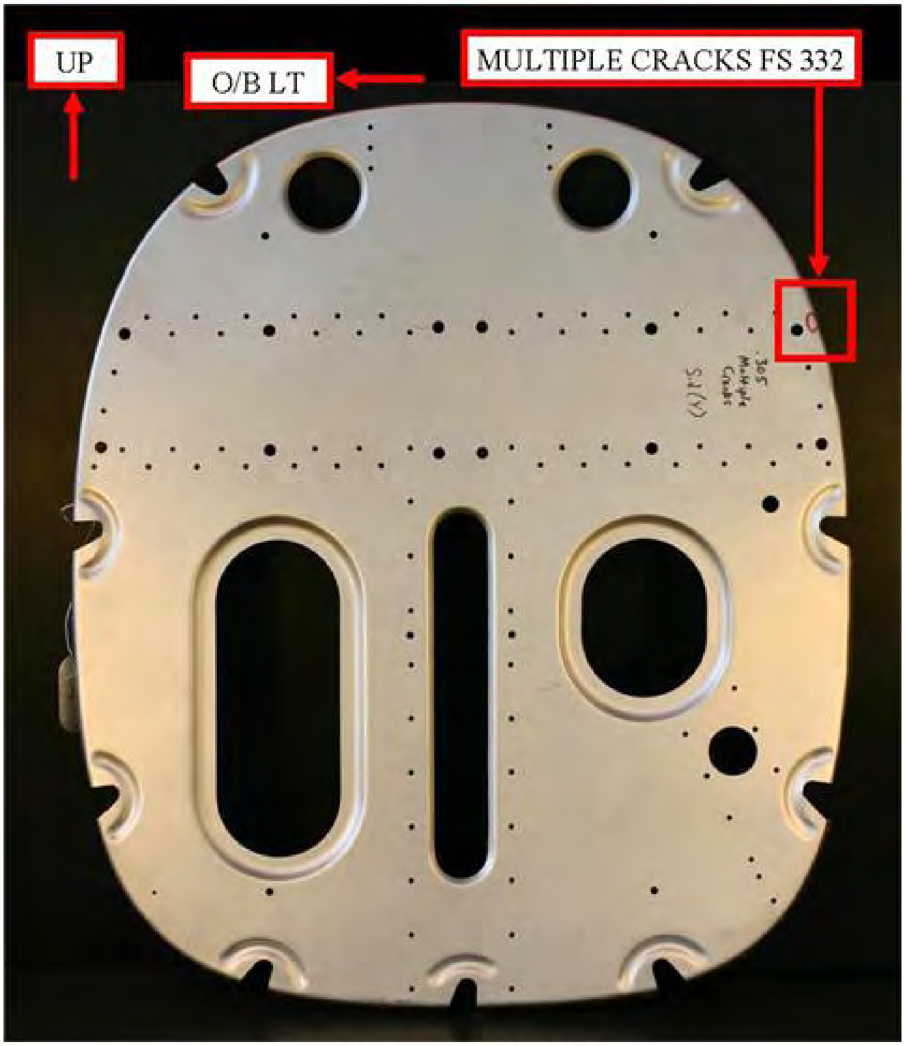
Detailed characterization of each defect found on the fuselage from FS 318 to FS 379 is shown in table 40.

Table 40. Inspection Results From FS 318 Through FS 379

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Fuselage bulkhead assembly, figure 535	Multiple cracks	FS 332	0.305 inch	Crack indication	Bend radii	536 537
Bulkhead assembly, figure 538	Multiple cracks	FS 342.25	0.437 inch	Crack indication	Surface crack	539 540
Aft body lower channel, figure 541	Crack	FS 342.25	0.906 inch	Not inspected[1]	Fatigue	542 543
	Crack	FS 342.25	0.931 inch	Not inspected[1]	Fatigue	544 545
	Crack	FS 347.5	0.337 inch	Crack indication	Surface crack	546
	Crack	FS 348	1.044 inches	Not inspected[1]	Fatigue	547 548
Bulkhead assembly, figure 549	Gouge	FS 342.25	1.92 inches by 0.06 inch	No indication	Gouge 8% thickness loss	550
	Crack	FS 342.25	0.058 inch	Crack indication	Surface crack	550 551
Fuselage bulkhead assembly, figures 552 and 553	Crack	FS 352	0.105 inch	Crack indication	Surface crack	554
	Crack	FS 352	0.189 inch	Crack indication	Surface crack	555 556
	Crack	FS 352	0.9875 inch	Crack indication	Surface crack	557 558
	Crack	FS 352	0.1095 inch	Crack indication	Hole crack	559 560
	Crack	FS 352	0.14 inch	Crack indication	Hole crack	561 562

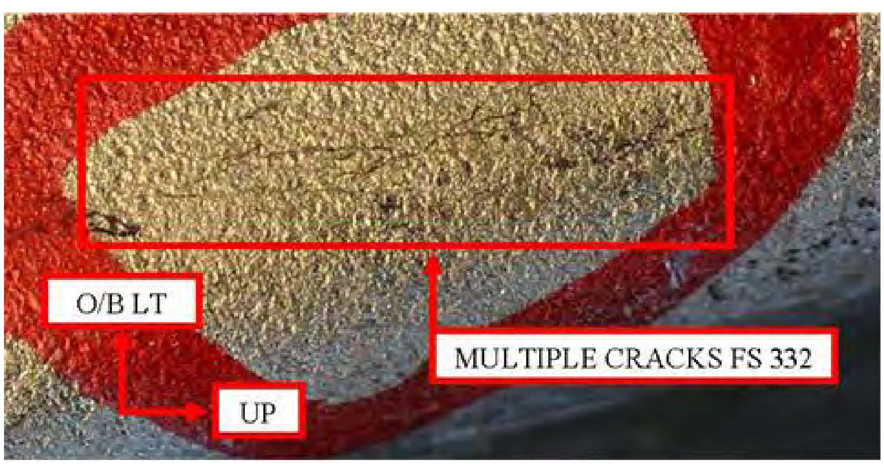
[1] Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 534 shows the location of multiple part-through cracks on the fuselage bulkhead assembly, part number 40655-00. These cracks, which are shown macroscopically in figure 535 had a maximum length of 0.305 inch and were located at FS 332. A microscopic view of these cracks is shown in figure 536. The location of multiple part-through cracks on the bulkhead assembly, part number 40639-004, is shown in figure 537. These cracks had a maximum length of 0.437 inch and were located at FS 342.25. A macroscopic view of these cracks is shown in figure 538, and a microscopic view is shown in figure 539.



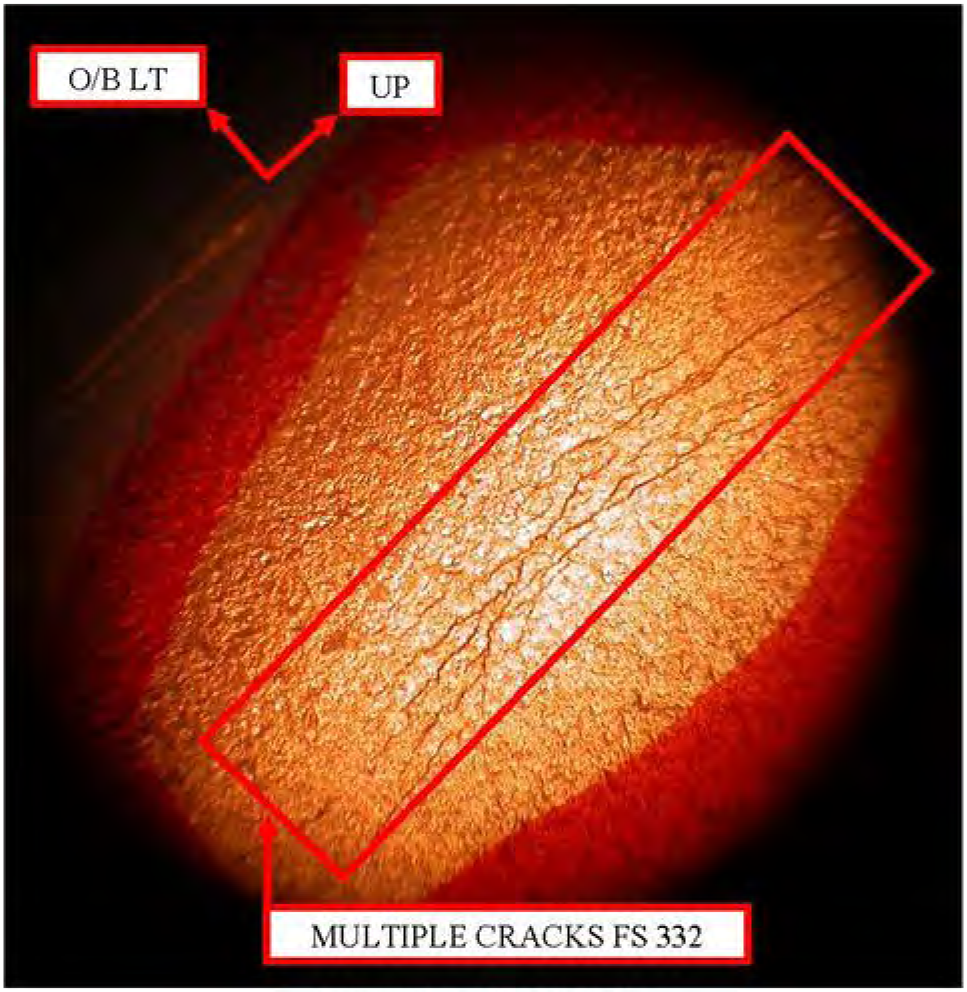
40655-00 (1 of 5) CRK FS 332 OV

Figure 534. Location of Cracks on the Fuselage Bulkhead Assembly FS 332



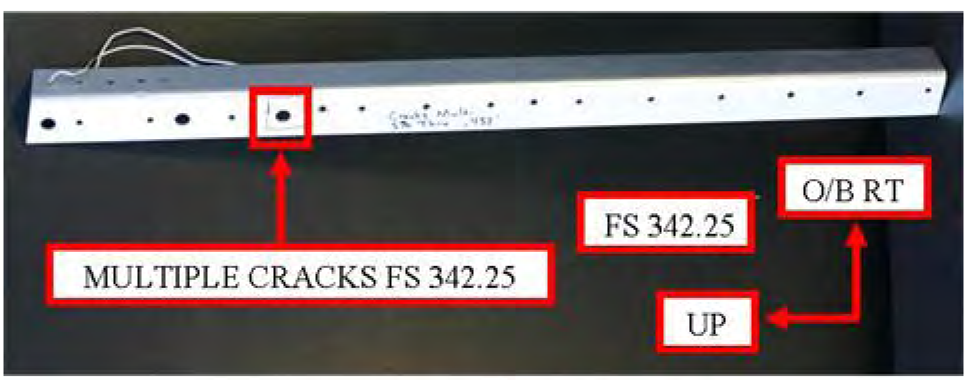
40655-00 (1 of 5) CRK FS 332 Mac

Figure 535. Macroscopic View of Cracks on the Fuselage Bulkhead Assembly FS 332



40655-00 (1 of 5) CRK FS 332 Mic

Figure 536. Microscopic View of Cracks on the Fuselage Bulkhead Assembly FS 332



40639-04 CRK FS342.25 OV

Figure 537. Location of Multiple Cracks on the Bulkhead Assembly FS 342.25

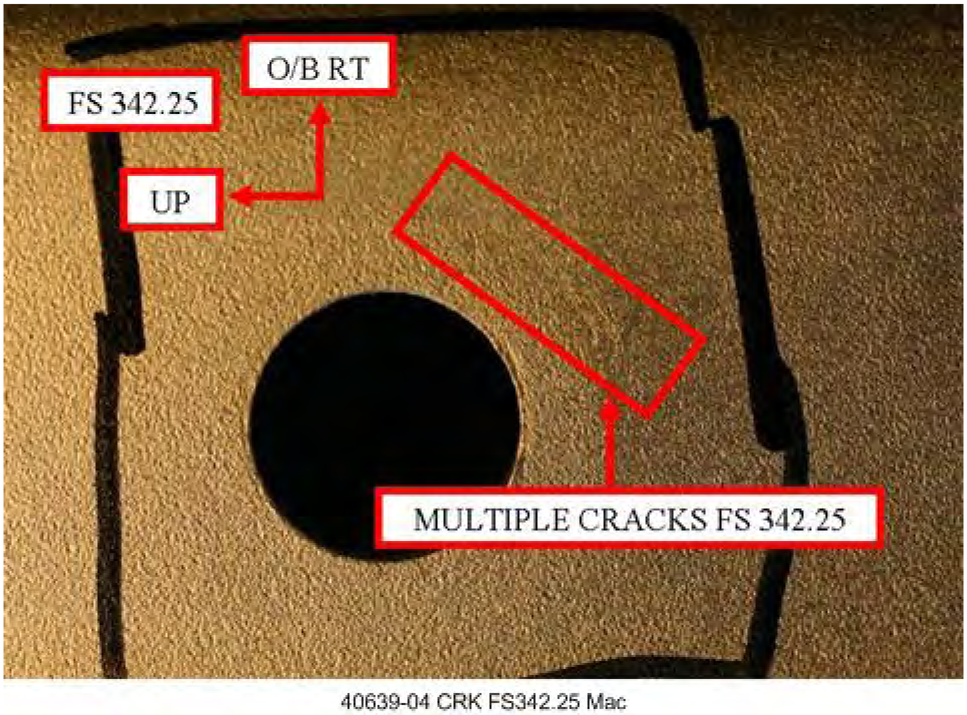


Figure 538. Macroscopic View of Multiple Cracks on the Bulkhead Assembly FS 342.25

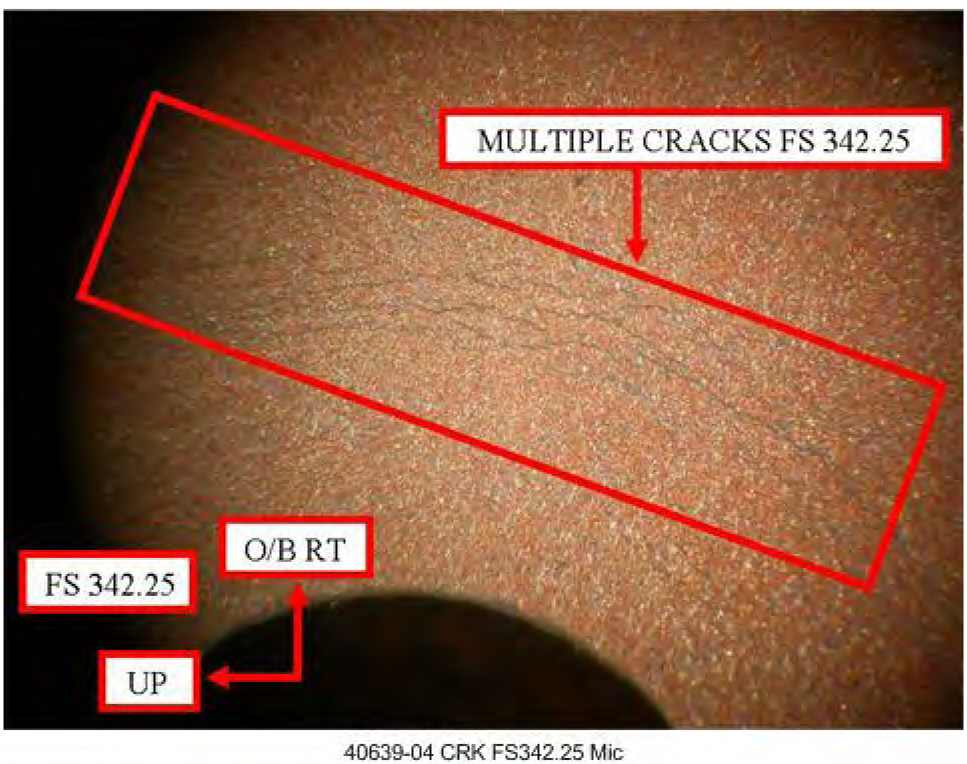


Figure 539. Microscopic View of Multiple Cracks on the Bulkhead Assembly FS 342.25

The location of four cracks found on the aft body lower channel, part number 40792-00, is shown in figure 540. A macroscopic view of crack 1, which was located at FS 342.25 and measured 0.906 inch, is shown in figure 541, and the fracture face is shown in figure 542. Figure 543 shows a macroscopic view of crack 2, which measures 0.931 inch in length and is located at FS 342.25. The fracture face of crack 2 is shown in figure 544. Crack 3, which measured 0.337 inch and was located at FS 347.5, is shown macroscopically in figure 545. A macroscopic view of crack 4, which measured 1.044 inches and was located at FS 348, is shown in figure 546, and figure 547 shows the fracture face for crack 4. From the fracture face photographs for cracks 1, 2, and 4, it was determined that these cracks were caused by fatigue.

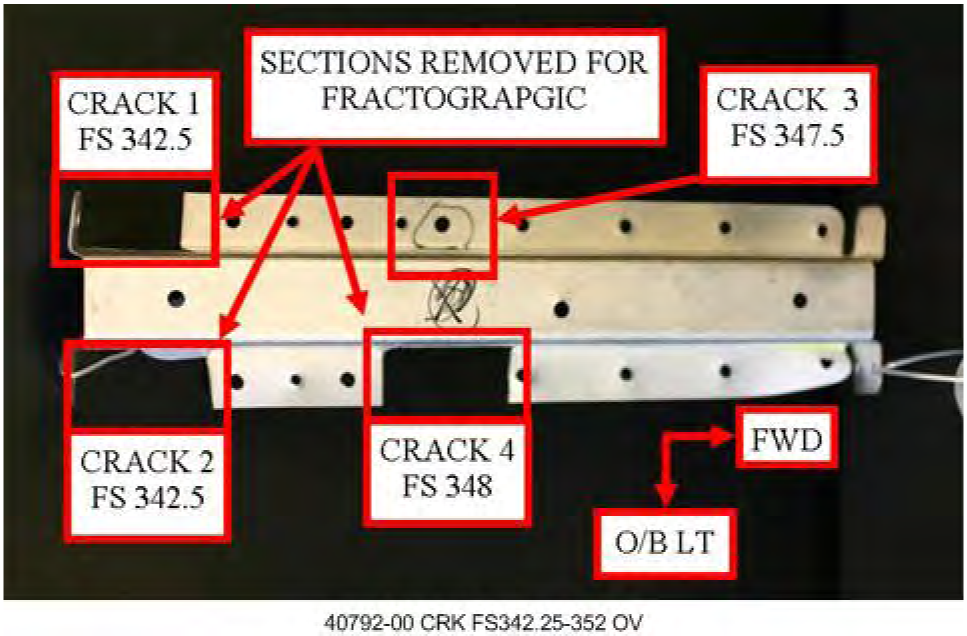


Figure 540. Location of Four Cracks on the Aft Body Lower Channel FS 342.25 Through FS 352

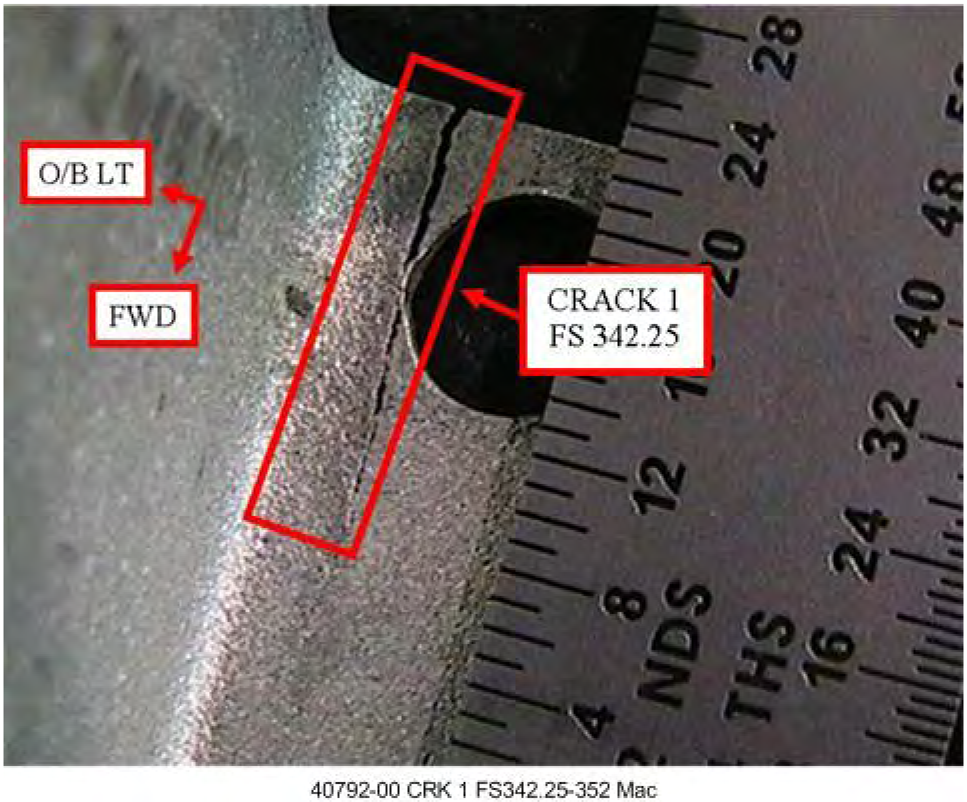


Figure 541. Macroscopic View of Crack 1 on the Aft Body Lower Channel FS 342.25

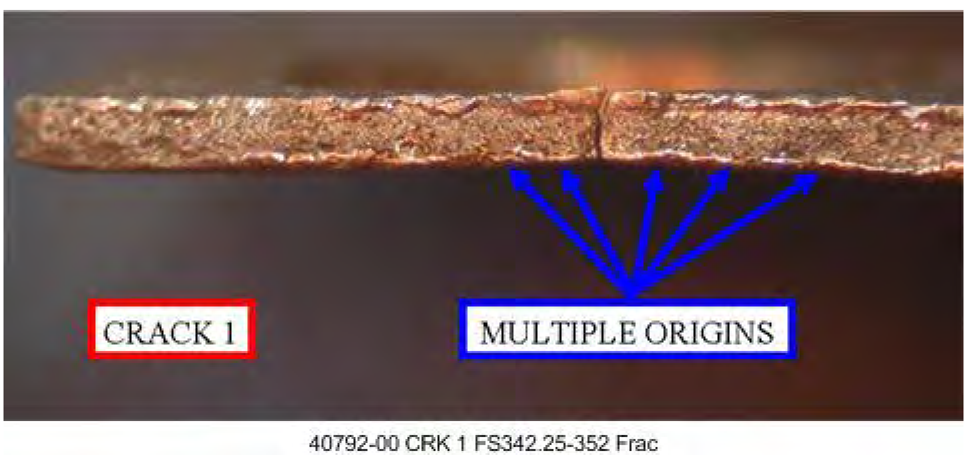


Figure 542. Fracture Face of Crack 1 on the Aft Body Lower Channel FS 342.25

Figure 543. Macroscopic View of Crack 2 on the Aft Body Lower Channel FS 342.25

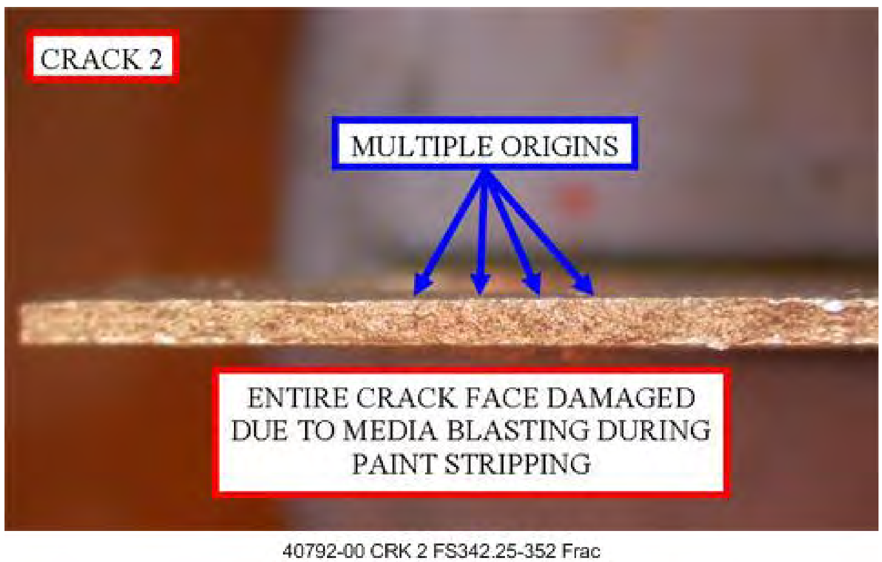


Figure 544. Fracture Face of Crack 2 on the Aft Body Lower Channel FS 342.25

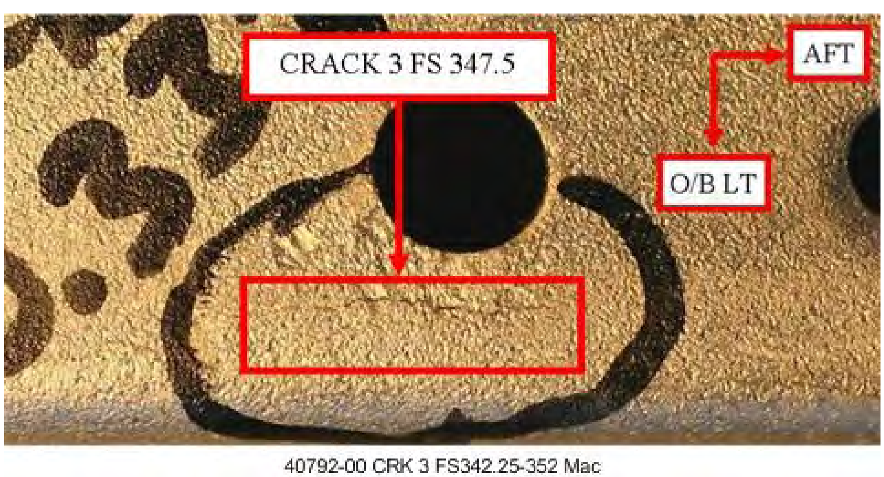


Figure 545. Macroscopic View of Crack 3 on the Aft Body Lower Channel FS 347.5

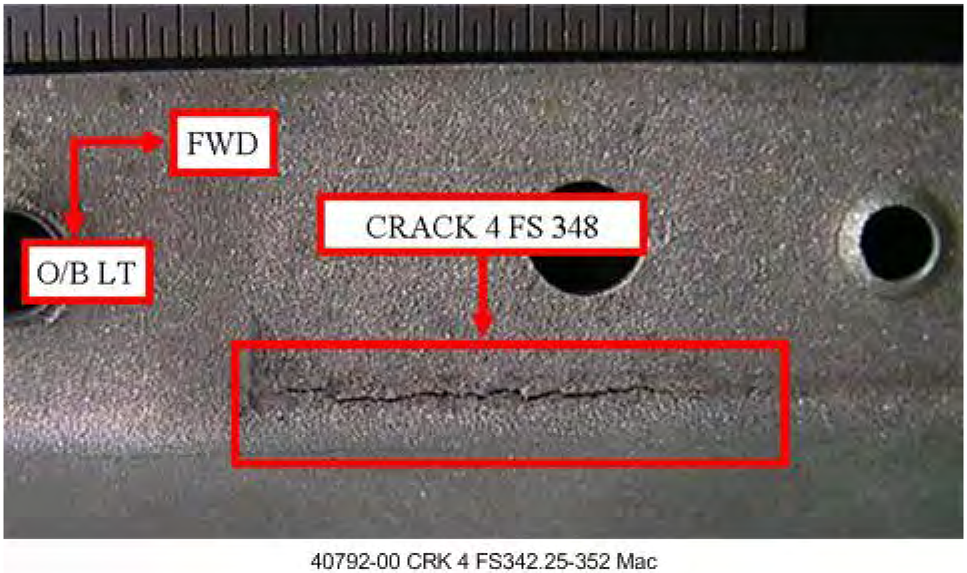


Figure 546. Macroscopic View of Crack 4 on the Aft Body Lower Channel FS 348

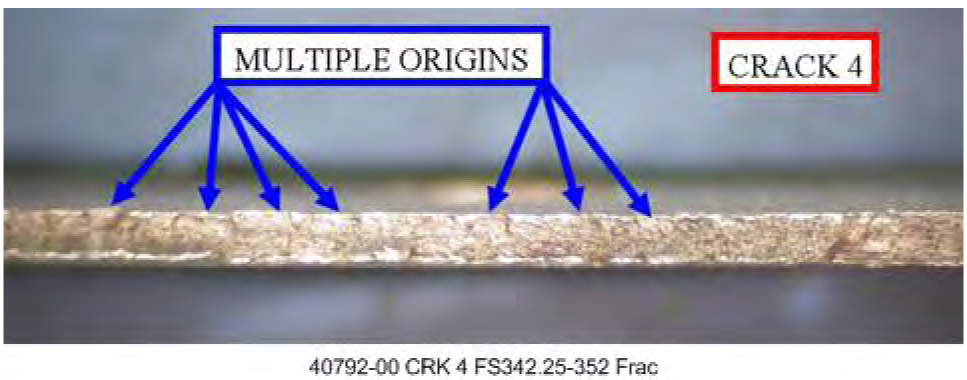


Figure 547. Fracture Face of Crack 4 on the Aft Body Lower Channel FS 348

The location of a gouge and a crack on the bulkhead assembly, part number 40639-03, at FS 342.25, is shown in figure 548. A macroscopic view of the 0.12-square-inch gouge and 0.058-inch crack is shown in figure 549. This gouge caused a localized thickness loss of 8%. A microscopic view of the crack is shown in figure 550.

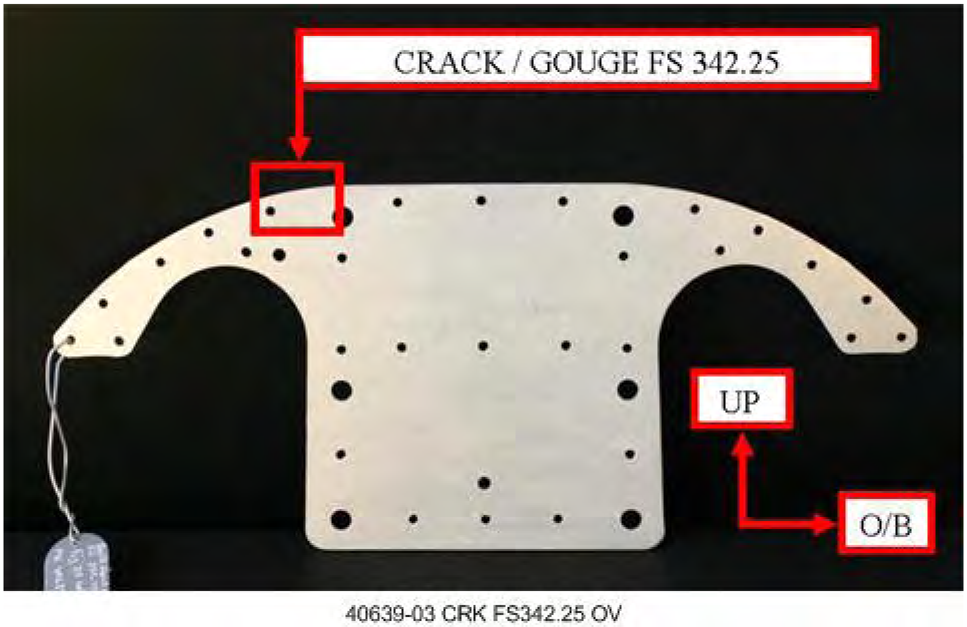


Figure 548. Location of Gouge and Crack on the Bulkhead Assembly FS 342

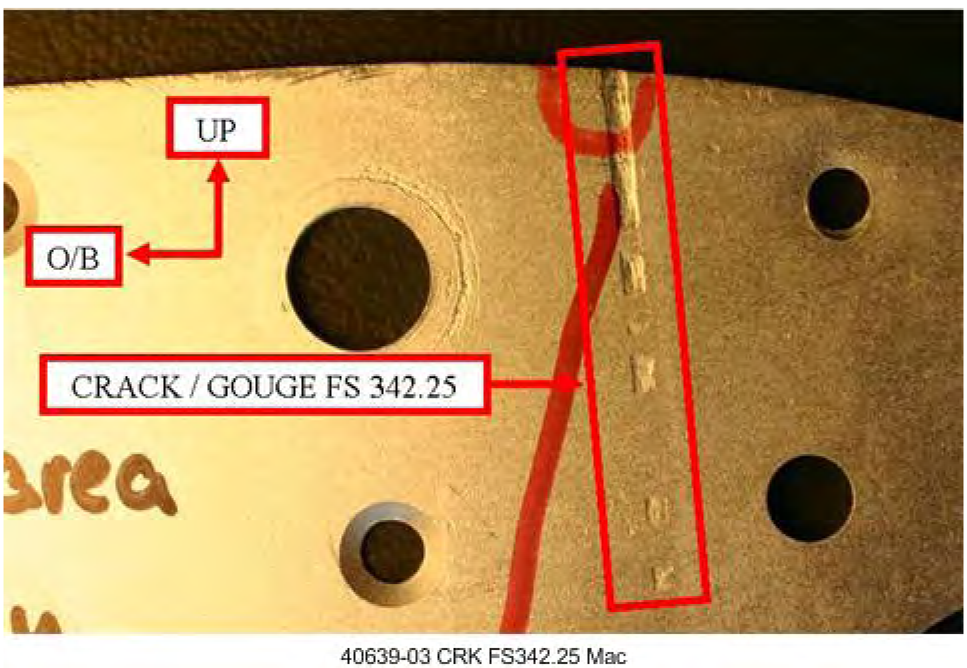


Figure 549. Macroscopic View of Crack and Gouge on the Bulkhead Assembly FS 342.25

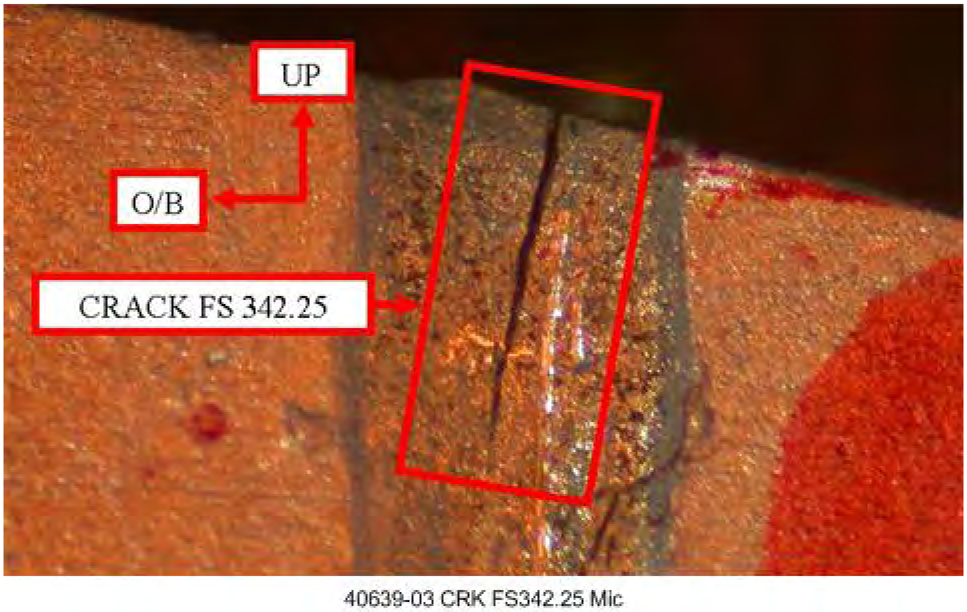
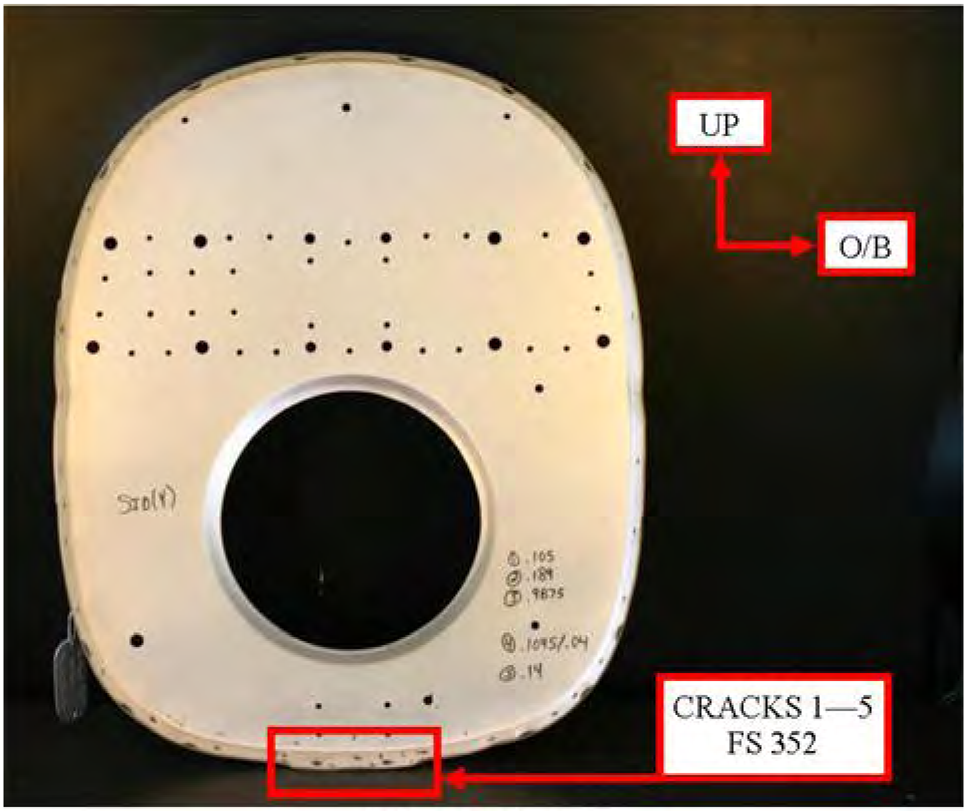


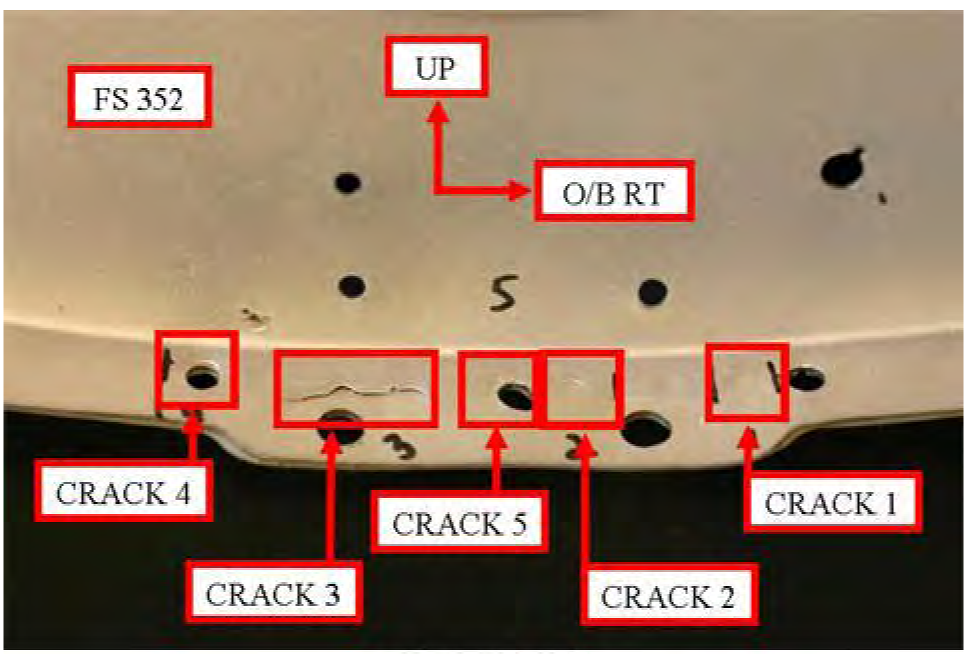
Figure 550. Microscopic View of Crack on the Bulkhead Assembly FS 342.25

An overview of the location of five cracks on the fuselage bulkhead assembly, part number 40638-00, at FS 352, is shown in figure 551, and a detailed overview is shown in figure 552. A macroscopic view of crack 1, which measured 0.105 inch, is shown in figure 553. A macroscopic view of crack 2, which measured 0.189 inch in length, is shown in figure 554. A microscopic view of crack 2 is shown in figure 555. Figure 556 shows a macroscopic view of crack 3, which measured 0.9875 inch, and figure 557 shows a microscopic view of this crack. Crack 4, which measured 0.1095 inch in length, is shown macroscopically in figure 558 and microscopically in figure 559. Figure 560 shows a macroscopic view of crack 5, and figure 561 shows a microscopic view of this 0.14-inch-long crack.



40638-00 CRK FS352 OV

Figure 551. Overview of Five Cracks on the Fuselage Bulkhead Assembly FS 352



40638-00 CRK FS352 Mac

Figure 552. Location of Five Cracks on the Fuselage Bulkhead Assembly FS 352

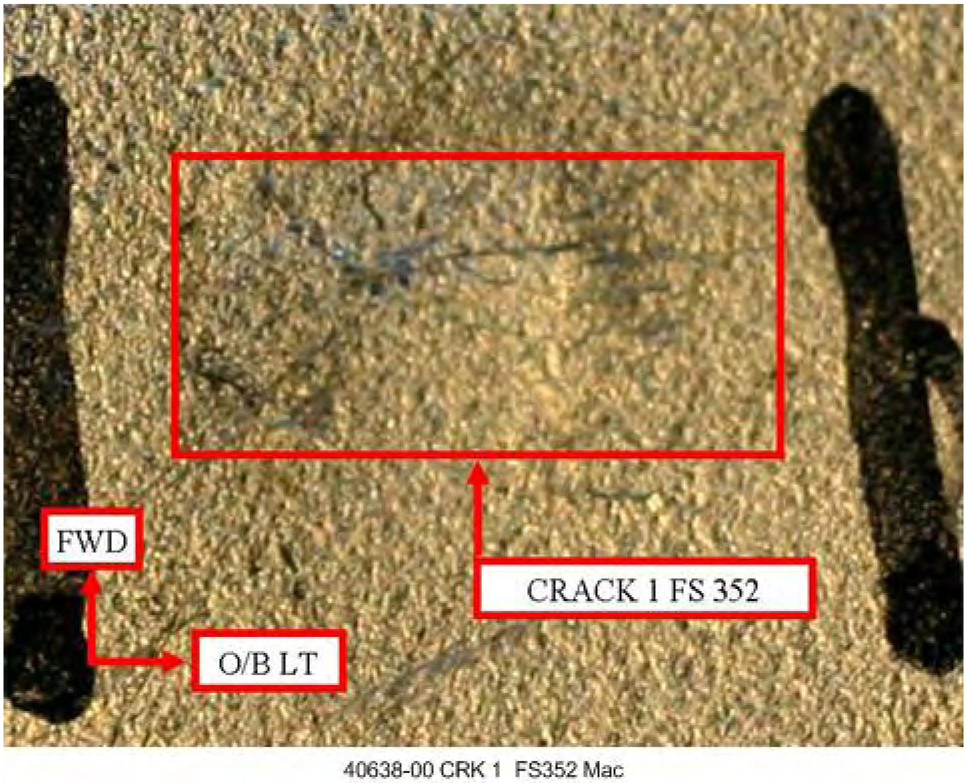


Figure 553. Macroscopic View of Crack 1 on the Fuselage Bulkhead Assembly FS 352

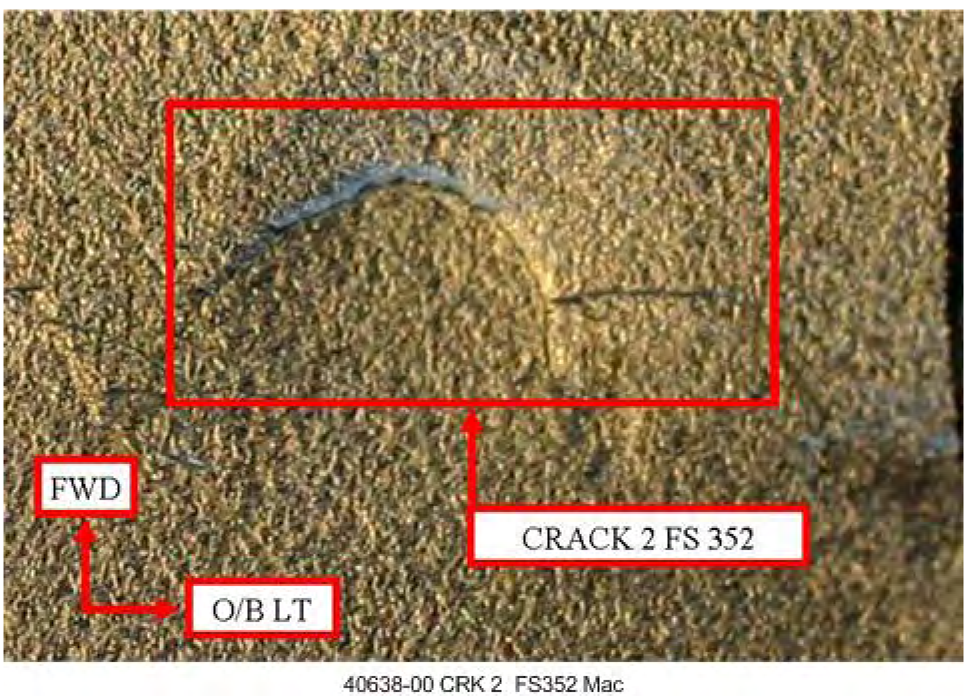


Figure 554. Macroscopic View of Crack 2 on the Fuselage Bulkhead Assembly FS 352

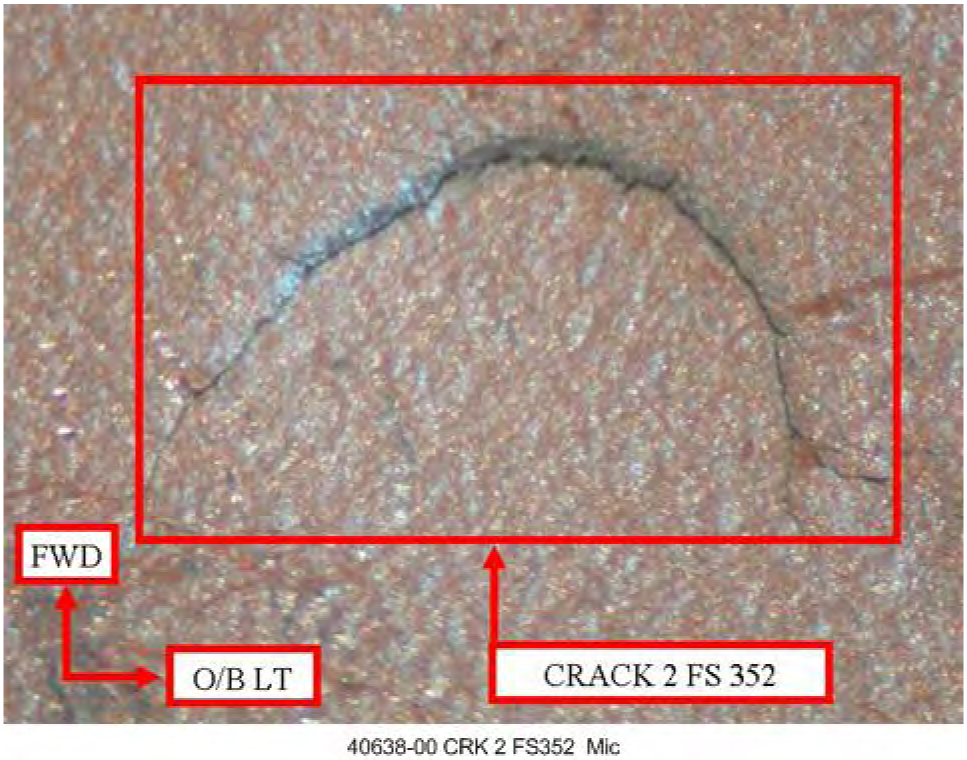


Figure 555. Microscopic View of Crack 2 on the Fuselage Bulkhead Assembly FS 352

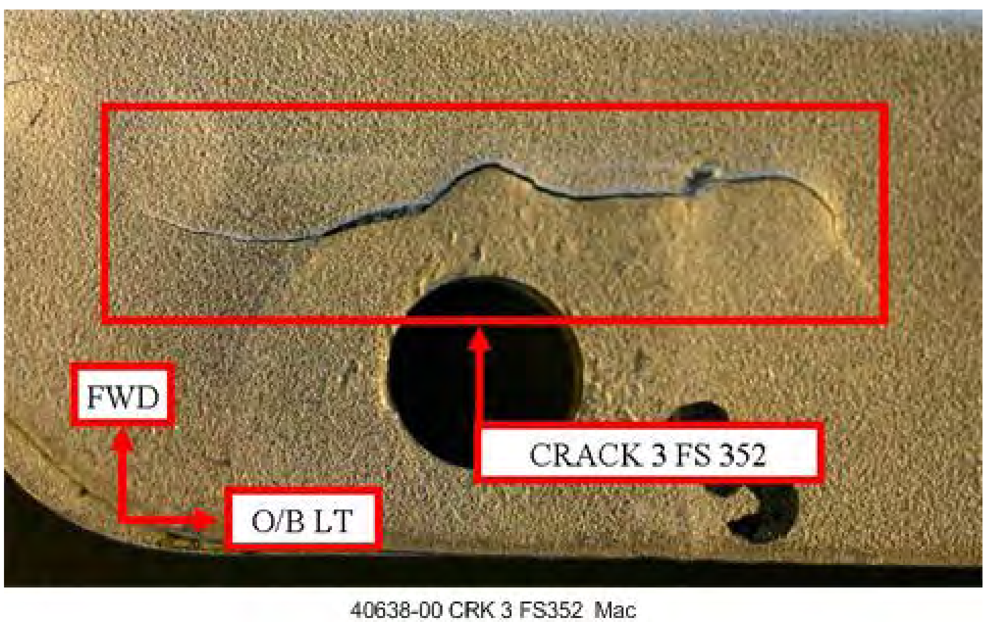


Figure 556. Macroscopic View of Crack 3 on the Fuselage Bulkhead Assembly FS 352

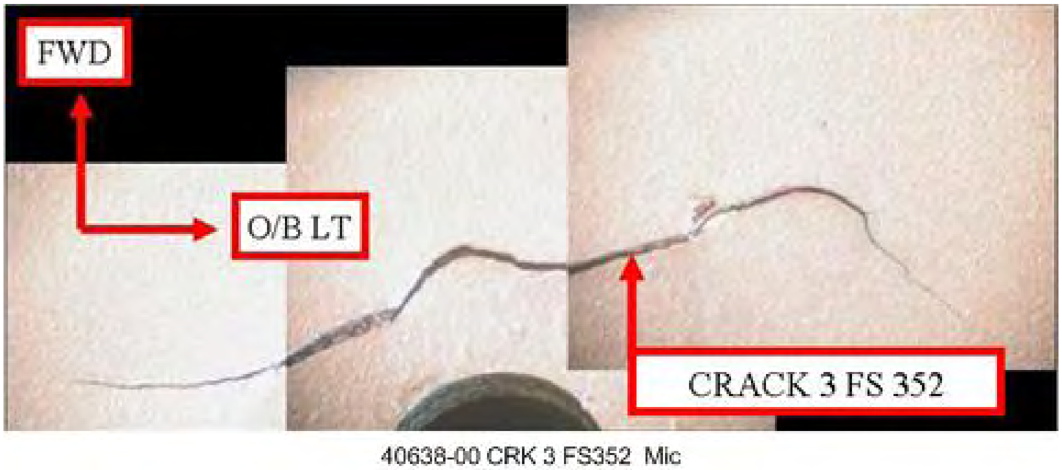


Figure 557. Microscopic View of Crack 3 on the Fuselage Bulkhead Assembly FS 352

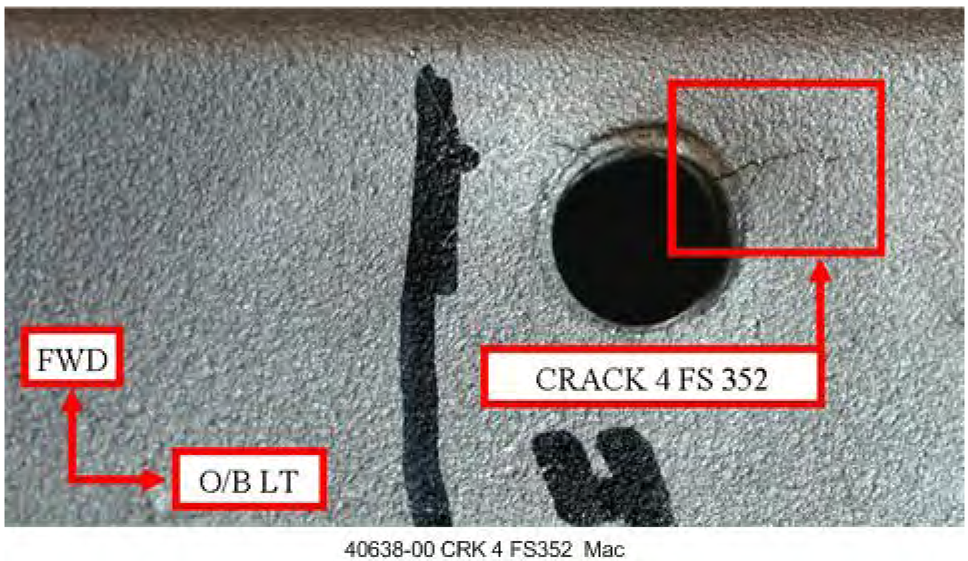


Figure 558. Macroscopic View of Crack 4 on the Fuselage Bulkhead Assembly FS 352

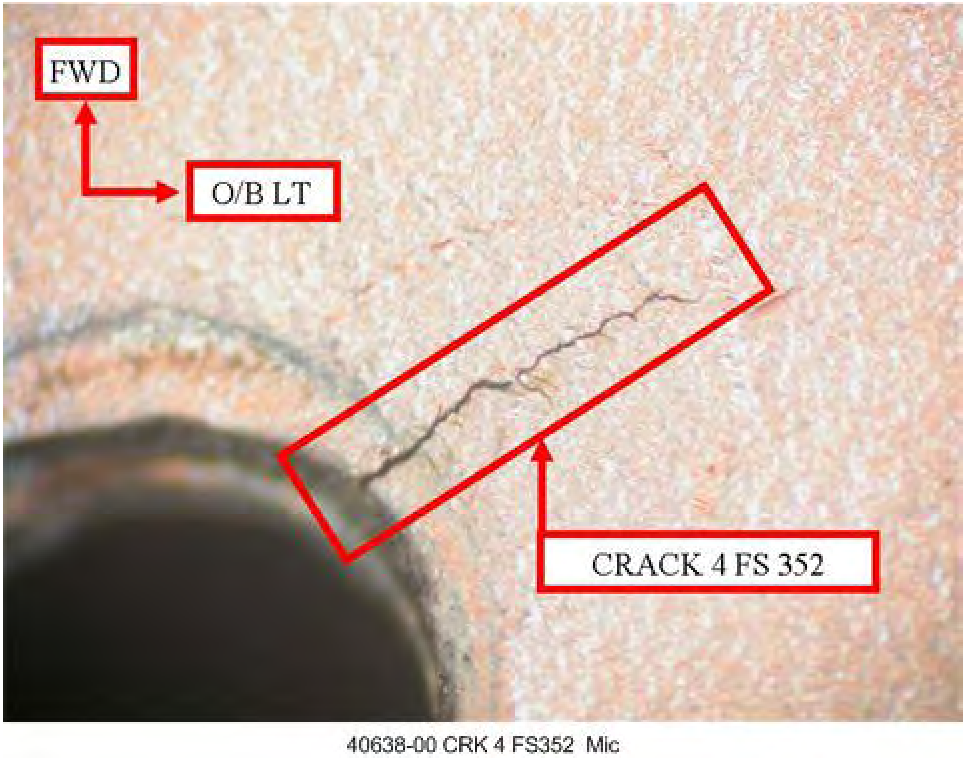


Figure 559. Microscopic View of Crack 4 on the Fuselage Bulkhead Assembly FS 352

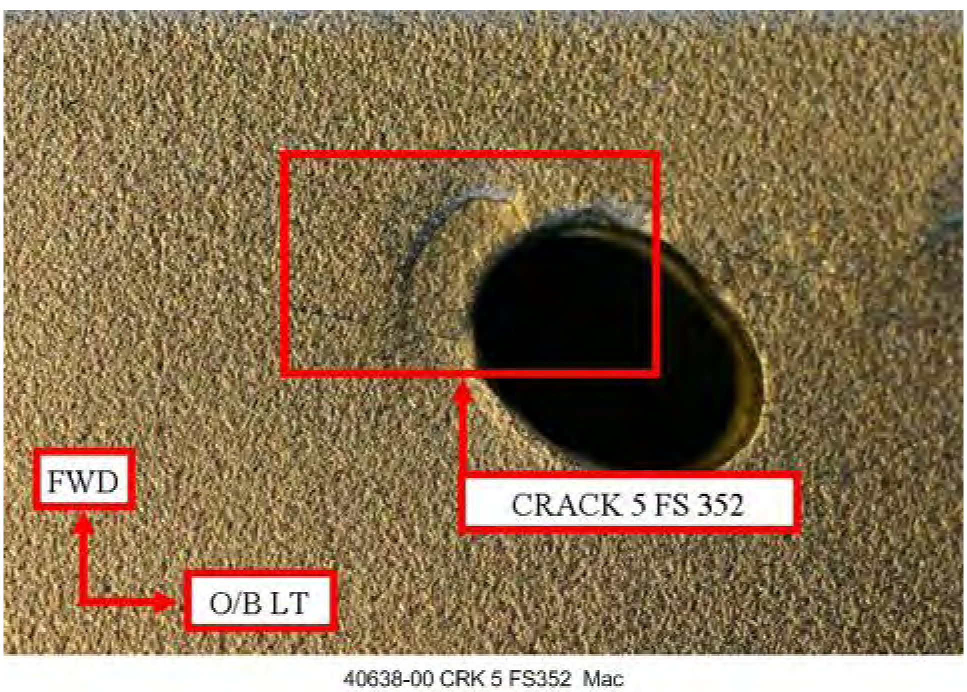


Figure 560. Macroscopic View of Crack 5 on the Fuselage Bulkhead Assembly FS 352

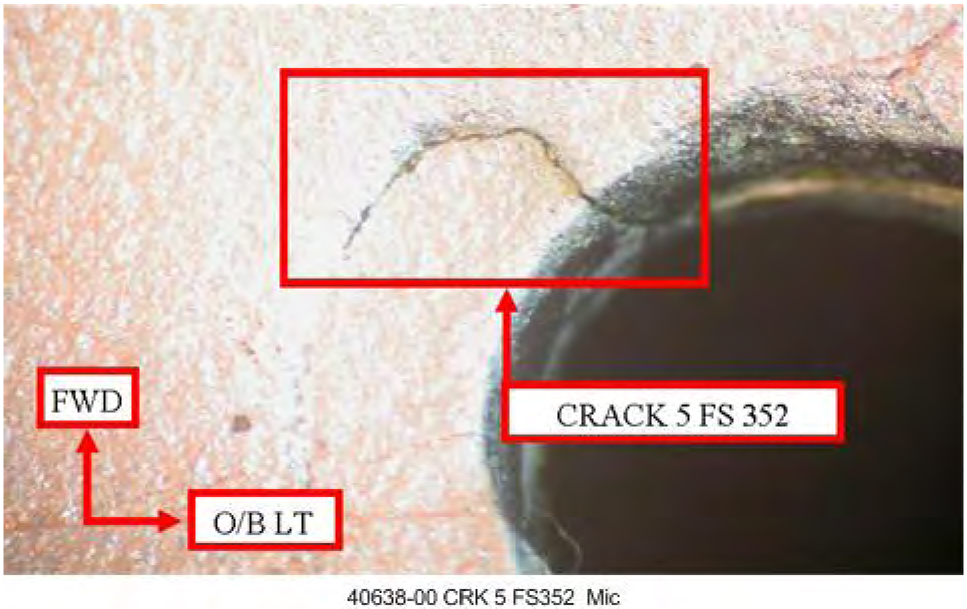
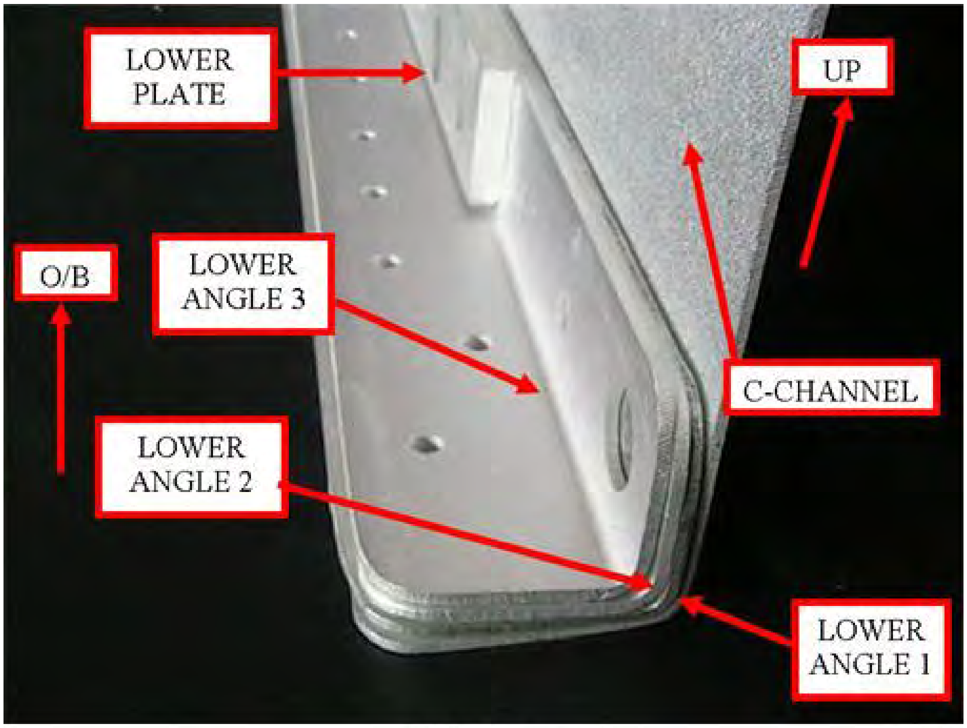


Figure 561. Microscopic View of Crack 5 on the Fuselage Bulkhead Assembly FS 352

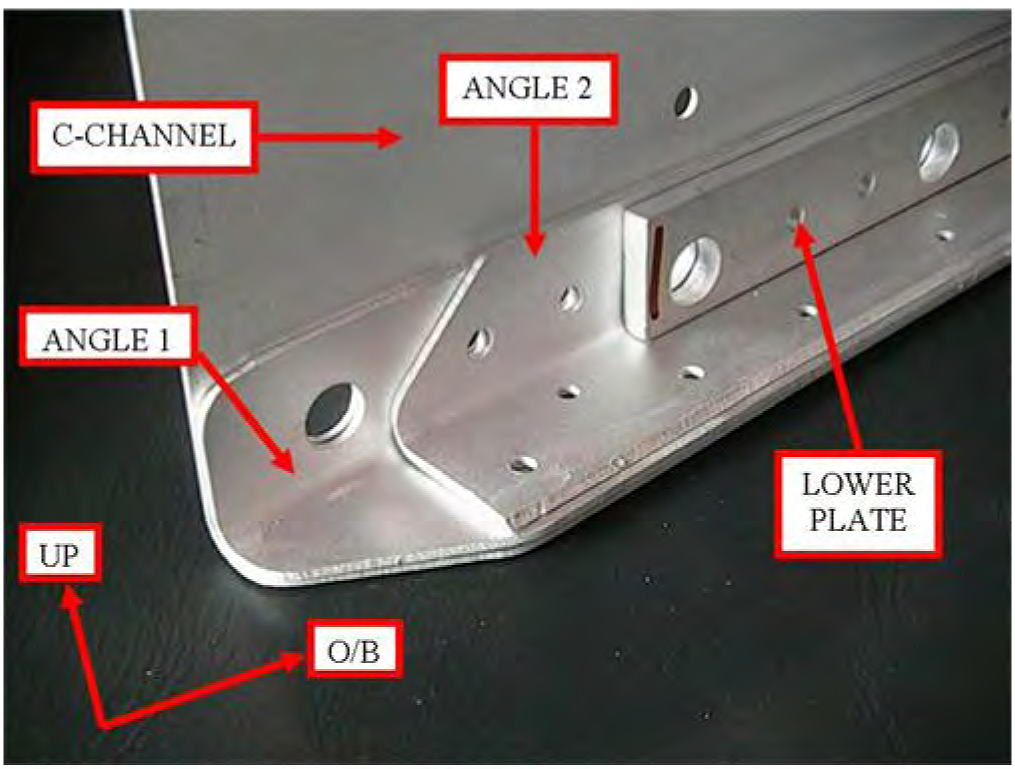
3.4.2.4 Horizontal Stabilizer.

Figure 562 shows the structural stackup for the horizontal stabilizer front spar, and the structural stackup for the horizontal stabilizer rear spar is shown in figure 563. Table 41 shows detailed characterizations for the 12 defects identified and characterized during the teardown evaluation on the horizontal stabilizer. Of the 136 parts inspected on the horizontal stabilizer, only 12 defects were identified. Four areas of corrosion, seven cracks, and one area of other damage were reported on the horizontal stabilizer. Most cracks occurred on ribs or the leading-edge skin assemblies. The corrosion was categorized as light scattered corrosion and occurred on the components of the spar cap assemblies.



HOR FS Stackup

Figure 562. Structural Stackup of the Horizontal Stabilizer Front Spar



HOR RS Stackup

Figure 563. Structural Stackup of the Horizontal Stabilizer Rear Spar

Table 41. Inspection Results From the Horizontal Stabilizer

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left horizontal stabilizer rear spar lower plate, figure 565	Corrosion	BL 3 through BL 24.5	Scattered over entire part	Corrosion indication	Light corrosion 1.7% thickness loss	103
Right horizontal stabilizer rear spar upper plate, figure 566	Corrosion	BL 3 through BL 24.5	Scattered over entire part	Corrosion indication	Light corrosion 1% thickness loss	103
Right horizontal stabilizer rear spar lower plate, figure 567	Corrosion	BL 3 through BL 24.5	Scattered over entire part	Corrosion indication	Light corrosion 2% thickness loss	103
Right horizontal stabilizer front spar upper plate, figure 568	Corrosion	BL 3.5 through BL 18.75	Scattered over entire part	No indication	Light corrosion less than 1% thickness loss	103
Left horizontal stabilizer rib, figure 569	Crack	BL 11	0.089 inch	Crack indication	Hole crack	570
Right horizontal stabilizer gusset-front spar, figure 571	Crack multiple	BL 12.6	0.026 inch	Crack indication	Bend radii	572 573
Right horizontal stabilizer stringer, figure 574	Damage	BL 16.25	0.723 inch by 0.25 inch	No indication	Damage 100% thickness loss	575
Right horizontal stabilizer leading-edge skin assembly, figure 576	Crack multiple	BL 18.5	0.106 inch	No indication	Hole crack	577

Table 41. Inspection Results From the Horizontal Stabilizer (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Left horizontal stabilizer leading-edge skin, figure 578	Crack	BL 94	0.599 inch	Not inspected[1]	Puncture	578
	Crack	BL 96	0.524 inch	Not inspected[1]	Puncture	580
	Crack	BL 100	0.549 inch	Not inspected[1]	Puncture	581
	Crack	BL 104.75	0.313 inch	Crack indication	Surface crack	582 583 584

[1]Cracks selected for fractographic analysis during the disassembly phase were extracted from the part prior to postdisassembly NDI.

Figure 564 shows the location of light scattered corrosion on the left horizontal stabilizer rear spar lower plate form BL 3 to BL 24.5. This corrosion caused a maximum reduction in thickness of 1.7%. The location of light corrosion, scattered across the surface of the right horizontal stabilizer rear spar upper plate from BL 3 to BL 24.5, is shown in figure 565. This corrosion caused a thickness loss of 1%. Figure 566 shows the location of light scattered corrosion, which caused a reduction in thickness of 2%, on the right horizontal stabilizer rear spar lower plate from BL 3 to BL 24.5. The location of light scattered corrosion on the right horizontal stabilizer front spar upper plate is shown in figure 567. This corrosion was scattered across the surface of the part form BL 3.5 to BL 18.75 and caused a reduction in thickness of less than 1%.

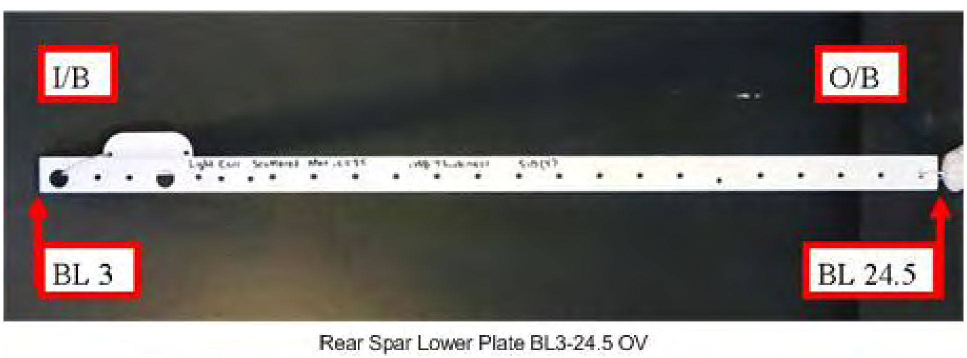
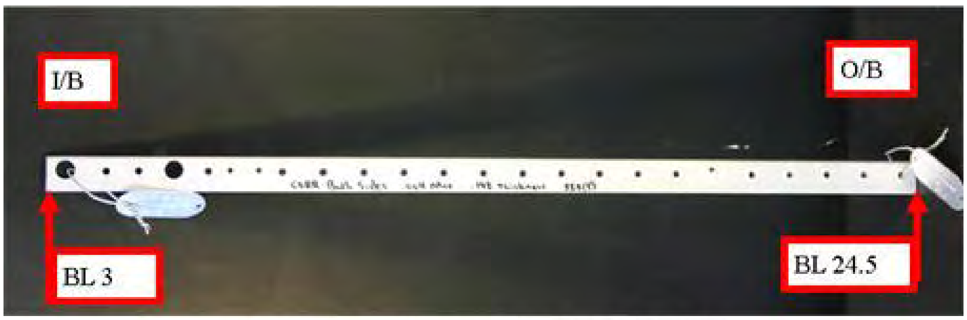


Figure 564. Location of Light Corrosion on the Left Horizontal Stabilizer Rear Spar Lower Plate BL 3 Through BL 24.5

Right Horiz Rear Spar Upper Plate Corr BL 3-24.5 OV

Figure 565. Location of Light Corrosion on the Right Horizontal Stabilizer Rear Spar Upper Plate BL 3 Through BL 24.5



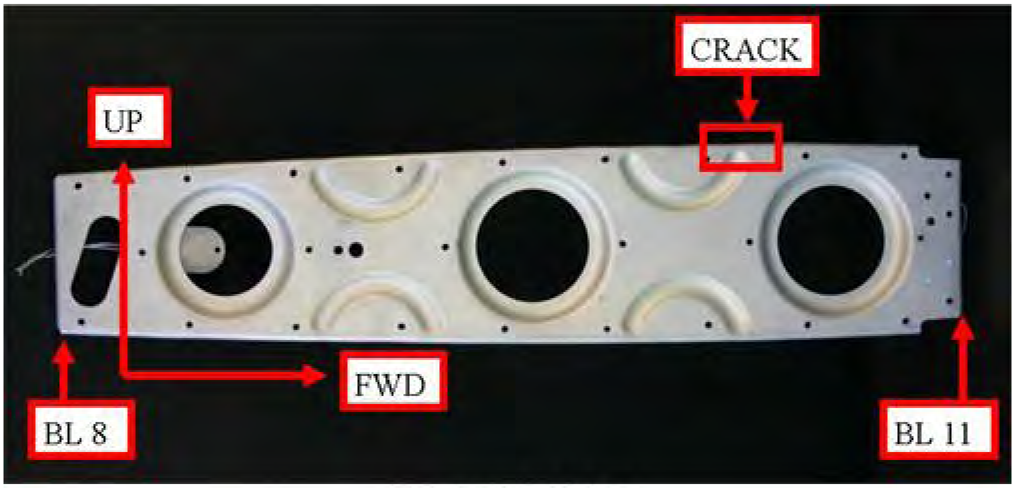
Right Horiz Rear Spar Lower Plate Corr BL 3-24.5 OV

Figure 566. Location of Light Corrosion on the Right Horizontal Stabilizer Rear Spar Lower Plate BL 3 Through BL 24.5

Right Horiz Front Spar Upper Plate Corr BL 35-18.75 OV

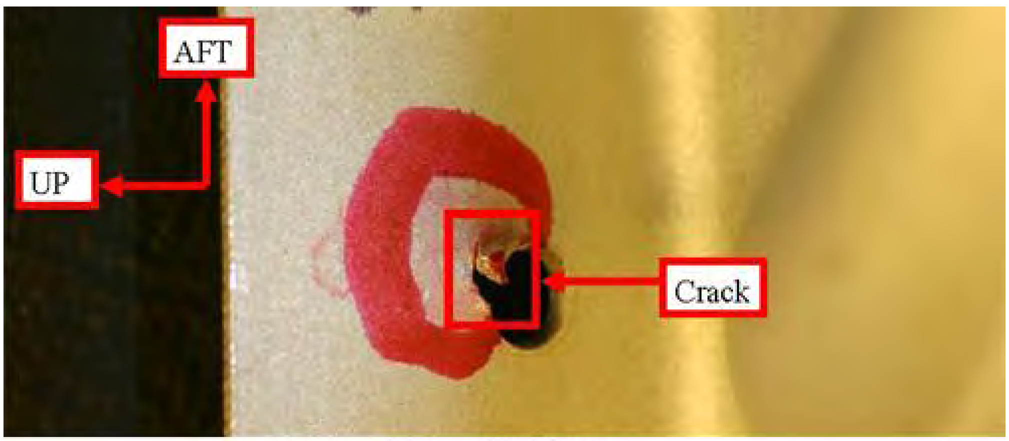
Figure 567. Location of Light Corrosion on the Right Horizontal Stabilizer Front Spar Upper Plate BL 3.5 Through BL 18.75

The location of a crack on the left horizontal stabilizer rib, part number 42270-2, is shown in figure 568. A macroscopic view of this 0.089-inch crack that is located at BL 11 is shown in figure 569. Figure 570 shows the location of a 0.026-inch crack at BL 12.6 on the right horizontal stabilizer gusset-front spar, part number 40170-15. A macroscopic view of this crack is shown in figure 571, and the fluorescent liquid penetrant indication is shown in figure 572.



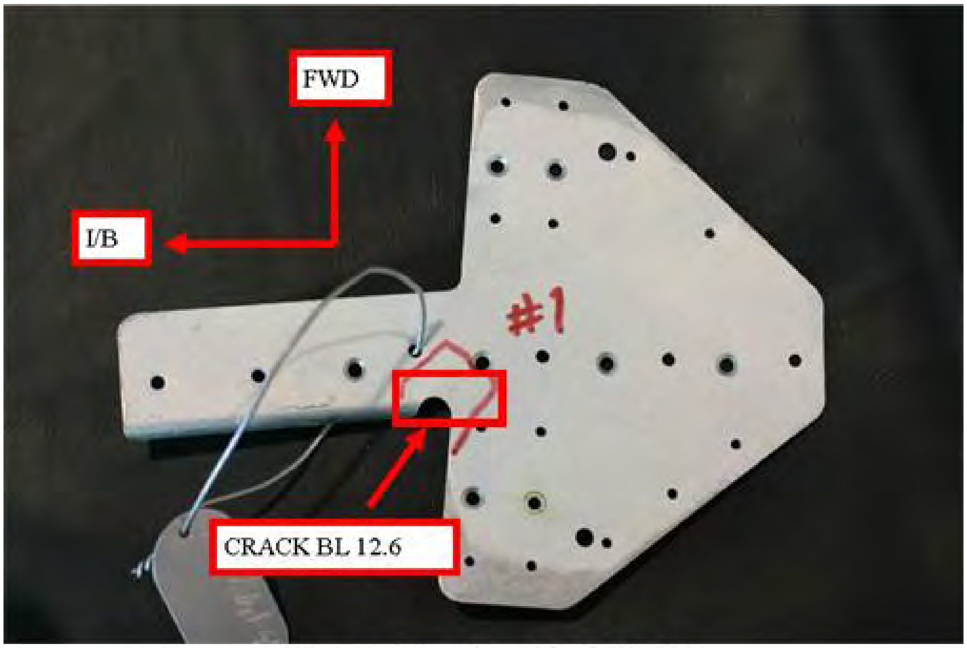
42230-2 CRK BL 8-11 OV

Figure 568. Location of Crack on the Left Horizontal Stabilizer Rib BL 8 Through BL 11



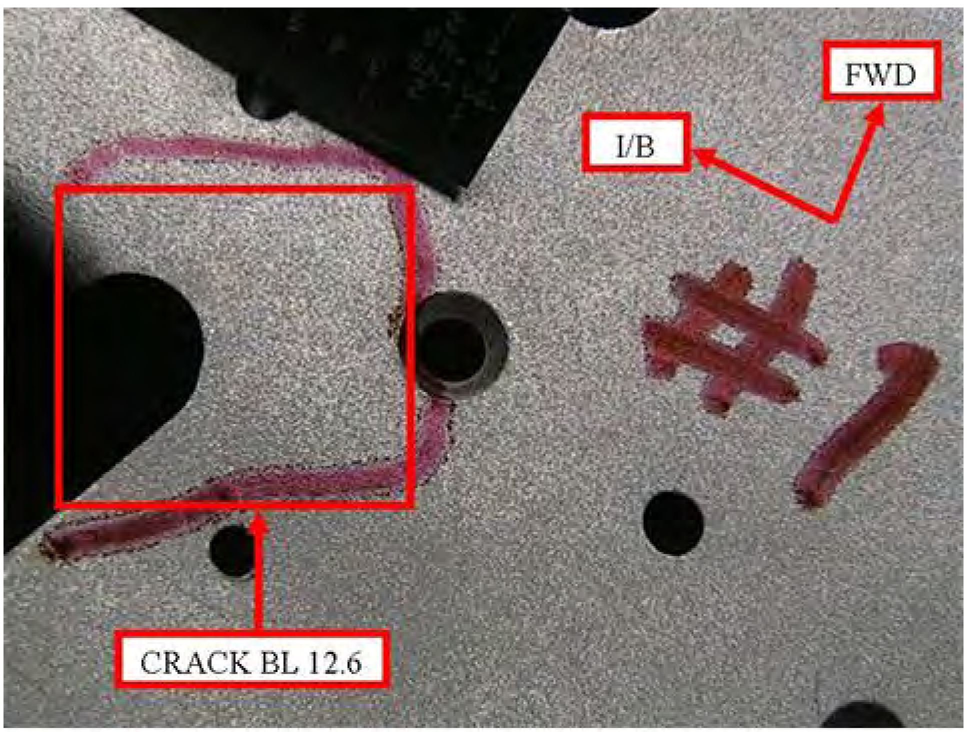
42230-2 CRK BL 8-11 Mac

Figure 569. Macroscopic View of Crack on the Left Horizontal Stabilizer Rib BL 11



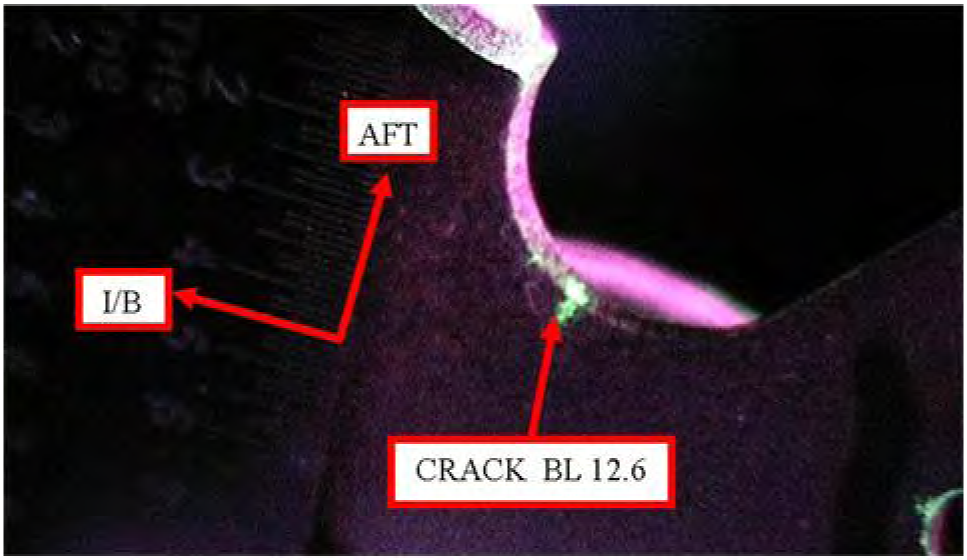
RT Hor FT SP Upper Gusset CRK BL 12.6 OV

Figure 570. Location of Crack on the Right Horizontal Stabilizer Gusset—Front Spar BL 12.6



RT Hor FT SP Upper Gusset CRK BL 12.6 Mac

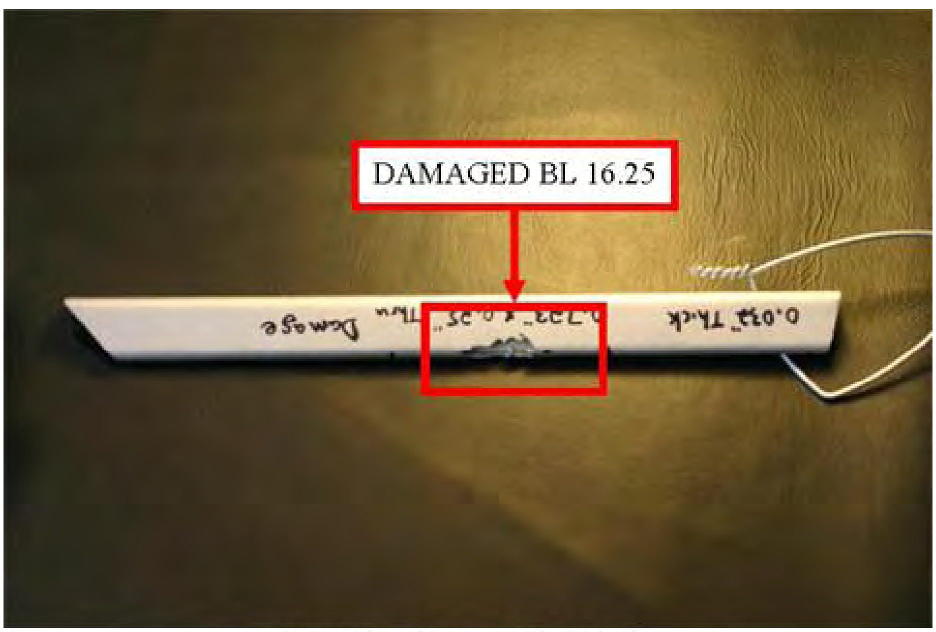
Figure 571. Macroscopic View of Crack on the Right Horizontal Stabilizer Gusset—Front Spar BL 12.6



RT Hor FT SP Upper Gusset CRK BL 12.6 FLP

Figure 572. Fluorescent Liquid Penetrant Indication of Crack on the Right Horizontal Stabilizer Gusset—Front Spar BL 12.6

Figure 573 shows the location of damage on the right horizontal stabilizer stringer. A macroscopic view of the 0.18-square-inch area of damage that caused areas of complete localized thickness loss is shown in figure 574. The location of multiple cracks on the right horizontal stabilizer leading-edge skin assembly, part number 40155-15, is shown in figure 575. A macroscopic view of these cracks, which measure 0.106 inch in length and are located at BL 18.5, are shown in figure 576.



RH STRINGER Damage BL 16.25 OV

Figure 573. Location of Damage on the Right Horizontal Stabilizer Stringer BL 16.25

Figure 574. Macroscopic View of Severe Damage on Right Horizontal Stabilizer Stringer BL 16.25

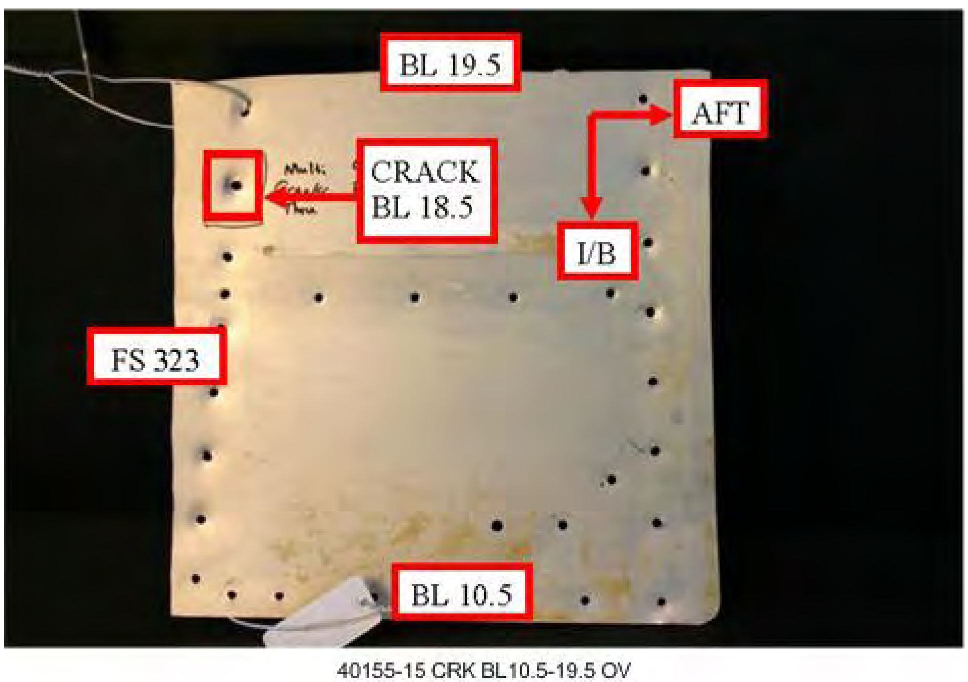


Figure 575. Location of Multiple Cracks on the Right Horizontal Stabilizer Leading-Edge Skin Assembly BL 18.5

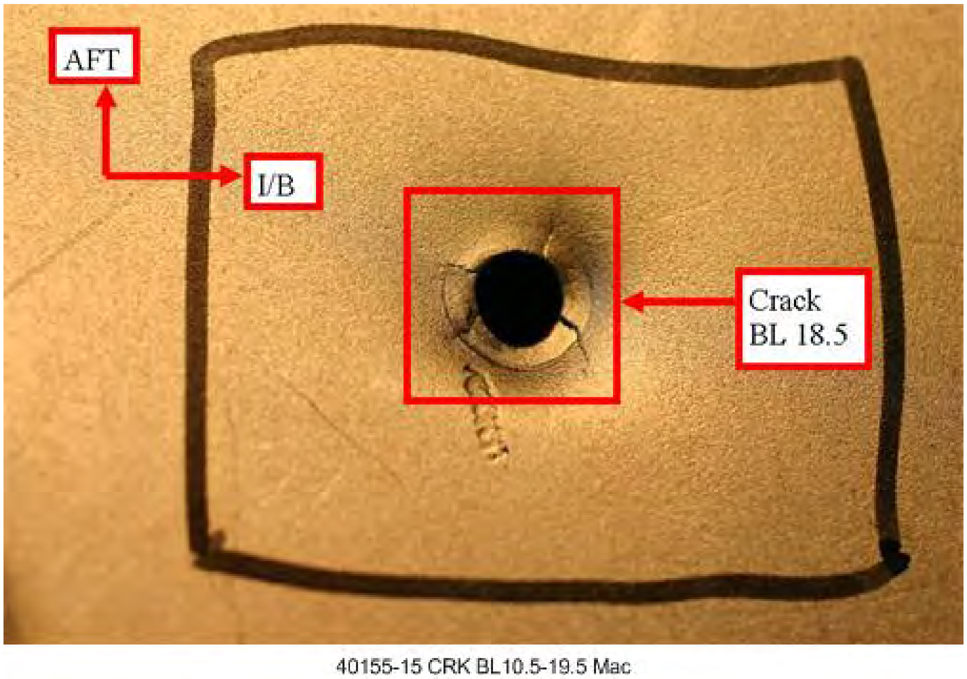


Figure 576. Macroscopic View of Multiple Cracks on the Right Horizontal Stabilizer Leading-Edge Skin Assembly BL 18.5

Figure 577 shows the location of four cracks on the left horizontal stabilizer leading-edge skin, part number 40155-14. Figure 578 shows a macroscopic view of crack 1, which measured 0.599 inch and was located at BL 94. A macroscopic view of crack 2, which measured 0.524 inch and was located at BL 96, is shown in figure 579. Figure 580 shows a macroscopic view of crack 3, which measured 0.549 inch and was located at BL 100. Cracks 1 through 3 were caused by a puncture to the leading-edge skin. Crack 4, which is shown macroscopically in figure 581, was located at BL 104.75 and measured 0.313 inch in length. The fluorescent liquid penetrant indication is shown in figure 582. A macroscopic view of crack 4 is shown in figure 583.

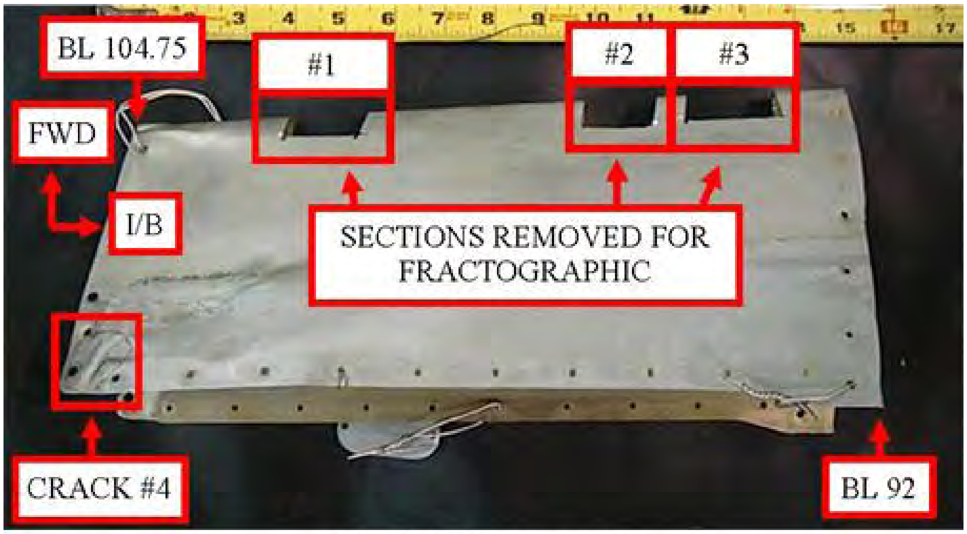


Figure 577. Location of Cracks on the Left Horizontal Stabilizer Leading-Edge Skin
BL 92 Through BL 104.75

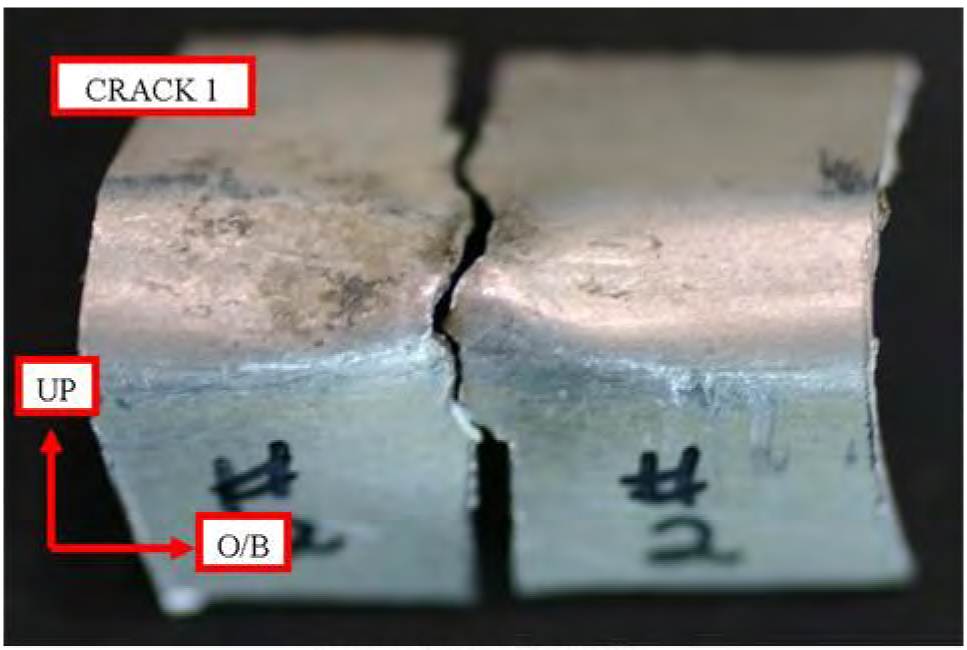
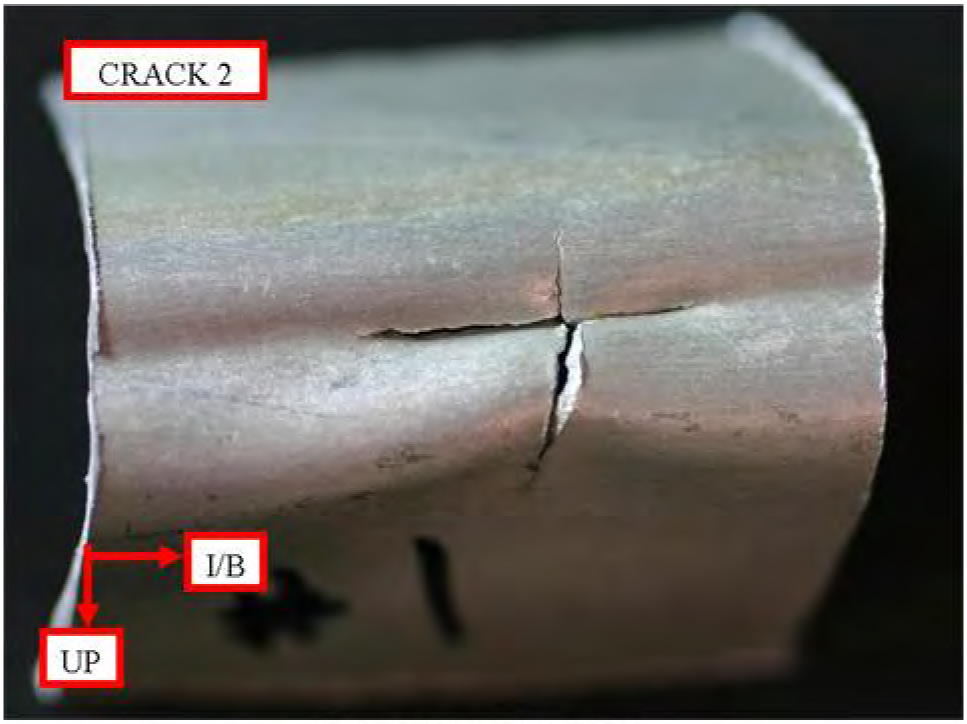
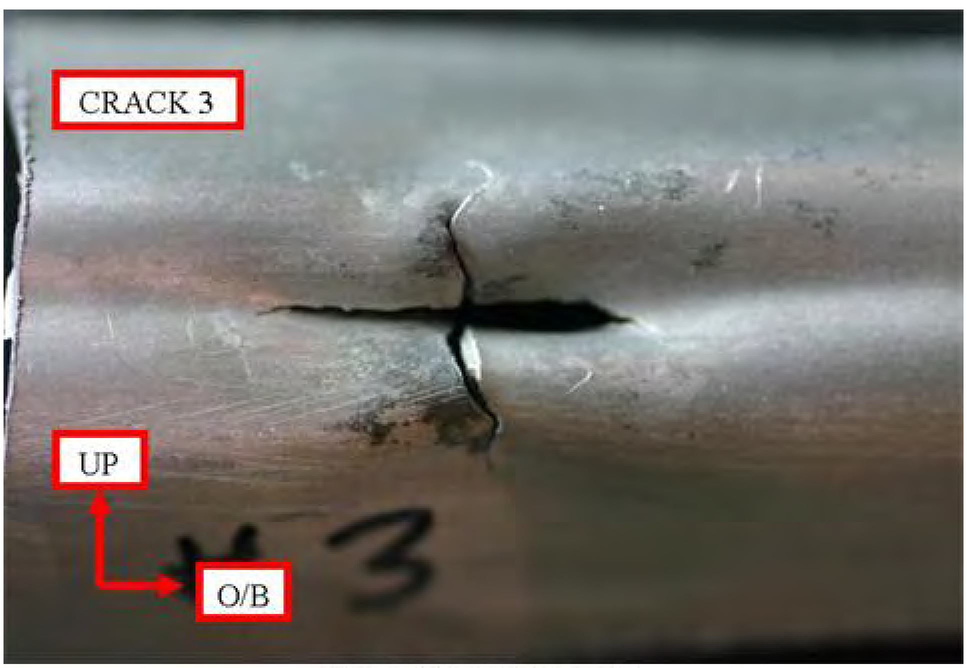


Figure 578. Macroscopic View of Crack 1 on the Left Horizontal Stabilizer Leading-Edge
Skin BL 94



40155-14 CRK 2 BL 92-104.75 Mac

Figure 579. Macroscopic View of Crack 2 on the Left Horizontal Stabilizer Leading-Edge Skin BL 96



40155-14 CRK 3 BL 92-104.75 Mac

Figure 580. Macroscopic View of Crack 3 on the Left Horizontal Stabilizer Leading-Edge Skin BL 100

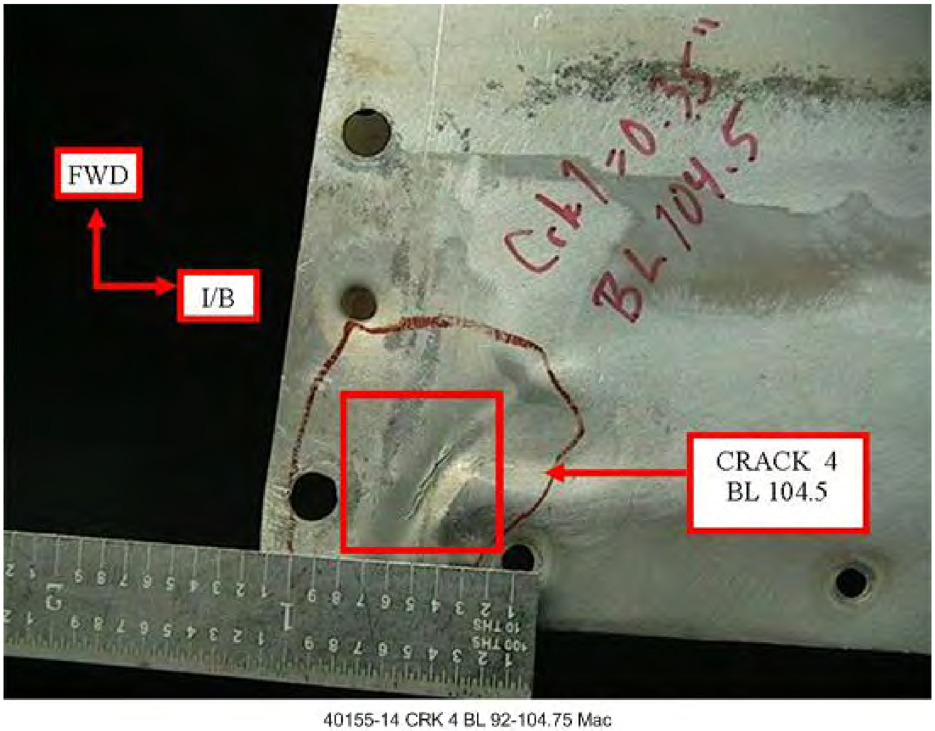


Figure 581. Macroscopic View of Crack 4 on the Left Horizontal Stabilizer Leading-Edge Skin BL 104.75

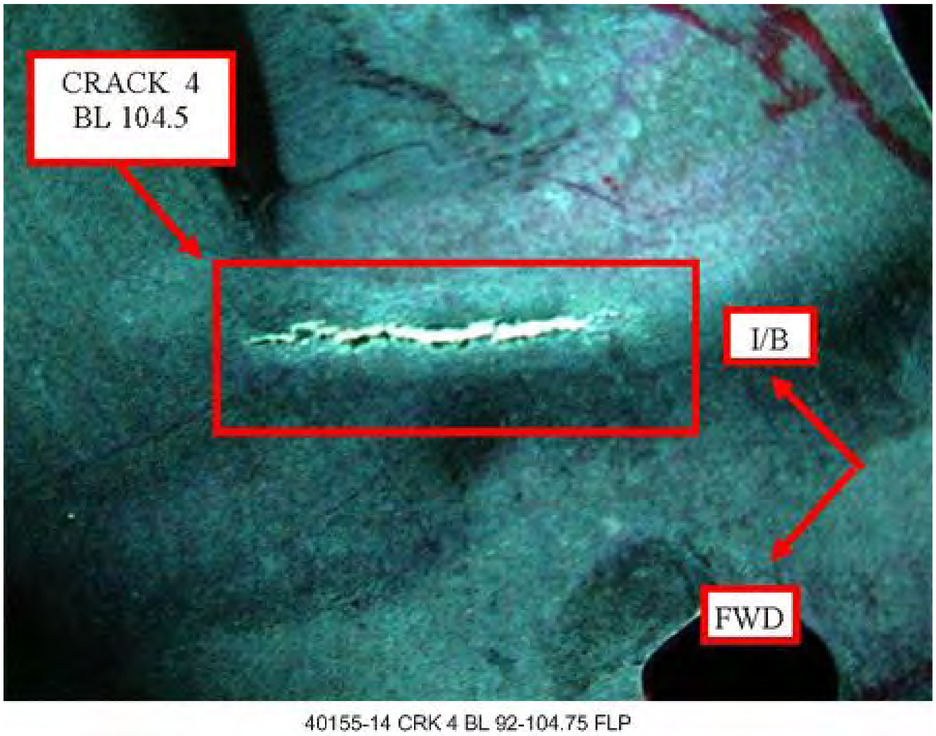


Figure 582. Fluorescent Liquid Penetrant Indication of Crack 4 on the Left Horizontal Stabilizer Leading-Edge Skin BL 104.5

Figure 583. Microscopic View of Crack 4 on the Left Horizontal Stabilizer Leading-Edge Skin at BL 104.5

3.4.2.5 Vertical Stabilizer.

Of the 43 parts inspected on the vertical stabilizer, 34 defects were identified. There were 27 cracks, 5 areas of corrosion, and 2 areas of damage recorded and characterized during the teardown evaluation. Multiple cracks occurred on each of the following structural members:

vertical stabilizer front spar, front spar left and right angles, rear spar, rear spar right angle, left and right side gussets, main bottom rib assembly, leading-edge rib, and the rudder leading-edge upper skin. Severe corrosion was found on the rudder torque tube assembly and rudder butt rib.

3.4.2.5.1 Vertical Stabilizer Front Spar.

Figure 584 shows the structural stackup of the vertical stabilizer front spar. Table 42 shows a detailed characterization of all defects found on the vertical stabilizer front spar.

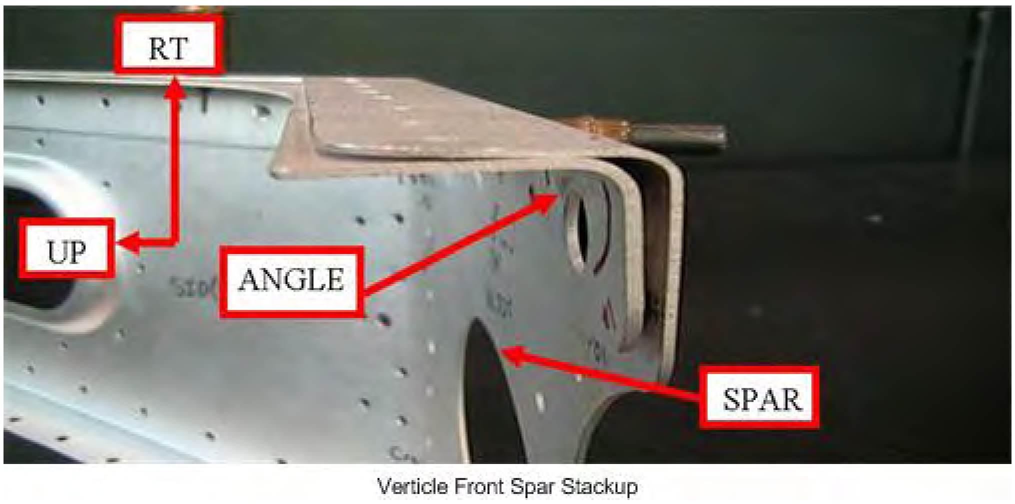


Figure 584. Structural Stackup of the Vertical Stabilizer Front Spar

Table 42. Inspection Results From the Vertical Stabilizer Front Spar

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Vertical stabilizer front spar, figure 586	Multiple cracks	WL 21	0.700 inch	Crack indication	Surface crack	587 588
	Multiple cracks	WL 21	0.810 inch	Crack indication	Surface crack	589 590
	Multiple cracks	WL 23.75	0.373 inch	Crack indication	Surface crack	591 592
	Multiple cracks	WL 23.75	0.340 inch	Crack indication	Surface crack	593 594
	Multiple cracks	WL 27.25	0.138 inch	Crack indication	Bend radii	595 596
	Multiple cracks	WL 27.25	0.086 inch	Crack indication	Bend radii	597 598
Vertical stabilizer front spar left angle, figure 599	Multiple cracks	WL 20.5	0.692 inch	Crack indication	Surface crack	600 601
	Multiple cracks	WL 23	0.35 inch	Crack indication	Surface crack	602 603

Table 42. Inspection Results From the Vertical Stabilizer Front Spar (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Vertical stabilizer front spar right angle, figure 604	Crack	WL 20.25	0.602 inch	Crack indication	Surface crack	605 606
	Crack	WL 23.5	0.414 inch	Crack indication	Surface crack	607 608
	Crack	WL 26.5	0.400 inch	Crack indication	Surface crack	609 610

Figure 585 shows the location of six areas of cracking on the vertical stabilizer front spar from WL 20.25 to WL 46.75. A macroscopic view of crack 1, which measured 0.70 inch and was located at WL 21, is shown in figure 586, and a microscopic view of this cracks is shown in figure 587. Crack 2 located at WL 21, which measured 0.810 inch, is shown macroscopically in figure 588 and microscopically in figure 589. A macroscopic view of crack 3, which measured 0.373 inch and was located on the front spar at WL 23.75, is shown in figure 590. A microscopic view of this crack is shown in figure 591. Crack 4, which is located at WL 23.75 on the front spar, measured 0.340 inch and is shown macroscopically in figure 592. A microscopic view of crack 4 is shown in figure 593.

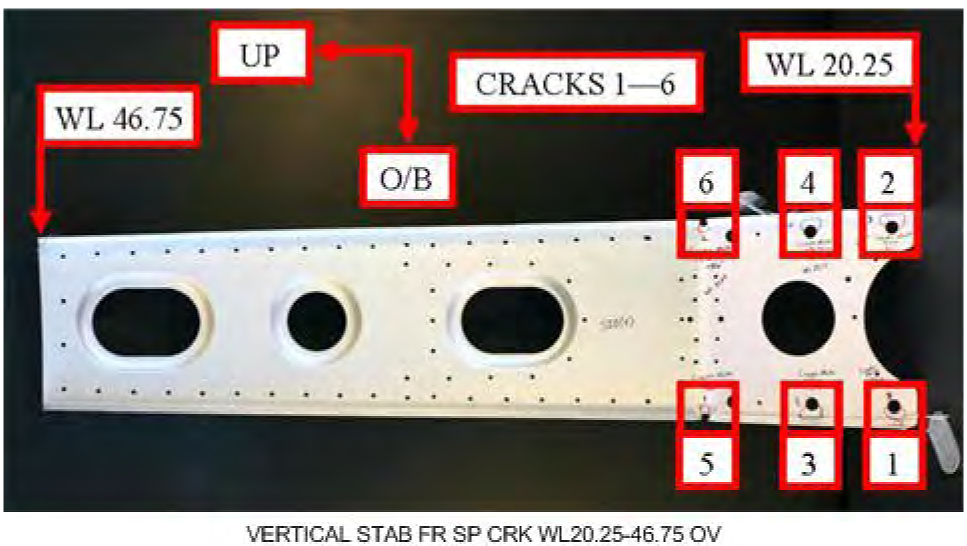


Figure 585. Location of Six Cracks on the Vertical Stabilizer Front Spar WL 20.25 Through WL 46.75

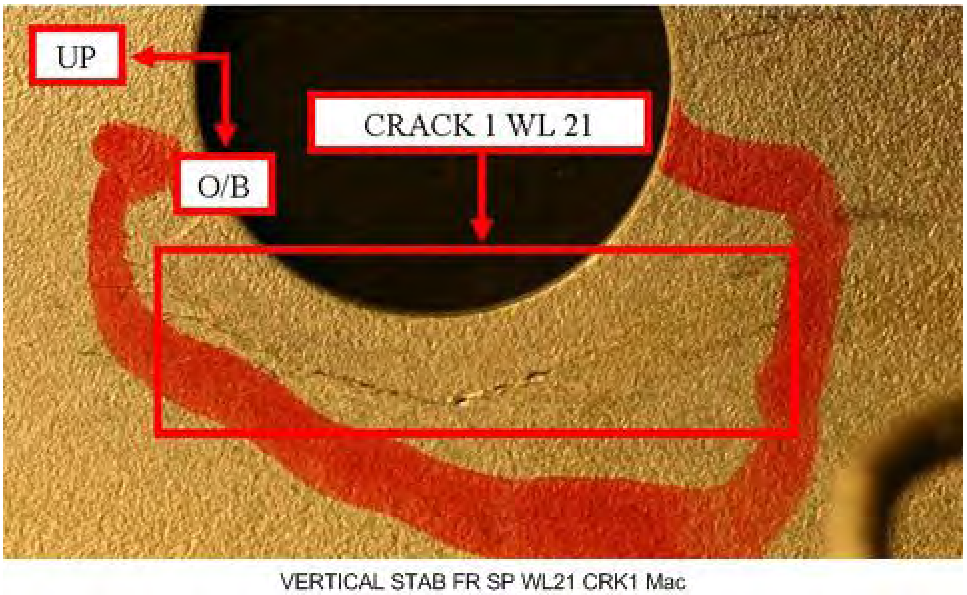


Figure 586. Macroscopic View of Crack 1 on the Vertical Stabilizer Front Spar WL 21

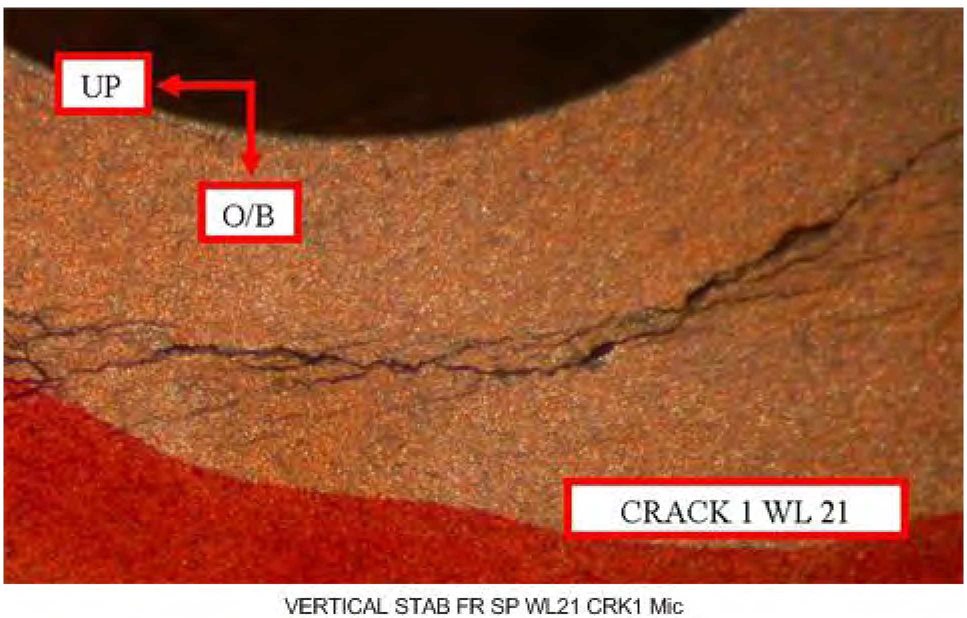


Figure 587. Microscopic View of Crack 1 on the Vertical Stabilizer Front Spar WL 21

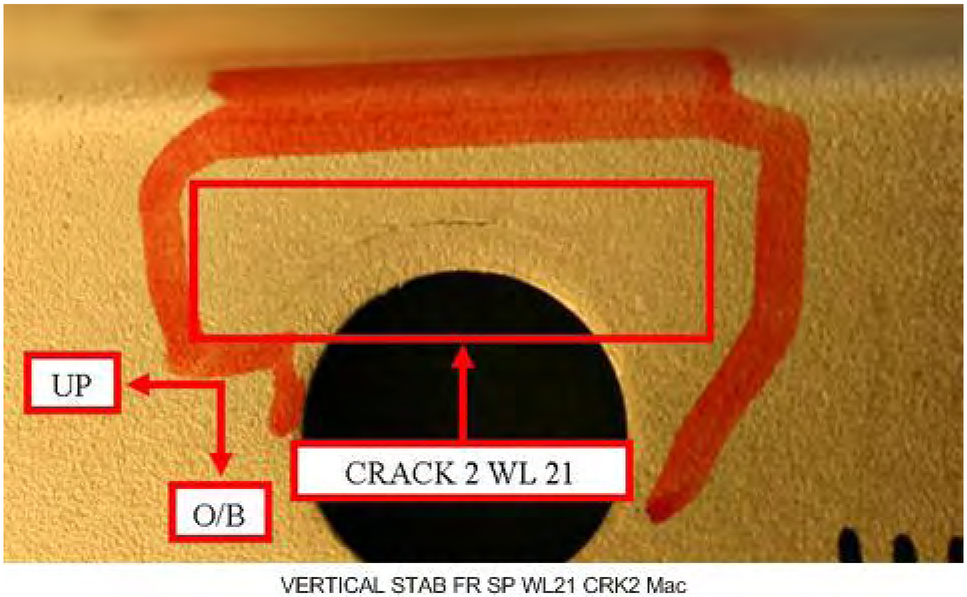


Figure 588. Macroscopic View of Crack 2 on the Vertical Stabilizer Front Spar WL 21

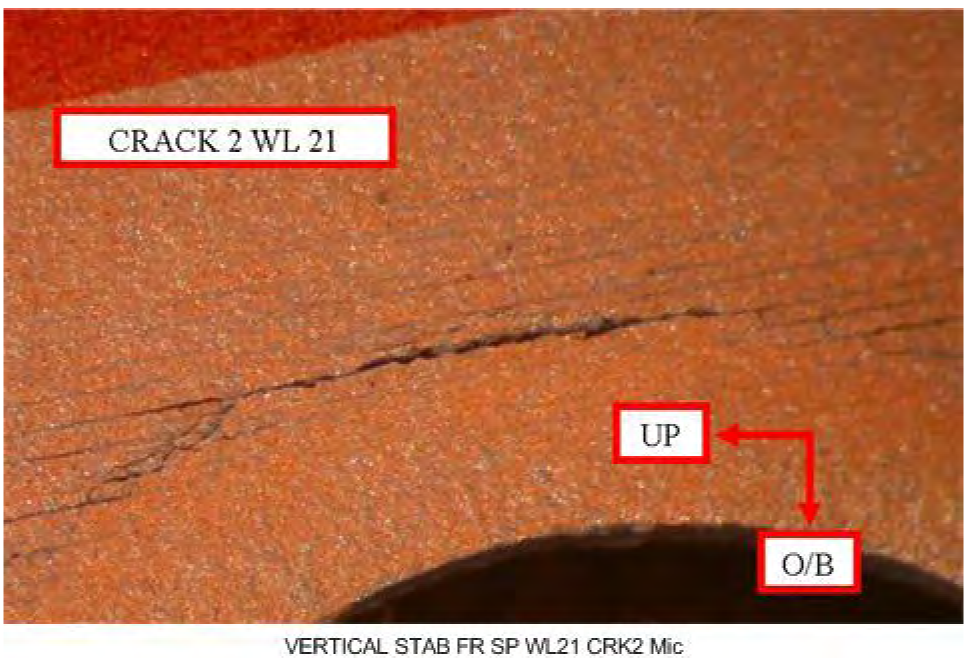
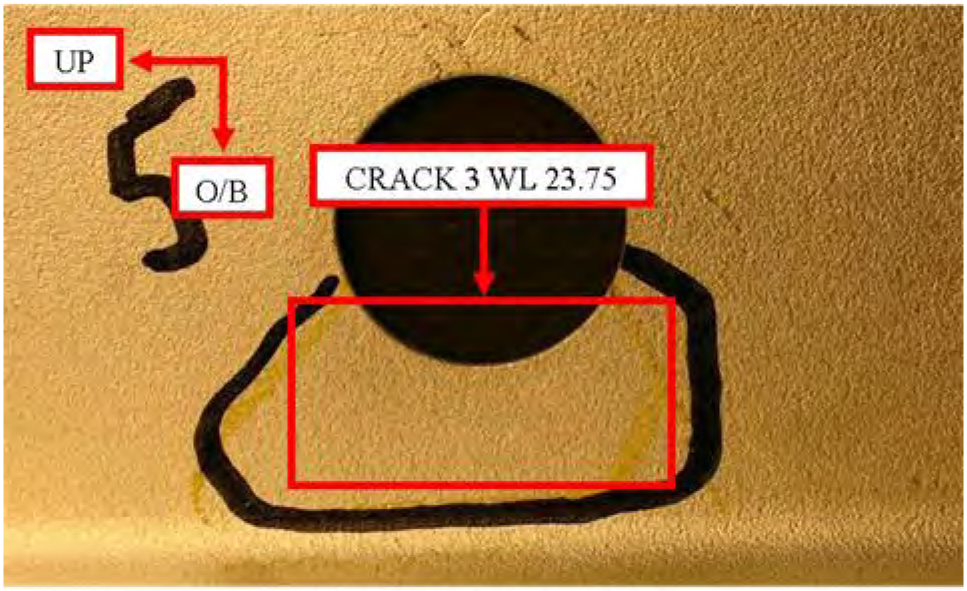
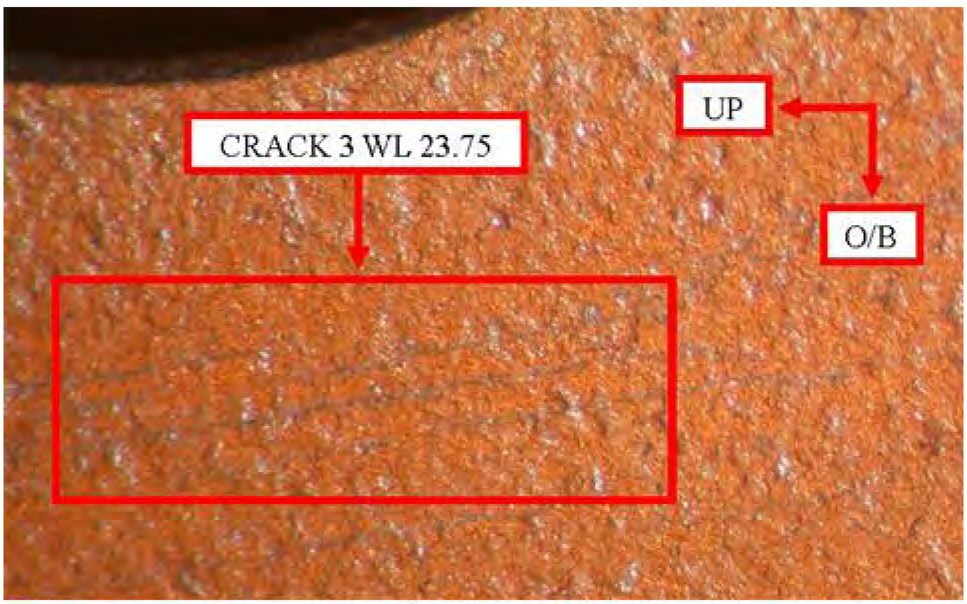


Figure 589. Microscopic View of Crack 2 on the Vertical Stabilizer Front Spar WL 21



VERTICAL STAB FR SP WL21 CRK3 Mac

Figure 590. Macroscopic View of Crack 3 on the Vertical Stabilizer Front Spar WL 23.75



VERTICAL STAB FR SP WL21 CRK3 Mic

Figure 591. Microscopic View of Crack 3 on the Vertical Stabilizer Front Spar WL 23.75

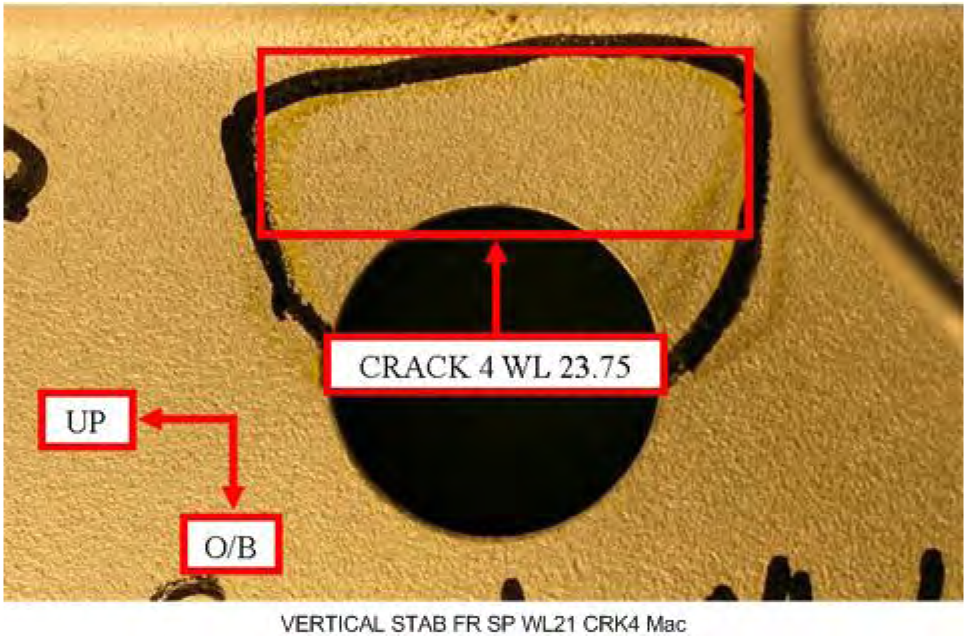


Figure 592. Macroscopic View of Crack 4 on the Vertical Stabilizer Front Spar WL 23.75

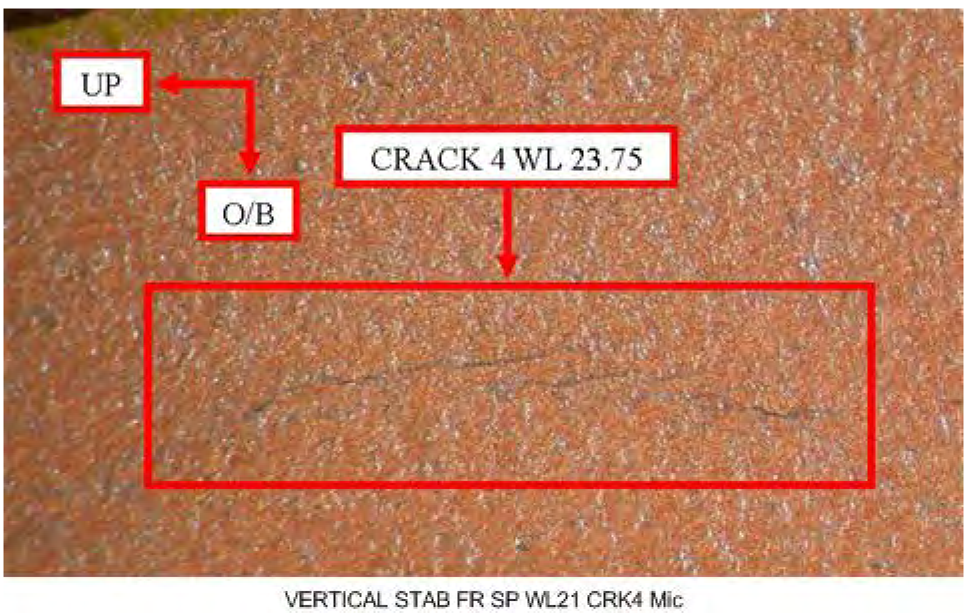


Figure 593. Microscopic View of Crack 4 on the Vertical Stabilizer Front Spar WL 23.75

Crack 5, which measured 0.138 inch and was located at WL 27.25 on the vertical stabilizer front spar, is shown macroscopically in figure 594. A microscopic view of crack 5 is shown in figure 595. Figure 596 shows a macroscopic view of crack 6, which was located at FS 27.25. This 0.086-inch crack is also shown microscopically in figure 597.

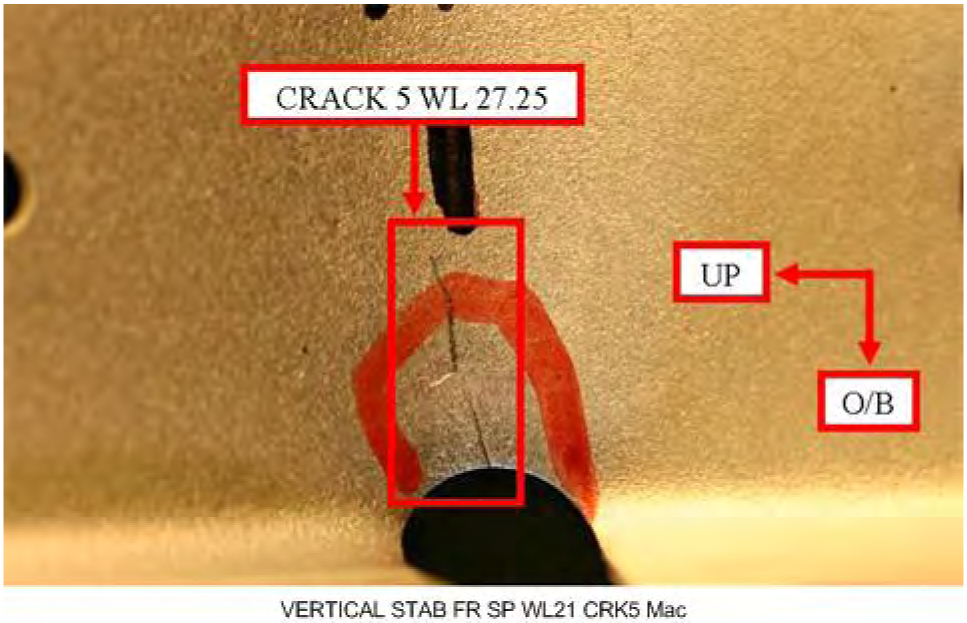


Figure 594. Macroscopic View of Crack 5 on the Vertical Stabilizer Front Spar WL 27.25

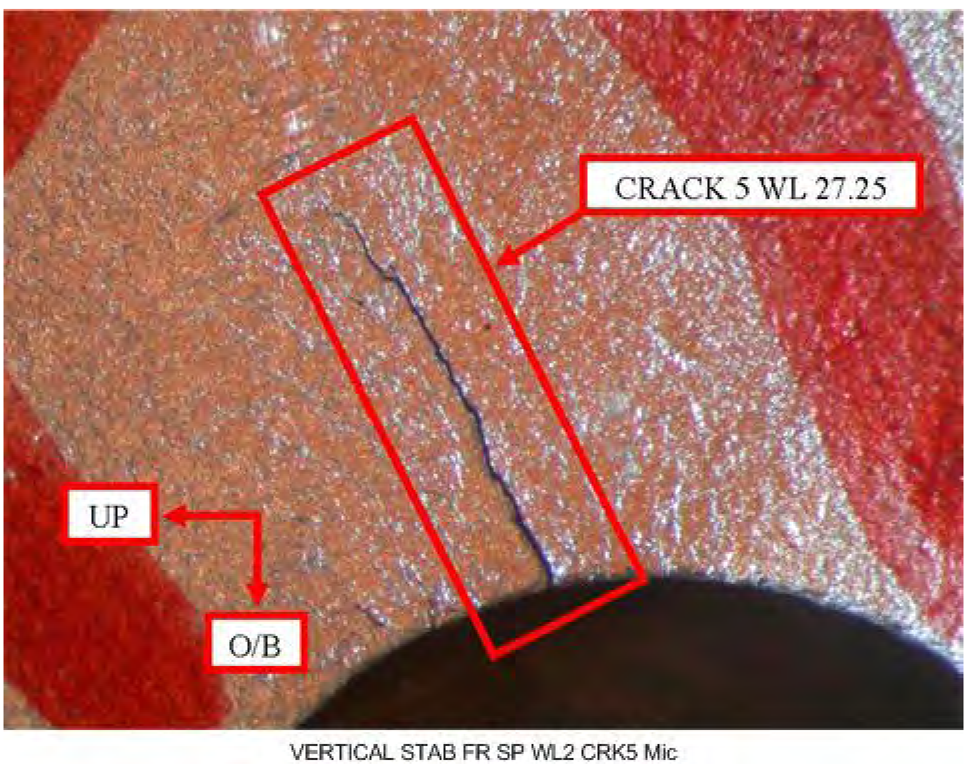


Figure 595. Microscopic View of Crack 5 on the Vertical Stabilizer Front Spar WL 27.25

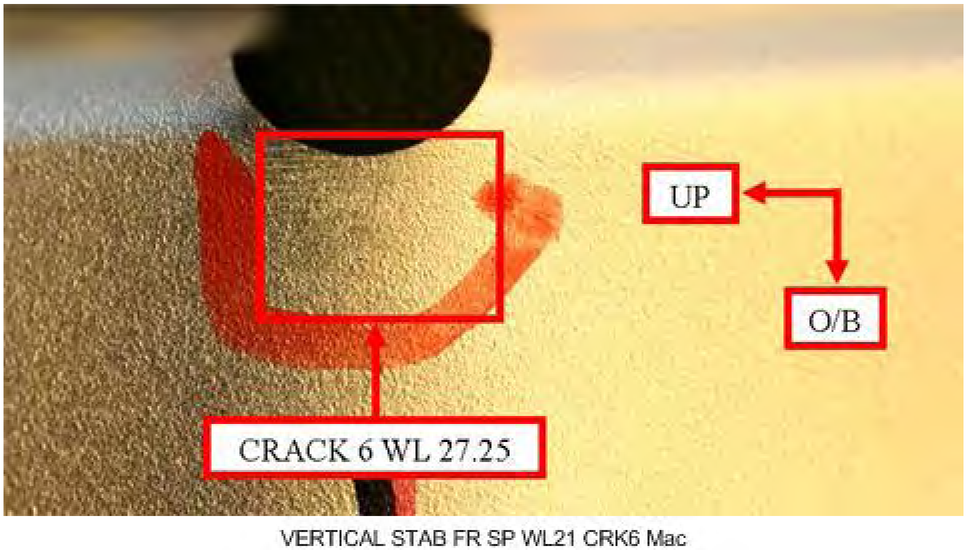


Figure 596. Macroscopic View of Crack 6 on the Vertical Stabilizer Front Spar WL 27.25

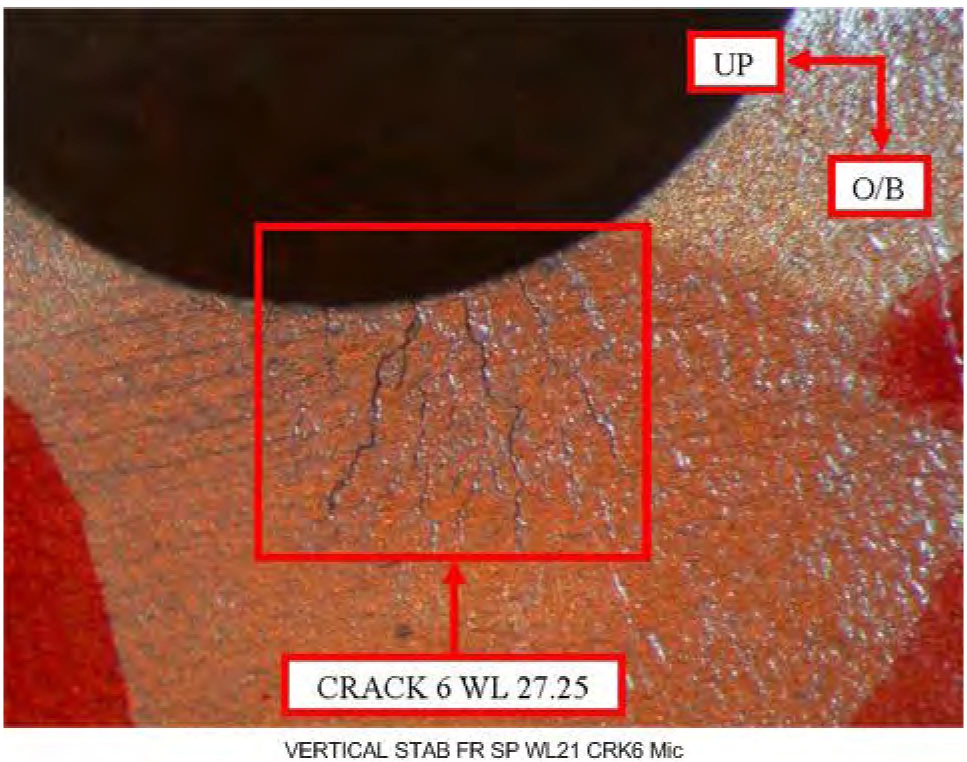
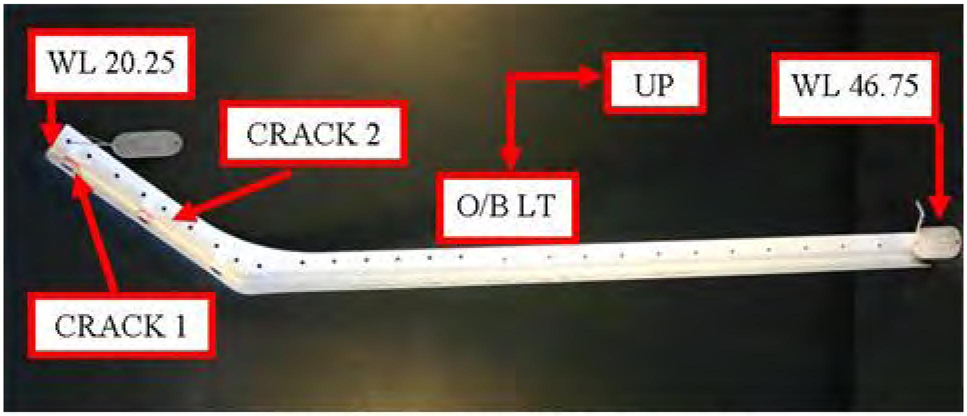


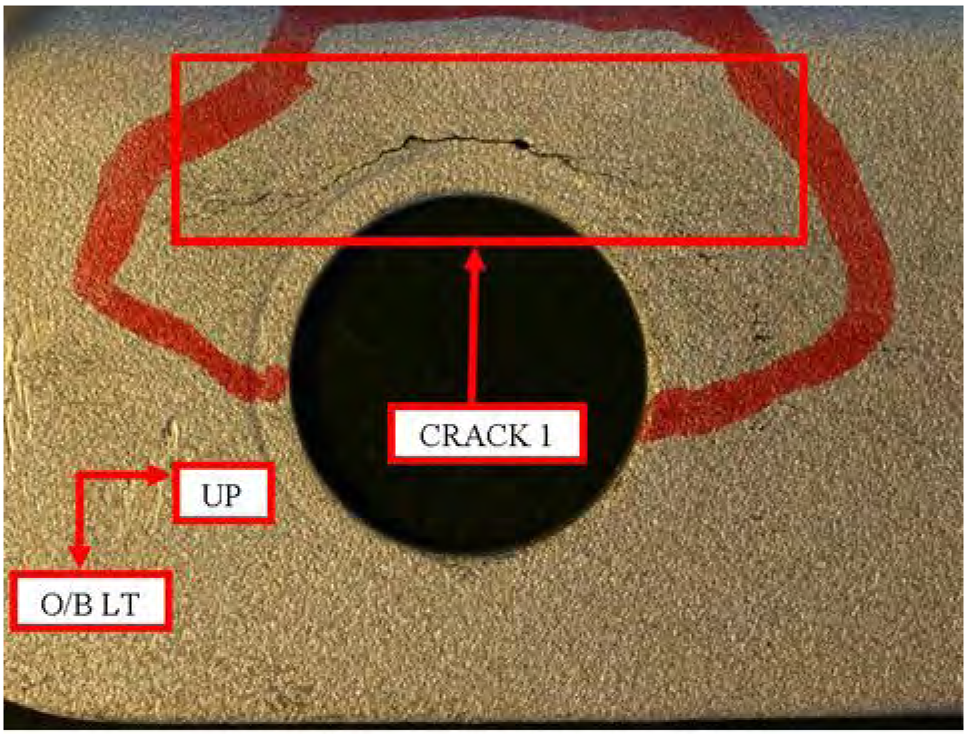
Figure 597. Microscopic View of Crack 6 on the Vertical Stabilizer Front Spar WL 27.25

The location of two cracks on the vertical stabilizer front spar left angle is shown in figure 598. Crack 1, which is shown macroscopically in figure 599, was located at WL 20.5 and measured 0.692 inch. A microscopic view of this crack is shown in figure 600. Figure 601 shows a macroscopic view of crack 2, which measured 0.35 inch and was located at WL 23 on the front spar left angle. This crack is shown microscopically in figure 602.



VERTICAL STAB FR SP LEFT ANGLE #1 WL20.25 OV

Figure 598. Location of Two Cracks on the Vertical Stabilizer Front Spar Left Angle WL 20.25 Through WL 46.75



VERTICAL STAB FR SP LT ANGLE #1 WL20.25 CRK1 Mac

Figure 599. Macroscopic View of Crack 1 on the Vertical Stabilizer Front Spar Left Angle WL 20.25

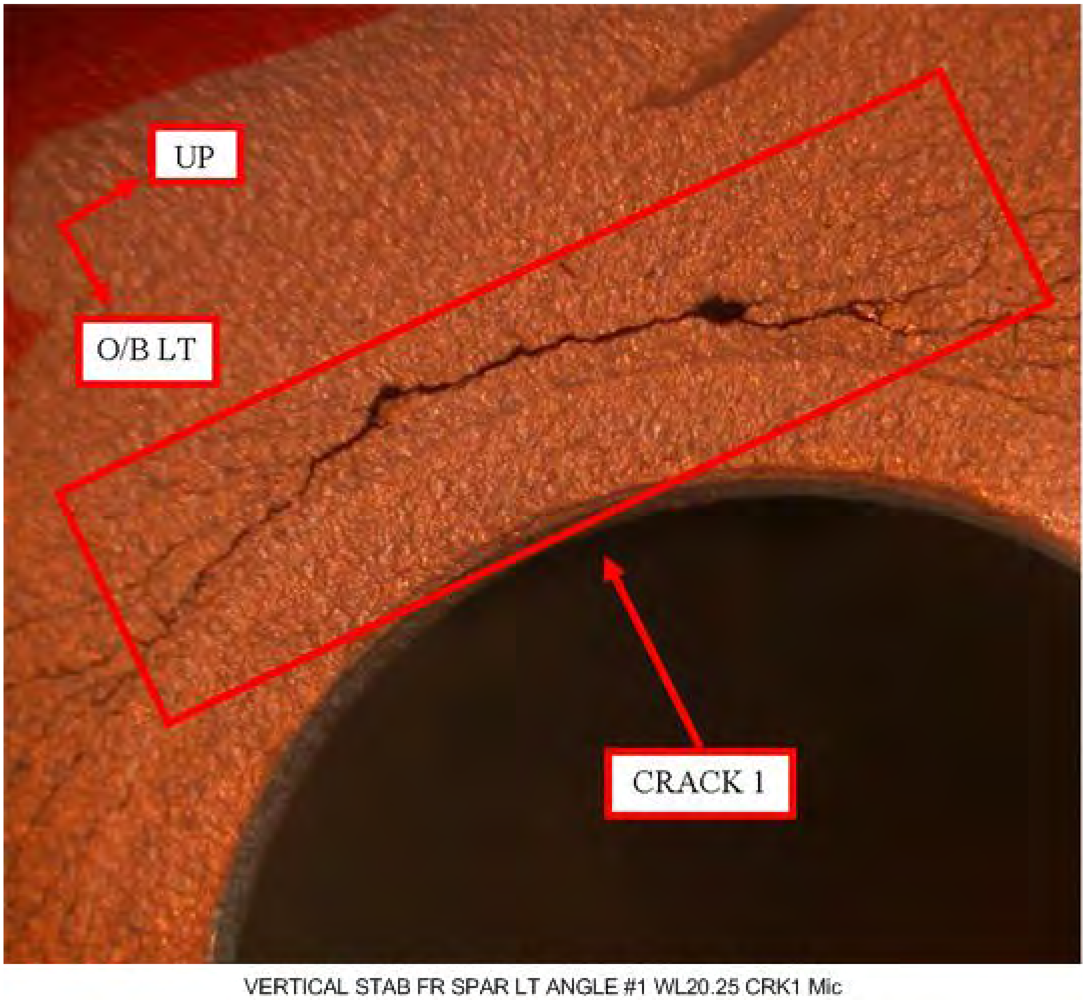


Figure 600. Microscopic View of Crack 1 on the Vertical Stabilizer Front Spar Left Angle WL 20.25

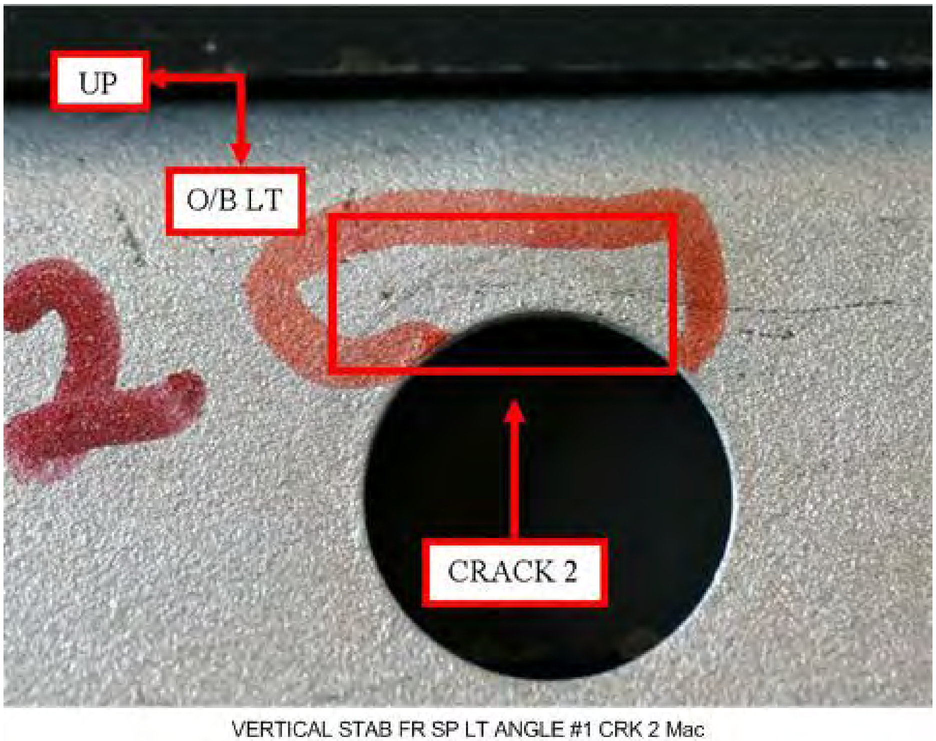


Figure 601. Macroscopic View of Crack 2 on the Vertical Stabilizer Front Spar Left Angle WL 23

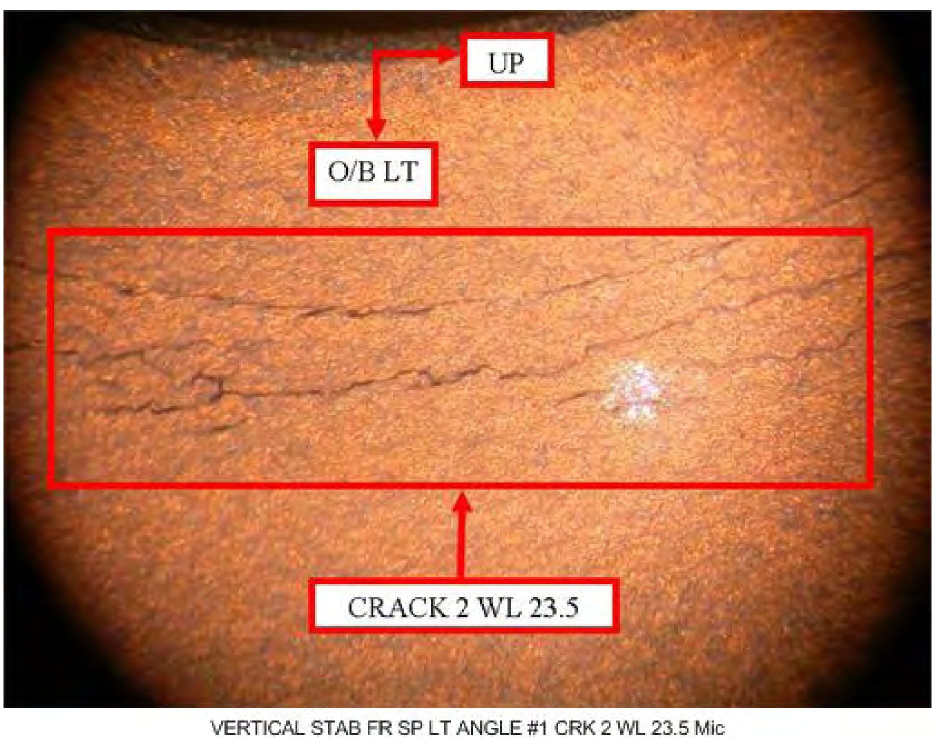
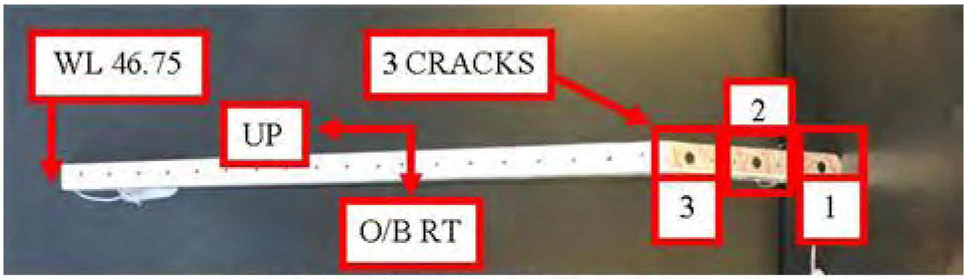


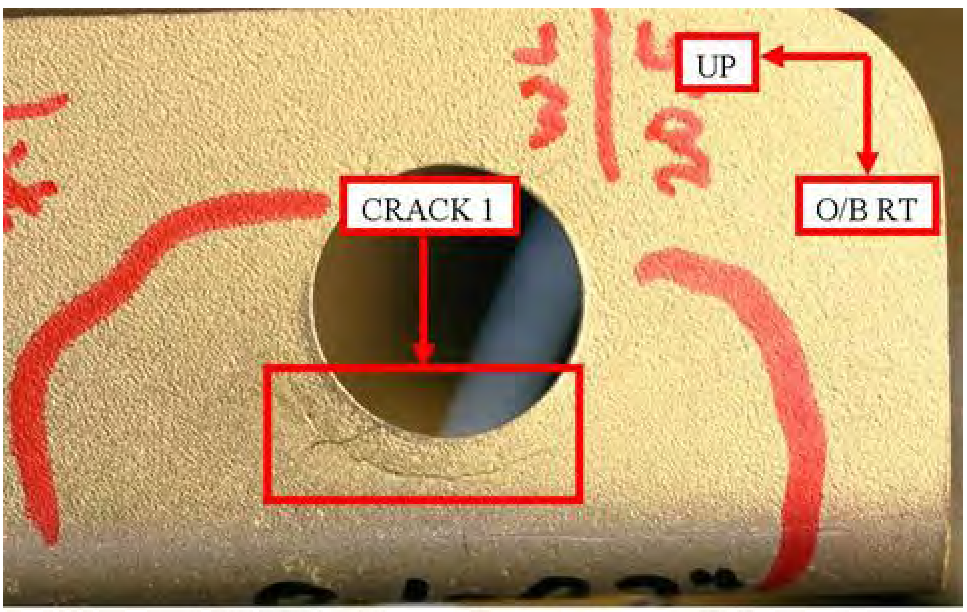
Figure 602. Microscopic View of Crack 2 on the Vertical Stabilizer Front Spar Left Angle WL 23.5

Figure 603 shows the location of three cracks on the vertical stabilizer front spar right angle. Crack 1, which measured 0.602 inch and was located at WL 20.25, is shown macroscopically in figure 604 and microscopically in figure 605. Figure 606 shows a macroscopic view of crack 2 at WL 23.5, and a microscopic view of this 0.414-inch crack is shown in figure 607. A macroscopic view of crack 3, which measured 0.40 inch and was located at WL 26.5, is shown in figure 608. Figure 609 shows a microscopic view of this crack.



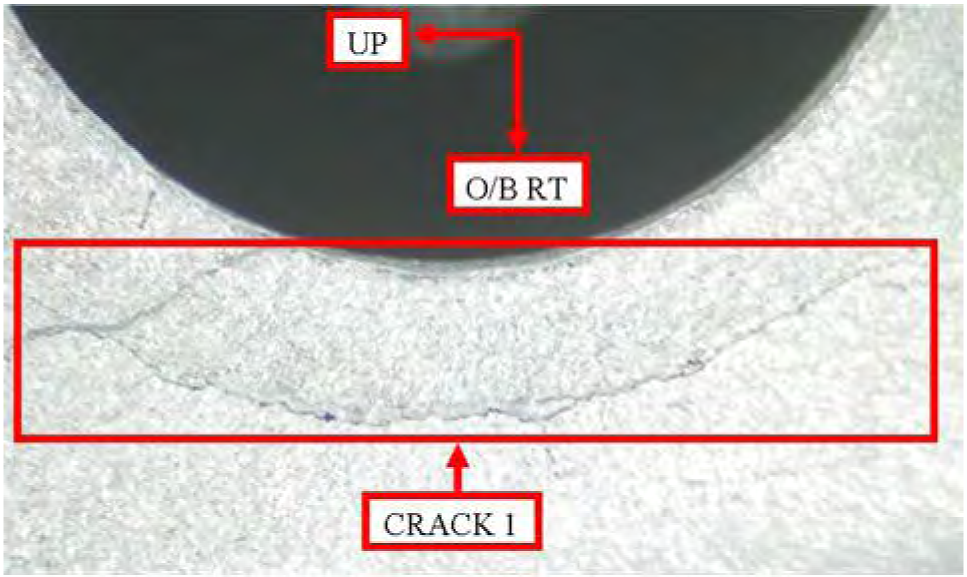
VERTICAL STAB FR SP RT ANGLE WL 20.25-46.75 OV

Figure 603. Location of Three Cracks on the Vertical Stabilizer Front Spar Right Angle WL 20.25 Through WL 46.75



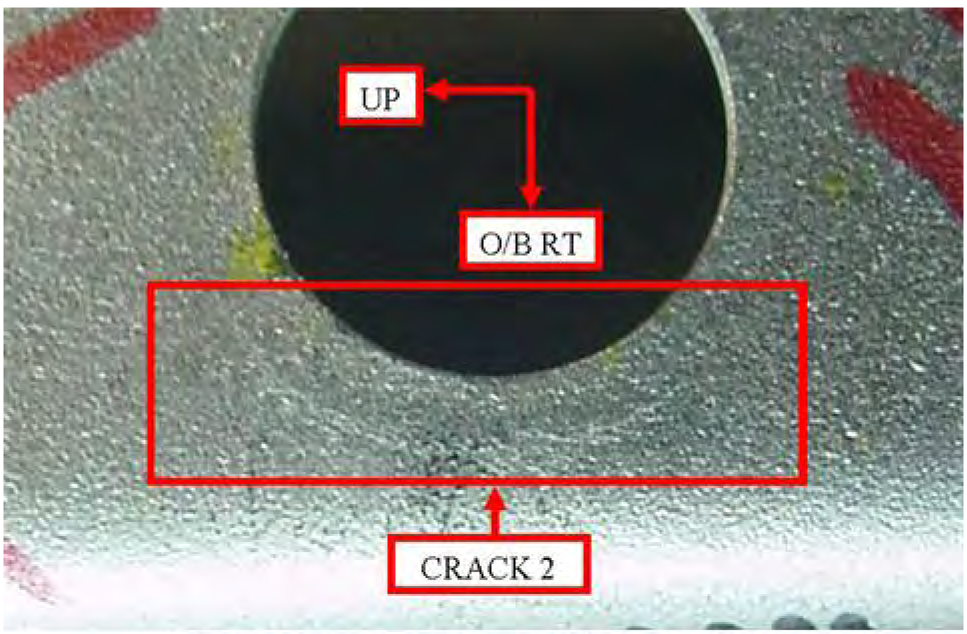
VERTICAL STAB RT FT SP ANGLE #1 WL20.25 CRK1 Mac

Figure 604. Macroscopic View of Crack 1 on the Vertical Stabilizer Front Spar Right Angle WL 20.25



VERTICAL STAB RT FT SP ANGLE #1 WL20.25 CRK1 Mic

Figure 605. Microscopic View of Crack 1 on the Vertical Stabilizer Front Spar Right Angle WL 20.25



VERTICAL STAB RT FT SP ANGLE #1 WL20.25-46.75 CRK2 Mac

Figure 606. Macroscopic View of Crack 2 on the Vertical Stabilizer Front Spar Right Angle WL 23.5

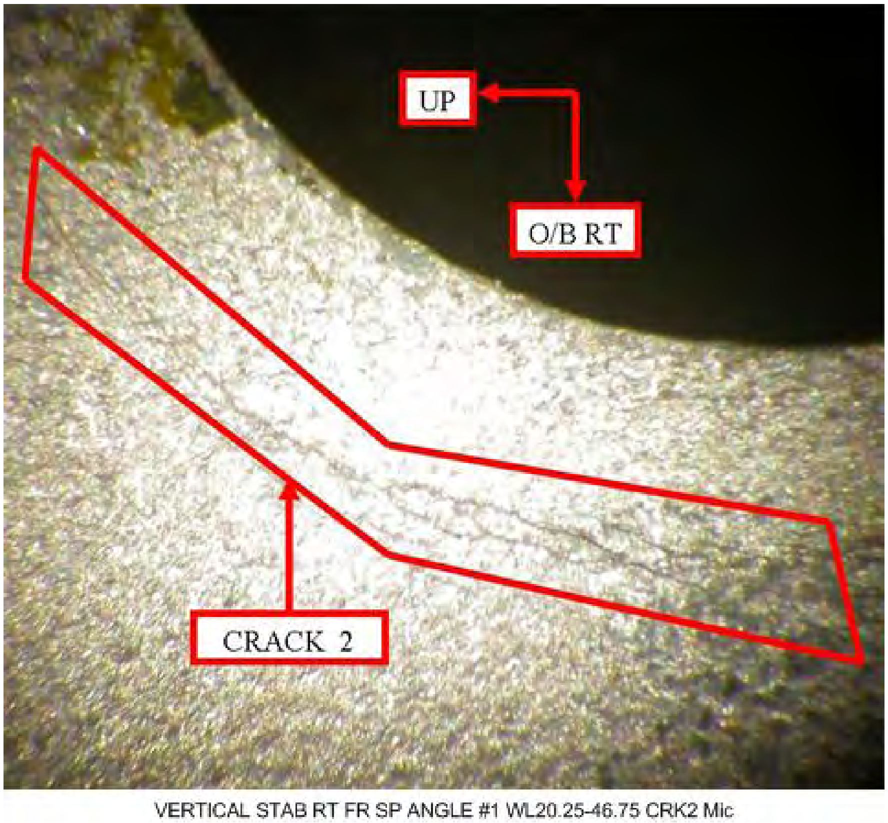


Figure 607. Microscopic View of Crack 2 on the Vertical Stabilizer Front Spar Right Angle WL 23.5

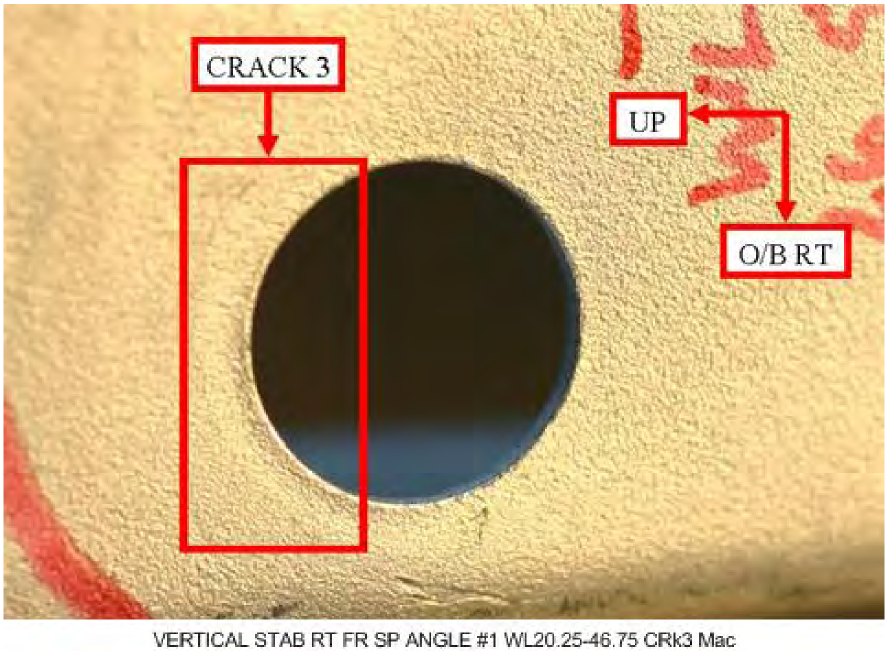


Figure 608. Macroscopic View of Crack 3 on the Vertical Stabilizer Front Spar Right Angle WL 26.5

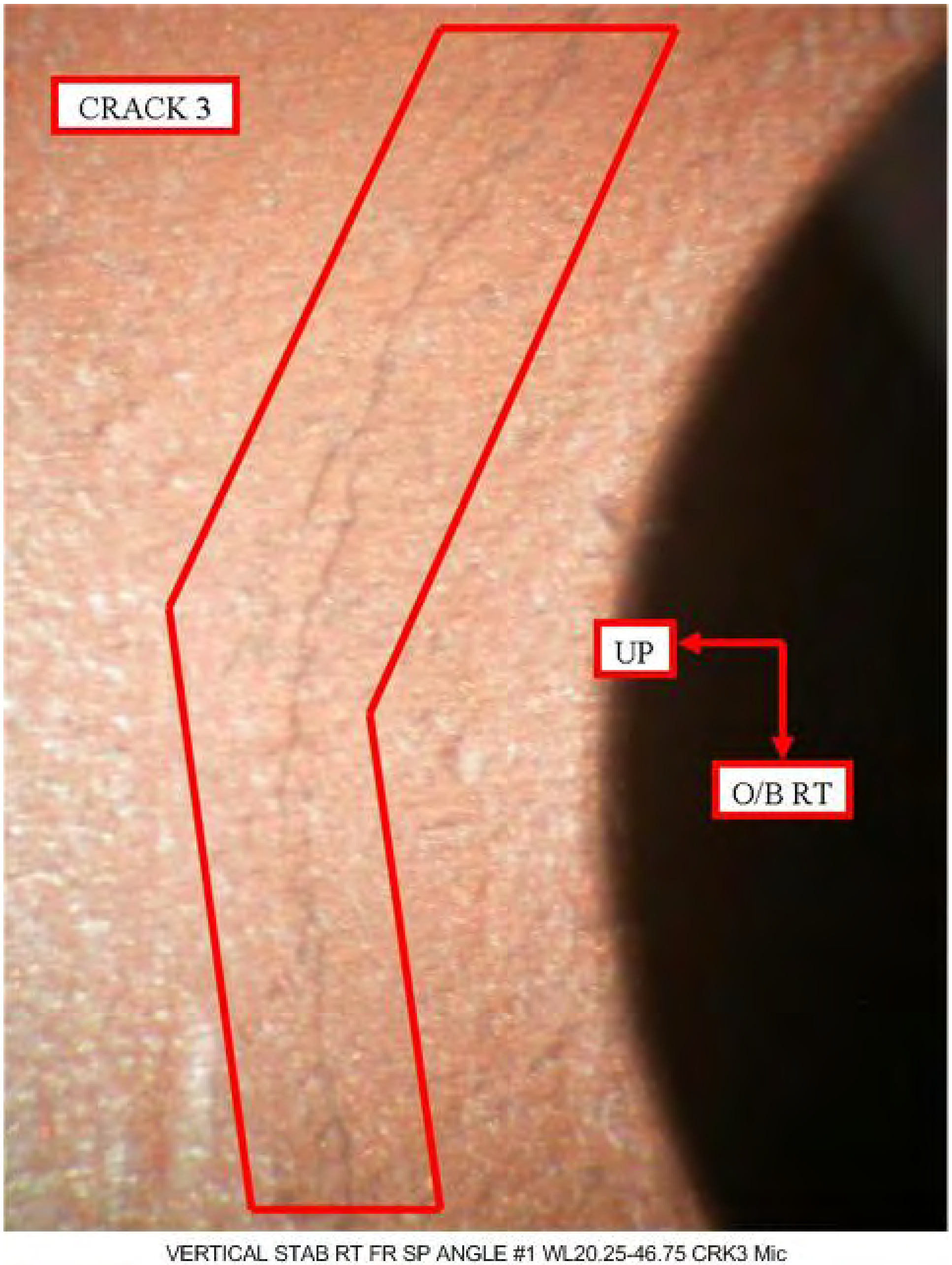
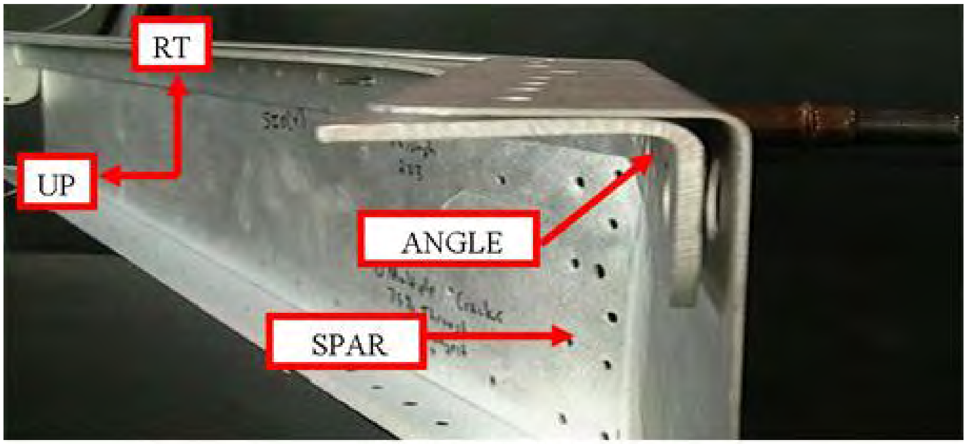


Figure 609. Microscopic View of Crack 3 on the Vertical Stabilizer Front Spar Right Angle WL 26.5

<u>3.4.2.5.2 Vertical Stabilizer Rear Spar</u>.

Figure 610 shows the structural stackup of the vertical stabilizer rear spar. Table 43 shows detailed characterizations of each defect identified during the teardown evaluation on the rear spar.



Vertical Rear Spar Stackup

Figure 610. Structural Stackup of the Vertical Stabilizer Rear Spar

Table 43. Inspection Results From the Vertical Stabilizer Rear Spar

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indications	Defect Classification	Defect Figure Number
Vertical stabilizer rear spar right angle, figure 612	Multiple cracks	WL 20.25	0.335 inch	Crack indication	5% Through-the-thickness cracks	613 614
	Multiple cracks	WL 24.75	0.497 inch	Crack indication	5% Through-the-thickness cracks	615 615
Vertical stabilizer rear spar, figure 617	Crack	WL 26	0.102 inch	Crack indication	Bend radii	618 620
	Crack	WL 26	0.203 inch	Crack indication	Bend radii	623

The location of two cracks on the vertical stabilizer rear spar right angle is shown in figure 611. A macroscopic view of multiple cracks grouped together and identified as crack 1 is shown in figure 612. The maximum length of this group of 5% through-the-thickness cracks is 0.335 inch. A microscopic view of crack 1, which is located on the rear spar right angle at WL 20.25, is shown in figure 613. Another cluster of 5% through-the-thickness cracks, crack 2, were identified at WL 24.75 and had a length of 0.497 inch. A macroscopic view of crack 2 is shown in figure 614, and a microscopic view is shown in figure 615.

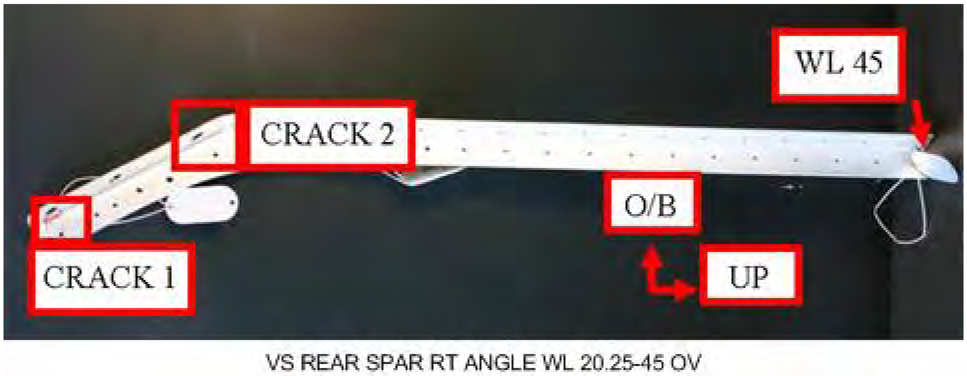


Figure 611. Location Two Cracks on the Vertical Stabilizer Rear Spar Right Angle WL 20.25 Through WL 45

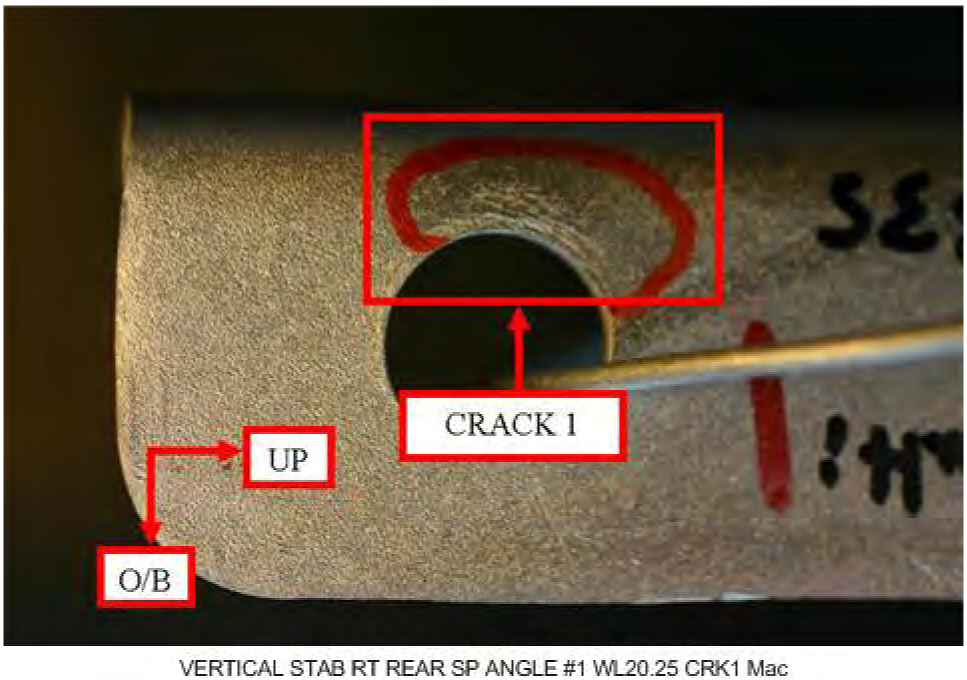


Figure 612. Macroscopic View of Crack 1 on the Vertical Stabilizer Rear Spar Right Angle WL 20.25

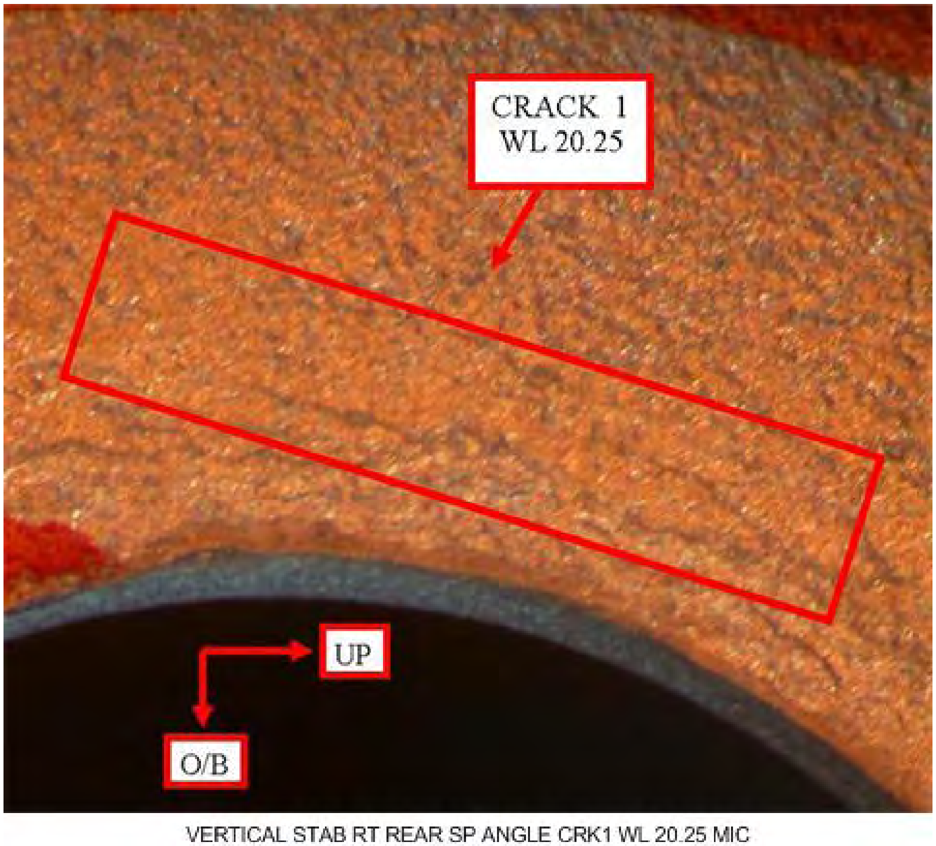


Figure 613. Microscopic View of Crack 1 on the Vertical Stabilizer Rear Spar Right Angle WL 20.25

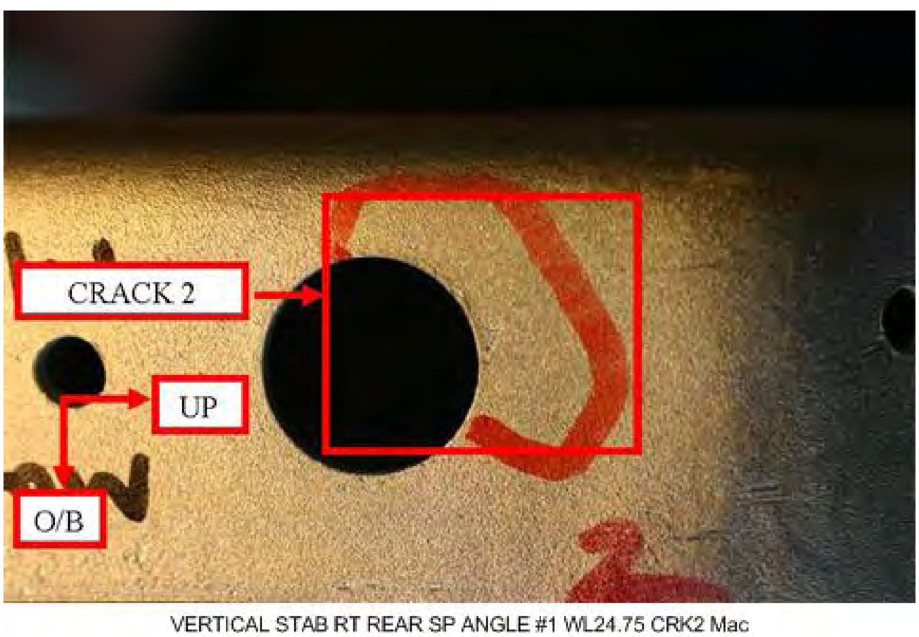


Figure 614. Macroscopic View of Crack 2 on the Vertical Stabilizer Rear Spar Right Angle WL 24.75

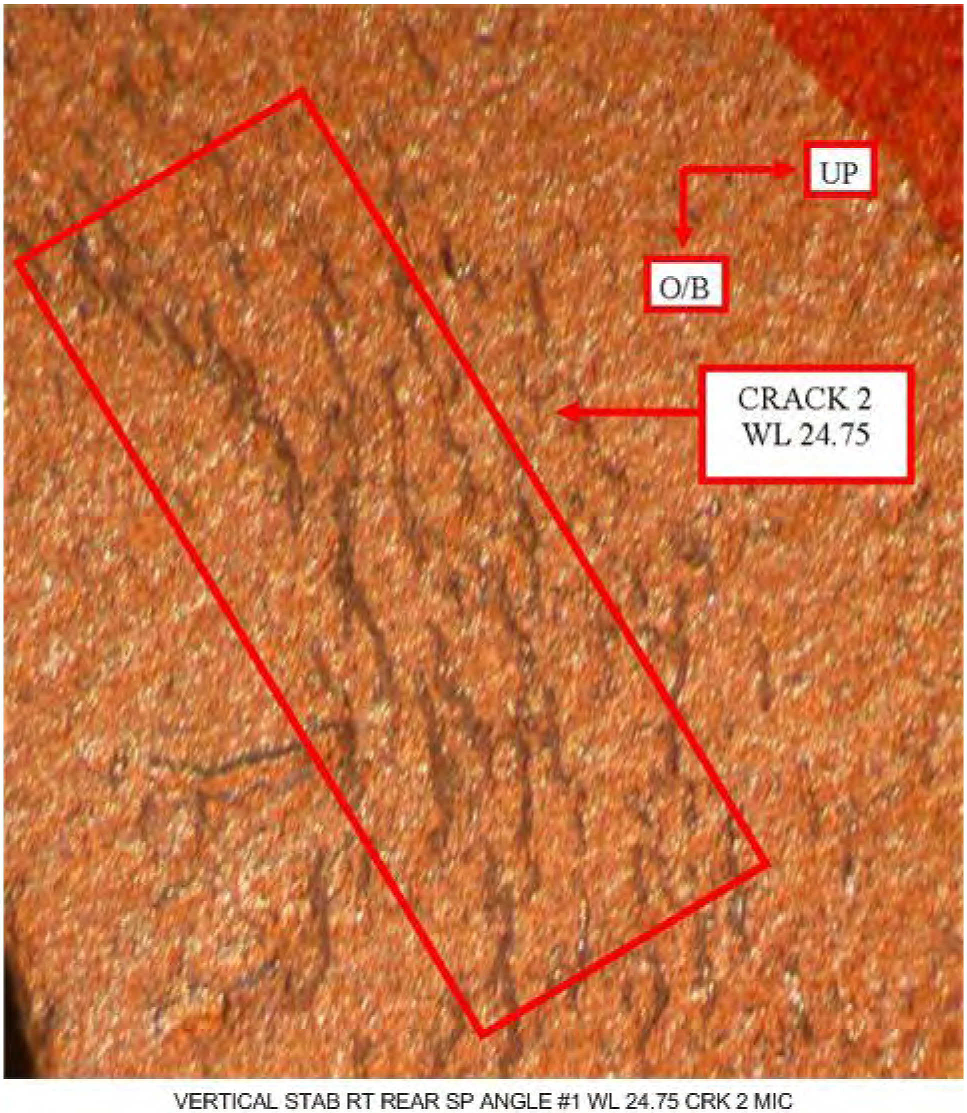
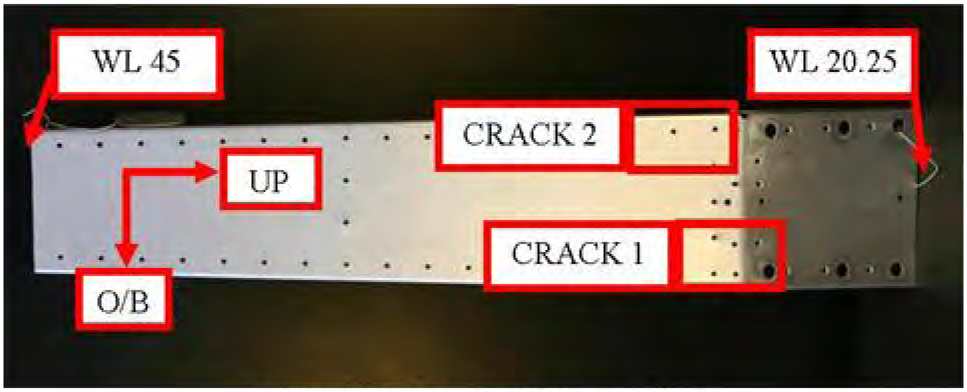


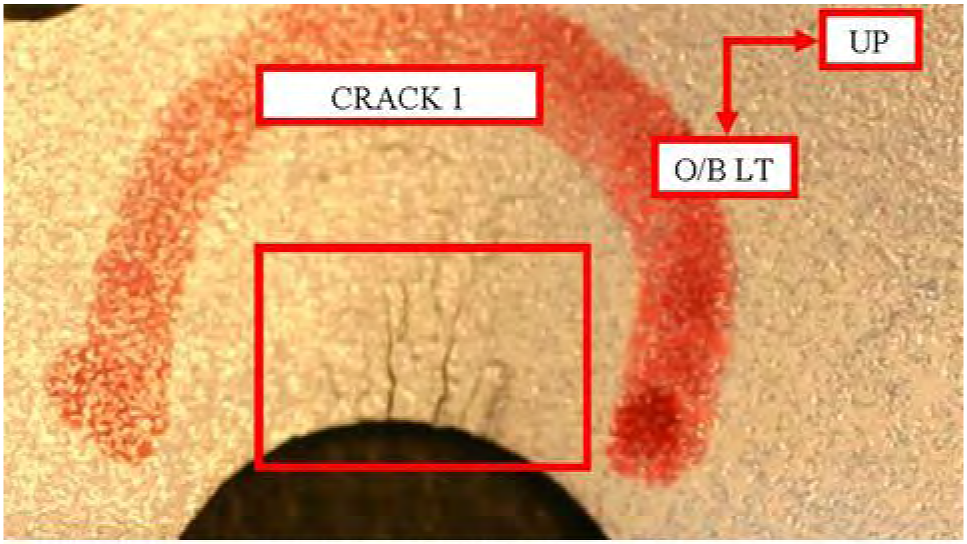
Figure 615. Microscopic View of Crack 2 on the Vertical Stabilizer Rear Spar Right Angle WL 24.75

Figure 616 shows the location of two cracks on the vertical stabilizer rear spar at WL 26. Crack 1, which is shown macroscopically in figure 617, measured 0.102 inch. The fluorescent penetrant indication for crack 1 is shown in figure 618. A microscopic view of this crack is shown in figure 619. A macroscopic view of crack 2 is shown in figure 620. The fluorescent liquid penetrant indication for this 0.203-inch crack is shown in figure 621. A microscopic view is shown in figure 622.



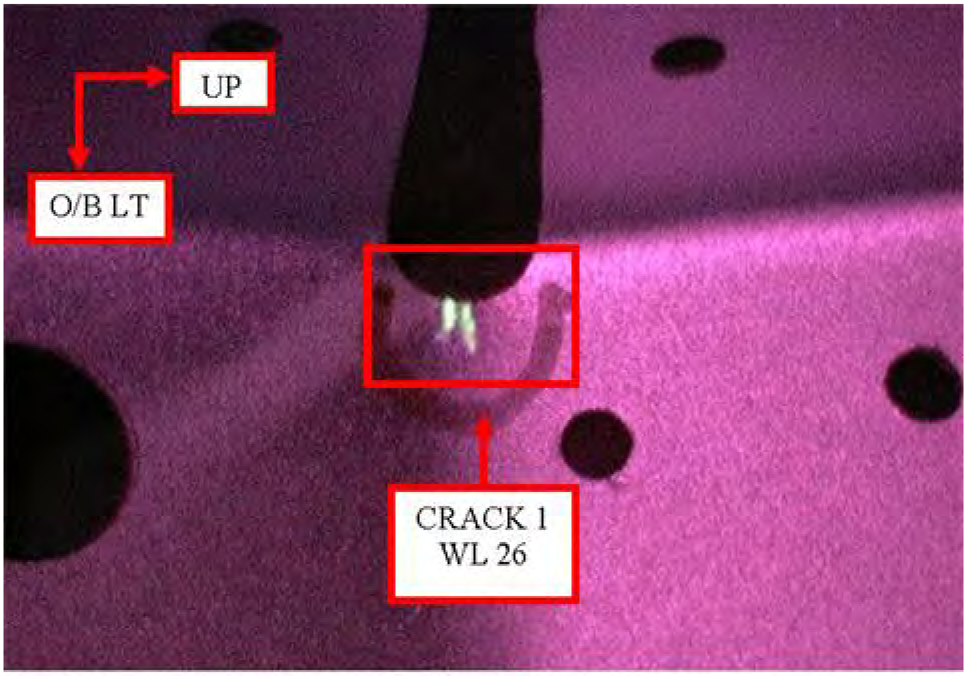
VERTICAL STAB REAR SPAR WL 20.25-45 OV

Figure 616. Location of Two Cracks on the Vertical Stabilizer Rear Spar WL 20.25 Through WL 45



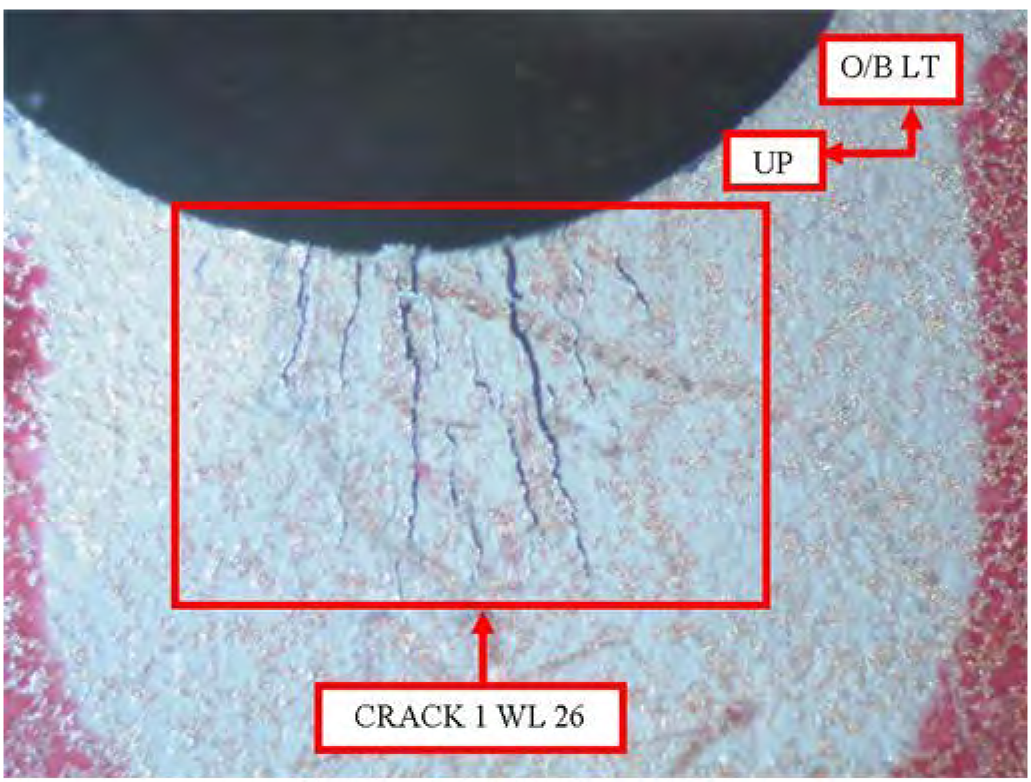
VERTICAL STAB REAR SP WL26 CRK1 Mac

Figure 617. Macroscopic View of Crack 1 on the Vertical Stabilizer Rear Spar WL 26



VERTICAL STAB REAR SP WL26 CRK 1 FLP

Figure 618. Fluorescent Liquid Penetrant Indication of Crack 1 on the Vertical Stabilizer Rear Spar WL 26



VERTICAL STAB REAR SP WL 26 CRK 1 MIC

Figure 619. Microscopic View of Crack 1 on the Vertical Stabilizer Rear Spar WL 26

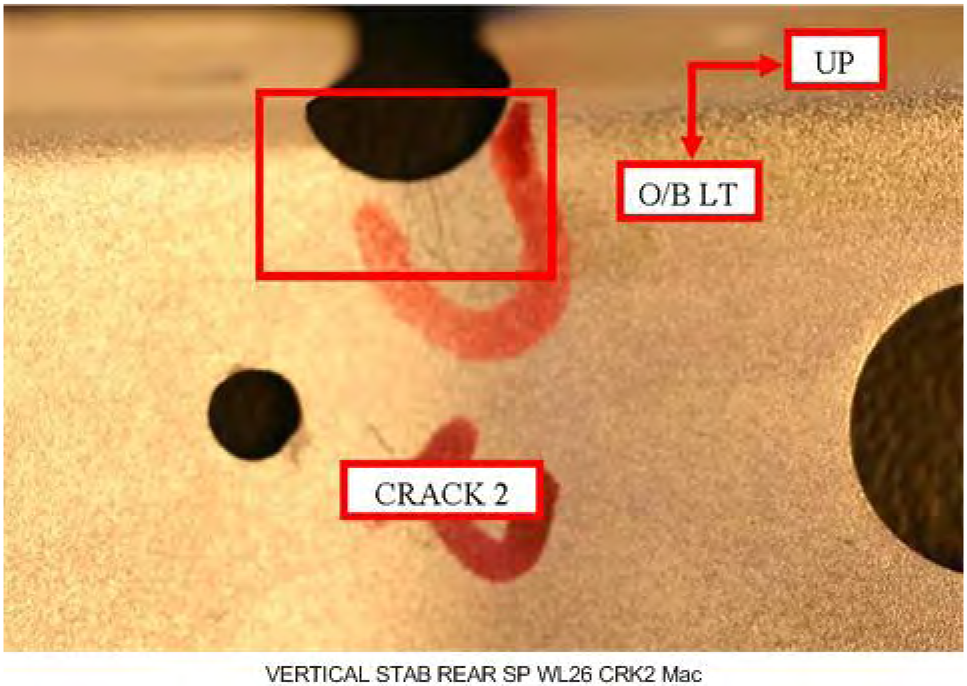


Figure 620. Macroscopic View of Crack 2 on the Vertical Stabilizer Rear Spar WL 26

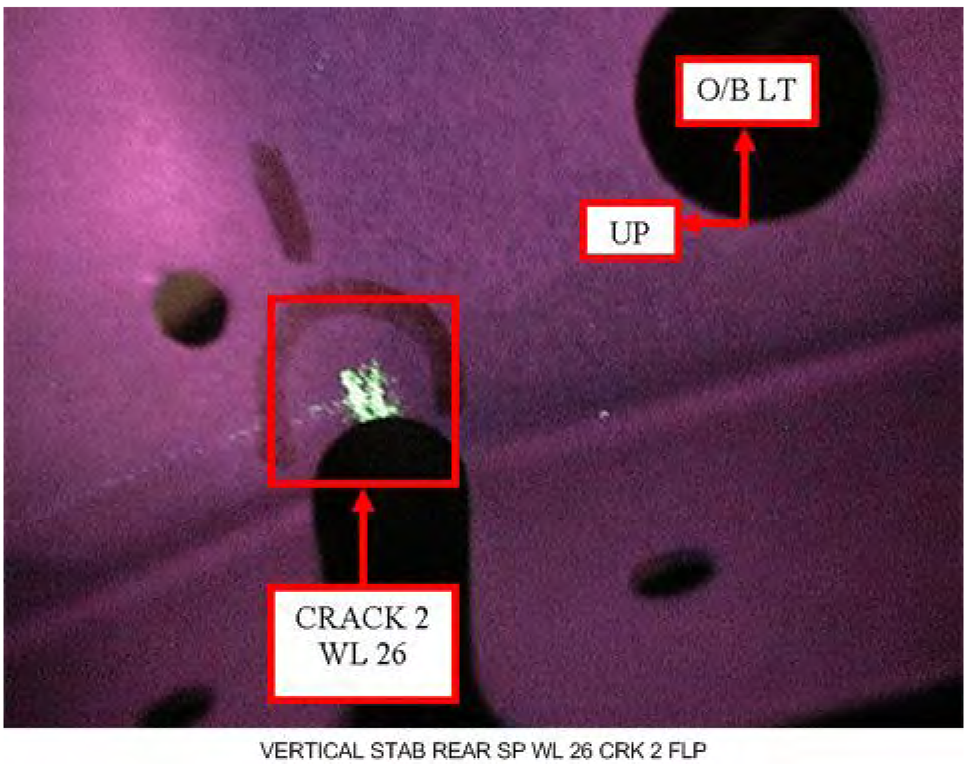


Figure 621. Fluorescent Liquid Penetrant Indication of Crack 2 on the Vertical Stabilizer Rear Spar WL 26

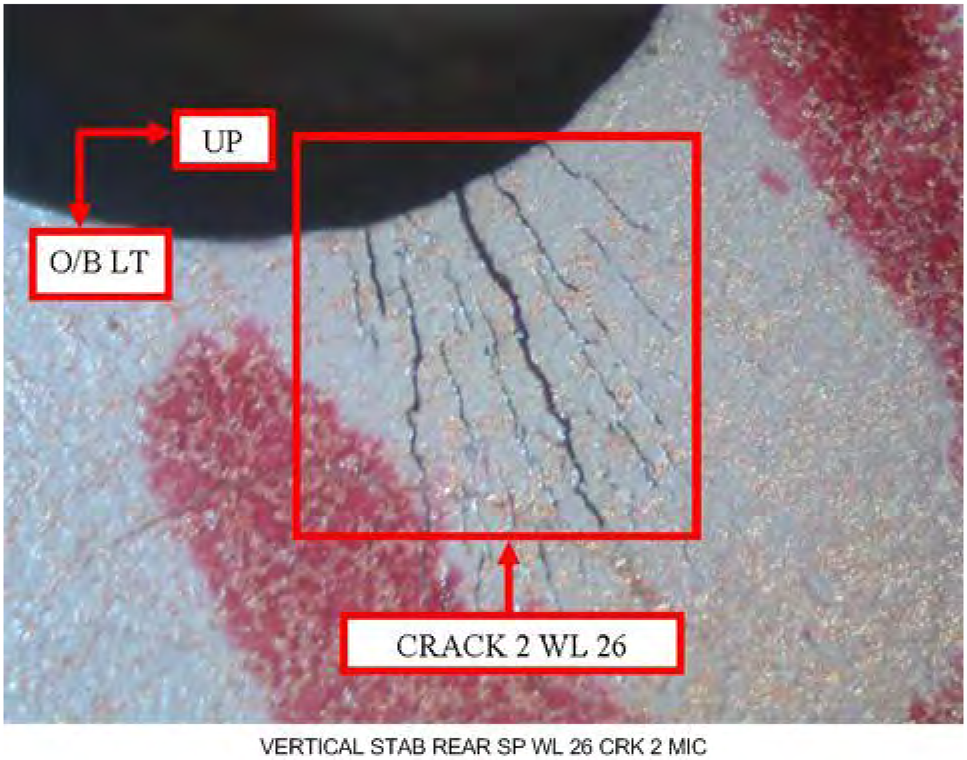


Figure 622. Microscopic View of Crack 2 on the Vertical Stabilizer Rear Spar WL 26

3.4.2.5.3 Vertical Stabilizer Various Indications.

Table 44 provides a detailed characterization of all defects found at locations on the vertical stabilizer other than the spars during the teardown evaluation.

Table 44. Inspection Results From the Remainder of the Vertical Stabilizer

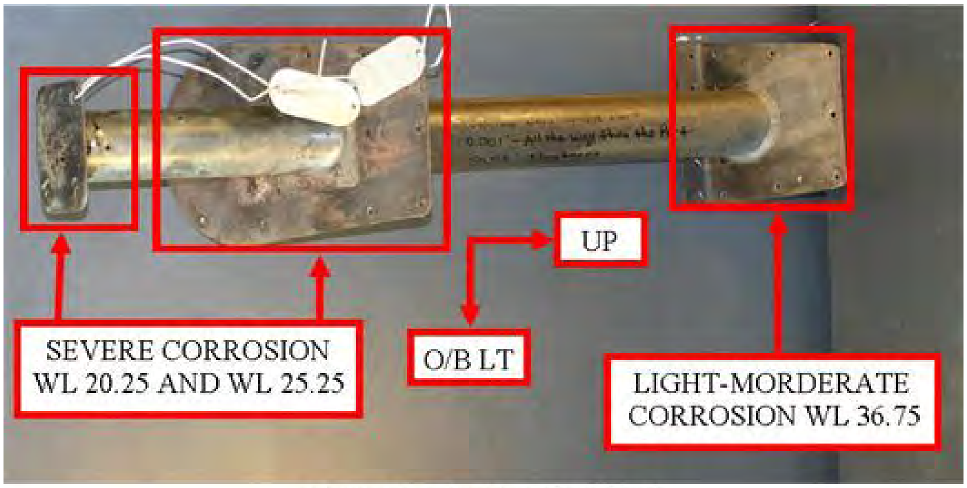
Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Rudder torque tube assembly, figure 624	Corrosion	WL 20.25	2.25 inches by 1.5 inches	Corrosion indication	Severe corrosion 100% thickness loss	625
	Corrosion	WL 25.25	6 inches by 4 inches	Corrosion indication	Severe corrosion 40% thickness loss	626
	Corrosion	WL 36.75	4 inches by 3.5 inches	Corrosion indication	Light-moderate corrosion 4% thickness loss	105
Rudder butt rib, figure 627	Corrosion	Wl 25.75	7.5 inches by 4 inches	Corrosion indication	Severe corrosion 15.6% thickness loss	628
Rudder horn, figure 629	Corrosion	Wl 20.25	1.7 inches by 1.5 inches	No indication	Light-moderate corrosion 2.7% thickness loss	105

Table 44. Inspection Results From the Remainder of the Vertical Stabilizer (Continued)

Part Name and Figure Number	Defect Type	Defect Location	Defect Extent	NDI Indication	Defect Classification	Defect Figure Number
Vertical stabilizer left side gusset, figure 630	Multiple cracks	WL 27	0.054 inch	Crack indication	Bend radii surface crack	537 631
Right rear vertical stabilizer gusset, figure 633	Crack	WL 27	0.147 inch	Crack indication	Surface crack	636
	Multiple cracks	WL 27	0.032 inch	Crack indication	Bend radii Surface crack	637
Vertical stabilizer main bottom rib assembly, figure 638	Crack	WL 25.75	0.397 inch	Crack indication	Bend radii	639 640
	Crack	WL 25.75	0.107 inch	Crack indication	Bend radii	641 642
	Crack	WL 25.75	0.083 inch	Crack indication	Bend radii	641 642
Vertical stabilizer leading edge rib, figure 643	Multiple cracks	WL 30.125	0.410 inch	Crack indication	Bend radii	644 645
	Multiple cracks	WL 30.125	0.370 inch	Crack indication	Surface crack	644 646
	Multiple cracks	WL 30.125	0.2535 inch	Crack indication	Bend radii	647 648
	Multiple cracks	WL 30.125	0.2125 inch	Crack indication	Surface crack	647 649
Main rudder rib, figure 650	Crack	WL 36.75	0.047 inch	Crack indication	Hole crack	651
Rudder lower hinge assembly, figure 652	Hole in weld bead	WL 51.75	0.085 inch	Round indication	N/A	653
Rudder upper hinge, figure 654	Hole in weld bead	WL 82.5	0.055 inch	Round indication	N/A	655
Rudder leading edge upper skin, figure 656	Multiple cracks	WL 96	0.042 inch	No indication	Hole crack	657 658

N/A = Not applicable

The location of light-moderate and severe corrosion on the rudder torque tube assembly, part number 40040-09, is shown in figure 623. A 14-square-inch area of light-moderate corrosion was located at WL 36.75 and caused a localized thickness loss of 4%, and an area of severe corrosion, located at WL 20.25 and shown macroscopically in figure 624, caused areas of complete localized thickness loss. This corrosion covered a surface area of 3.375 square inches. Another area of severe corrosion, covering a surface area of 24-square-inches and resulting in a localized reduction in thickness of 40%, was observed at WL 25.25 and is shown microscopically in figure 625.



RUDDER TORQUE TUBE SV CORR OV

Figure 623. Location of Light-Moderate and Severe Corrosion on the Rudder Torque Tube Assembly WL 20.25 Through WL 36.75

RUDDER TORQUE TUBE SV CORR WL20.25 MAC

Figure 624. Macroscopic View of Severe Corrosion on the Rudder Torque Tube Assembly WL 20.25

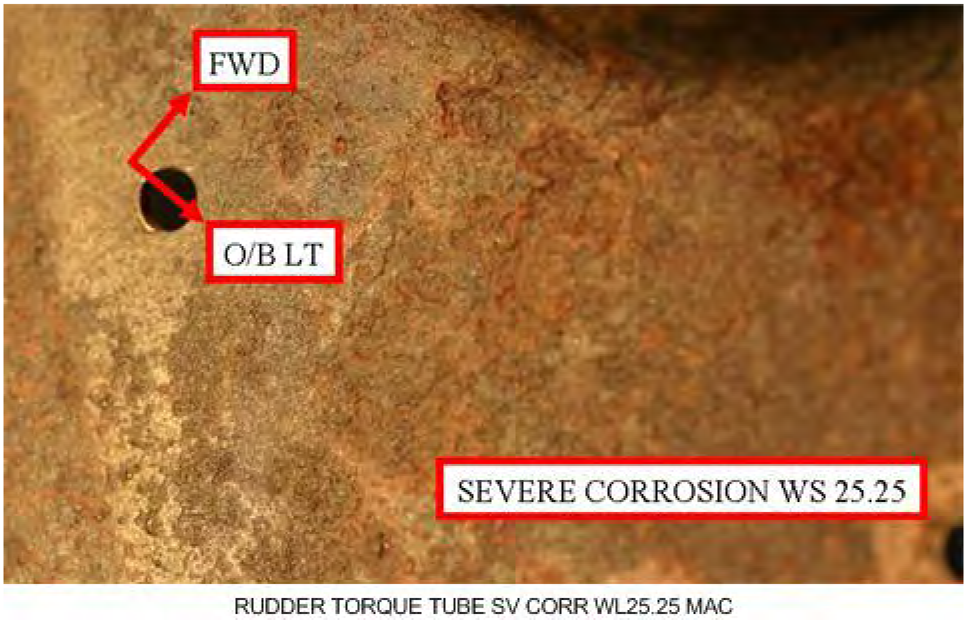


Figure 625. Microscopic View of Severe Corrosion on the Rudder Torque Tube Assembly WL 25.25

The location of severe corrosion on the rudder butt rib, part number 40039-00, is shown in figure 626. This 30-square-inch area of corrosion, shown macroscopically in figure 627, was located at WL 25.75 and caused a maximum reduction in thickness of 15.6%. The location of a 30-square-inch area of light-moderate corrosion on the rudder horn, part number 40253-00, is shown in figure 628. This corrosion caused a localized thickness loss of 2.7%. Figure 629 shows the location of a group of cracks on the vertical stabilizer left side gusset, part number 40170-16. A macroscopic view of this 0.054-inch crack is shown in figure 630, and a microscopic view is shown in figure 631 at WL 27.

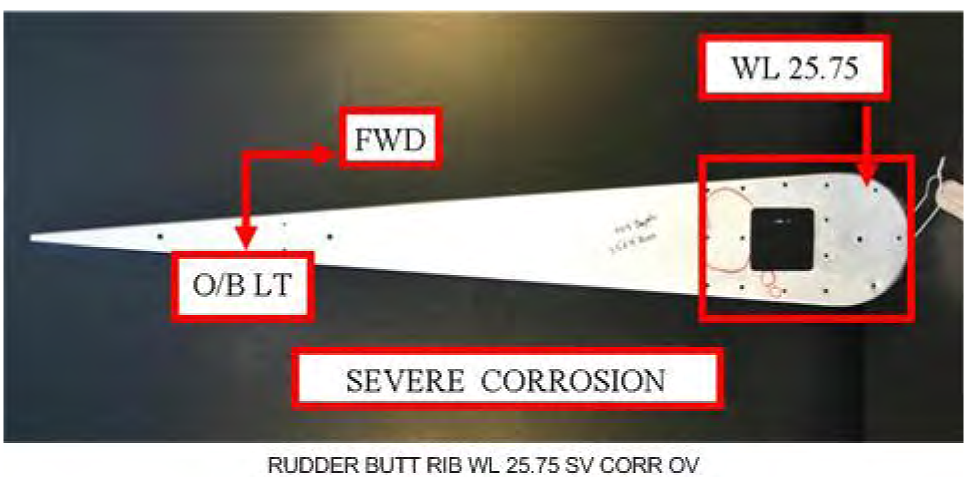


Figure 626. Location of Severe Corrosion on the Rudder Butt Rib WL 25.75

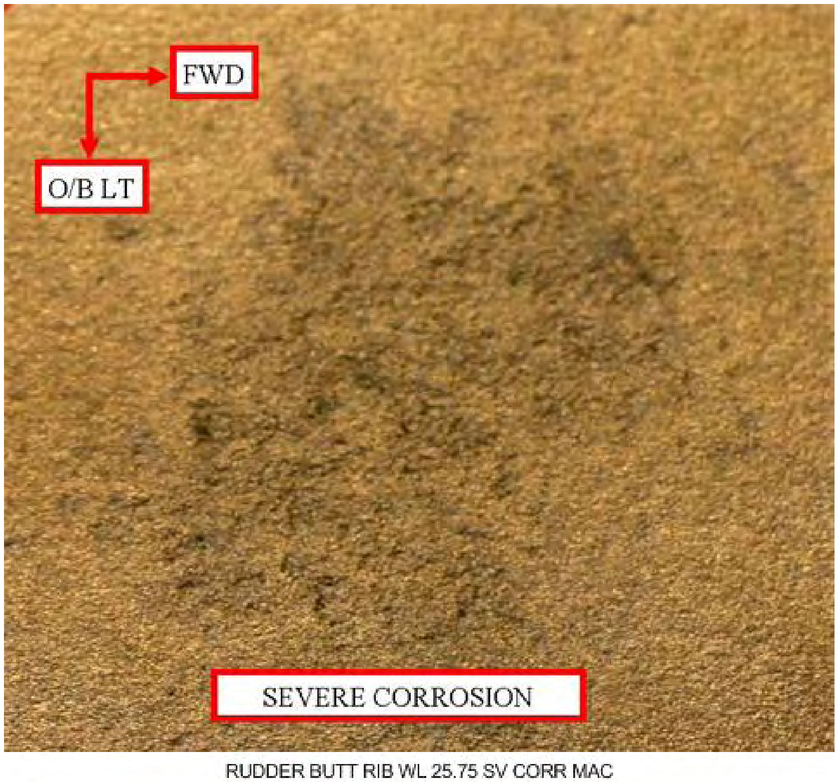


Figure 627. Macroscopic View of Severe Corrosion on the Rudder Butt Rib WL 25.75

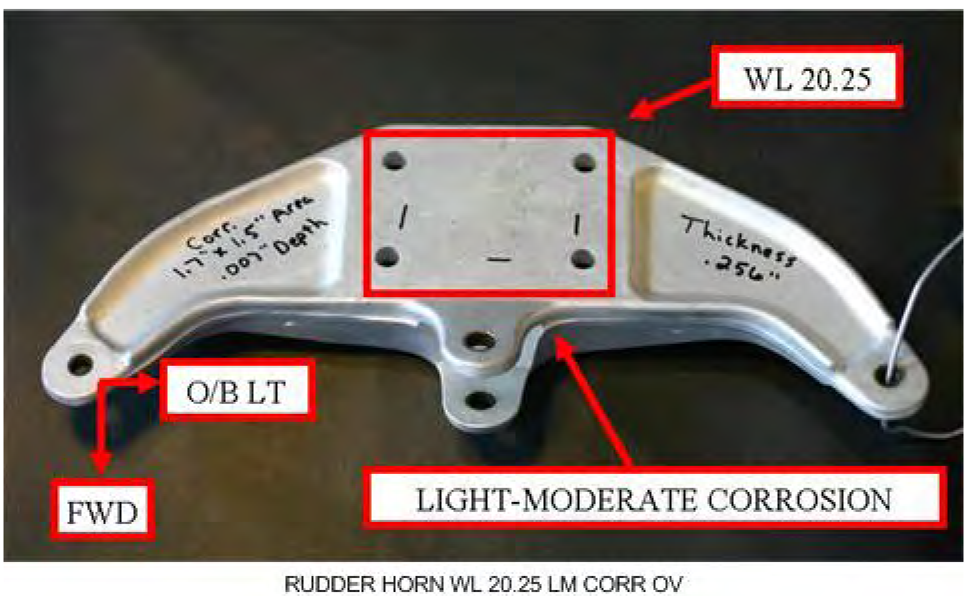
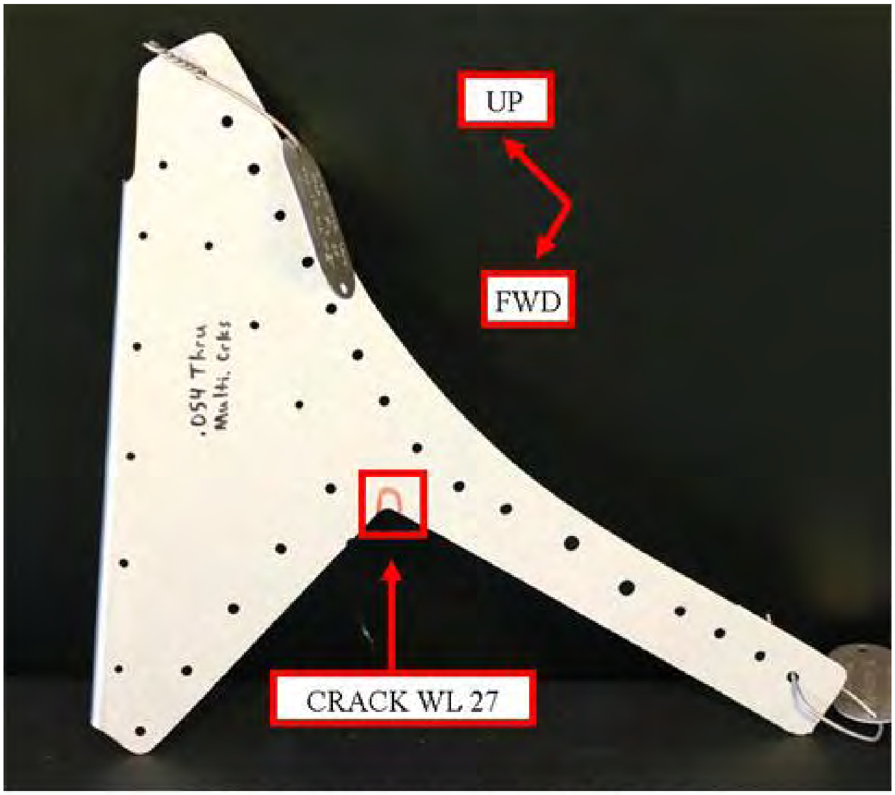
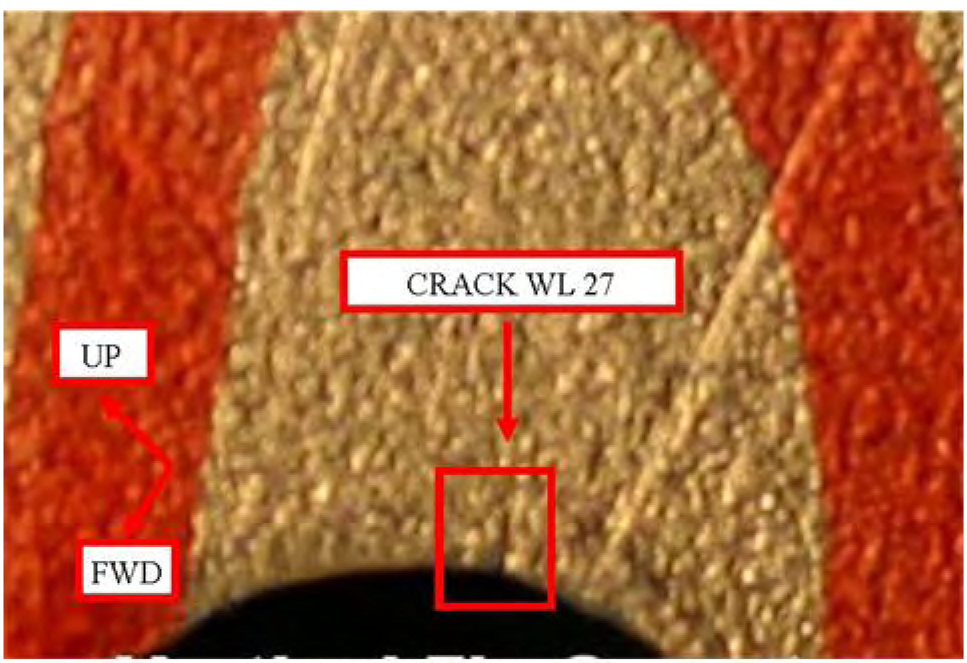


Figure 628. Location of Light-Moderate Corrosion on the Rudder Horn WL 20.25



VERTICAL STAB LT SIDE GUSSET WL 20.25-31.75 OV

Figure 629. Location of Cracks on the Vertical Stabilizer Left Side Gusset WL 20.25 Through WL 31.75



VERTICAL STAB LT SIDE GUSSET WL 27 CRK MAC

Figure 630. Macroscopic View of Crack on the Vertical Stabilizer Left Side Gusset WL 27

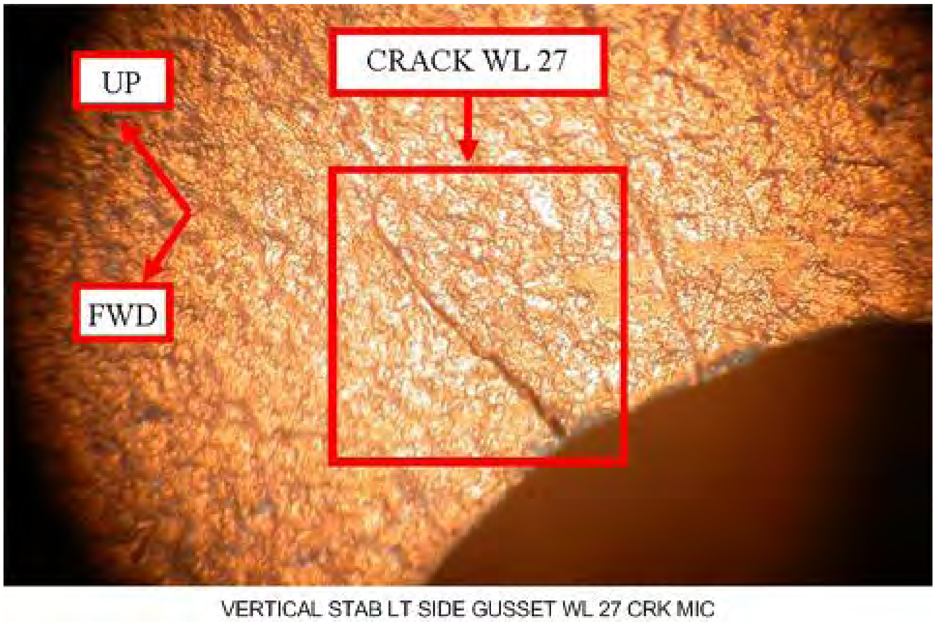


Figure 631. Microscopic View of Crack on the Vertical Stabilizer Left Side Gusset WL 27

Figure 632 shows the location of two cracks on the right rear vertical stabilizer gusset, part number 40170-15. A macroscopic view of crack 1, which measured 0.147 inch, and crack 2, which measured 0.032 inch, is shown in figure 633. The fluorescent liquid penetrant indications for both cracks is shown in figure 634. A microscopic view of crack 1 is shown in figure 635. A microscopic view of crack 2, which was located at WL 27, is shown in figure 637.

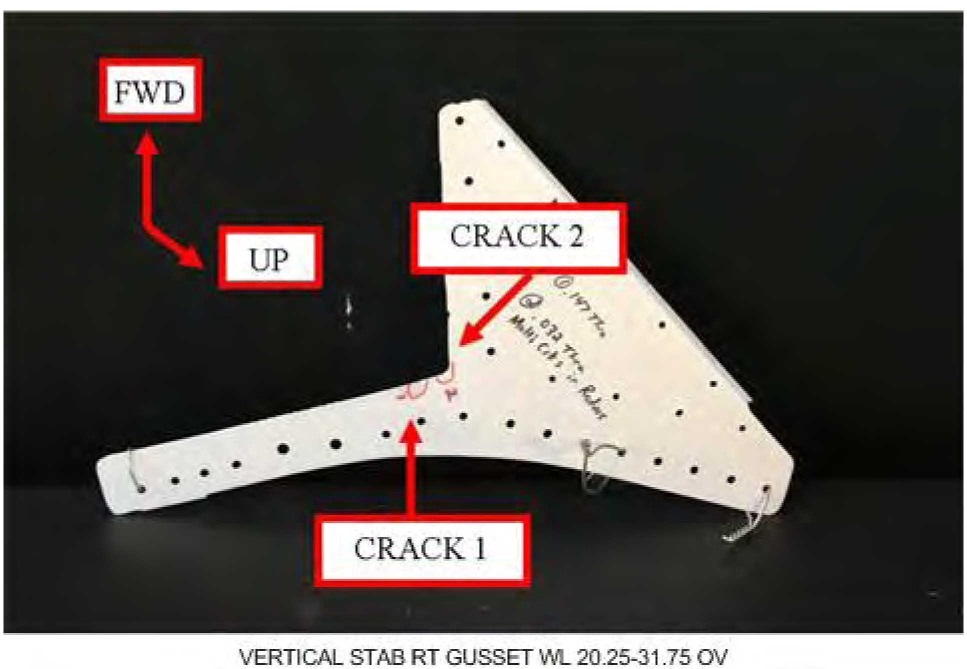
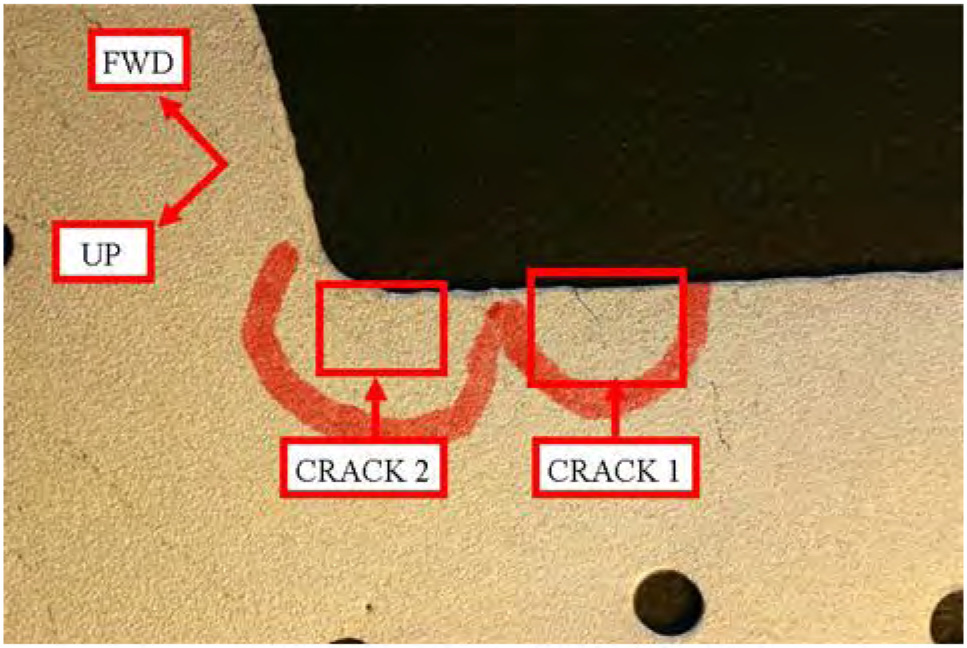
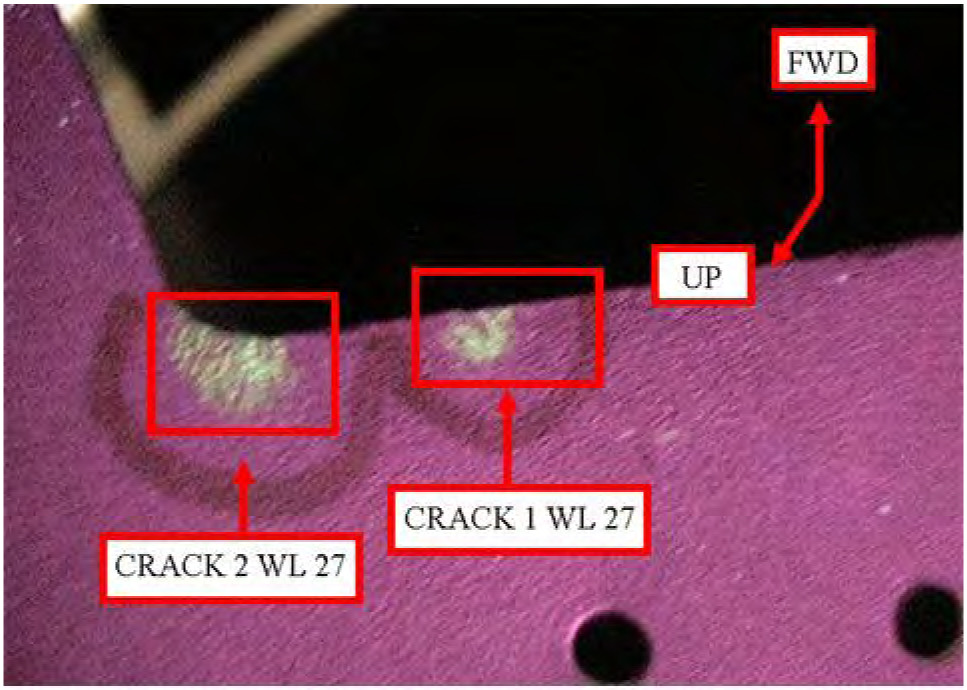


Figure 632. Location of Two Cracks on the Right Rear Vertical Stabilizer Gusset WL 20.25 Through WL 31.75



VERTICAL STAB GUSSET Rt WL27 CRK1&2 MAC

Figure 633. Macroscopic View of Cracks 1 and 2 on the Right Rear Vertical Stabilizer Gusset WL 27



VERTICAL STAB RT GUSSET WL 27 FLP

Figure 634. Fluorescent Liquid Penetrant Indication on the Right Rear Vertical Stabilizer Gusset WL 27

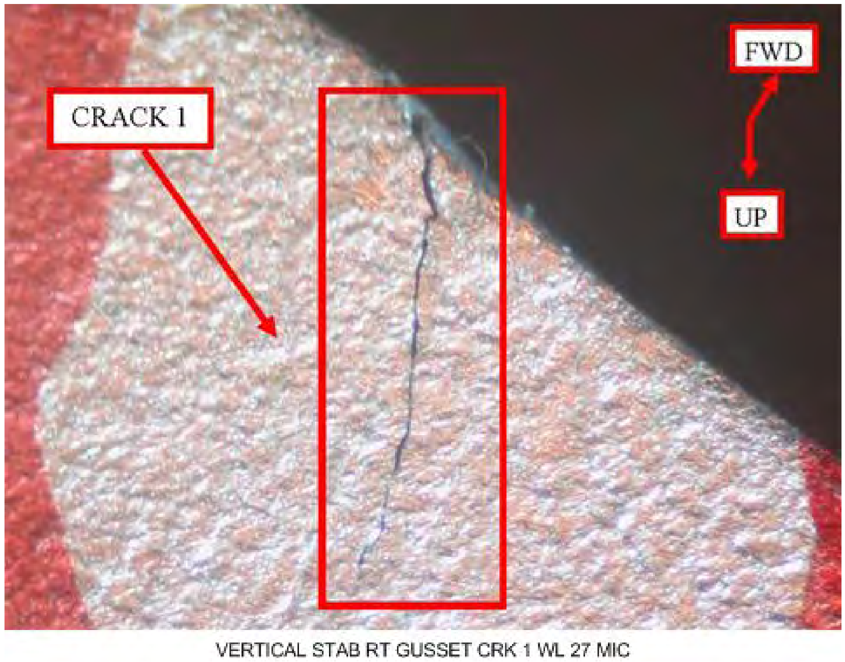


Figure 635. Microscopic View of Crack 1 on the Right Rear Vertical Stabilizer Gusset WL 27

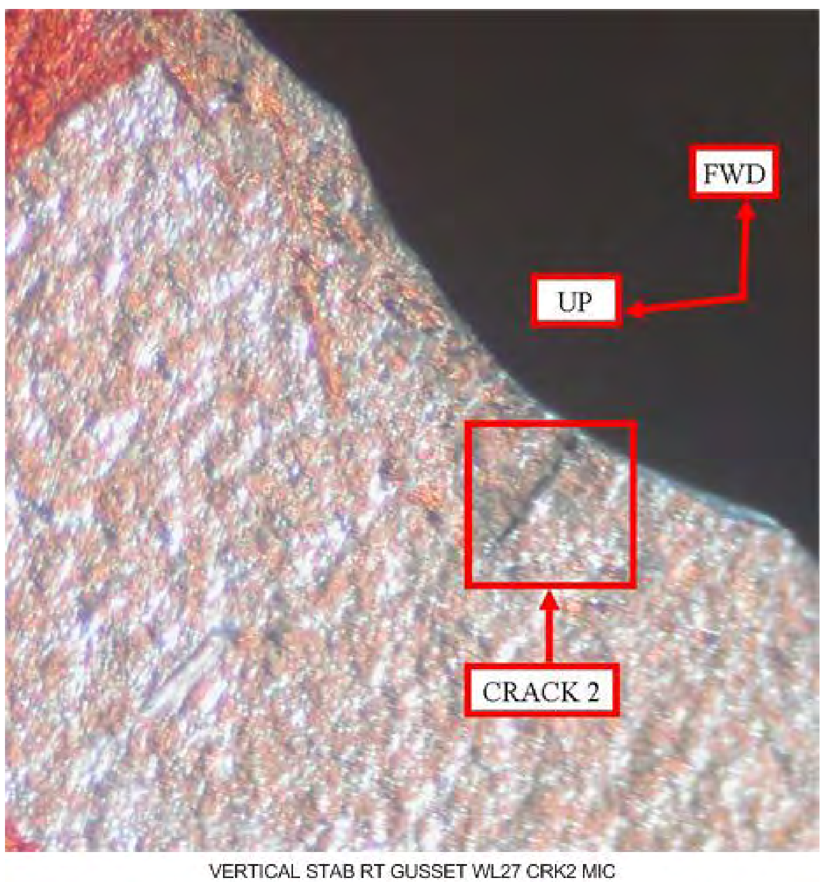


Figure 636. Microscopic View of Crack 2 on the Right Rear Vertical Stabilizer Gusset WL 27

The location of three cracks on the vertical stabilizer main bottom rib assembly, part number 44294-00, is shown in figure 637 at WL 25.75. A macroscopic view of crack 1 is shown in figure 638, and a microscopic view of this 0.397-inch crack is shown in figure 639. Figure 640 shows a macroscopic view of cracks 2 and 3 on the main bottom rib assembly. A microscopic view of crack 2, which measured 0.107 inch, and crack 3, which measured 0.083 inch is shown in figure 641.

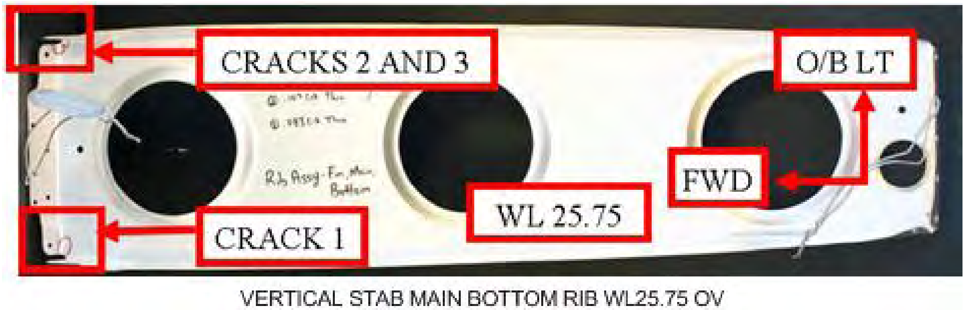


Figure 637. Location of Three Cracks on the Vertical Stabilizer Main Bottom Rib Assembly WL 25.75

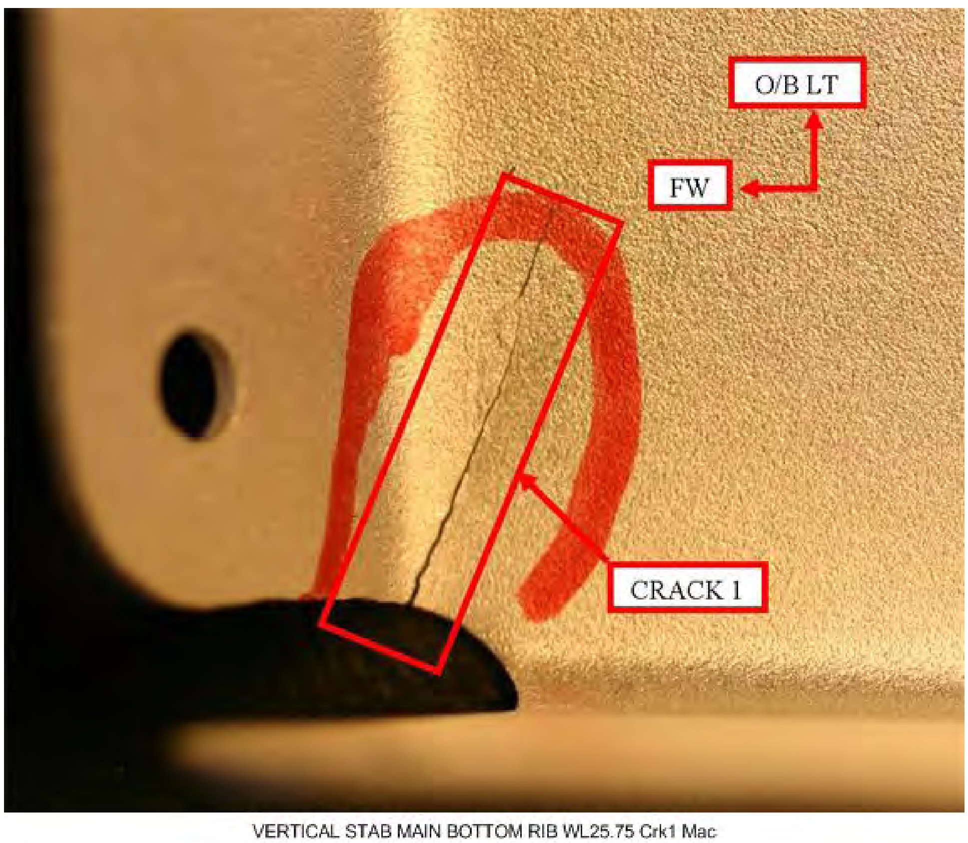
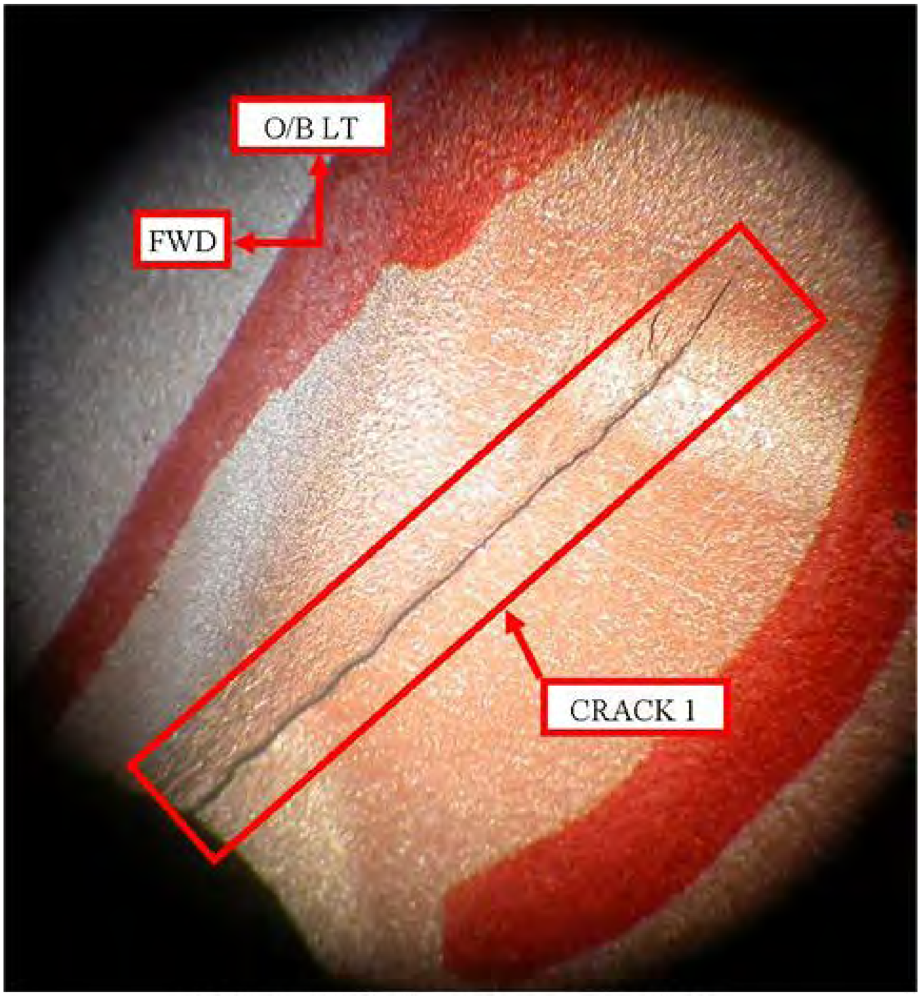
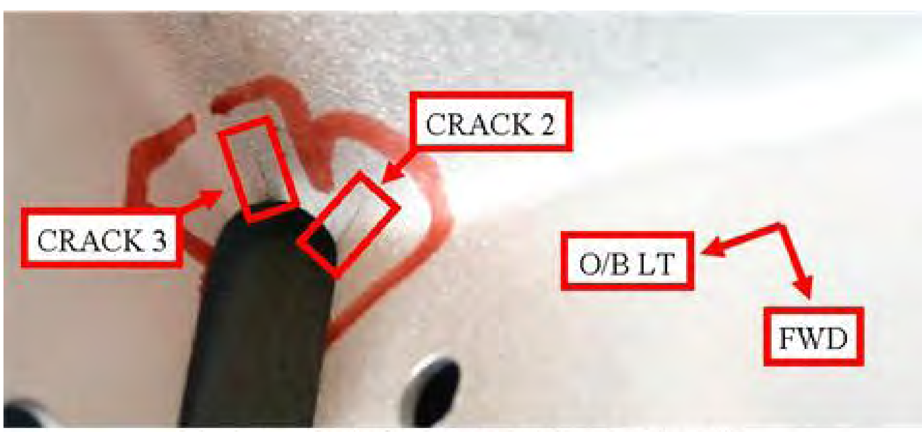


Figure 638. Macroscopic View of Crack 1 on the Vertical Stabilizer Main Bottom Rib Assembly WL 25.75



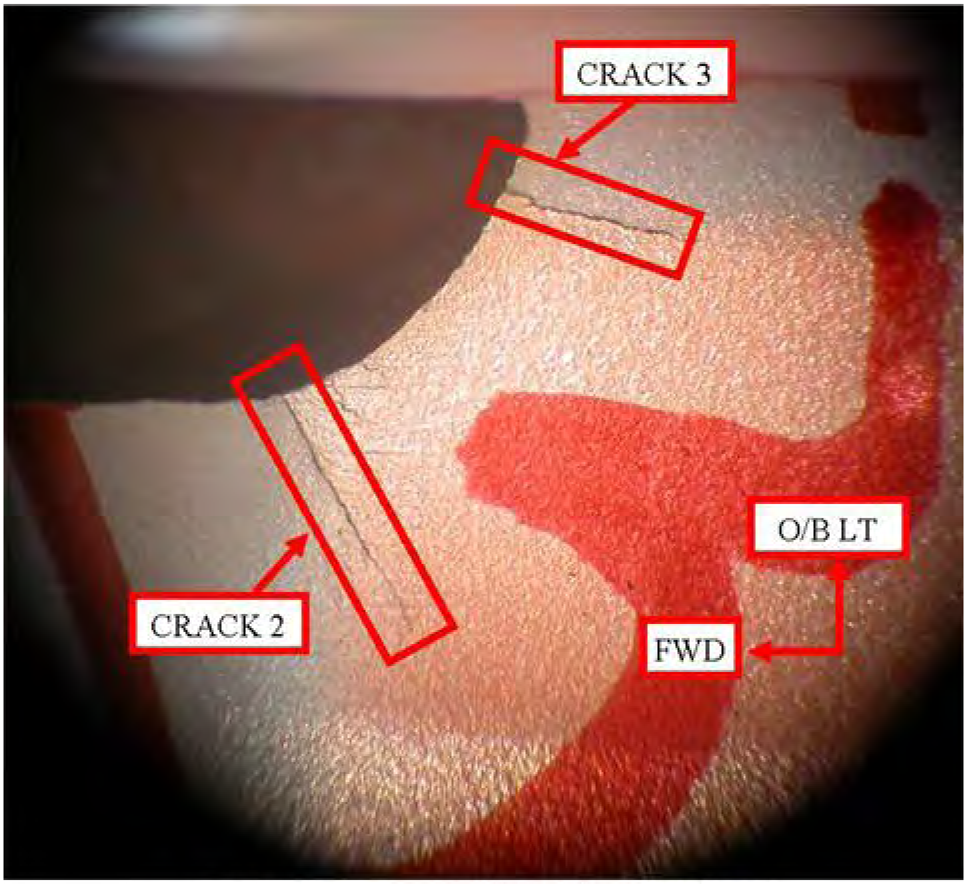
VERTICAL STAB MAIN BOTTOM RIB WL25.27 CRK1 MIC

Figure 639. Microscopic View of Crack 1 on the Vertical Stabilizer Main Bottom Rib Assembly WL 25.75



VERTICAL STAB MAIN BOTTOM RIB WL25.75 CRK2&3 MAC

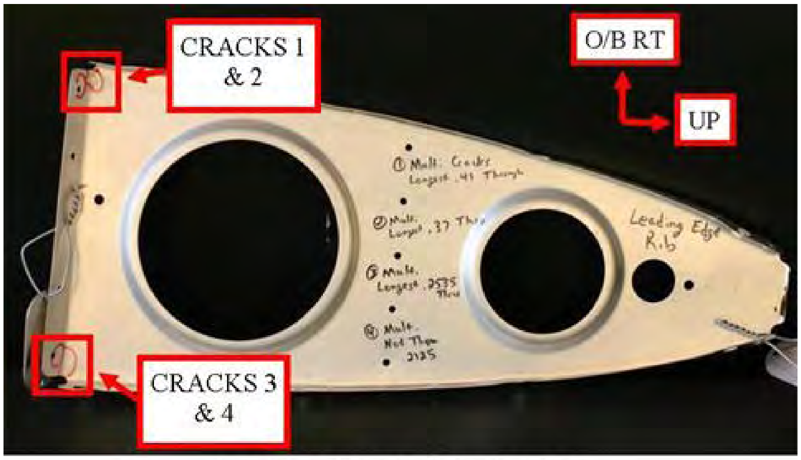
Figure 640. Macroscopic View of Cracks 2 and 3 on the Vertical Stabilizer Main Bottom Rib Assembly WL 25.75



VERTICAL STAB MAIN BOTTOM RIB WL25.75 CRK 2&3 MIC

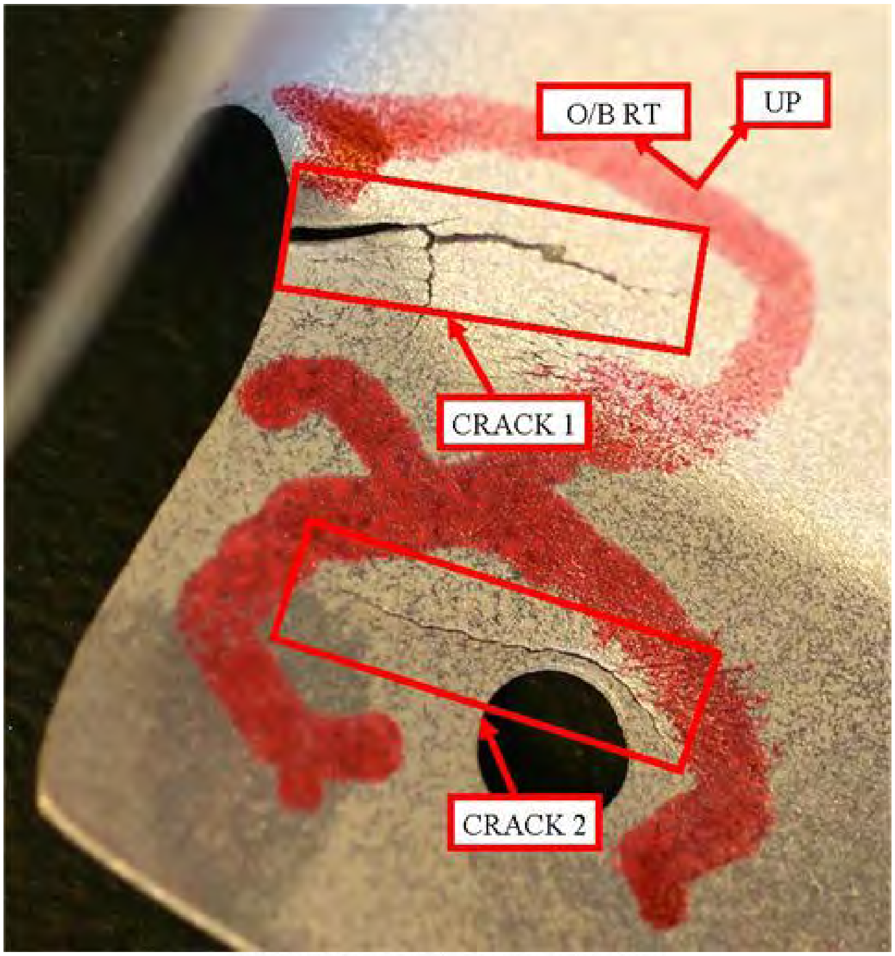
Figure 641. Microscopic View of Cracks 2 and 3 on the Vertical Stabilizer Main Bottom Rib Assembly WL 25.75

The location of four cracks on the vertical stabilizer leading-edge rib, part number 40096-00, is shown in figure 642 at WL 30.125. A macroscopic view of cracks 1 and 2 is shown in figure 643. Figure 644 shows a microscopic view of crack 1, which measures 0.410 inch, and figure 645 shows a microscopic view of crack 2, which measures 0.370 inch. A macroscopic view of cracks 3 and 4 is shown in figure 646. A microscopic view of crack 3, which measures 0.2535 inch, is shown in figure 647. Figure 648 shows a 0.2125-inch crack, crack 4, microscopically.



VERTICAL STAB LE RIB WL30.125-44.75 OV

Figure 642. Location of Four Cracks on the Vertical Stabilizer Leading-Edge Rib WL 30.125 Through WL 44.75



VERTICAL STAB LE Rib WL30.125 Crk1-2 MAC

Figure 643. Macroscopic View of Cracks 1 and 2 on the Vertical Stabilizer Leading-Edge Rib WL 30.125

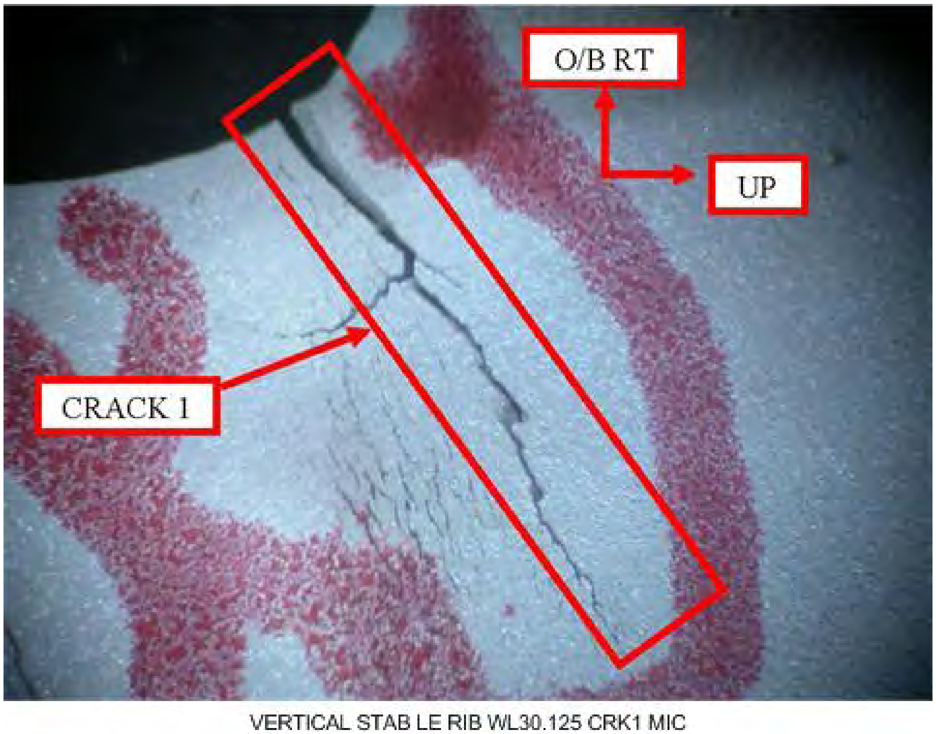


Figure 644. Microscopic View of Crack 1 on the Vertical Stabilizer Leading-Edge Rib WL 30.125

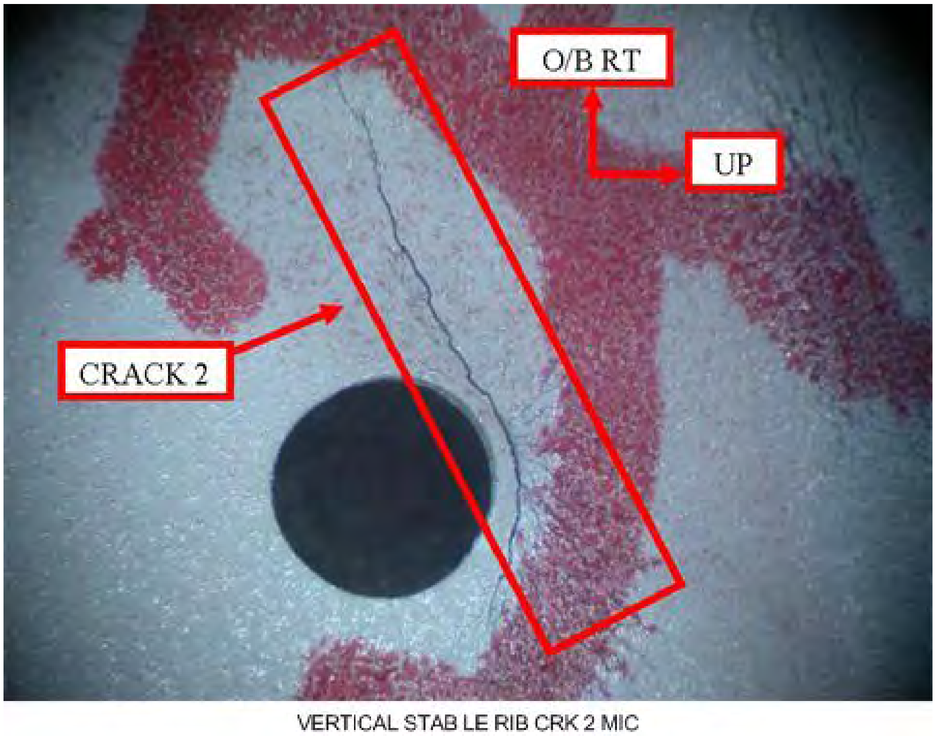


Figure 645. Microscopic View of Crack 2 on the Vertical Stabilizer Leading-Edge Rib WL 30.125

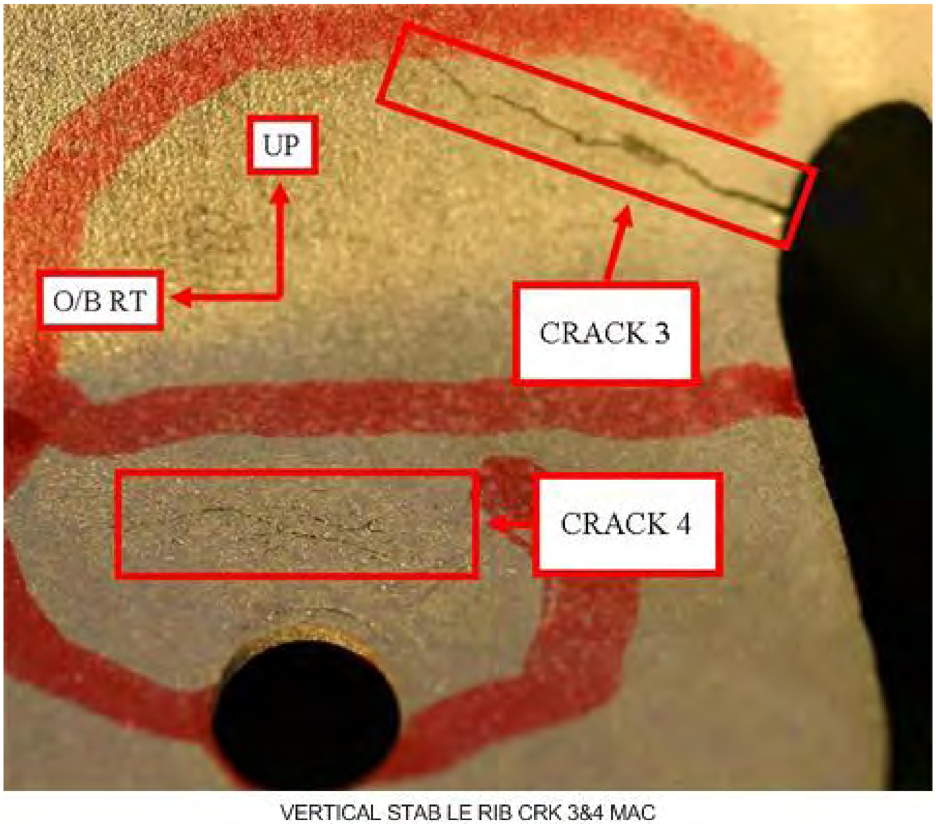


Figure 646. Macroscopic View of Cracks 3 and 4 on the Vertical Stabilizer Leading-Edge Rib WL 30.125

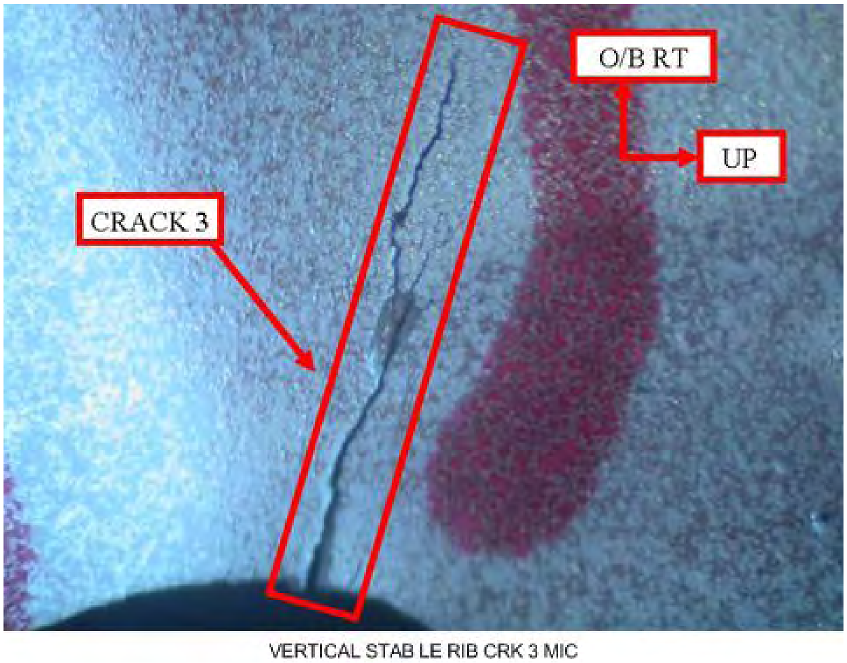


Figure 647. Microscopic View of Crack 3 on the Vertical Stabilizer Leading-Edge Rib WL 30.125

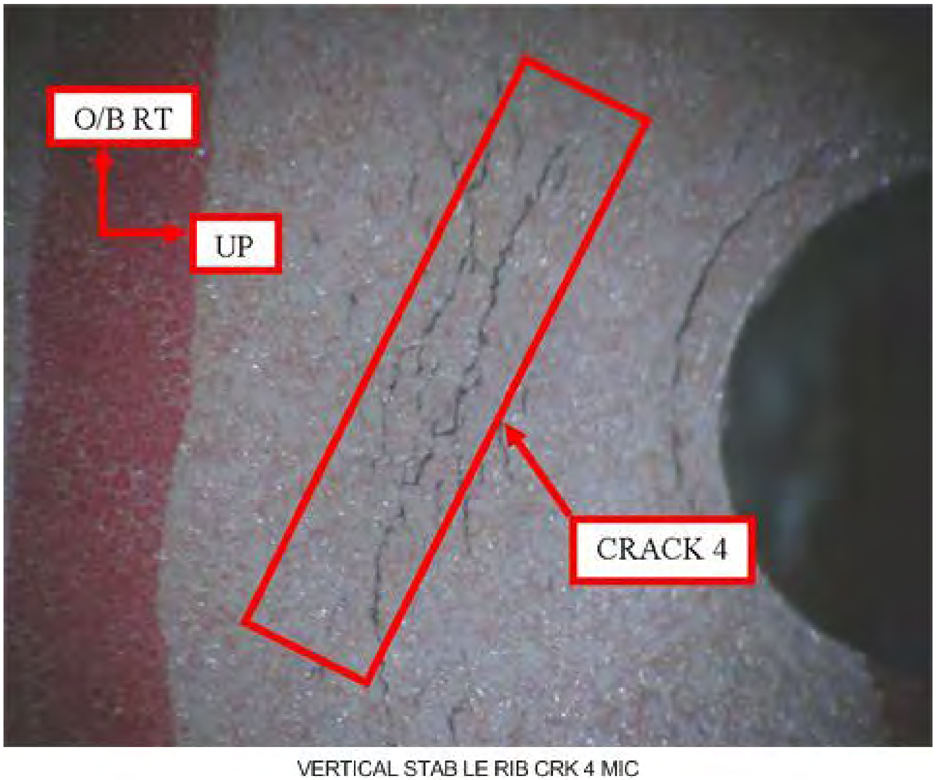


Figure 648. Microscopic View of Crack 4 on the Vertical Stabilizer Leading-Edge Rib WL 30.125

The location of a 0.047-inch crack on the main rudder rib, part number 40078-3, is shown in figure 649. This crack, which was located at WL 36.75, is shown macroscopically in figure 650. The location of a hole in the weld bead of the rudder lower hinge assembly, part number 51109-04, is shown in figure 651, and a macroscopic view of the 0.085-inch-diameter inclusion is shown in figure 652. Figure 653 shows the location of a similar hole in a weld bead, with a diameter of 0.055 inch, in the rudder upper hinge, part number 51108-04. A microscopic view of this inclusion, located at WL 82.5, is shown in figure 654. The location of multiple cracks grouped together and identified as a single crack on the rudder leading-edge upper skin, part number 40046-04, is shown in figure 655. A macroscopic view of this crack group is shown in figure 656. The maximum length of the crack was measured as 0.042 inch and was located at WL 96. A microscopic view of this crack group is shown in figure 657.

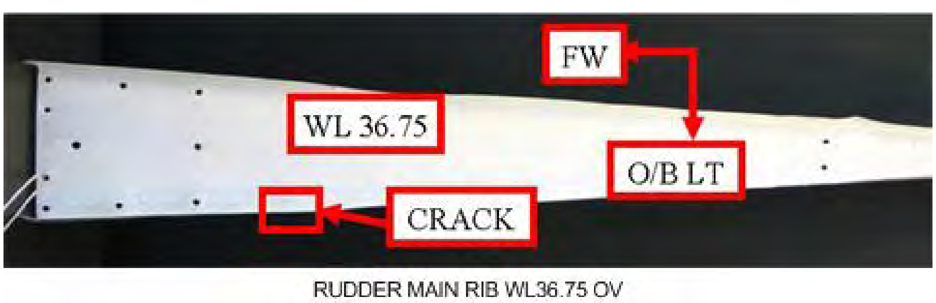


Figure 649. Location of Crack on the Main Rudder Rib WL 36.75

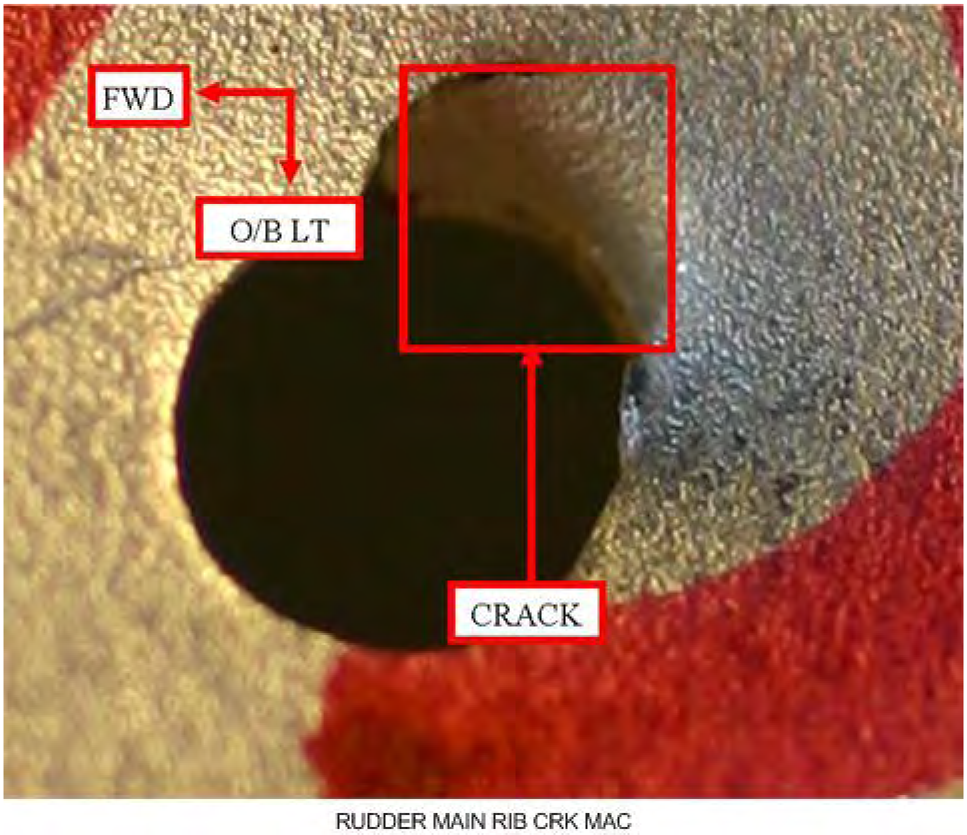


Figure 650. Macroscopic View of Crack on the Main Rudder Rib WL 36.75

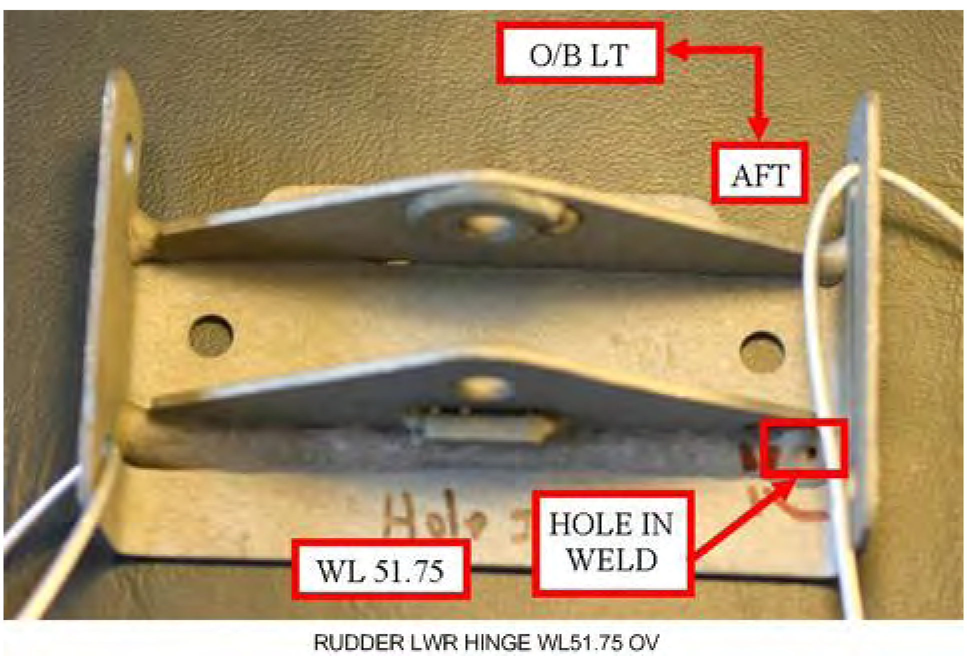


Figure 651. Location of Hole in the Weld Bead on the Rudder Lower Hinge Assembly WL 51.75

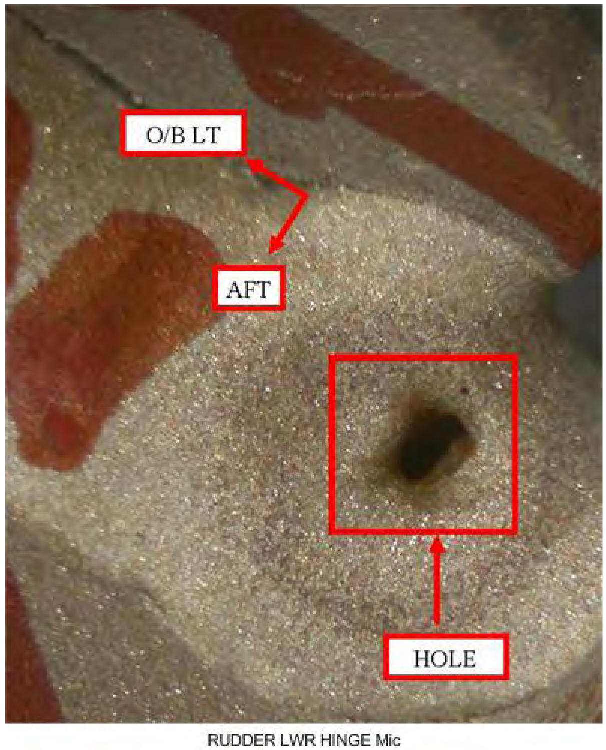


Figure 652. Macroscopic View of Hole in the Weld Bead on the Rudder Lower Hinge Assembly WL 51.75

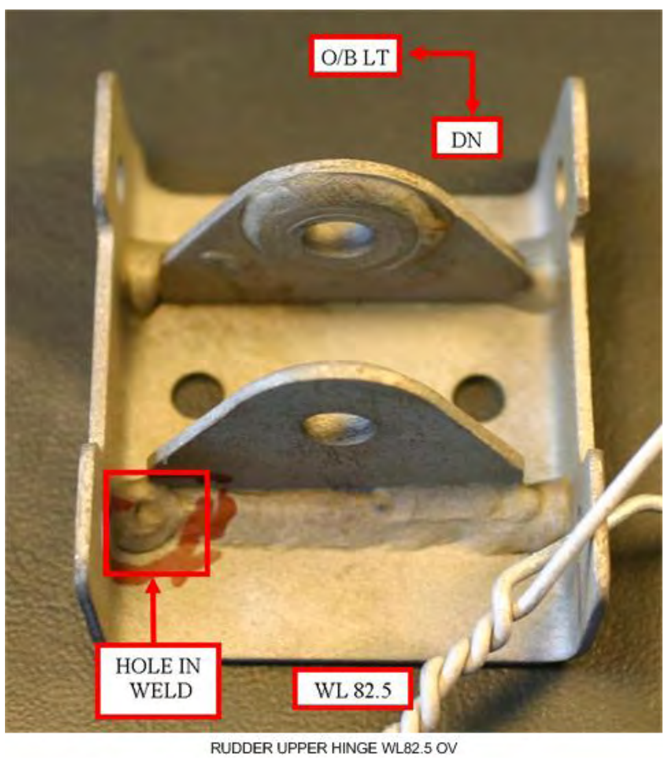
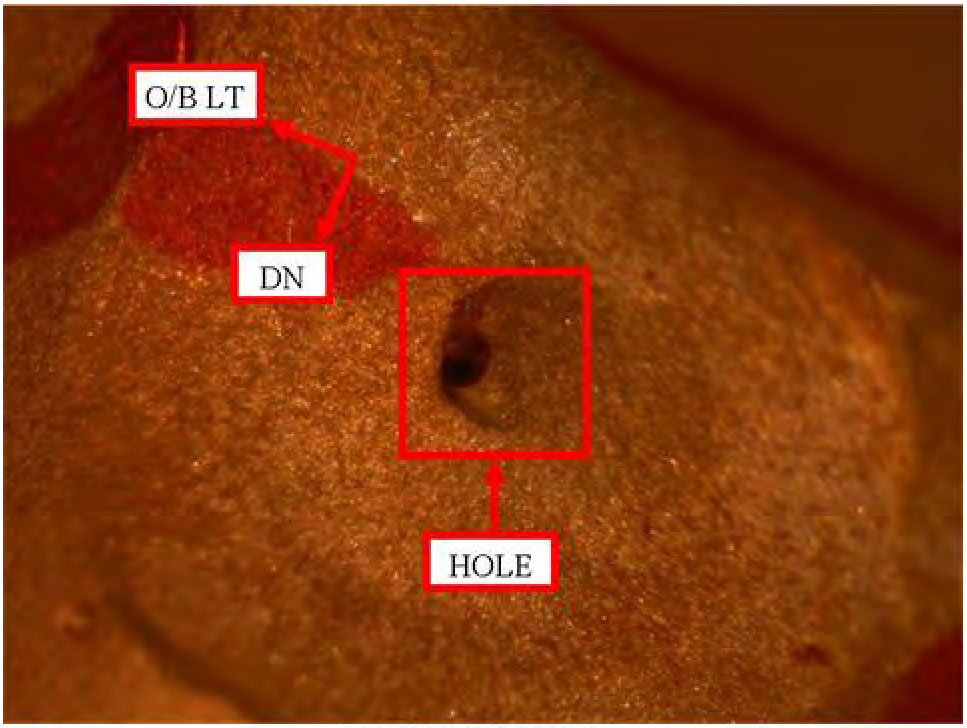
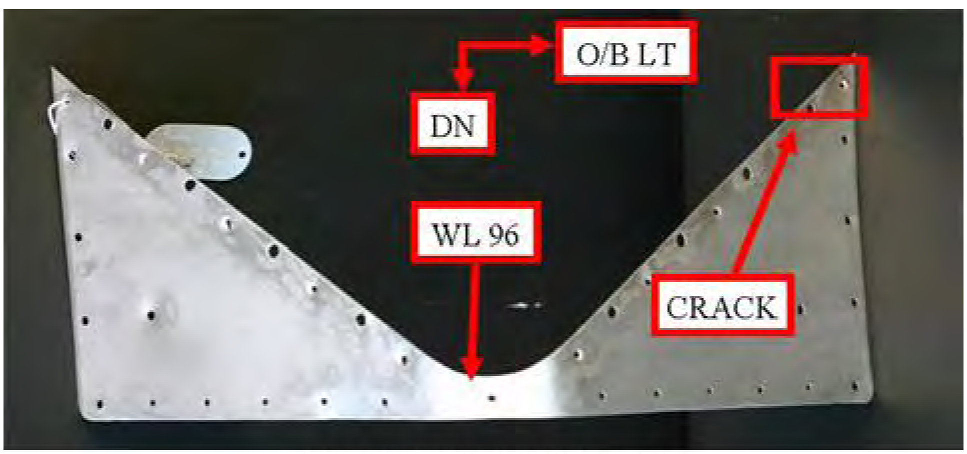


Figure 653. Location of Hole in the Weld Bead on the Rudder Upper Hinge WL 82.5



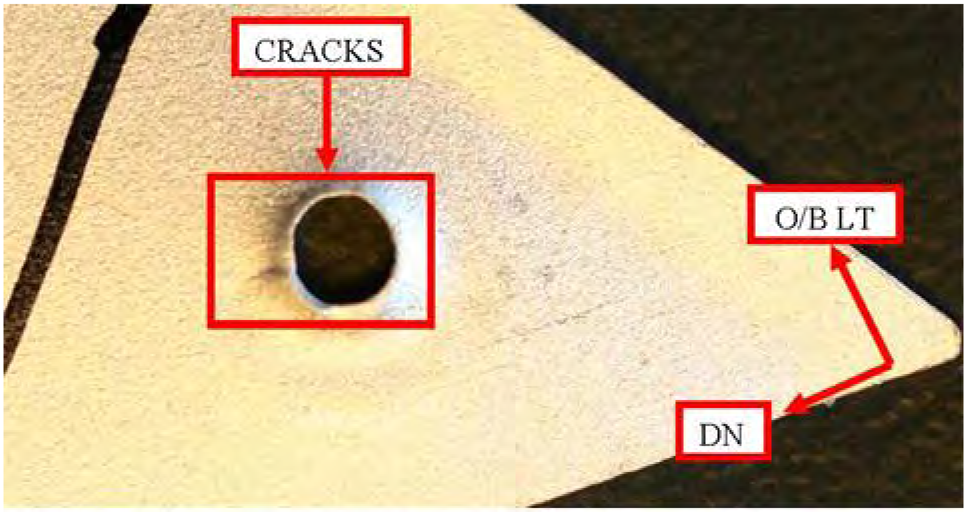
RUDDER UPPER HINGE Mac

Figure 654. Macroscopic View of Hole in the Weld Bead on the Rudder Upper Hinge WL 82.5



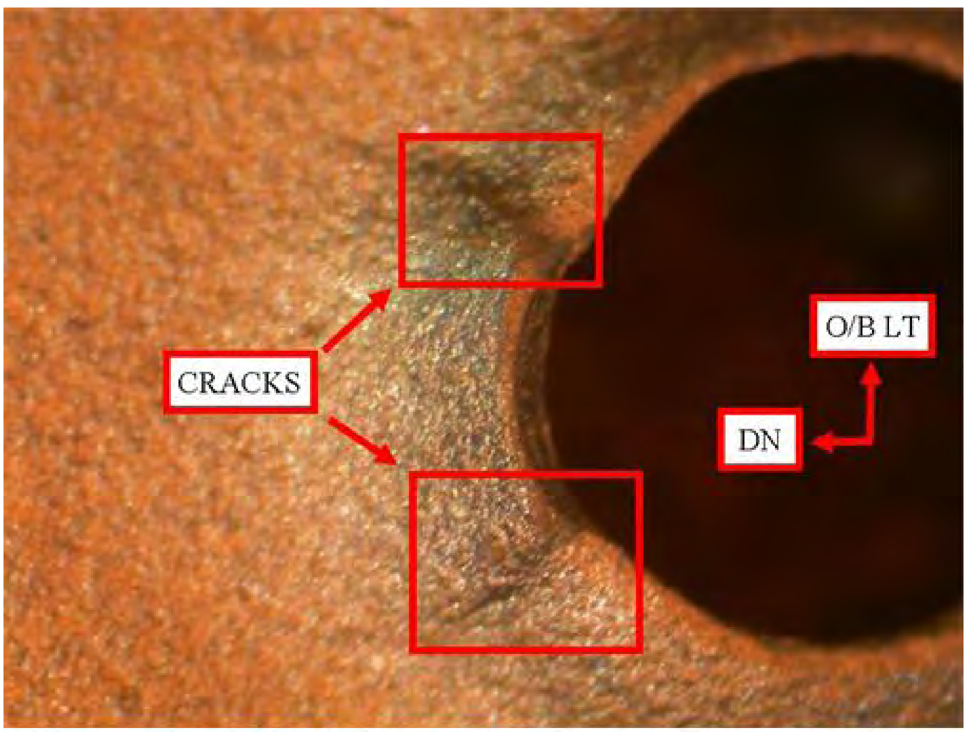
RUDDER LE UPPER SKIN WL96 OV

Figure 655. Location of Crack on the Rudder Leading-Edge Upper Skin WL 96



RUDDER LE UPPER SKIN CRK MAC

Figure 656. Macroscopic View of Crack on the Rudder Leading-Edge Upper Skin WL 96



RUDDER LE UPPER SKIN WL96 CRK MIC

Figure 657. Microscopic View of Cracks on the Rudder Leading-Edge Upper Skin WL 96

3.5 INTRUSIVE WIRING VISUAL INSPECTION.

The goal of performing an intrusive visual inspection on the wiring of this airplane is to evaluate the condition of the wiring located in inaccessible areas for flaws or findings. Coupled with the routine visual inspections performed during the inspection phase, the intrusive visual inspections allow all wiring of the airplane to be assessed for condition, since all wires were removed from the airplane during the teardown evaluation phase. This also allows for a comparison of which findings could be found during routine maintenance and which findings were inaccessible to maintenance personnel.

3.5.1 Inspection Procedures.

Once all wires and wiring harnesses were removed from the airplane during detailed disassembly, each wire was visually inspected for wiring condition, repair, termination, and connector findings. Every finding on each wire was labeled with the wire identification number and a finding code. Table 45 lists all finding codes, which were designed in accordance with the wiring inspection/practices training presentation of the FAA academy to aid in the cataloging of multiple findings on a particular wire, as well as a description of each finding code. Occasionally, it was necessary to cut a wire or wire harness to remove it from the airplane. A visual inspection was performed in the affected area prior to any cut. All inspections were performed in accordance with Advisory Circular 43.13.1B and the Piper Navajo Chieftain Service Manuals.

Table 45. Intrusive Wiring Visual Inspection Finding Codes

Finding Type	Finding Description	Finding Code
Wiring Condition	Rubbing/chafing of outer insulation	01A
	Cutting through outer insulation	01B
	Exposed shield	01C
	Damaged shield	01D
	Chafing/cutting of inner insulation	01E
	Exposed inner conductor	01F
	Damaged inner conductor	01G
	Heat damage	01H
	Fluid/chemical/dust contamination	01J
	Corroded shield/conductors	01K
	Label not readable	01L
	Insulation brittle and cracking	01M
	Wire crushed	01P
Repair Condition	Spliced corroded	02A
	Fluid/chemical/dust contamination	02B
	Splice inner conductor exposed	02C
	Spliced improperly	02D
	Loose crimp	02E
	+10% of conductor cut or broken	02F
	Cold solder joint	02G
	Solder joint exposed	02H
	Solder joint corroded	02J
	Not labeled properly	02K
	Unused wires improperly terminated	02L

Table 45. Intrusive Wiring Visual Inspection Finding Codes (Continued)

Finding Type	Finding Description	Finding Code
Termination Condition	Loose/broken terminals	03B
	Corroded terminals	03C
	Improperly termination	03E
Connector Condition	Insert damage/deterioration	04A
	Contact arcing/fretting	04B
	Missing/damaged/loose back shells	04C
	Missing hardware	04D
	Corroded	04E
	Fluid/chemical/dust contamination	04F

3.5.2 Inspection Results.

During the intrusive visual inspections, 1022 findings were documented; some were minor findings, and others were critical findings. Critical findings included exposed shields, exposed inner conductors, damaged inner conductions, heat-damaged wires, spliced inner conductors exposed, cold solder joints, exposed solder joints, loose or broken terminals, and contact arcing or fretting. Of the 1022 findings documented during the intrusive visual inspections, 117 were categorized as critical findings, as shown in figure 658. Figure 659 shows a breakdown of wiring findings by finding code. Eighteen percent of the wiring findings were rubbing/chafing of the outer insulation, with an additional eighteen percent classified as fluid contamination and another 17 percent identified as brittle and cracking insulation. Figure 660 shows the intrusive wiring visual inspection of findings by airplane systems. Twenty-two percent of the findings were related to the heater-ventilating and deicing system, and fifteen percent were part of the airplane power system. Another fifteen percent of the findings were related to the landing gear system.

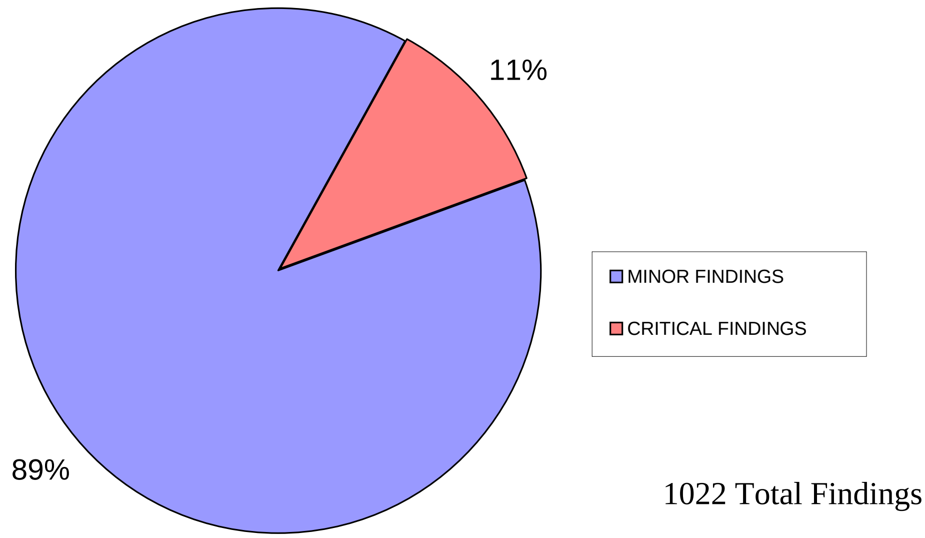


Figure 658. Critical Versus Minor Findings From the Intrusive Wiring Visual Inspections

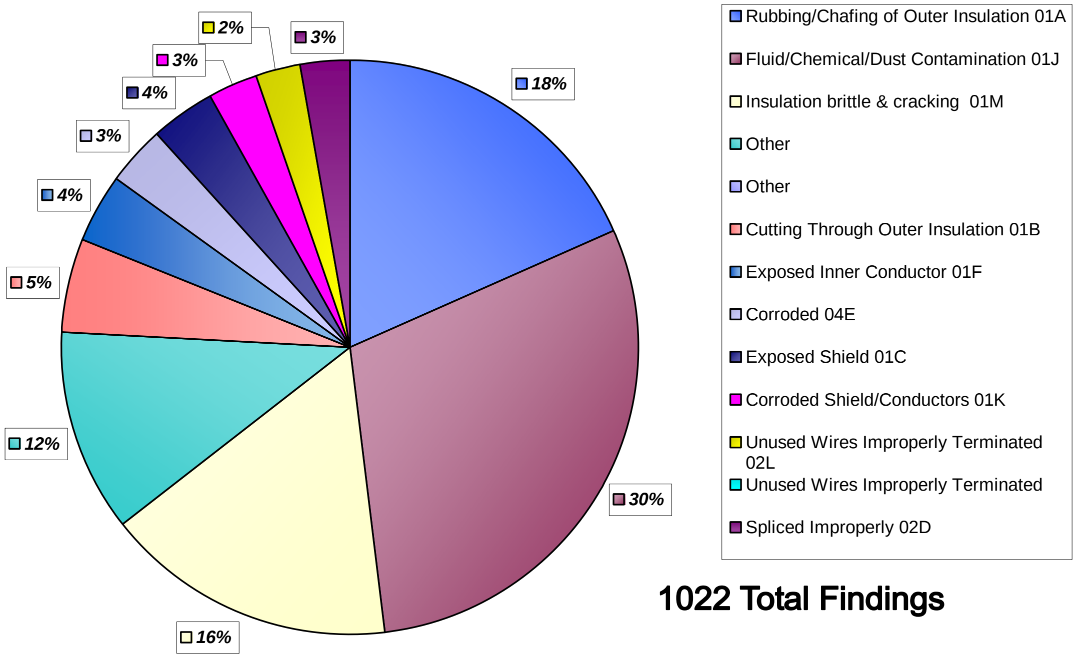


Figure 659. Breakdown of Wiring Findings by Finding Code

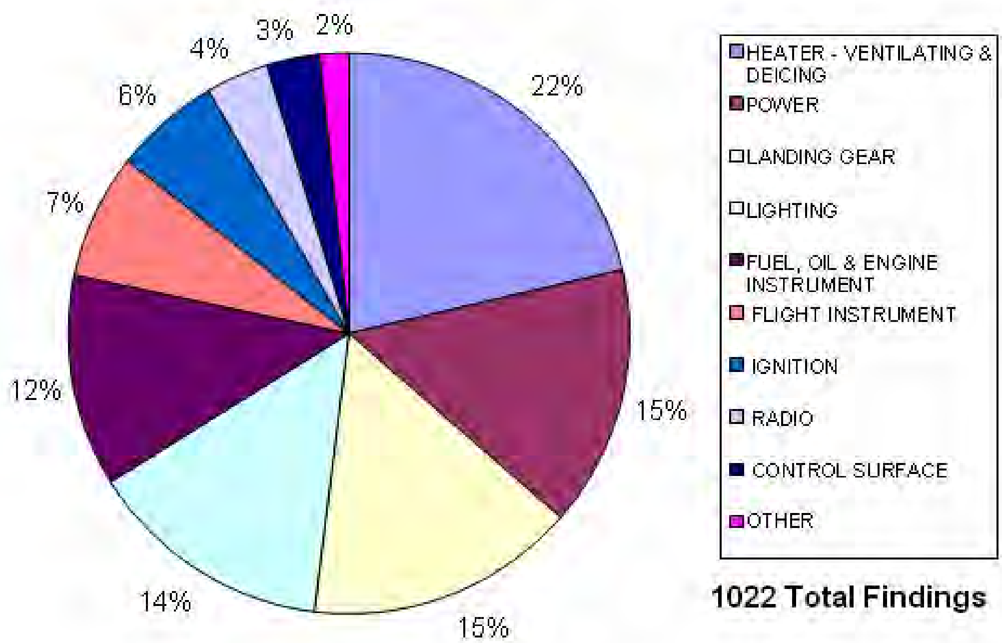


Figure 660. Intrusive Wiring Visual Inspection Findings by Airplane System

Examples of the more common wiring findings are shown in figures 661 to 670. Figure 661 shows heat-damaged insulation, and figure 662 shows the cracked insulation. Figure 663 shows broken outer insulation due to chafing of the wires. Figure 664 shows an example of a corroded terminal. An exposed inner conductor due to improper splicing is shown in figure 665. An example of a damaged terminal is shown in figure 666. A connector with fluid contamination is shown in figure 667. Contact arcing fretting is shown in figure 668. A worn and brittle shrink wrapping is shown in figure 669. Figure 670 shows brittle and broken pilot yoke interphone insulation.

Figure 661. Heat-Damaged Insulation

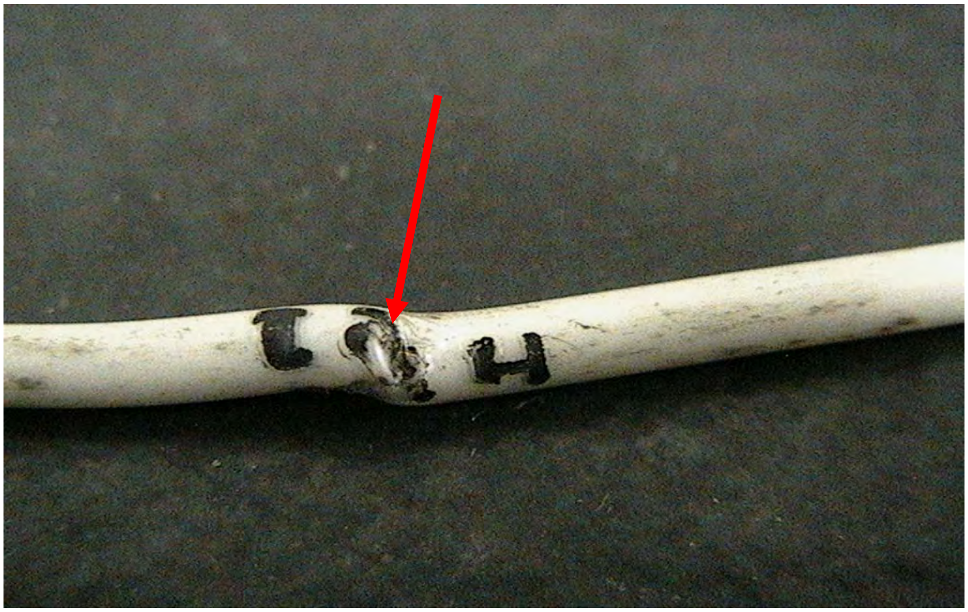


Figure 662. Cracked Insulation

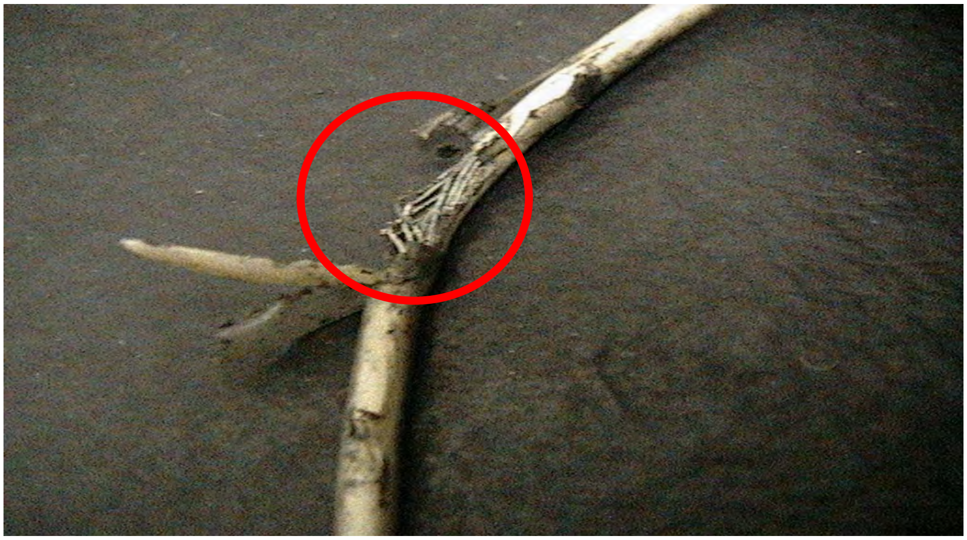


Figure 663. Insulation Broken Due to Chafing

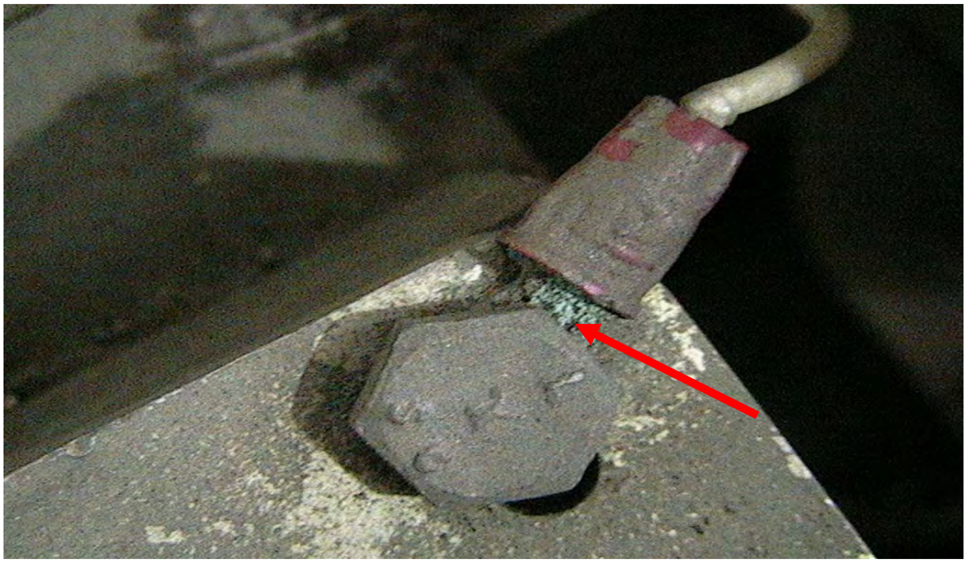


Figure 664. Corroded Terminal

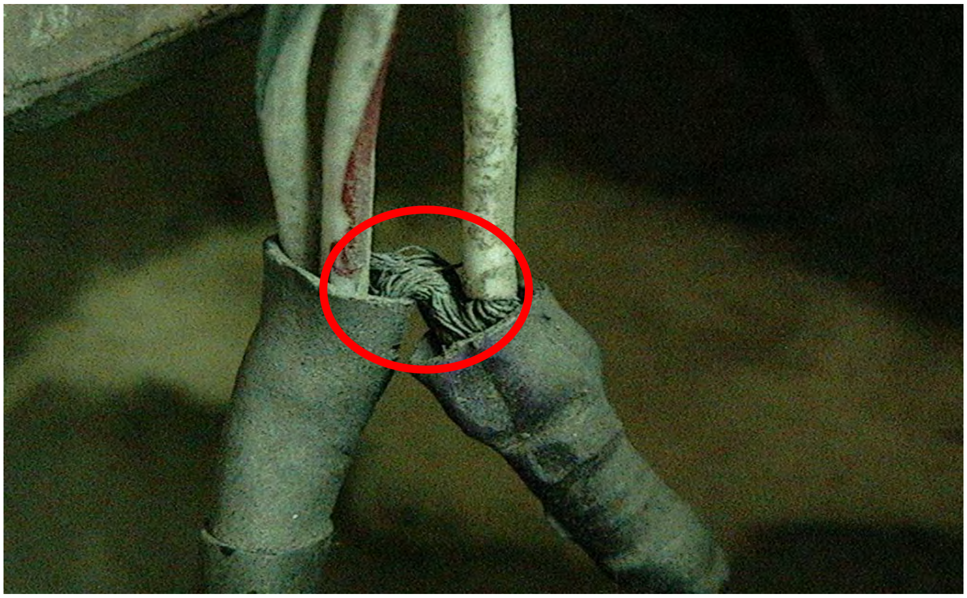


Figure 665. Inner Conductor Exposed Due to Improper Splicing

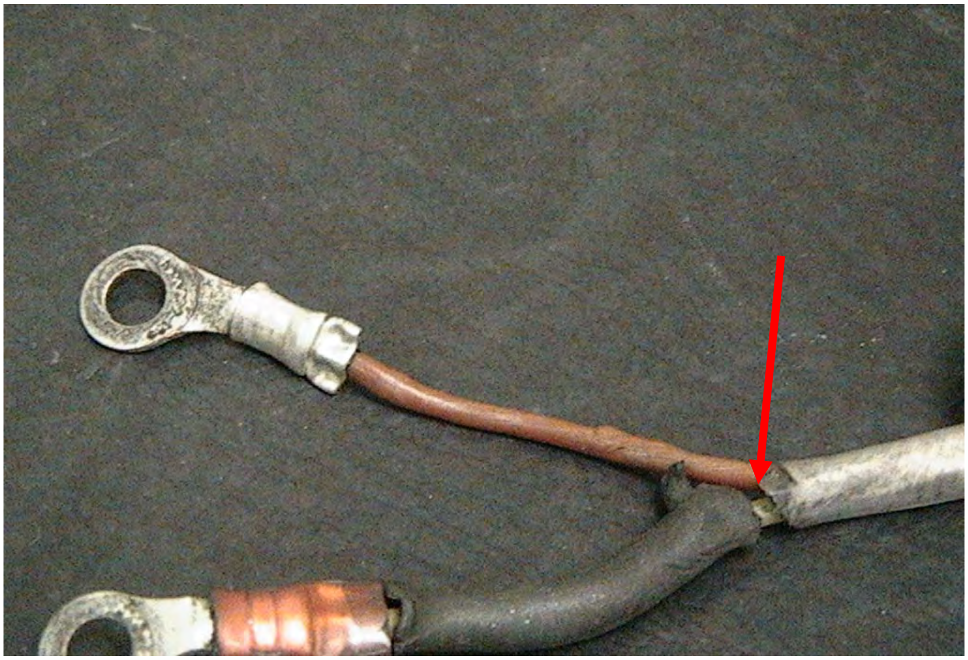


Figure 666. Damaged Terminal Insulation

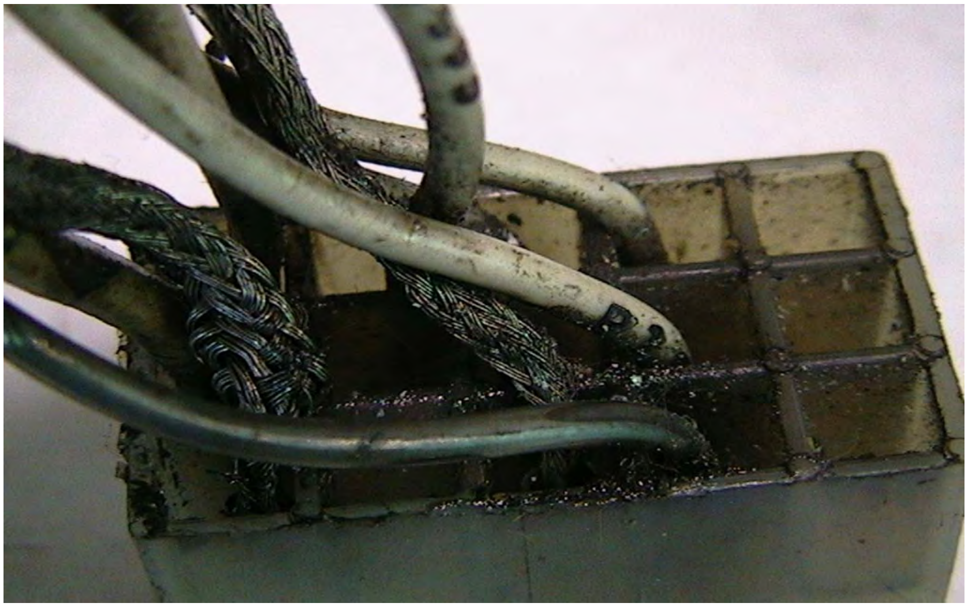


Figure 667. Fluid Contamination on Connector

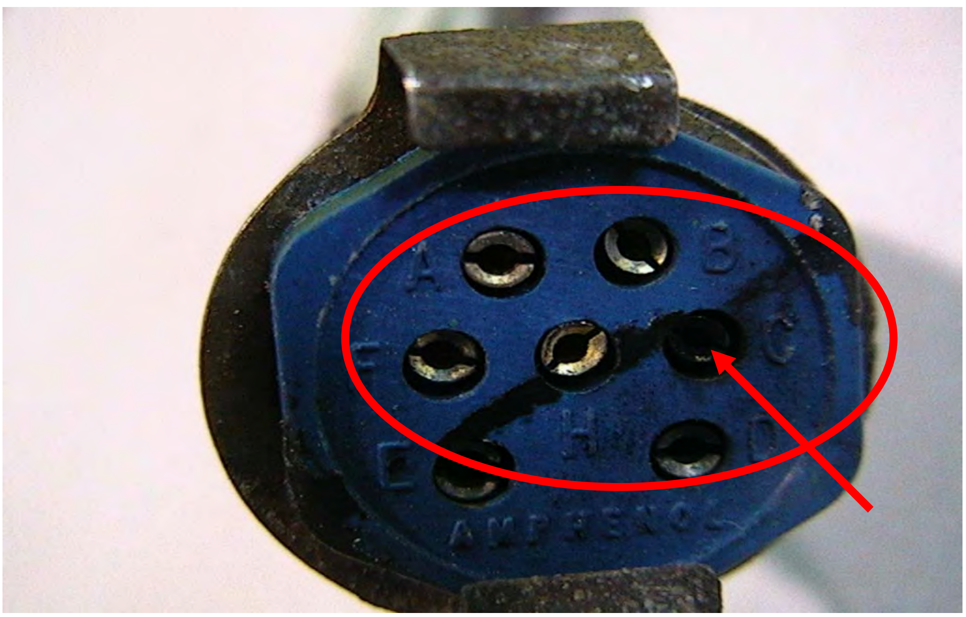


Figure 668. Contact Arcing Fretting

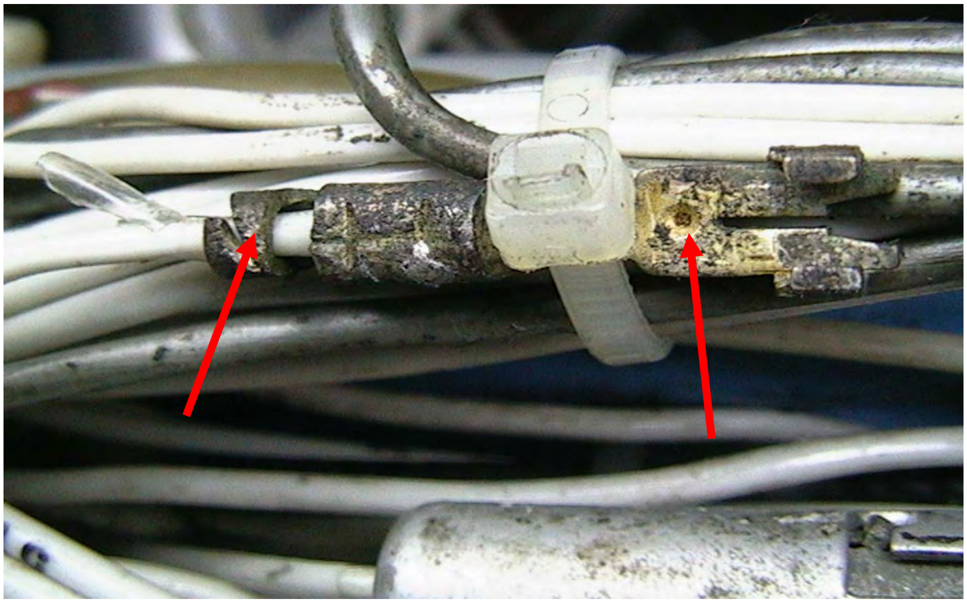


Figure 669. Shrink Wrap Worn and Brittle

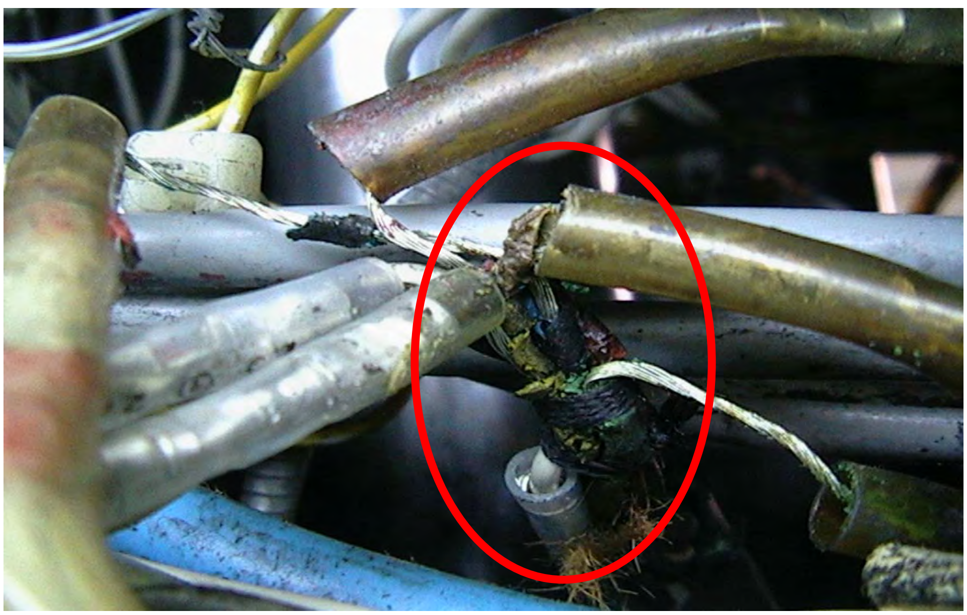


Figure 670. Pilot Yoke Interphone Insulation Brittle and Broken

3.6 CIRCUIT BREAKER TESTING.

Circuit breakers are specifically designed to protect airplane wiring. A circuit breaker will trip off when the current going through it exceeds its rating, which provides primary electrical protection. The functionality of each of the Piper Navajo Chieftain circuit breakers was tested during the initial operational checks. The circuit breakers were then removed from the airplane and tested in the laboratory to verify current trip time specifications. Figure 671 shows the circuit breaker panel and the removed circuit breakers that were tested.

Figure 671. Circuit Breaker Panel and Removed Circuit Breakers

<u>3.6.1 Testing Procedure</u>.

The following test procedure was used to check functionality and compliance to manufacturer's specifications. Testing was also conducted to examine the effects of manually cycling the circuit breakers periodically.

- The circuit breakers were removed from the panel prior to testing.

- The current rating and physical condition of each circuit breaker was noted.

- Conductivity was measured and recorded before applying current to each breaker.

- During each test, the circuit breakers were tested at two current levels—200% and 300% overload:

  - In the 200% overload test, 200% of the rated current was applied to the breaker and the time required for it to trip was recorded.

  - In the 300% overload test, 300% of the rated current was applied to the breaker and the time required for it to trip was recorded.

- Initially, four tests were performed. If the circuit breaker was within specifications and stable, no further testing was done. If the circuit breaker was functional, but outside specifications or unstable, six additional tests were performed.

- For tests 1 through 4, each circuit breaker was reset after each test.
- For tests 5 through 10, the circuit breakers that were unstable or outside specifications were manually cycled five times before each test.
- If a circuit breaker failed to trip during a test, it was manually cycled before the next test.
- Each circuit breaker was given at least half an hour to cool off between tests.

The data was tabulated and compared with the circuit breaker specifications.

3.6.2 Circuit Breaker Findings.

Table 46 shows the relevant information of the circuit breakers subjected to testing. The information includes the manufacturer's part number, the current rating of the circuit breaker, the condition of the breaker, and what system the circuit breaker was used for. Even though there were 35 circuit breakers on the panel, only 33 were tested because 2 circuit breakers (numbers 26 and 33) were defective and did not function. Due to the high current rating of four circuit breakers (numbers 3, 7, 13, and 20), they were not tested at 300% overload because the equipment available could not supply the required current.

Tables 47 and 48 show the results for the 200% and 300% overload tests, respectively. The trip time in seconds is listed in these tables for the ten tests performed and are compared with the maximum trip time specified on the manufacturer's datasheet, which are 22 seconds for 200% overload and 6 seconds for 300% overload, respectively. Not applicable (N/A) listed in the tables means that these breakers were not further tested due to the fact that the circuit breaker met the manufacturer's specifications for the 200% and 300% overload tests by test 4 and were stable, or that it was an open circuit for all four tests. An overcurrent (OC) listing instead of a value in the tables indicates that the corresponding circuit breaker open-circuited during the respective test.

As seen in table 47, for 200% overload, the initial test (test 1) shows that 6 out of 29 (20.5%) of the functional breakers failed to trip within the specified time of 22 seconds. Three (10%) of the circuit breakers had open-circuit faults, a failure in a safe manner, which was also considered nonfunctional when calculating the percent of circuit breakers out-of-spec. As seen in table 48, for 300% overload, the initial test (test 1) shows that 3 out of the 24 (12.5%) functional circuit breakers failed to trip within the specified time of 6 seconds. After test 4, 5 out of 28 (18%) functional circuit breakers (a few began to work when cycled) failed to meet timing specifications, and at the end of the tenth test only one out of 28 (4%) of the circuit breakers did not meet specifications. Five out of 28 (18%) had open-circuit faults (test 1), which is a failure in a safe manner.

Additional tests revealed interesting characteristics of the circuit breakers. Analysis of tests 1 through 4 indicated that some the circuit breakers' performances improved as the number of the test increased. For the 200% overload test, four circuit breakers (numbers 1, 5, 12, and 24) went from either failing or open-circuit in test 1 to passing by test 4. For the 300% overload test, three circuit breakers (numbers 1, 5, and 12) went from open-circuiting in test 1 to passing by test 4.

Additional tests (tests 5 through 10) were performed on selected circuit breakers whose trip times were inconsistent to allow the circuit breaker to stabilize, and also, to verify that the improvement in performance was, in fact, a result of manual cycling. At the final test (test 10), the number of circuit breakers that failed to trip within the specified time had dropped from six (test 1) to three (test 10) for 200% overload, providing a 50% reduction in failures. The number of circuit breakers with an open-circuit fault also dropped from three (test 1) to one (test 10), resulting in 67% reduction. For the 300% overload, the number of circuit breakers that failed dropped from three (test 1) to one (test 10), resulting in 67% reduction in failures. The number of circuit breakers with open-circuit fault also dropped from five (test 1) to one (test 10), resulting in 80% reduction.

As seen from the data in this report, the percent of circuits breakers on aging airplane that fail to meet specifications is relatively high (20.5% for 200% overload and 12.5% for 300% overload) in this study. However, the data in this study also suggests the circuit breakers can extend their functional lifetimes if manual cycling of the circuit breakers is part of the routine maintenance. This is seen in the data by the reduction of failures as the circuit breaker is cycled. It should be noted, however, that in the 300% overload case, the number of out-of-specification circuit breakers increased, before decreasing and, even though there was a significant reduction in out-of-specification circuit breakers from test 1 to test 10, the amount of change was not that significant, only a few seconds. Typically, the circuit breakers that were identified as out-of-specification, were just slightly out of specification.

Table 46 provides a list of circuit breakers subjected to testing. The table includes the manufacturer's part number, the current rating of the circuit breaker, the condition of the breaker, and what system the circuit breaker was used for. Even though there were 35 circuit breakers on the panel, only 33 were tested because two circuit breakers (numbers 26 and 33) were defective and did not function.

Table 46. Circuit Breakers Tested

Number	Name	Part Number	Rating	Condition
1	Unreadable	W23-X1A1G-5	5A	
2	Unreadable	W23-X1A1G-5	5A	
3	Heate blower	W23-X1A1G-20	20A	
4	Pitot heat left	W23-XIA1G-10	10A	
5	Pitot heat right	W23-X1A1G-10	10A	Corroded
6	Windshield wiper	W23-X1A1G-10	10A	
7	Left windshield heat	W23-X1A1G-25	25A	
8	Air control	W23-X1A1G-10	10A	New/corroded
9	Fuel boost pumps	W23-X1A1G-5	5A	New
10	Fuel boost pumps	W23-X1A1G-5	5A	New
11	Anticollision	W23-X1A1G-10	10A	
12	Position	W23-X1A1G-5	5A	
13	Land, taxi, wing, insp	W23-X1A1G-20	20A	
14	Panel	W23-X1A1G-5	5A	
15	Cabin door ajar	W23-X1A1G-5	5A	

Table 46. Circuit Breakers Tested (Continued)

Number	Name	Part Number	Rating	Condition
16	Gear oil temperature fuel quantity	W23-X1A1G-5	5A	New
17	Trim cowl	W23-X1A1G-5	5A	
18	Large safety	W23-X1A1G-5	5A	
19	Start/flap solar right fuel dump	W23-X1A1G-10	10A	
20	Wing flap	W23-X1A1G-25	25A	
21	Cowl flap	W23-X1A1G-10	10A	
22	Heater/light	W23-X1A1G-15	15A	
23	Stall warn	W23-X1A1G-5	5A	New
24	Fuel pumps	W23-X1A1G-10	10A	New
25	Voltage regulator left	W23-X1A1G-5	5A	New
26	Voltage regulator right	W23-X1A1G-5	5A	New/in oper
27	Access/speaker amplifier	W23-X1A1G-5	5A	
28	Navigation/communication 1	W23-X1A1G-10	10A	New
29	Navigation/communication 2 adf2	W23-X1A1G-10	10A	
30	Automatic direction finder/magnetic bearing	W23-X1A1G-5	5A	
31	Distance measuring equipment	W23-X1A1G-5	5A	
32	Electric trim	W23-X1A1G-5	5A	
33	Radar	W23-X1A1G-3	3A	New/broken
34	Transmitter	W23-X1A1G-7.5	7.5A	Cracked
35	High-frequency communication/personal headset amplifier	W23-X1A1G-5	5A	

Table 47 shows experimental results for 200% current overload. Shown in the table is the current rating for each circuit breaker and the trip times for ten tests. The numbers in bold indicate a trip time that exceeds the manufacturer's specification of 22 seconds. Each circuit breaker was tested four times, and those that had unstable trip times or that were out of specification were tested an additional six times. Two circuit breakers (numbers 26 and 33) were defective and did not function.

Table 47. Experimental Results 200% Overload

Number	Current Rating	Test 1	Test 2	Test 3	Test 4	Test 5	Test 6	Test 7	Test 8	Test 9	Test 10
1	5A	**>600**	**OC**	19	19	15	13	14	13	14	14
2	5A	17	14	15	17	N/A	N/A	N/A	N/A	N/A	N/A
3	20A	11	12	12	12	N/A	N/A	N/A	N/A	N/A	N/A
4	10A	12	13	11	11	N/A	N/A	N/A	N/A	N/A	N/A
5	10A	**OC**	**OC**	21	18	15	15	14	16	16	19
6	10A	**27**	**37**	**33**	**36**	**25**	19	21	20	18	18

Table 47. Experimental Results 200% Overload (Continued)

Number	Current Rating	Test 1	Test 2	Test 3	Test 4	Test 5	Test 6	Test 7	Test 8	Test 9	Test 10
7	25A	9	10	10	10	N/A	N/A	N/A	N/A	N/A	N/A
8	10A	**257**	**OC**	**OC**	**OC**	N/A	N/A	N/A	N/A	N/A	N/A
9	5A	14	10	12	13	N/A	N/A	N/A	N/A	N/A	N/A
10	5A	6	7	11	6	N/A	N/A	N/A	N/A	N/A	N/A
11	10A	**OC**	**127**	**44**	**50**	**30**	**28**	**31**	**47**	**40**	21
12	5A	**OC**	**OC**	19	15	N/A	N/A	N/A	N/A	N/A	N/A
13	20A	18	18	**25**	21	19	18	**24**	21	**23**	**24**
14	5A	17	16	**27**	18	19	19	19	16	17	16
15	5A	13	12	10	11	N/A	N/A	N/A	N/A	N/A	N/A
16	5A	12	11	12	11	N/A	N/A	N/A	N/A	N/A	N/A
17	5A	15	**OC**	7	13	N/A	N/A	N/A	N/A	N/A	N/A
18	5A	19	15	14	16	N/A	N/A	N/A	N/A	N/A	N/A
19	10A	17	20	21	**23**	18	14	14	13	13	13
20	25A	**OC**	**OC**	**OC**	**OC**	N/A	N/A	N/A	N/A	N/A	N/A
21	10A	14	12	11	12	N/A	N/A	N/A	N/A	N/A	N/A
22	15A	12	11	14	14	N/A	N/A	N/A	N/A	N/A	N/A
23	5A	13	12	14	13	N/A	N/A	N/A	N/A	N/A	N/A
24	10A	**39**	18	18	17	17	18	19	20	16	17
25	5A	10	7	7	7	N/A	N/A	N/A	N/A	N/A	N/A
26	5A	**OC**	**OC**	**OC**	**OC**	N/A	N/A	N/A	N/A	N/A	N/A
27	5A	15	15	16	15	N/A	N/A	N/A	N/A	N/A	N/A
28	10A	15	20	19	16	N/A	N/A	N/A	N/A	N/A	N/A
29	10A	16	16	14	16	N/A	N/A	N/A	N/A	N/A	N/A
30	5A	**35**	**62**	**>180**	**>180**	**>180**	**>180**	**>180**	**>180**	**>180**	**>180**
31	5A	14	14	14	14	N/A	N/A	N/A	N/A	N/A	N/A
32	5A	**45**	**26**	22	**27**	**32**	20	**>180**	**40**	**35**	**39**
33	3A	**OC**	**OC**	**OC**	**OC**	N/A	N/A	N/A	N/A	N/A	N/A
34	7.5A	16	**25**	23	20	**29**	12	16	17	15	15
35	5A	12	10	11	11	N/A	N/A	N/A	N/A	N/A	N/A

Table 48 shows experimental results for 300% current overload. Shown in the table is the current rating for each circuit breaker and the trip times for ten tests. The numbers in bold indicate a trip time that exceeds the manufacturer's specification of 6 seconds. Each circuit breaker was tested four times, and those that had unstable trip times or that were out of specification were tested an additional six times. Due to the high current rating of four circuit breakers (numbers 3, 7, 13, and 20), they were not tested at 300% overload due to equipment limitations. Two circuit breakers (numbers 26 and 33) were defective and did not function.

Table 48. Experimental Results 300% Overload

Number	Current Rating	Test 1	Test 2	Test 3	Test 4	Test 5	Test 6	Test 7	Test 8	Test 9	Test 10
1	5A	**OC**	6	5	4	4	4	4	5	4	4
2	5A	5	5	5	5	N/A	N/A	N/A	N/A	N/A	N/A
3	20A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
4	10A	4	4	4	4	N/A	N/A	N/A	N/A	N/A	N/A
5	10A	**OC**	**OC**	5	6	5	5	5	5	5	5
6	10A	**7**	**7**	**8**	**8**	6	5	6	5	5	5
7	25A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
8	10A	**OC**	**OC**	**OC**	**OC**	N/A	N/A	N/A	N/A	N/A	N/A
9	5A	4	4	4	4	N/A	N/A	N/A	N/A	N/A	N/A
10	5A	3	2	3	3	N/A	N/A	N/A	N/A	N/A	N/A
11	10A	**OC**	5	**8**	**7**	**7**	**7**	**7**	**7**	**7**	6
12	5A	**OC**	**OC**	5	5	N/A	N/A	N/A	N/A	N/A	N/A
13	20A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
14	5A	5	5	6	6	6	6	5	6	5	4
15	5A	5	4	4	4	N/A	N/A	N/A	N/A	N/A	N/A
16	5A	4	3	4	4	N/A	N/A	N/A	N/A	N/A	N/A
17	5A	4	**OC**	4	4	N/A	N/A	N/A	N/A	N/A	N/A
18	5A	6	5	5	5	N/A	N/A	N/A	N/A	N/A	N/A
19	10A	5	6	**7**	**7**	5	4	4	4	4	4
20	25A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
21	10A	4	4	3	4	N/A	N/A	N/A	N/A	N/A	N/A
22	15A	4	4	3	3	N/A	N/A	N/A	N/A	N/A	N/A
23	5A	4	5	5	4	N/A	N/A	N/A	N/A	N/A	N/A
24	10A	5	5	5	5	5	6	5	5	5	5
25	5A	3	3	3	2	N/A	N/A	N/A	N/A	N/A	N/A
26	5A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
27	5A	5	5	5	5	N/A	N/A	N/A	N/A	N/A	N/A
28	10A	5	5	5	5	N/A	N/A	N/A	N/A	N/A	N/A
29	10A	5	5	3	5	N/A	N/A	N/A	N/A	N/A	N/A
30	5A	**12**	**12**	**14**	**8**	**14**	**14**	**12**	**12**	**9**	**14**
31	5A	5	4	5	5	N/A	N/A	N/A	N/A	N/A	N/A
32	5A	**7**	**7**	6	**7**	6	**7**	**9**	**8**	**7**	6
33	3A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
34	7.5A	6	6	5	5	4	4	5	4	6	6
35	5A	3	3	3	3	N/A	N/A	N/A	N/A	N/A	N/A

4. SUMMARY.

To determine if potential continuing airworthiness problems exist for the small airplane fleet as a function of the aging process and to assess the ability of current inspection programs to detect defects that might compromise airworthiness assurance, the FAA established a research program to conduct a destructive evaluation of four aged airplanes (two Cessna 402s, a Piper Navajo Chieftain, and a Beechcraft 1900D) used in commuter service. The intent of the program was to provide insight into the condition of a typical aged airplane, to see if a correlation exists between its maintenance history and current condition from a safety of flight perspective, and to assess maintenance and inspection program's capabilities to detect defects that might pose a threat to the airworthiness of an airplane. This document provided findings in a summary report for the teardown evaluation of a 1975 Piper Navajo Chieftain airplane in support of the research program. The results provided information for use in future investigations into the aged small airplane fleet and to determine if additional research is required to address specific problems observed (if any). Specific observations were made regarding findings discovered during the teardown evaluation on the particular airplane selected.

The destructive evaluation of this commuter-class airplane was divided into two main tasks: (1) inspection of the airframe, systems, and wiring, and (2) teardown evaluation of the airframe and airplane systems. During the inspection phase, four subtasks were performed: a survey of the airplane maintenance records, routine visual inspections of the airframe and airplane systems as prescribed by the service manual, routine visual inspection of all accessible airplane wiring, and airworthiness inspections developed for accessible structural locations. The maintenance records survey provided information on the airplane maintenance history for correlation of maintenance practices to airplane condition, and the inspections determined the condition of the airplane based on normal maintenance activity. The teardown evaluation involved disassembly of the airframe and major airplane sections, inspection of the airplane system components, a structural assessment using alternative NDI techniques, circuit breaker testing, microscopic examination of critical and suspect areas, and fractographic analysis of selected cracks and areas of corrosion. All airplane systems' components and wiring were removed during disassembly. The disassembly was performed to provide full access to all critical structural areas on the airplane. Inspection of the airplane systems' components assists in determining if any signs of aging effects are apparent on the airplane systems. The structural assessment using alternative NDI techniques was conducted on the primary structure prior to disassembly. The microscopic examination of suspect and critical structural areas provided verification and detailed quantification of the extent of damage found during the airworthiness inspections, structural assessment using NDI, and disassembly of the entire airframe. Fractographic analysis was used to determine the failure mode of selected cracks as well as to provide a more detailed characterization of cracks and areas of corrosion.

4.1 SUMMARY OF INSPECTION PHASE.

In the inspection phase, the survey of the airplane maintenance records provided information on the airplane maintenance history for correlation of maintenance practices to airplane condition. The routine visual inspections of the airframe, airplane systems, and wiring along with the airworthiness airframe inspections determined the condition of the airplane based on normal maintenance activity.

4.1.1 Maintenance Record Review.

A survey of the airplane maintenance records was conducted in an effort to correlate airplane condition with airplane usage and maintenance history. Maintenance log books, SBs, ADs, and

FAA 337 forms for major repairs and modifications were reviewed in this survey. The survey of the airplane records for the Piper Navajo Chieftain PA31-350 serial number 31-7552019 was completed using data obtained from the following resources:

- Airframe log books
- Approved airplane inspection program records
- FAA databases
- NTSB databases
- Piper airplane maintenance and service publications
- Civil Aviation Registry records

The records review showed that during the airplane's 29 years of operation, it was registered under only one tail number to 19 different owners/operators. The airplane spent approximately 3 years in Alaska, 6 years in the corrosive environment of the East Coast, and about 12 years in Arizona and Nevada, likely performing Grand Canyon tours in a fatigue conducive environment. All ADs other than AD 2003-24-07, which requires an inspection of the rudder torque tube, were complied with. AD 2003-24-07 was not complied with since the NIAR purchased the airplane in October 2004. All applicable SBs, excluding SB 1007 and 1105A, were performed on the airplane. A review of FAA Form 337 showed the airplane has undergone significant repairs to its belly due to a gear-up landing in 1982, which was not required to be reported and is not listed in the NTSB's accident/incident database. Minor repairs listed in the maintenance logs also document the replacement of the landing gear strut housing in 1992. A review of the Type Certificate Data Sheet A20SO and the Supplemental Type Certificates indicates the airplane was in compliance with requirements for airworthiness at the time of purchase.

Approximately 6200 records from the SDR database from 1974 to 2004 were reviewed to identify common fleetwide issues encountered on the Piper Navajo Chieftain. This review revealed that 17% of all records were related to the landing gear and an additional 15% were related to the reciprocating engines. It is important to note that since the SDR reporting system is not mandatory, common problems may exist that are not reflected in the SDR database.

4.1.2 Routine Visual Inspections.

The general condition of the airframe and system components were evaluated through routine visual inspections prescribed in the Piper Navajo Chieftain Service Manual. All accessible wiring was also visually inspected for wiring condition defects, installation defects, and termination defects. The intent of these inspections was to discover all possible visual defects and document each defect's severity and location. Each inspection was duplicated by independent inspectors. The results of these inspections were reviewed by a licensed airframe mechanic to determine which findings were noteworthy. Noteworthy findings are defined as defects that would require further maintenance action.

There were 101 different visual inspection locations performed on the Piper Navajo Chieftain, with the intent of finding every detectable flaw on the airplane. This inspection methodology led to a large number of documented flaws. Only 11 of the total findings were determined to be noteworthy, and the remaining findings were deemed minor by airframe mechanics. With aged airplanes, flaws such as minor cracks, scratches, dents, paint chips, and slight corrosion are to be expected and pose no immediate threat to the safety of the airplane. In addition to dents and areas of missing paint, slight to moderate corrosion was noted on a majority of the screws on the airplane. It was also noted that some disposable parts, such as seals and hoses, were due to be replaced. The following defects found during the routine visual structures and systems inspections were deemed noteworthy:

- Corrosion and cracking of the battery box and tray at FS 2, WL 13, and BL 0
- ELT Antenna wire severed at FS 261, WL 40, and RBL 6
- Wear of fuel cell transmitter float arms
- Chafing of fuel tube by engine control cable at FS 110, WL 15, and LWS 64
- Damaged repairs to the ribs in the left and right gear wheel wells at FS 110, WL 15, LWS 39.5-49, and RWS 39.5-49
- Cracked stringers in left and right main gear wheel wells at FS 150, WL 16, and LWS 39.5
- Damaged repair to floor plate gusset assembly in nose baggage compartment at FS 28, WL 13, and RBL 12
- Cabin floor bulkhead assembly cracked/torn at FS 162.6, WL 16, RBL 6, and LBL 6
- Corrosion of rudder torque tube at FS 342.25, WL 20, and BL 0
- Chafing of coaxial cable between trim control cables at FS 84, WL 17, and BL 0
- Nose gear oleo strut housing damaged at FS 25, WL 32, and RBL 2

The purpose of performing the routine visual inspections on the wiring systems of this airplane was to check the condition of the wires for any defects or flaws, which could be hazardous to the airplanes in flight. This is the first step performed prior to any other wiring inspections that are done to assess the maintenance activities performed on the airplane.

The wiring routine visual inspection was performed without disturbing the condition of the wires in the airplane. The inspection was performed to assess the general wiring condition (chafing, rubbing, burning, or tearing of the wires), installation defects, terminations, connectors, groundings, and circuit breakers. The condition of the wires was documented and photographed during the inspection process. The most frequent issues found in the routine visual inspections were broken terminals, chafing, rubbing, cutting, exposed shield, exposed inner conductor, cutting though the outer insulation, improper terminations of end terminals, improper termination of wires (no cap), unused wires improperly stowed, and corroded terminals.

Of the 199 wiring defects documented during the routine visual inspections, 84 were wiring condition defects. Of the wiring condition defects, 51% consisted of rubbing or chafing of the outer insulation, and 30% of the wires had exposed inner conductors. Of the 97 installation defects noted, 28% were missing or deteriorated grommets, with an additional 35% with inadequate clearance to structure. Eighteen termination defects were noted, with seven being loose or broken terminals. No connector defects were observed on this airplane. Defects in the passenger compartment accounted for 35% of the total wiring defects observed. Another 22% of the defects were located in the crew compartment with an additional 16% located in the left wing.

4.1.3 Airworthiness Inspections of Accessible Structural Locations.

Twenty-three PSEs located on the wings, fuselage, horizontal stabilizer, vertical stabilizer, landing gear, and engine compartment were identified and inspected for airworthiness of the airframe. These airworthiness inspections target specific PSEs that were accessible on the airframe, using standard NDI techniques, and they are typically performed once the airplane's flight hours approaches or exceeds its design life. Using engineering judgment, PSEs were determined to include all accessible major attachment fittings of the airplane and all accessible structural members reacting high stresses, such as the wing root and the engine beam rib. Four other significant areas were determined based on maintenance history. These areas included the rudder torque tube, elevator torque tube, landing gear strut housing attachment points, and the horizontal stabilizer butt rib. Structural areas that were not accessible, such as nacelle bulkheads, were not inspected since destructive disassembly would have been required to access these areas for inspection. Detailed NDI procedures were developed for each PSE inspected during the airworthiness inspections based on the type of defect to be found, access to the inspection area, and the type of material in the inspection area. The NDI methods used for these airworthiness inspections included visual inspection, fluorescent liquid penetrant inspection, magnetic particle inspection, surface scan eddy-current inspection, and bolthole eddy-current inspection.

A crack was found on the nose landing gear strut housing during the visual inspections. Corrosion resulting in localized areas of complete thickness loss was observed during the visual inspections on the rudder torque tube. Corrosion was found on the left and right wing rear spar attachment points during the fluorescent liquid penetrant inspections. During the eddy-current inspections, ten gouges were reported on the wing main spar splice attachment points.

4.2 SUMMARY OF TEARDOWN EVALUATION PHASE.

The goal of the teardown evaluation phase was to identify, locate, and characterize all defects on the airplane. An inspection of systems' components was performed to identify cracking, corrosion, and wear on the components of the airplane's systems, and a structural assessment using alternative NDI techniques, aimed at identifying additional defects embedded in the airplane's primary structure prior to disassembly, was also performed. Then, the airplane was disassembly into components, visually inspected at the component level, inspected postdisassembly using traditional NDI techniques, and microscopically examined to identify and characterize all defects on the airplane's primary structure. Fractographic analysis was performed on selected cracks to determine failure mode. Airplane wiring was also assessed visually after removal from the airplane during the intrusive wiring visual inspections. The functionality and performance of the circuit breakers were also tested and compared to design specifications.

4.2.1 Inspection of Systems' Components.

The following airplane systems' components were removed from the airplane, disassembled, and visually inspected per Advisory Circulars 43.13.1B and 43.13.2A, and the Piper Service Manual, to determine the current condition of each component:

- Hydraulic actuator
- Pitot probe
- Nose steering tube assembly
- Hydraulic assembly
- Right starter solenoid
- Torsion link assembly
- Heat duct switch
- Air conditioning unit
- Battery box
- Right engine hydraulic pump

The findings from these components included minor surface corrosion on the right starter solenoid, air conditioning unit, and the battery box, cracks on the air conditioning unit and battery box, and corrosion on the mounting bracket and bots of the right engine hydraulic pump. The visual inspection of the right pitot probe revealed minor corrosion around both attachment holes, and moderate to severe corrosion was discovered around both sensor pins, which could possibly affect pitot probe heating capabilities. Leak checks and visual inspections were performed on 203 lines. Besides minor scratches, nicks, gouges, flared tube ends, and minor corrosion, which are expected on an airplane of this age, a 1-inch area of chafing was discovered on a manifold pressure line with a 0.06-inch-diameter hole.

4.2.2 Structural Assessment Using Alternative NDI Techniques.

A structural assessment using alternative NDI techniques was performed on the wing horizontal and vertical stabilizer spars in an effort to identify additional embedded defects in the primary structure prior to disassembly. Two eddy-current-based methods were implemented for these inspections: the spot probe and sliding probe. Difficulties were encountered while performing these inspections due to the type of structure to be inspected. Areas of the spar cap were too narrow causing edge effects to mask possible defect signals. Also, the presence of vortex generators caused inspection accessibility issues on the vertical stabilizer. Three crack indications were identified using the sliding probe on the left wing main and rear spar, and five indications were found on the right wing main spar. Nine crack indications were identified on the horizontal stabilizer front and rear spars using the sliding probe and two sliding probe crack indication were documented on the vertical stabilizer front spar.

4.2.3 Teardown Evaluation.

Of the 697 parts inspected during the teardown evaluation, 303 defects were identified and characterized, as shown in table 49.

Table 49. Summary of Findings From the Teardown Evaluation

Airplane Section	Total Number of Parts Inspected	Cracks	Corrosion	Wear	Other Damage	Total Number of Defects
Left wing	196	27	39	12	1	79
Right wing	186	21	23	11	2	57
Fuselage	136	79	33	5	4	121
Horizontal stabilizer	136	7	4	0	1	12
Vertical stabilizer	43	27	5	0	2	34
Total	697	161	104	28	10	303

Of the 196 parts subjected to the teardown evaluation from the left wing, 79 defects were noted and characterized. There were 27 cracks, 39 areas of corrosion, 12 areas of wear, and 1 damaged area identified on the left wing. Light to light-moderate scattered corrosion was observed on the surface of many left wing main spar cap assembly components, three areas of wear were identified on the front spar, and one on the rear spar. Eight cracks were also identified on the left wing bulkhead wing fillets, and multiple areas of wear were observed on the rear spar assembly lower aft angle. Four areas of corrosion were also noted on the nacelle floor. Areas of corrosion causing localized reductions in thickness of greater than 5% were noted on the left wing main spar lower aft reinforcement, main spar lower aft angle, plate-wing panel reinforcement assembly, and the nacelle front locker floor.

Of the 186 parts inspected during the teardown evaluation phase of the right wing, 57 defects were noted and characterized. There were 21 cracks, 23 areas of corrosion, 11 instances of wear, and 2 areas of damage noted during the teardown evaluation of the right wing. No defects were identified on the right wing front or rear spars. Similar to the left wing, scattered light to light-moderate corrosion was observed frequently on the main spar cap assembly components and five cracks, ranging in length from 0.458 to 2.25 inches, were observed on the bulkhead wing fillets. Areas of wear were observed on the plate-wing panel reinforcement assembly, rear spar assembly main gear forward, and the rear spar lower aft assembly. Cracks and areas of corrosion were frequently observed on the nacelle bulkhead components.

During the teardown evaluation of the fuselage, 136 parts were inspected and 121 defects were identified and characterized. There were 79 cracks recorded on the fuselage, along with 33 areas of corrosion, 5 instances of wear, and 4 occurrences of other damage. Multiple cracks were characterized in each of the following areas: windshield bracket, stiffener angle, various bulkhead assemblies, fuselage web-beam assembly lower forward frame right side, fuselage left lower forward web, fuselage lower plate assembly, floorboard support structure, left and right front spar attachments, and the door sill support aft body. Multiple areas of corrosion were found on the floorboard support structure and various bulkheads.

Of the 136 parts inspected on the horizontal stabilizer, only 12 defects were identified. Four areas of corrosion, seven cracks, and one area of other damage were reported on the horizontal stabilizer. Most cracks occurred on ribs or the leading-edge skin assemblies, and the corrosion was categorized as light scattered corrosion and occurred on the components of the spar cap assemblies.

During the teardown evaluation, 43 parts were inspected on the vertical stabilizer, and 34 defects were identified. There were 27 cracks, 5 areas of corrosion, and 2 areas of damage recorded and characterized during the teardown evaluation. Multiple cracks occurred on each of the following structural members: vertical stabilizer front spar, front spar left and right angles, rear spar, rear spar right angle, left and right side gussets, main bottom rib assembly, leading-edge rib, and the rudder leading-edge upper skin. Severe corrosion was found on the rudder torque tube assembly and rudder butt rib.

4.2.4 Intrusive Wiring Visual Inspections.

Once all wires and wiring harnesses were removed from the airplane during detailed disassembly, each wire was visually inspected for wiring condition, repair, termination, and connector findings. Every finding on each wire was labeled with the wire identification number and a finding code. All inspections were performed in accordance with Advisory Circular 43.13.1B and the Piper Service Manuals.

During the intrusive visual inspections, 1022 findings were documented; some were minor findings others were critical findings. Of the 1022 findings documented during the intrusive visual inspections, 117 were categorized as critical findings. Eighteen percent of the wiring findings were rubbing/chafing of the outer insulation with an additional eighteen percent classified as fluid contamination and another seventeen percent identified as brittle and cracking insulation. Twenty-two percent of the findings were related to the heater-ventilating and deicing system, fifteen percent were part of the airplane power system, and an additional fifteen percent of the findings were related to the landing gear system.

4.2.5 Circuit Breaker Testing.

The following test procedure was used to check functionality and compliance to manufacturer's specifications. Testing was also conducted to examine the effects of manually cycling the circuit breakers periodically.

- The circuit breakers were removed from the panel prior to testing.
- The current rating and physical condition of each circuit breaker was noted.
- Conductivity was measured and recorded before applying current to each breaker.

- During each test, the circuit breakers were tested at two current levels—200% and 300% overload:

  - In the 200% overload test, 200% of the rated current was applied to the breaker, and the time required for it to trip was recorded.

  - In the 300% overload test, 300% of the rated current was applied to the breaker, and the time required for it to trip was recorded.

- Initially, four tests were performed. If the circuit breaker was within specifications and stable, no further testing was done. If the circuit breaker was functional, but outside specifications or unstable, six additional tests were performed.

- For tests 1 through 4, each circuit breaker was reset after each test.

- For tests 5 through 10, the circuit breakers that were unstable or outside specifications were manually cycled five times before each test.

- If a circuit breaker failed to trip during a test, it was manually cycled before the next test.

- Each circuit breaker was given at least half an hour to cool off between tests.

The data was tabulated and compared with the circuit breaker specifications.

As seen from the data in this report, the percent of circuits breakers on aging airplanes that fail to meet specifications is relatively high (20.5% for 200% overload and 12.5% for 300% overload) in this study. However, the data in this study also suggests the circuit breakers can extend their functional lifetimes if manual cycling of the circuit breakers is part of the routine maintenance. This is seen in the data by the reduction of failures as the circuit breaker is cycled. It should be noted, however, that in the 300% overload case, the number of out-of-specification circuit breakers increased before decreasing and, even though there was a significant reduction in out-of-specification circuit breakers from test 1 to test 10, the amount of change was not that significant, only a few seconds. Typically, the circuit breakers that were identified as out-of-specification, were just slightly out of specification.

APPENDIX A—SUPPLEMENTAL INSPECTION DOCUMENT PIPER PA-31-350 NAVAJO CHIEFTAIN

Inspection Methods and Requirements

General Requirements

A. Facilities performing nondestructive inspection (NDI) as defined in this Supplemental Inspection Document (SID) shall have a written practice that meets the minimum intent of The American Society for Non-Destructive Testing Recommended Practice Number SNT-TC-1A or National Aerospace Standard NAS 410, NAS Certification and Qualification of Non-Destructive Test Personnel.

B. Personnel performing NDI defined in this SID shall be certified to a minimum of a Level II in the appropriate inspection method as defined in a written practice that meets the minimum intent of The American Society for Non-Destructive Testing Recommended Practice Number SNT-TC-1A or National Aerospace Standard NAS 410, NAS Certification and Qualification of Non-Destructive Test Personnel.

C. Organizations and personnel engaged in the application of NDI and operating under the jurisdiction of a foreign government shall use the appropriate documents issued by the applicable regulatory agency complying with the above requirements.

D. Facilities performing NDI as defined in this SID, must own or have access to the appropriate test equipment and be capable of performing the inspection and reporting the test results as defined in this document.

General Eddy-Current Inspection

General

The eddy-current (EC) method is used to detect discontinuities in parts that are conductors of electricity. An EC is a circulating electrical current induced in a conductor by an alternating magnetic field. A coil of copper wire is placed in a holder called a “probe”. The probe produces the alternating magnetic field used in EC. The ECs induced in an electrical conductor vary in magnitude and distribution based on specimen properties such as electrical conductivity, magnetic permeability, geometry, and discontinuities. When ECs encounter an obstacle, such as a crack, the normal path and strength of the currents are changed. This change is detected on a display or a meter.

Equipment

The EC equipment listed in each procedure was used during the development of the inspection technique. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity. When equipment is substituted, it may be necessary to make appropriate adjustments to the established techniques.

Instrument Requirements

A. Certain inspection techniques require the use of instruments that provide both phase and amplitude information on a storage cathode ray tube for impedance plane analysis. Impedance plane instruments may be used as a substitute for metered instruments. Metered instruments shall not be substituted for impedance plane instruments where the ability to distinguish phase information is required.

B. The instrument shall demonstrate a repeatable signal response that has a signal-to-noise ratio of greater than 3:1 for the test in which it is to be used.

Probe Requirements

A. The probe may have an absolute or differential coil arrangement. The probe may be shielded or unshielded; however, a shielded probe is normally recommended. Bolthole EC may be performed using either automated scanners or manual probes.

B. The probe shall have an operating frequency that produces the required test sensitivity and depth of penetration.

C. Since smaller coil diameters are more effective in detecting cracks, a coil diameter of 0.125 inches was used for the development of surface scan inspections prescribed in the SID, while a coil diameter of 0.080 inch was used for bolthole EC inspection development. All coils contained a ferrite core as is typical of this class of probes.

D. The probe shall not give interfering responses from scanning or normal operating pressure variations that cause the signal-to–noise ratio to be less than 3:1.

E. Teflon® tape may be used to decrease wear on the EC probe coil. When Teflon tape is used, the instrument calibration must be verified.

Calibration Standard Requirements

A. In some cases, specially fabricated reference standards will be necessary to simulate a part’s geometry, configuration, and/or a specific discontinuity location. If a general purpose surface or bolthole reference standard is indicated, substitution is not permitted.

B. Reference standards should be of an alloy having the same major base material, basic temper, and approximate electrical conductivity of the material to be inspected.

C. Reference standards shall have a minimum surface finish of 150 roughness height rating (RHR).

D. An Electrical Discharge Machined (EDM) surface notch no deeper than 0.020 inch shall be used for surface scan EC inspection calibration. An EDM corner notch of no larger than 0.050 inch in surface length shall be used for bolthole EC inspection calibration. The dimensional accuracy of the notch shall be documented and traceable to the National Institute of Standards and Technology (NIST).

Inspection

General Considerations

A. Inspections shall not be preformed until the temperature of the probe, the standard, and the material to be inspected has been allowed to equalize.

B. EC inspection requires that good contact be made between the probe and the part, unless a specific procedure requires a certain amount of liftoff. The inspection area shall be free of dirt, grease, oil, or any other contaminants that may interfere with the inspection. Mildly corroded parts must be cleaned lightly with emery cloth; heavily corroded parts must be lightly abraded and cleaned locally in the inspection area. Paint removal is not required as long as the coating is relatively uniform and not loose or flaking. If the paint thickness is such that it will interfere with the inspection, the paint must be removed to maintain inspection sensitivity.

Instrument Calibration

A. The instrument shall be calibrated and operated in accordance with the manufacturer's instructions. Calibration shall be performed using the reference standard indicated in the inspection technique.

B. Instrument calibration shall be performed prior to inspection. Calibration shall be checked at intervals necessary to maintain calibration during continuous use and at the end of the inspection. The instrument shall be recalibrated if any part of the system is replaced or if any calibrated control settings are changed.

C. For surface scan EC, the instrument settings should be adjusted to achieve a minimum separation of three major screen divisions or 30 percent full screen height between the null/balance point and the signal from the appropriate reference notch. A minimum separation of eight major screen divisions or 80 percent full screen height between the null/balance point and the signal from the appropriate reference notch is recommended for bolthole EC inspections. For a differential probe, the signal amplitude should be considered as peak to peak. Filters may be used to improve the signal-to-noise ratio as necessary.

Inspection Performance

A. When performing surface scan EC, the inspection area shall be scanned in two directions with perpendicular scan paths, when feasible.

B. While performing surface scan EC, scan the inspection area at index increments that do not exceed the width of the EC test coil. The part edge shall be scanned as long as the response from the edge effect does not mask the calibration notch response. Areas where edge effect is greater than the calibration notch signal shall not be inspected using EC.

C. Whenever possible, fillets and radii should be scanned both transverse and parallel to the axis of the radius with surface scan EC. The edge of the fillet or radius shall be scanned transverse to the axis of the radius.

D. When performing bolthole EC inspections, the entire depth of a hole shall be inspected unless otherwise stated. Be aware that the hole may have more than a single layer of material.

<u>Inspection Interpretation</u>

A. If a surface scan EC indication is detected, carefully repeat the inspection in the opposite direction of probe movement to verify the indication. If the indication persists, carefully monitor the probes movement to verify the indication or rotation required to cause the signal to move off maximum indication response.

B. If performing bolthole EC inspection with the probe centered on a crack, the signal will be at a maximum and movement of the probe will cause the signal's amplitude to decrease and return to the baseline signal. Corrosion pits, foreign material, and out of round holes can cause an instrument response of 20 to 30 degrees of bolthole probe rotation before the indication begins to return to the baseline reading.

C. Unless otherwise specified, cracks shall be considered unacceptable.

D. The end of a crack is determined using the 50 percent method. Scan the probe slowly across the end of the crack until a point is reached where the crack signal's amplitude has been reduced by 50 percent. The center of the probe coil at this location is considered the end of the crack.

<u>NOTE</u>:

Perform a comprehensive visual inspection on all areas where access is too limited to perform EC inspections.

NOTE:

Due to accessibility issues, an inspector may not be able to perform all bolthole EC inspections. Therefore, surface scan EC procedures are provided. It is not necessary to perform both the bolthole EC and the surface scan EC inspections at the same location.

General Fluorescent Liquid Penetrant Inspections

General

Fluorescent liquid penetrant (FLP) inspection is effective in detecting small cracks and discontinuities open to the surface that may not be evident by normal visual inspection. Penetrant inspection can be used on most airframe parts and assemblies accessible for its application. The inspection is performed by applying a liquid that penetrates into surface discontinuities. The penetrant on the surface is removed and a suitable developer is applied to draw the remaining penetrant from the surface discontinuities. Indications are obtained visually by fluorescence of the penetrant when exposed to ultraviolet light.

Materials and Equipment

General

The equipment and material listed in each procedure are those used in the development of the inspection technique. Equivalent equipment and materials may be used if they provide equivalent or heightened sensitivity.

Materials

A. Only materials approved for listing on the latest revision to QPL-SAE-AMS-2644; Qualified Products List of Products Qualified under SAE Aerospace Material Specification AMS 2644 Inspection Materials, Penetrant; or an equivalent shall be used for penetrant inspection. All materials shall be from the same family group. Interchanging or mixing penetrant cleaners, developers, or other materials from different manufactures is prohibited.

WARNING:

Due to the oily nature of most penetrants, they SHALL NOT be used on parts such as assemblies where they cannot be completely removed and will subsequently come in contact with gaseous or liquid oxygen. Oils, even residual quantities, may explode or burn very rapidly in the presence of oxygen. Only materials specifically approved for this application SHALL be used if penetrant inspection is required and complete removal of the residue is not possible. Each application of

these special oxygen-compatible materials SHALL be directed by the applicable technical manual and/or upon direction by the responsible NDI engineering agency.

B. Penetrant materials are defined by specific classifications per SAE AMS 2644; Inspection Materials, Penetrant; or an equivalent and must meet or exceed the classifications listed below. This list assumes a portable inspection system for use on the airplane.

Type:	Type I (Fluorescent)
Method:	Method C (Solvent Removable)
Sensitivity Level:	High Sensitivity
Developer:	Form “d” Nonaqueous (Wet; for Type I)
Solvent Remover:	Class II Non-halogenated

C. Visible dye penetrants (Type II) shall not be used for inspections on this airplane or its components. This penetrant type has poor sensitivity compared to fluorescent liquid penetrant. It is extremely difficult to completely clean visible penetrant dyes from surface discontinuities under field conditions. Dye build up can prevent subsequent penetrant inspections from entering or indicating surface discontinuities.

CAUTION:

Type II (visible) penetrant shall not be used for the inspection of the airplane or its components, except those parts with specific engineering approval.

Lighting Requirements

A. Penetrant inspections shall be performed in a darkened environment where the ambient white light intensity does not exceed 2 lumen.

B. Ultraviolet lights used for penetrant inspection shall operate at a wavelength in the range of 320-380 nanometers. Light intensity shall be at least 1200 microwatts per square centimeter at the part surface or 1000 microwatts per square centimeter at a distance of fifteen inches. Ultraviolet lights shall be energized for at least 10 minutes before use.

C. The ultraviolet light and the ambient light intensities shall be measured with a calibrated light meter prior to each inspection.

Inspection

General

A. Fluorescent liquid penetrant shall be accomplished in accordance with the procedures contained or referenced in the SID. ASTM E1417, Standard Practice for Liquid Penetrant Examination, or an equivalent shall be consulted for the general requirements for penetrant inspection. In the event of a conflict between the text of the SID and ASTM E1417, the text of the SID shall take precedence.

B. Paint removal from the inspection area is required to allow penetration into surface discontinuities. In addition, the inspection area must be clean, dry, and free of dirt, grease, oil, paint, or any contaminants which would interfere with the liquid penetrant inspection. Cleaning and paint removal methods selected for a particular component shall be consistent with the contaminants to be removed and shall not be detrimental to the component or its intended function.

NOTE:

All cleaning materials must be approved for use by the appropriate Piper Airplane Maintenance Manual, Structural Repair Manual, Component Maintenance Manual, or NDI Manual.

NOTE:

Mechanical methods of cleaning and paint removal should be avoided where practical. Take care when using mechanical methods of cleaning and paint removal to avoid filling in or sealing the entrance to a surface discontinuity. Penetrant inspection can not show discontinuities that are not open at the surface.

CAUTION:

Halogenated solvents shall not be used on titanium or high nickel alloy materials.

C. Throughout the penetrant inspection process, the materials, equipment, and area to be inspected shall maintain a temperature within the range of 40 to 125 degrees Fahrenheit. For temperatures between 40 to 50 degrees Fahrenheit, the dwell time should be a minimum of 20 minutes.

Penetrant Application

Completely cover the inspection area with the penetrant. Allow penetrant to remain on the area (dwell) for a minimum of 10 minutes for temperatures above 50 degrees Fahrenheit or 20 minutes for temperature under 50 degrees Fahrenheit. Maximum dwell times should not exceed one hour except under special circumstances.

NOTE:

If penetrant is allowed to dry on the inspection surface, it shall be completely removed and the cleaning and inspection re-accomplished.

Penetrant Removal

Initially, remove the penetrant by wiping with a clean, dry, lint free cloth. Then remove the remaining penetrant using a clean, lint free cloth dampened with the penetrant cleaner. Examine the inspection area with the ultraviolet light to ensure removal of the surface penetrant. This process is complete when all the excess surface penetrant has been removed from the area.

NOTE:

Do not flush the surface or saturate the cloth with the cleaner as it may mask smaller discontinuities, preventing their detection.

Developer Application

A. Inspection shall occur after a minimum dwell time of ten minutes, but not after a maximum dwell time of 1 hour.

B. The best result is obtained by applying the developer to achieve the minimum coating thickness possible. The coating should be slightly translucent with the color of the inspection area visible through the developer.

Interpretation

A. Personnel shall not wear light sensitive (photochromatic) lenses during the evaluation process.

B. Personnel shall allow a minimum of three minutes for dark adaptation of the eyes prior to performing inspections.

General Magnetic Particle Inspections

General

A. Magnetic particle inspection (MPI) is an NDI method for revealing surface and near surface discontinuities in parts made of magnetic materials. Alloys that contain a high percentage of iron and can be magnetized make up the ferromagnetic class of metals. The MPI method consists of three basic operations:

1) Establishment of suitable magnetic field
2) Application of magnetic particles
3) Examination and evaluation of the particle accumulations

B. Electrical current is used to create or induce magnetic fields into the material. The direction of the magnetic field can be altered and is controlled by the direction of the magnetizing current. When the magnetic field within a part is interrupted by a discontinuity, some of the field is forced out into the air above the discontinuity. The presence of a discontinuity is detected by the application of finely divided fluorescent ferromagnetic particles to the surface of the part. Some of the particles will be gathered and held by the leakage field. The magnetically held collection of particles forms an outline of the discontinuity and indicates its location, size, and shape.

Material and Equipment

A. Fluorescent MPI has a high sensitivity and the ability to detect small cracks. Visible dry magnetic particles do not have the required sensitivity.

CAUTION:

Visible dry magnetic particles shall not be used for inspection of airplanes or components.

B. The equipment and materials listed in each procedure were those used in the development of the inspection technique. Equivalent equipment and materials may be used if they provide equivalent or heightened sensitivity.

C. Magnetic particle inspection shall be accomplished in accordance with the procedures contained or referenced in the SID. ASTM E1444, Standard Practice for Magnetic Particle Examination, and ASTM E709, Standard Guide for Magnetic Particle Examination, or equivalents shall be consulted for general requirements of MPI. In the event of a conflict between the text of the SID and ASTM E1444 or ASTM E709, the text of the SID shall take precedence.

CAUTION:

Permanent magnets shall not be used for inspection of airplanes or components.

D. Permanent magnets shall not be used, as the intensity of the magnetic field can not be altered to suit inspection conditions.

CAUTION:

Contact prods shall not be used for inspection of airplanes or components.

E. Contact prods shall not be used due to concerns with localized heating of the surface and arcing of the electrical current.

Quality Control

Quality control of magnetic particle materials and equipment shall be accomplished per ASTM E1444, ASTM E709, or equivalent document. This section assumes the use of a portable magnetic particle system for use on an airplane (electromagnetic yoke, spray can type magnetic particles, and portable ultraviolet light).

Dead-Weight Check

The electromagnetic yoke shall demonstrate the ability to lift ten pounds with leg spacing of 2 to 4 inches while operating on alternating current (AC). It shall demonstrate the ability to lift either 30 pounds with a leg spacing of 2 to 4 inches or 50 pounds with leg spacing of 4 to 6 inches while operating on direct current (DC).

Lighting Requirements

A. Magnetic particle inspection shall be performed in a darkened environment where the ambient white light intensity does not exceed 2 lumen.

B. Ultraviolet lights used for magnetic particle inspection shall operate at a wavelength in the range of 320-380 nanometers. Light intensity shall be at least 1200 microwatts per square centimeter at the part surface or 1000 microwatts per square centimeter at a distance of 15 inches. Ultraviolet lights shall be energized for at least 10 minutes before use.

C. The ultraviolet light and the ambient light intensities shall be measured with a calibrated light meter prior to each inspection.

Inspection

A. Magnetic particle inspection can be accomplished through layers of paint not exceeding 0.003-inch thick. If the paint is too thick, it will interfere with the inspection, and shall be removed. Cleaning and paint removal methods selected for a particular component shall be consistent with the contaminants to be removed and shall not be detrimental to the component or its intended function.

NOTE:

Thin coatings such as cadmium, chromium, or a single coat of paint, if in good condition, will not interfere with the inspection process, and do not necessarily have to be removed. Parts that have been repainted or touched up may be thicker which requires stripping. Paint and plated coatings, if over 0.003-inch thick, have to be stripped. Tests have shown nonmagnetic coatings of any kind, in excess of 0.003 inch in thickness, can interfere with the formation of magnetic particle indications of small discontinuities.

NOTE:

All cleaning materials must be approved for use by the appropriate Piper Aircraft Maintenance Manual, Structure Repair Manual, Component Maintenance Manual or NDI Manual.

B. An adequate magnetic field for inspection shall be tested using a Hall Effect meter, field indicator, or equivalent detector. Quality indicators approved in ASTM E1444, ASTM E709, or equivalent equipment may be used to determine the presence of an adequate magnetic field.

NOTE:

Field indicators SHALL be kept away from fields strong enough to damage the needle because of rapid or violent deflection beyond full-scale reading. Field indicators SHALL NOT be stored within the influence of magnetizing or demagnetizing magnetic flux.

C. When possible, the preferred method of particle application is the continuous method.

D. A minimum of 3 minutes dark adaptation time shall be allowed before performing inspections.

E. Personnel shall not wear light sensitive (photochromatic) lenses during the evaluation process.

Inspections

TITLE

JASC 2710, Aileron Hinges and Attachment Fittings (P/N 45384-00, 45384-01, 45380-00, 45380-01, 54332-02, 54332-03)

DESCRIPTION

Inspect for defects in the aileron attachment fittings and hinges (WS 177, 222, 238)

REQUIRED DISASSEMBLY

1. Remove the wing tip.
2. Remove the wing tip aft attachment rib.
3. Disconnect the aileron control rod.
4. At the right aileron, disconnect the trim tab control rod.
5. Remove the hinge bolts.
6. Remove the aileron.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.
2. Remove paint from the aileron hinge assembly and attachment fittings using an approved chemical paint stripper.

INSPECTION METHOD

Fluorescent Liquid Penetrant

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent liquid penetrant materials may be used providing they meet the minimum of a Type I, Level III sensitivity capable of achieving the requirements listed in the General Section, FLP of the SID.

Manufacturer	Part Number
Met-L-Chek	FP-93A (M) Penetrant
Met-L-Chek	R-504 Cleaner/Remover
Met-L-Chek	D-70 Developer

Spectronics	DSE-100X Light Meter
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Surface Preparation – The aileron hinges and attachment fittings must be clean, dry, free of dirt, grease, oil, paint, or any contaminants which would fill, mask, or close a defect open to the surface.

    a) Remove the paint in the area to be inspected using an approved chemical stripper. The bearing areas around the inspection zone should be masked or protected.
    b) Thoroughly water rinse and dry the area prior to applying cleaner.
    c) Prepare the inspection area by scrubbing the inspection surface with a cloth that is damp with penetrant cleaner to remove any contaminants.
    d) Thoroughly dry the area before penetrant application.

2. Penetrant Application
   Penetrant shall be applied by spraying, dipping, or brushing to provide complete coverage of the aileron hinges and attachment fittings. The penetrant shall completely cover the area of interest for a minimum dwell time of 10 minutes. The penetrant shall not be allowed to dry on the part surface.

<u>CAUTION:</u>

Type II (visible dye) penetrant shall not be used for inspection of aircraft parts.

3. Penetrant Removal
   Remove the excess penetrant by first wiping the part surface with a dry, clean, lint-free cloth. The surface of the component shall not be flushed with solvent. Examine the inspection area under a black light to ensure the removal of all surface penetrant. Over-removal of the surface penetrant shall require that the components be cleaned and reprocessed. The part surface shall be dried by blotting with a clean, dry towel/cloth or by evaporation.

4. Application of Developer
   The aileron hinges and attachment fittings shall be completely dry before the application of developer. Nonaqueous developer shall be applied by spraying and allowed to dry at ambient temperature. Apply the developer as a uniform thin coating over the entire surface to be inspected. The minimum dwell time for nonaqueous developers is 10 minutes. The dwell time starts after the developer is dry on the component when using form "d" nonaqueous developers.

NOTE:

The aerosol nonaqueous developer shall be frequently agitated before and during application.

5. Interpretation
   The inspection area shall consist of a darkened booth or an area where the ambient white light does not exceed 2 lumen when measured by a radiometer. Viewing areas for portable FLP inspections shall use a dark canvas, photographer's black cloth, or other methods to reduce the white light background to the lowest level possible during inspection.

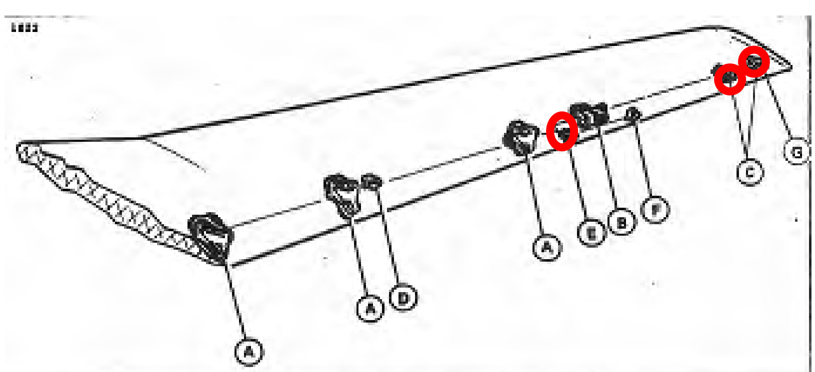
   The inspection area shall be viewed using a black light that provides a minimum of 1000 microwatts per square centimeter at the component surface. Do not position black lights closer than 6 inches from the inspection surface.

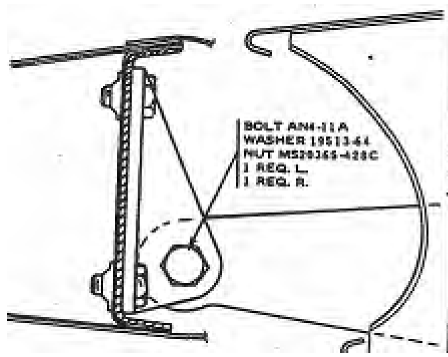
   All areas of fluorescence shall be interpreted. Components with excessive background or irrelevant indications that interfere with the detection of relevant indications shall be cleaned and reprocessed. Indications may be evaluated by wiping no more than twice. Ten power magnifiers may be used to interpret or evaluate indications.

6. Cracks detected during this inspection shall be documented and reported.

7. Post Cleaning
   Remove all developer and penetrant material from the part surface using the appropriate penetrant cleaner. Verification of adequate post cleaning shall be conducted using a black light.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.

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NUT MS20365-428C
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1 REQ. R.

TITLE

JASC 2720, Rudder Torque Tube (P/N 40094-02)

DESCRIPTION

Inspect for defects in the rudder torque tube (WL 23 and FS 345).

REQUIRED DISASSEMBLY

1. Relieve cable tension from the control system by removing the floor panel to the left of the control pedestal and loosening one of the rudder cable turnbuckles.
2. Remove the access panel located on top of the fuselage, aft of the vertical fin.
3. With the control cable tension relieved, disconnect the control cable from the rudder horn.
4. Disconnect the rudder trim control rod.
5. Swing the rudder and remove the hinge bolts.
6. Pull the rudder back and up removing the unit.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection. If bubbling of paint is observed, then remove paint with an approved paint stripper.

INSPECTION METHOD

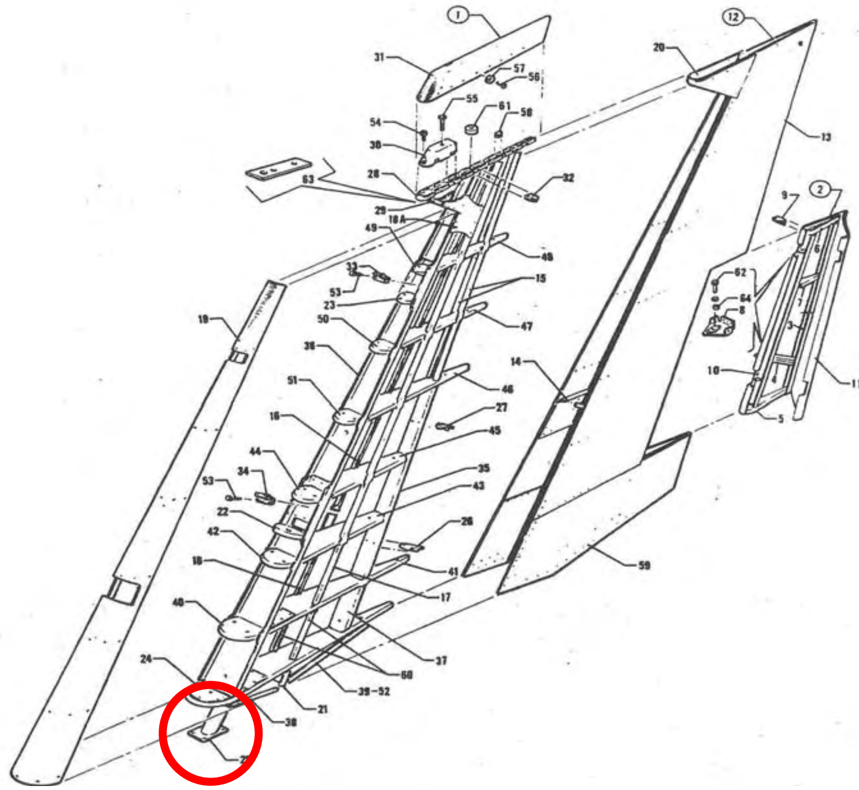
Visual

INSPECTION INSTRUCTIONS

1. Verify that AD 2003-24-7 has been complied with.

2. Visually inspect the rudder torque tube for condition, cracks, corrosion, and other defects.

3. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 2730, Elevator Torque Tube (P/N 40070-05)

DESCRIPTION

Inspect for defects in the elevator torque tube (FS 352, WL 23, BL 0 – BL 7).

REQUIRED DISASSEMBLY

1. Remove the screws that attach the fuselage tail cone, pull the cone back far enough to disconnect the navigation light wires, and then remove the tail cone.
2. At the right elevator, disconnect the trim tab control rod.
3. Remove the bolts that attach the elevator torque tube bracket to the elevator.
4. Remove hinge bolts and remove elevator.
5. To remove the elevator torque tube assembly, after the elevators have been removed, disconnect the elevator push-pull rod at the control arm.
6. Remove the hinge bolt and separate the torque tube assembly from its mating hinge bracket.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

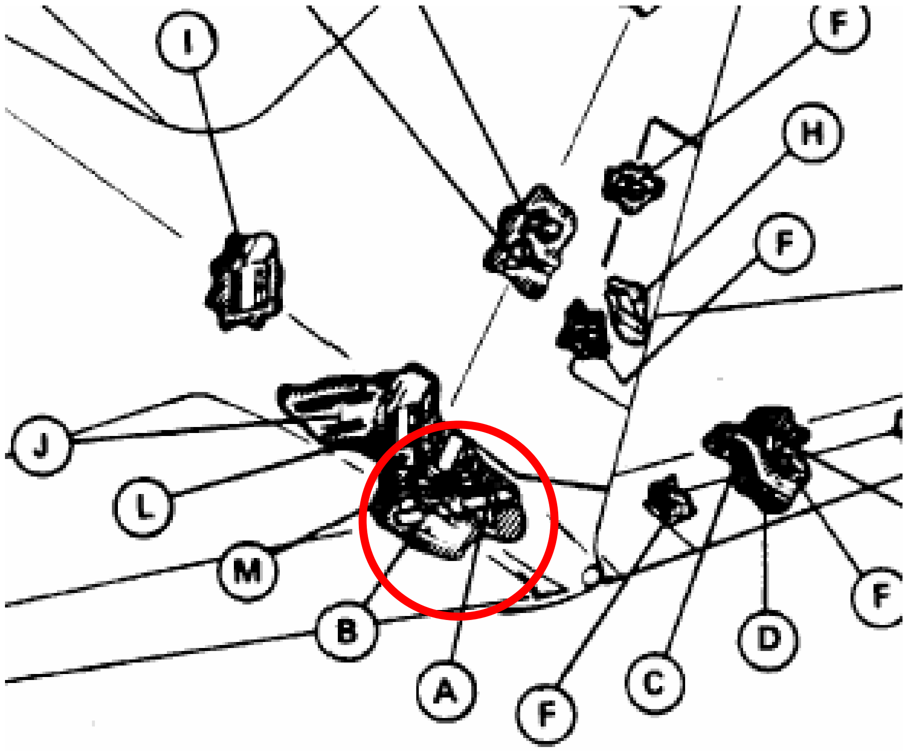
INSPECTION METHOD

Visual

INSPECTION INSTRUCTIONS

1. Visually inspect the elevator torque tube for condition, cracks, corrosion, and other defects.

2. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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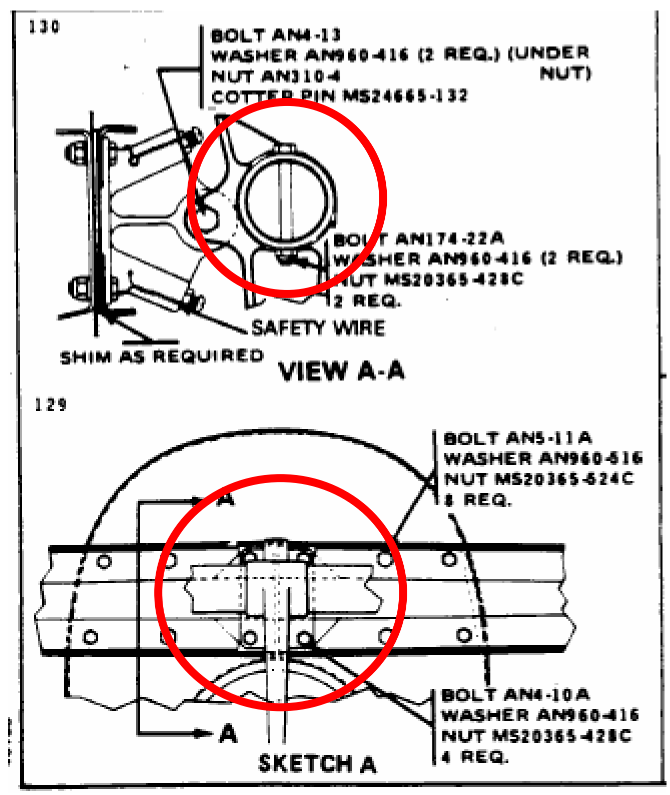
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WASHER AN960-416 (2 REQ.) (UNDER NUT)
NUT AN310-4
COTTER PIN MS24665-132
BOLT AN174-22A
WASHER AN960-416 (2 REQ.)
NUT MS20365-428C
2 REQ.
SAFETY WIRE
SHIM AS REQUIRED
VIEW A-A
129
BOLT AN5-11A
WASHER AN960-516
NUT MS20365-524C
8 REQ.
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BOLT AN4-10A
WASHER AN960-416
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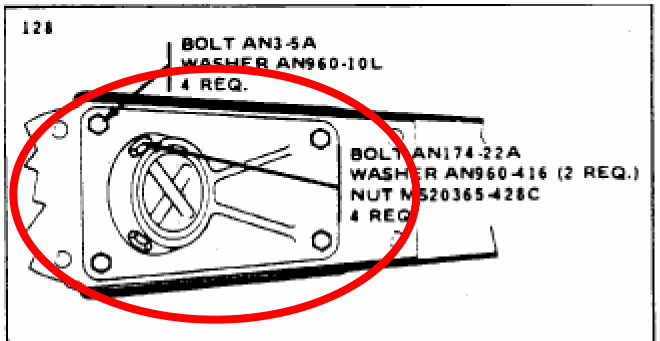
4 REQ.
SKETCH A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4 REQ.
BOLT AN174-22A
WASHER AN960-416 (2 REQ.)
NUT MS20365-428C
4 REQ.
SKETCH B

TITLE

JASC 3211, Landing Gear Strut Housing Attachment Points (P/N F 40273-00, L 45504-02, R 45504-03)—Magnetic Particle Inspection

DESCRIPTION

Inspect for defects around the landing gear strut housing attachment points.

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove the down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washers that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Remove paint from the inspection area using an approved chemical paint stripper.

INSPECTION METHOD

Magnetic Particle

EQUIPMENT

The following types of magnetic particle yokes may be used to accomplish this inspection. Equivalent substitutes may be used for the listed equipment.

1. Direct current electromagnetic yokes with a dead weight lifting capacity of at least 50 pounds with 4- to 6-inch spacing.

2. Alternating current electromagnetic yokes with a dead weight capacity of at least 10 pounds with leg spacing of 2 to 4 inches.

Manufacturer	Part Number
Magnaflux	Magnaglo 14AM Magnetic Particle Bath
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Remove all dirt, oil, grease, and paint from the inspection area.

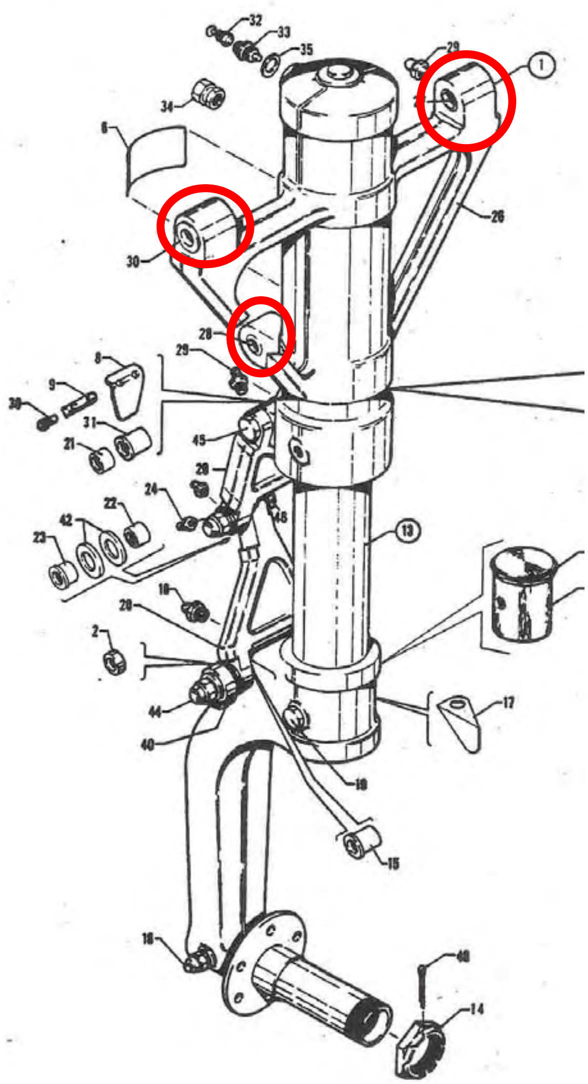
2. Position one leg on each side of the attachment point.

3. Apply the fluorescent magnetic particle bath to the inspection area. Stop the bath application, then immediately energize the yoke for approximately 1 second.

4. Inspect the attachment points for defects using a black light that has a minimum intensity of 1200 microwatts per square centimeter. The ambient light in the inspection area shall not exceed 2 lumen.

5. After completing the inspection, demagnetize the landing gear using the maximum alternating current. The residual magnetic field shall not exceed 3 gauss.

6. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 3211, Landing Gear Strut Housing Attachment Points (P/N F 40273-00, L 45504-02, R 45504-03)—Visual Inspection

DESCRIPTION

Inspect for defects in the landing gear strut housing attachment points.

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
    a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
    b. Remove the down-lock hook and spring by removing the pivot bolt.
    c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
    d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
    e. Remove the bolt that connects the upper and lower side brace links.
    f. Disconnect the aft link from its attachment plate.
    g. To remove the forward link, remove the nut with washers that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
    h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

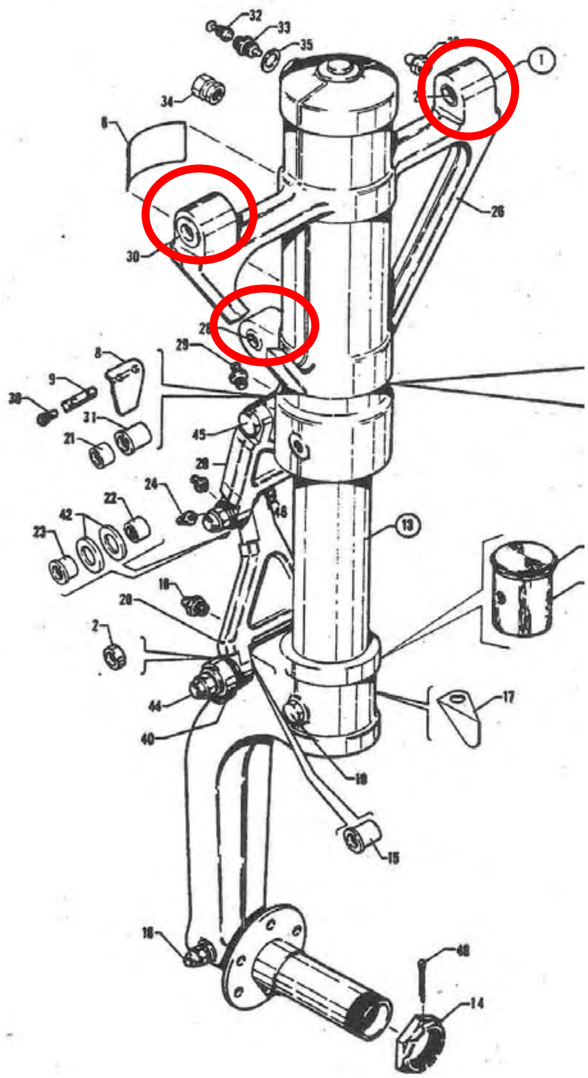
INSPECTION METHOD

Visual

INSPECTION INSTRUCTIONS

1. Visually inspect the landing gear strut housing attachment points for condition and defects.

2. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 3221, Nose Landing Gear Attachment Points Common To Fuselage (P/N 44677-00, 44677-01)

DESCRIPTION

Inspect for defects in the nose landing gear attachment points common to the fuselage (FS 23 and FS 48).

REQUIRED DISASSEMBLY

1. Remove the right and left access panels (FS 70) to the aft interior portion of the nose section. Remove the access plates located on the nose baggage compartment floor panel to gain access to the landing gear attachment bolts.
2. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
3. With the hand pump retract the nose gear slightly to relieve the gear from its down-locked position.
4. To remove the drag link assembly the following procedure may be used:
    a. Disconnect the gear retraction rod from the upper right drag link.
    b. Disconnect the lower drag link from the gear oleo housing.
    c. The upper and lower link assemblies may be removed as one unit by removing the upper drag link attachment bolts at their attachment plates.
5. With the lower drag link disconnected from the gear housing the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing. Note, if any, the number and location of spacer washers between the gear housing and attachment plates.
6. The idler link may be removed after the gear operating rod has been disconnected, by the following procedure:
    a. Remove the down-lock spring and the eye bolt that is attached to the idler link.
    b. Disconnect the gear actuating cylinder rod from the link.
    c. Remove the link pivot bolt by sliding the bolt out of the link, allowing the head to enter the hole in the side of the limit switch bracket. With the head through the bracket hole, the threaded end of the bolt can continue out of the link.
    d. Remove the idler link.
7. The uplock rod or cable may be removed by removing the nut from the actuating cylinder support bolt and sliding the rod or cable off the bolt. Retain the bolt in place to support the cylinder.
8. The uplock hook may be removed after the removal of the uplock rod or cable and the hook pivot bolt. Remove the hook with the uplock spring.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

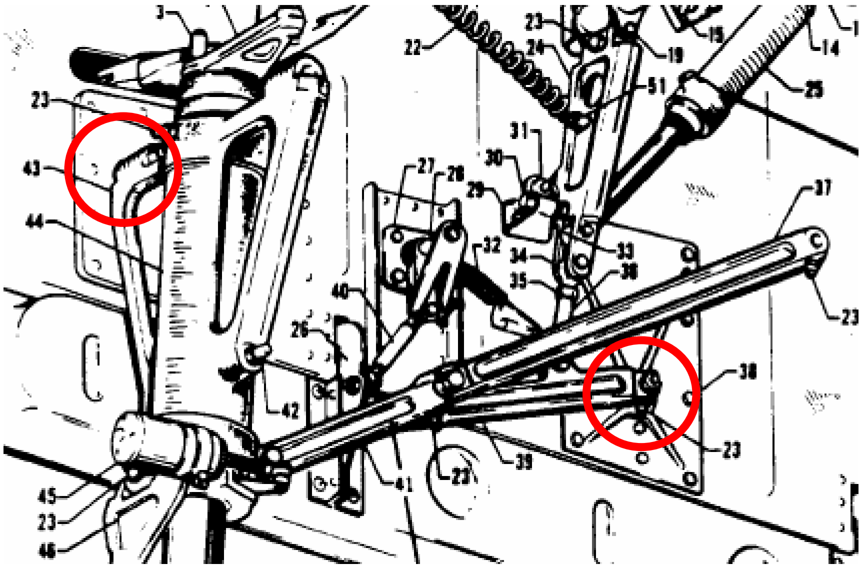
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

5. Inspect the area around and between the fasteners on the nose landing gear attachment points common to the fuselage. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 5511, Horizontal Stabilizer Butt Rib (P/N 42230-5, 42230-9)

DESCRIPTION

Inspect for defects on the horizontal stabilizer butt rib (BL 7 through BL 10).

REQUIRED DISASSEMBLY

1. Remove the left and right elevators.
2. Remove the access plates located on each side of the fuselage under the horizontal stabilizer and the panel located on top of the fuselage aft of the vertical fin.
3. Remove the access panel to the aft inside section of the fuselage.
4. To remove the right side of the stabilizer, locate the elevator trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation, and block the cables at one of the fuselage bulkheads and in the stabilizer to prevent the trim cables from unwinding.
5. Disconnect the trim cables.
6. Through the top access hole, remove the two elevator trim cable pulleys, spacer, and bolt. Draw the cables through the fuselage to this point.
7. Disconnect the elevator trim sender wires and deicer lines.
8. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
9. Remove the mounting bolts that attach the elevator torque tube hinge bracket and rear spar.
10. Pull the stabilizer directly away from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

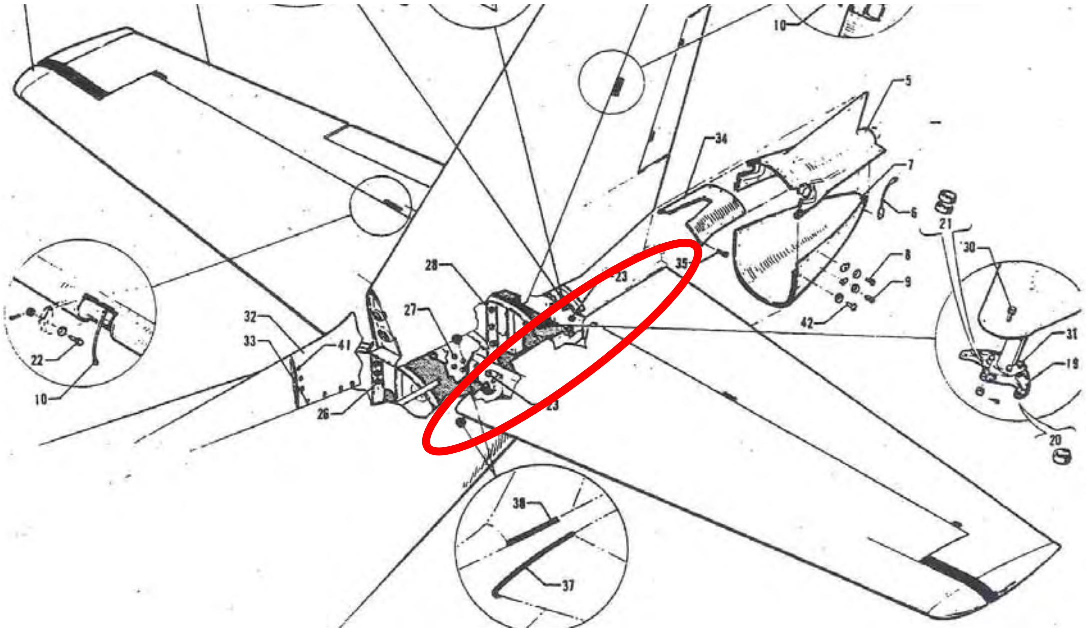
INSPECTION METHOD

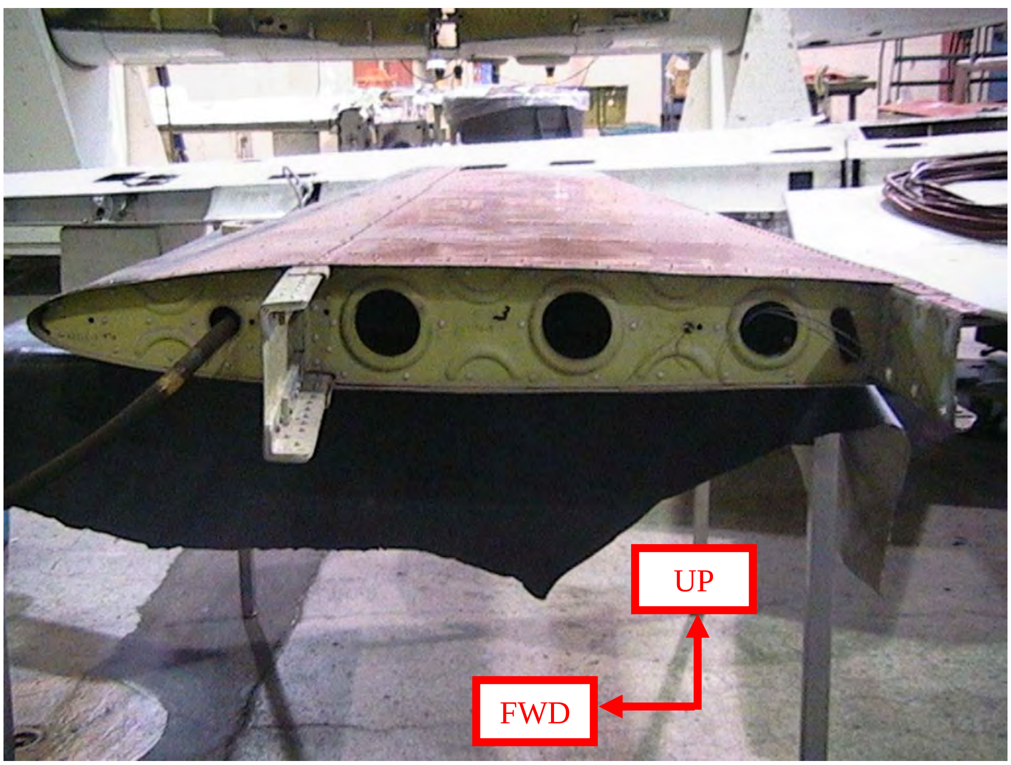
Visual

INSPECTION INSTRUCTIONS

1. Visually inspect the horizontal elevator butt rib for condition and defects. Verify that AD 99-12-05 has been complied with.

2. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for installation and rerigging procedures.





UP
FWD

TITLE

JASC 5524, Elevator Hinges and Attachment Fittings (P/N 40237-00, 42235-00, 54228-02, 71700-03)

DESCRIPTION

Inspect for defects in the elevator attachment fittings and hinges (BL 38, 90).

REQUIRED DISASSEMBLY

1. Remove the screws that attach the fuselage tail cone, pull the cone back far enough to disconnect the navigation light wires, and then remove the tail cone.
2. At the right elevator, disconnect the trim tab control rod.
3. Remove the bolts that attach the elevator torque tube bracket to the elevator.
4. Remove the hinge bolts and remove the elevator.
5. To remove the elevator torque tube assembly, after the elevators have been removed, disconnect the elevator push pull rod at the control arm.
6. Remove the hinge bolt and separate the torque tube assembly from its mating hinge bracket.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.
2. Remove paint from the elevator hinge assembly and attachment fittings using an approved chemical paint stripper.

INSPECTION METHOD

Fluorescent Liquid Penetrant

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent liquid penetrant materials may be used providing they meet the minimum of a Type I, Level III sensitivity capable of achieving the requirements listed in the FLP General Section.

Manufacturer	Part Number
Met-L-Chek	FP-93A (M) Penetrant
Met-L-Chek	R-504 Cleaner/Remover
Met-L-Chek	D-70 Developer

Spectronics	DSE-100X Light Meter
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Surface Preparation – The elevator hinges and attachment fittings must be clean, dry, free of dirt, grease, oil, paint, or any contaminants which would fill, mask, or close a defect open to the surface.

   a) Remove the paint in the area to be inspected using an approved chemical stripper. The bearing areas around the inspection zone should be masked or protected.
   b) Thoroughly water rinse and dry the area prior to applying cleaner.
   c) Prepare the inspection area by scrubbing the inspection surface with a cloth that is damp with penetrant cleaner to remove any contaminants.
   d) Thoroughly dry the area before penetrant application.

2. Penetrant Application
   Penetrant shall be applied by spraying, dipping, or brushing to provide completed coverage of the elevator hinges and attachment fittings. The penetrant shall completely cover the area of interest for a minimum dwell time of 10 minutes. The penetrant shall not be allowed to dry on the part surface.

CAUTION:

Type II (visible dye) penetrant shall not be used for the inspection of aircraft parts.

3. Penetrant Removal
   Remove the excess penetrant by first wiping the part surface with a dry, clean, lint free cloth. The surface of the component shall not be flushed with solvent. Examine the inspection area under a black light to ensure the removal of all surface penetrant. Over removal of the surface penetrant shall require that the components be cleaned and reprocessed. The part surface shall be dried by blotting with a clean, dry towel/cloth or by evaporation.

4. Application of Developer
   The elevator hinges and attachment fittings shall be completely dry before the application of developer. Nonaqueous developer shall be applied by spraying and allowed to dry at ambient temperature. Apply the developer as a uniform thin coating over the entire surface to be inspected. The minimum dwell time for nonaqueous developers is 10 minutes, with the maximum time of 1 hour. The dwell time starts after the developer is dry on the component when using form “d” nonaqueous developers.

NOTE:

The aerosol nonaqueous developer shall be frequently agitated before and during application.

5. Interpretation
   The inspection area shall consist of a darkened booth or an area where the ambient white light does not exceed 2 lumen when measured by a radiometer. Viewing areas for portable FLP inspections shall use a dark canvas, photographer's black cloth, or other methods to reduce the white light background to the lowest level possible during inspection.

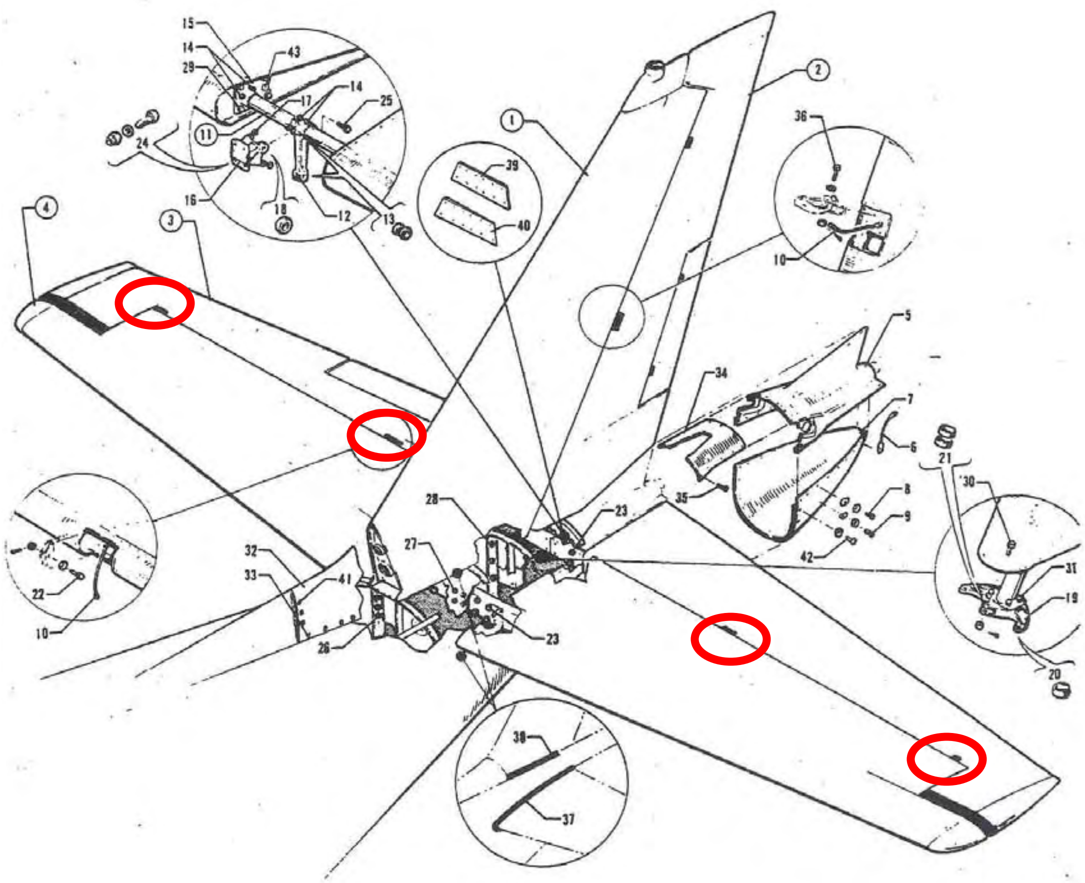
   The inspection area shall be viewed using a black light that provides a minimum of 1000 microwatts per square centimeter at the component surface. Do not position black lights closer than 6 inches from the inspection surface.

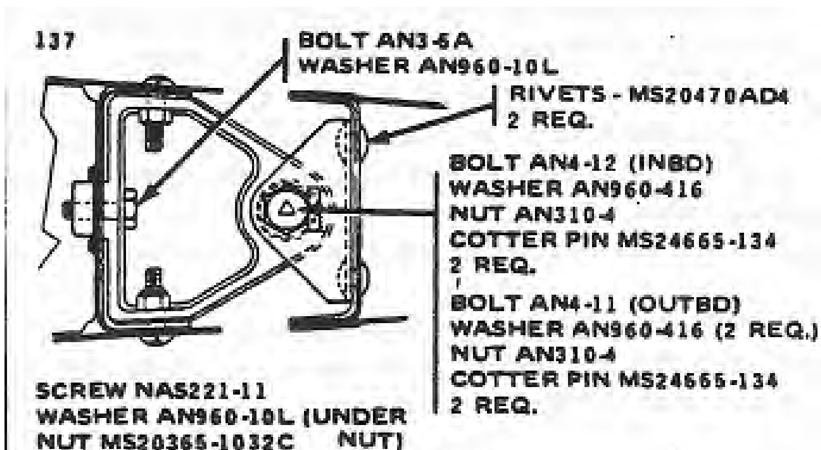
   All areas of fluorescence shall be interpreted. Components with excessive background or irrelevant indications which interfere with the detection of relevant indications shall be cleaned and reprocessed. Indications may be evaluated by wiping no more than twice. Ten power magnifiers may be used to interpret or evaluate indications.

6. Post Cleaning
   Remove all developer and penetrant material from the part surface using the appropriate penetrant cleaner. Verification of adequate post cleaning shall be conducted using a black light.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




137
BOLT AN3-6A
WASHER AN960-10L
RIVETS - MS20470AD4
2 REQ.
BOLT AN4-12 (INBD)
WASHER AN960-416
NUT AN310-4
COTTER PIN MS24665-134
2 REQ.
BOLT AN4-11 (OUTBD)
WASHER AN960-416 (2 REQ.)
NUT AN310-4
COTTER PIN MS24665-134
2 REQ.
SCREW NAS221-11
WASHER AN960-10L (UNDER NUT)
NUT MS20365-1032C
2 REQ.

SKETCH K

TITLE

JASC 5531, Vertical Stabilizer Front Spar Attachment Points (P/N 40026-02)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the vertical stabilizer front spar attachment points (BL 0-3).

REQUIRED DISASSEMBLY

1. Remove the air intake fairing, which is a portion of the dorsal fin.
2. Disconnect the rotating beacon wire, rudder tab sender wires, radio antenna cable, and deicer line.
3. Disconnect the antenna wire from the top of the stabilizer.
4. Remove the access plates located on each side of the fuselage, under the horizontal stabilizer and the panel located on top of the fuselage, aft of the vertical fin. The tail cone may be removed if desired.
5. Remove the access panel to the aft inside section of the fuselage.
6. Remove the rudder.
7. Locate the rudder trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation and block the cables in the aft section of the fuselage and in the rudder to prevent the cable from unwinding.
8. Disconnect the trim cables.
9. Through the right fuselage access holes, remove the two sets of trim cable pulleys, spacers, and bolts.
10. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
11. Remove the mounting bolts that attach the rear spar to the fuselage bulkhead.
12. Pull the stabilizer directly up from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

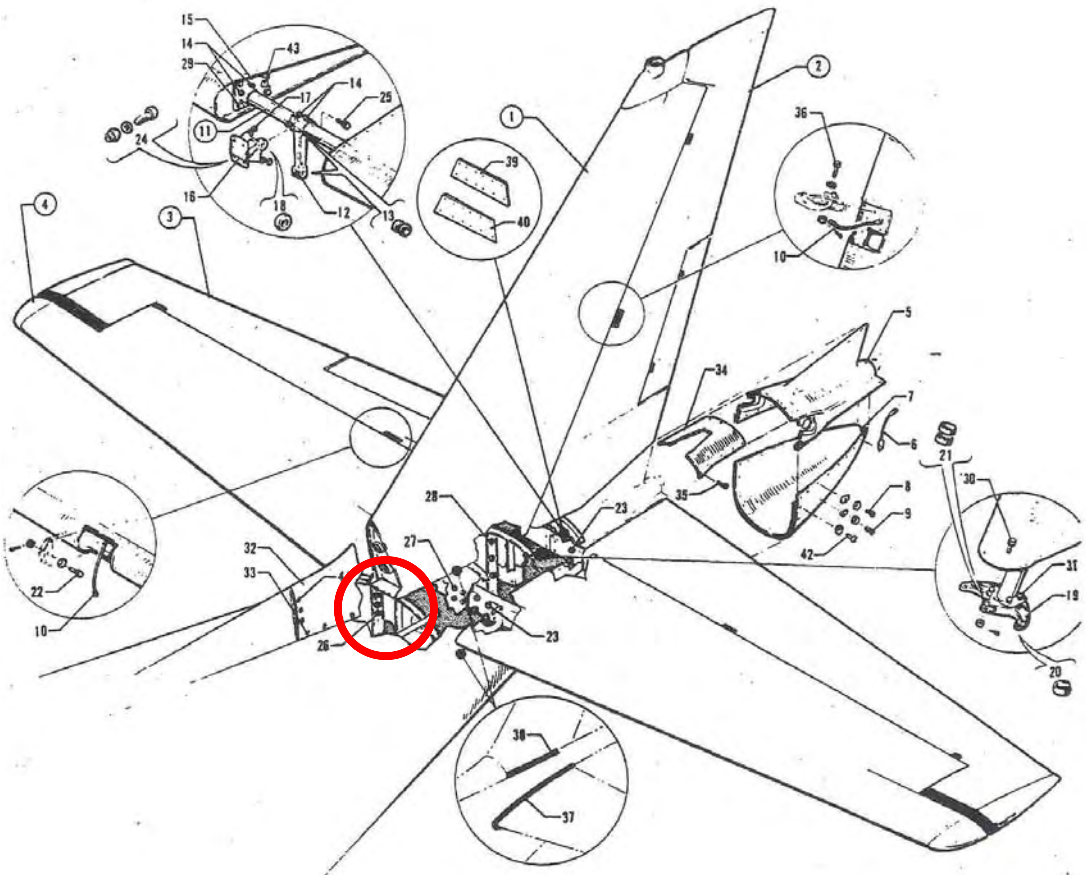
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

5. Inspect the area around and between the fasteners on the vertical stabilizer front spar and angle attachment points. Also inspect the area at the 45-degree bend in the front spar. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



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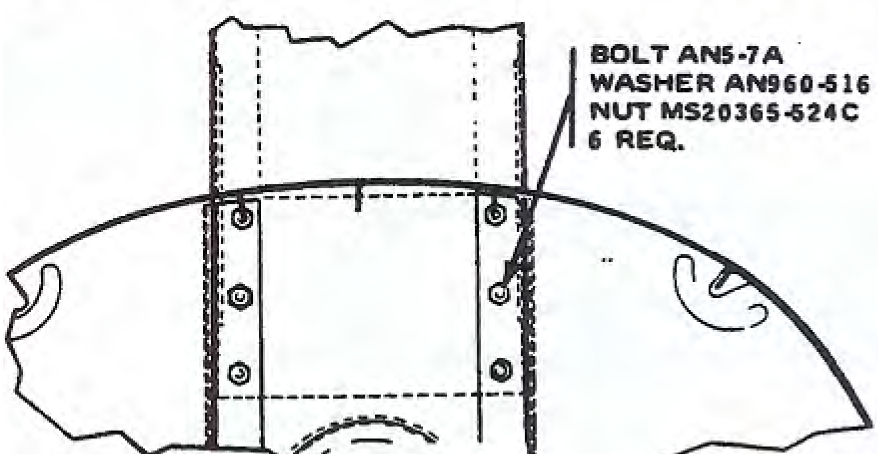


SKETCH I

TITLE

JASC 5531, Vertical Stabilizer Front Spar Attachment Points (P/N 40026-02)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the vertical stabilizer front spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the air intake fairing, which is a portion of the dorsal fin.
2. Disconnect the rotating beacon wire, rudder tab sender wires, radio antenna cable, and deicer line.
3. Disconnect the antenna wire from the top of the stabilizer.
4. Remove the access plates located on each side of the fuselage, under the horizontal stabilizer and the panel located on top of the fuselage, aft of the vertical fin. The tail cone may be removed if desired.
5. Remove the access panel to the aft inside section of the fuselage.
6. Remove the rudder.
7. Locate the rudder trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation and block the cables in the aft section of the fuselage and in the rudder to prevent the cable from unwinding.
8. Disconnect the trim cables.
9. Through the right fuselage access holes, remove the two sets of trim cable pulleys, spacers and bolts.
10. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
11. Remove the mounting bolts that attach the rear spar to the fuselage bulkhead.
12. Pull the stabilizer directly up from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

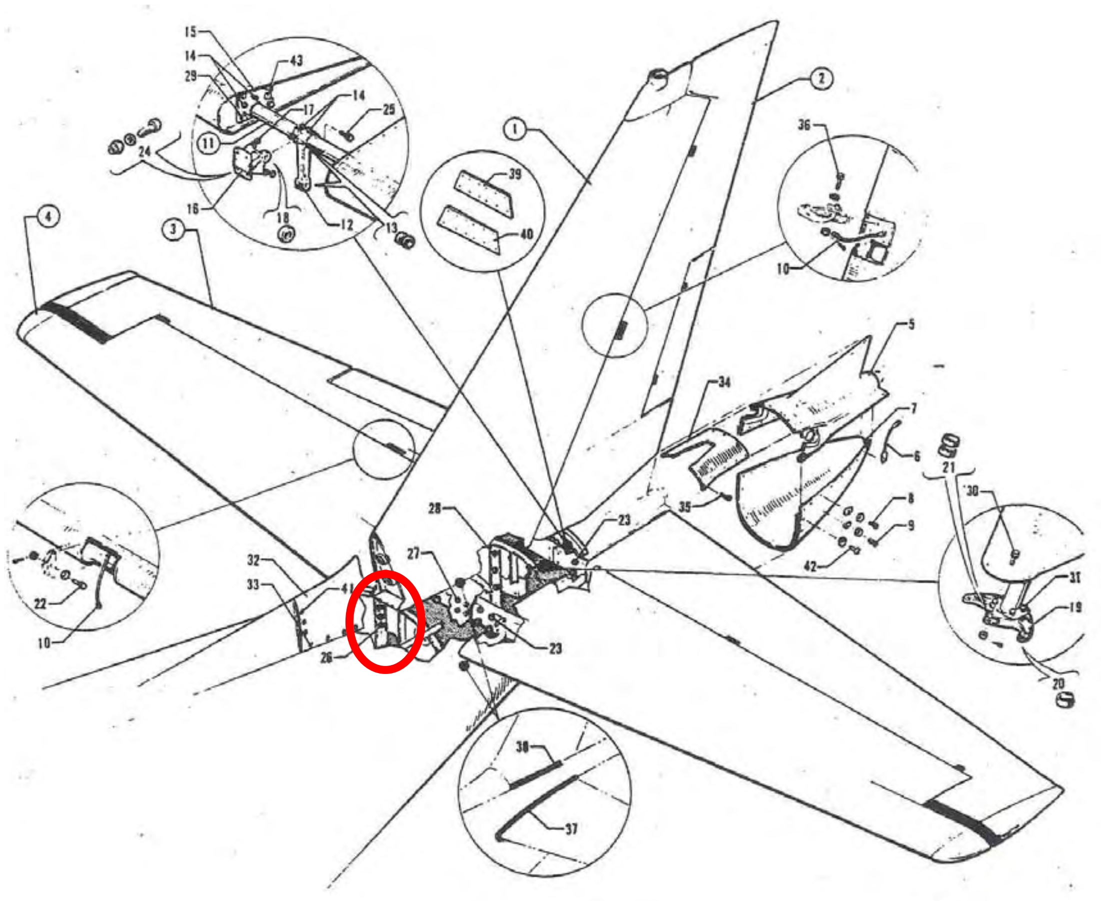
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

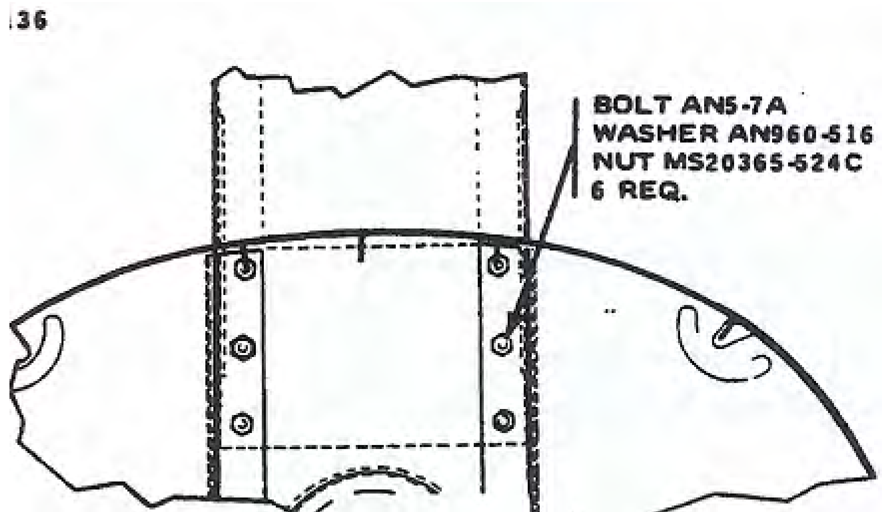
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





SKETCH I

TITLE

JASC 5531, Vertical Stabilizer Rear Spar Attachment Points (P/N 40026-03)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the vertical stabilizer rear spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the air intake fairing, which is a portion of the dorsal fin.
2. Disconnect the rotating beacon wire, rudder tab sender wires, radio antenna cable, and deicer line.
3. Disconnect the antenna wire from the top of the stabilizer.
4. Remove the access plates located on each side of the fuselage, under the horizontal stabilizer and the panel located on top of the fuselage, aft of the vertical fin. The tail cone may be removed if desired.
5. Remove the access panel to the aft inside section of the fuselage.
6. Remove the rudder.
7. Locate the rudder trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation and block the cables in the aft section of the fuselage and in the rudder to prevent the cable from unwinding.
8. Disconnect the trim cables.
9. Through the right fuselage access holes, remove the two sets of trim cable pulleys, spacers, and bolts.
10. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
11. Remove the mounting bolts that attach the rear spar to the fuselage bulkhead.
12. Pull the stabilizer directly up from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

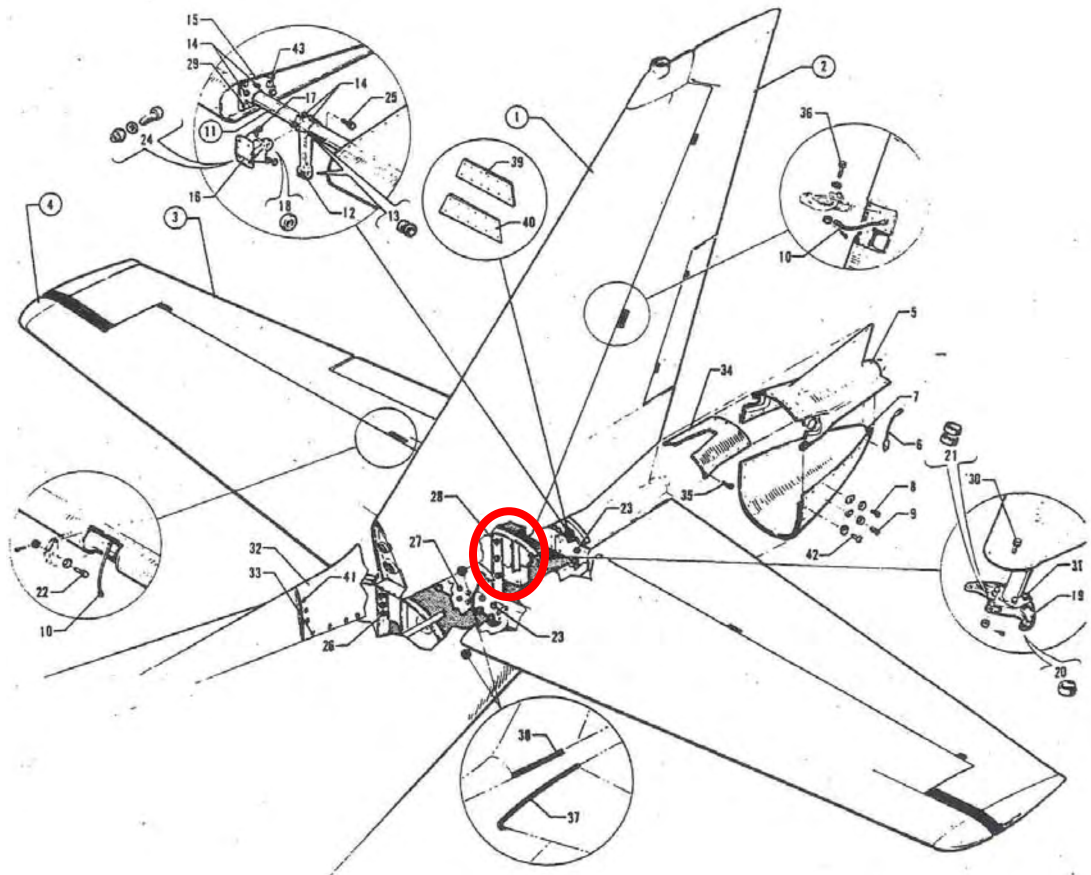
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

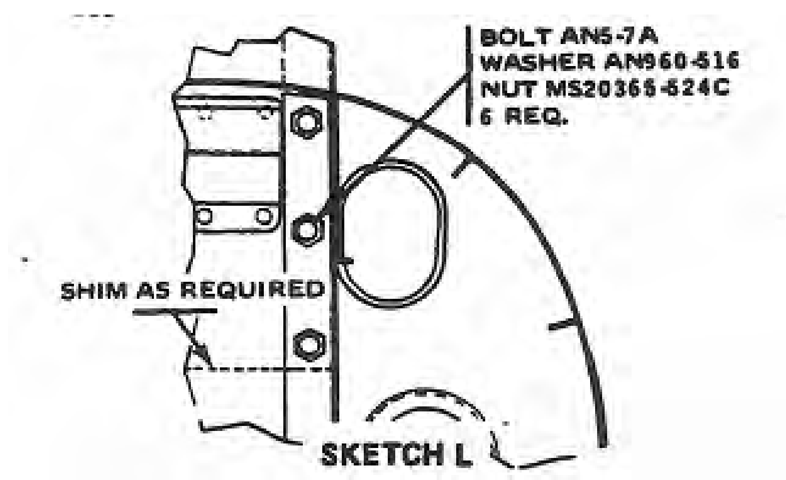
5. Inspect the area around and between the fasteners on the vertical stabilizer rear spar and angle attachment points. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




BOLT AN5-7A
WASHER AN960-516
NUT MS20365-524C
6 REQ.
SHIM AS REQUIRED
SKETCH L

TITLE

JASC 5531, Vertical Stabilizer Rear Spar Attachment Points (P/N 40026-03)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the vertical stabilizer rear spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the air intake fairing, which is a portion of the dorsal fin.
2. Disconnect the rotating beacon wire, rudder tab sender wires, radio antenna cable, and deicer line.
3. Disconnect the antenna wire from the top of the stabilizer.
4. Remove the access plates located on each side of the fuselage, under the horizontal stabilizer and the panel located on top of the fuselage, aft of the vertical fin. The tail cone may be removed if desired.
5. Remove the access panel to the aft inside section of the fuselage.
6. Remove the rudder.
7. Locate the rudder trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation and block the cables in the aft section of the fuselage and in the rudder to prevent the cable from unwinding.
8. Disconnect the trim cables.
9. Through the right fuselage access holes, remove the two sets of trim cable pulleys, spacers and bolts.
10. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
11. Remove the mounting bolts that attach the rear spar to the fuselage bulkhead.
12. Pull the stabilizer directly up from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

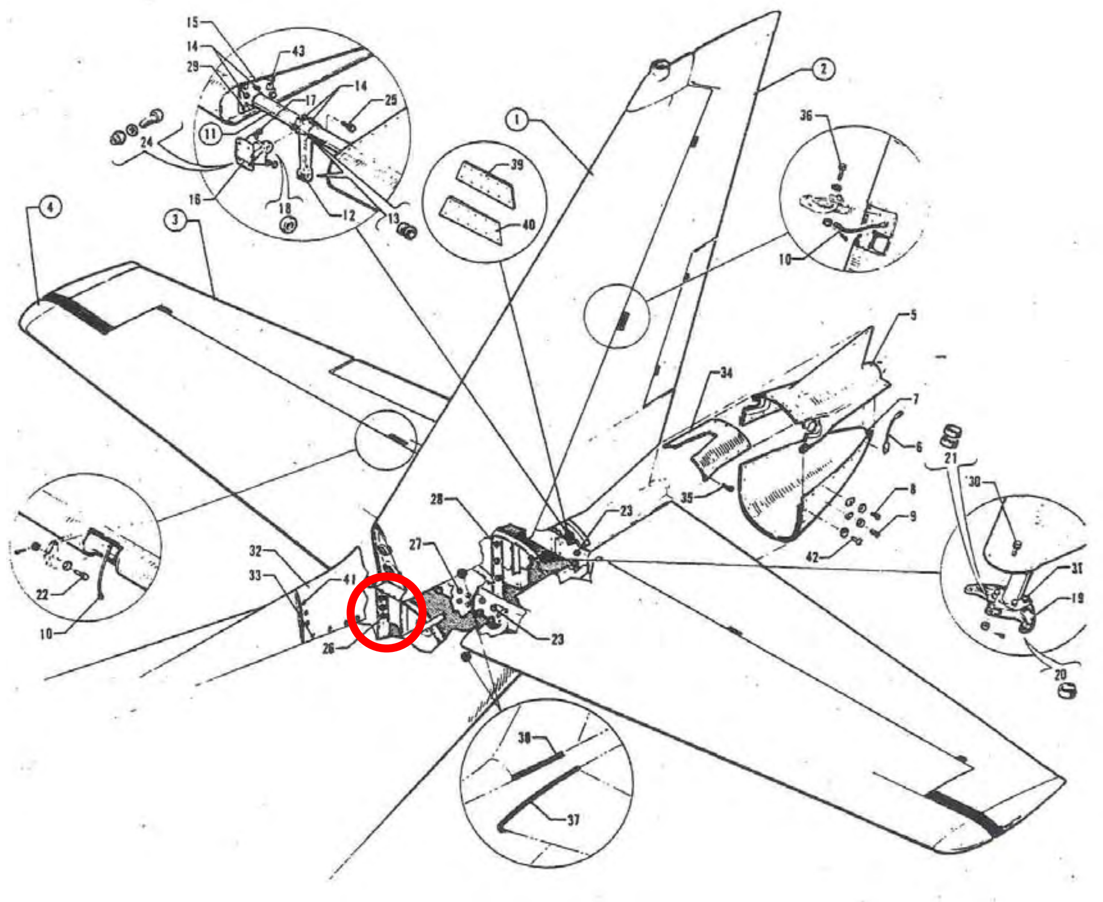
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

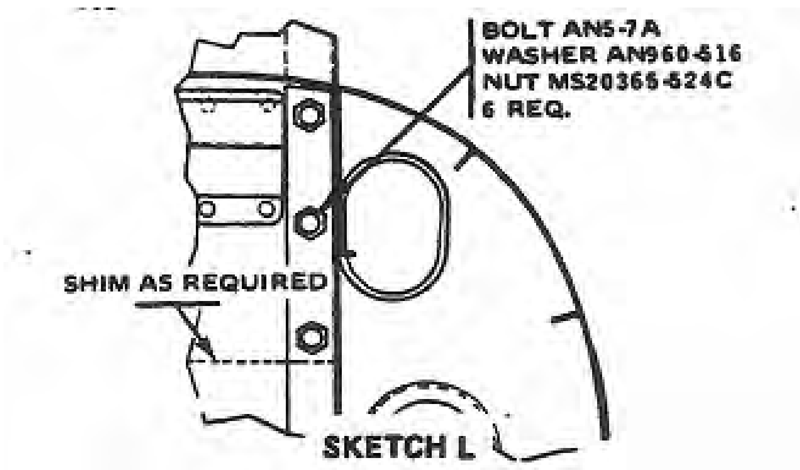
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




BOLT AN5-7A
WASHER AN960-516
NUT MS20365-524C
6 REQ.
SHIM AS REQUIRED
SKETCH L

TITLE

JASC 5544, Rudder Hinges and Attachment Fittings (P/N 40219-00, 40233-00, 48055-00)

DESCRIPTION

Inspect for defects in the rudder hinges and attachment fittings (WL 52, 83).

REQUIRED DISASSEMBLY

1. Relieve cable tension from the control system by removing the floor panel to the left of the control pedestal and loosen one of the rudder cable turnbuckles.
2. Remove the access panel located on top of the fuselage, aft of the vertical fin.
3. With the control cable tension relieved, disconnect the control cable from the rudder horn.
4. Disconnect the rudder trim control rod.
5. Swing the rudder and remove the hinge bolts.
6. Pull the rudder back and up removing the unit.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.
2. Remove paint from the rudder hinges and attachment fittings using an approved chemical paint stripper.

INSPECTION METHOD

Fluorescent Liquid Penetrant

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent liquid penetrant materials may be used providing they meet the minimum of a Type I, Level III sensitivity capable of achieving the requirements listed in the FLP General Section.

Manufacturer	Part Number
Met-L-Chek	FP-93A (M) Penetrant
Met-L-Chek	R-504 Cleaner/Remover
Met-L-Chek	D-70 Developer
Spectronics	DSE-100X Light Meter

Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Surface Preparation – The rudder hinges and attachment fittings must be clean, dry, free of dirt, grease, oil, paint, or any contaminants which would fill, mask, or close a defect open to the surface.

    a) Remove the paint in the area to be inspected using an approved chemical stripper. The bearing areas around the inspection zone should be masked or protected.
    b) Thoroughly water rinse and dry the area prior to applying cleaner.
    c) Prepare the inspection area by scrubbing the inspection surface with a cloth that is damp with penetrant cleaner to remove any contaminants.
    d) Thoroughly dry the area before penetrant application.

2. Penetrant Application
   Penetrant shall be applied by spraying, dipping, or brushing to provide complete coverage of the rudder hinges and attachment fittings. The penetrant shall completely cover the area of interest for a minimum dwell time of 20 to 30 minutes. The penetrant shall not be allowed to dry on the part surface.

CAUTION:

Type II (visible dye) penetrant shall not be used for inspection of aircraft parts.

3. Penetrant Removal
   Remove the excess penetrant by first wiping the part surface with a dry, clean, lint free cloth. The surface of the component shall not be flushed with solvent. Examine the inspection area under a black light to ensure the removal of all surface penetrant. Over removal of the surface penetrant shall require that the components be cleaned and reprocessed. The part surface shall be dried by blotting with a clean, dry towel/cloth or by evaporation.

4. Application of Developer
   The rudder hinges and attachment fittings shall be completely dry before the application of developer. Nonaqueous developer shall be applied by spraying and allowed to dry at ambient temperature. Apply the developer as a uniform thin coating over the entire surface to be inspected. The minimum dwell time for nonaqueous developers is ten minutes. The dwell time starts after the developer is dry on the component when using form “d” nonaqueous developers.

NOTE:

The aerosol nonaqueous developer shall be frequently agitated before and during application.

5. Interpretation
   The inspection area shall consist of a darkened booth or an area where the ambient white light does not exceed 2 lumen when measured by a radiometer. Viewing areas for portable (FLP) inspections shall use a dark canvas, photographer's black cloth, or other methods to reduce the white light background to the lowest level possible during inspection.

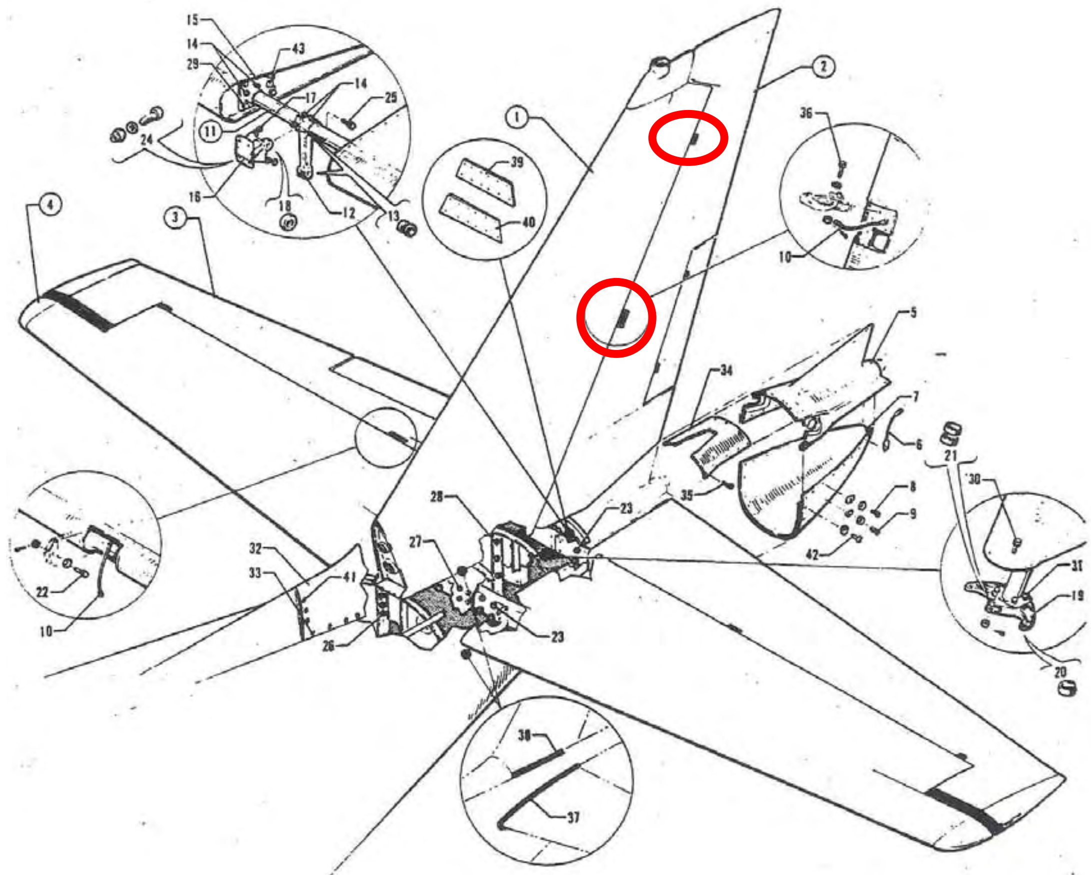
   The inspection area shall be viewed using a black light that provides a minimum of 1000 microwatts per square centimeter at the component surface. Do not position black lights closer than 6 inches from the inspection surface.

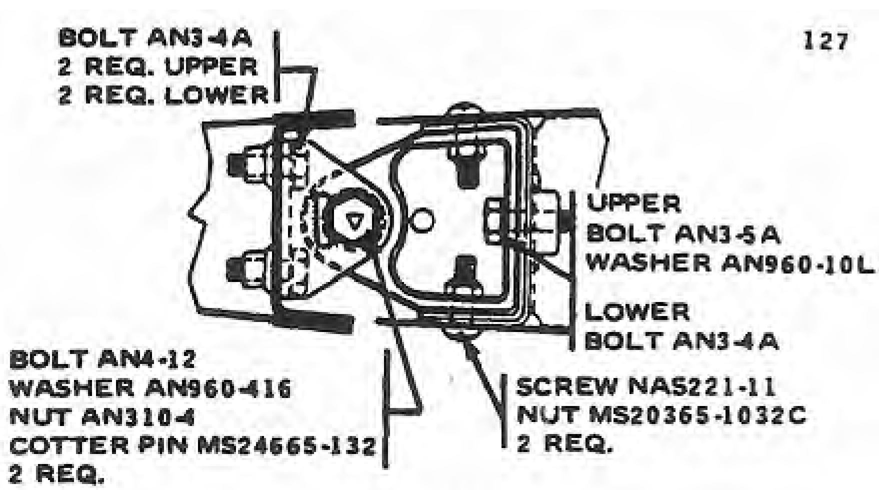
   All areas of fluorescence shall be interpreted. Components with excessive background or irrelevant indications which interfere with the detection of relevant indications shall be cleaned and reprocessed. Indications may be evaluated by wiping no more than twice. Ten power magnifiers may be used to interpret or evaluate indications.

6. Post Cleaning
   Remove all developer and penetrant material from the part surface using the appropriate penetrant cleaner. Verification of adequate post cleaning shall be conducted using a black light.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




BOLT AN3-4A
2 REQ. UPPER
2 REQ. LOWER
127
UPPER
BOLT AN3-5A
WASHER AN960-10L
LOWER
BOLT AN3-4A
BOLT AN4-12
WASHER AN960-416
NUT AN310-4
COTTER PIN MS24665-132
2 REQ.
SCREW NAS221-11
NUT MS20365-1032C
2 REQ.

SKETCH E

TITLE

JASC 5551, Horizontal Stabilizer Front Spar Attachment Points (P/N 40122-15, 40122-16)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the horizontal stabilizer front spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the left and right elevators.
2. Remove the access plates located on each side of the fuselage under the horizontal stabilizer and the panel located on top of the fuselage aft of the vertical fin.
3. Remove the access panel to the aft inside section of the fuselage.
4. To remove the right side of the stabilizer, locate the elevator trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation, and block the cables at one of the fuselage bulkheads and in the stabilizer to prevent the trim cables from unwinding.
5. Disconnect the trim cables.
6. Through the top access hole, remove the two elevator trim cable pulleys, spacer, and bolt. Draw the cables through the fuselage to this point.
7. Disconnect the elevator trim sender wires and deicer lines.
8. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
9. Remove the mounting bolts that attach the elevator torque tube hinge bracket and rear spar.
10. Pull the stabilizer directly away from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument

Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

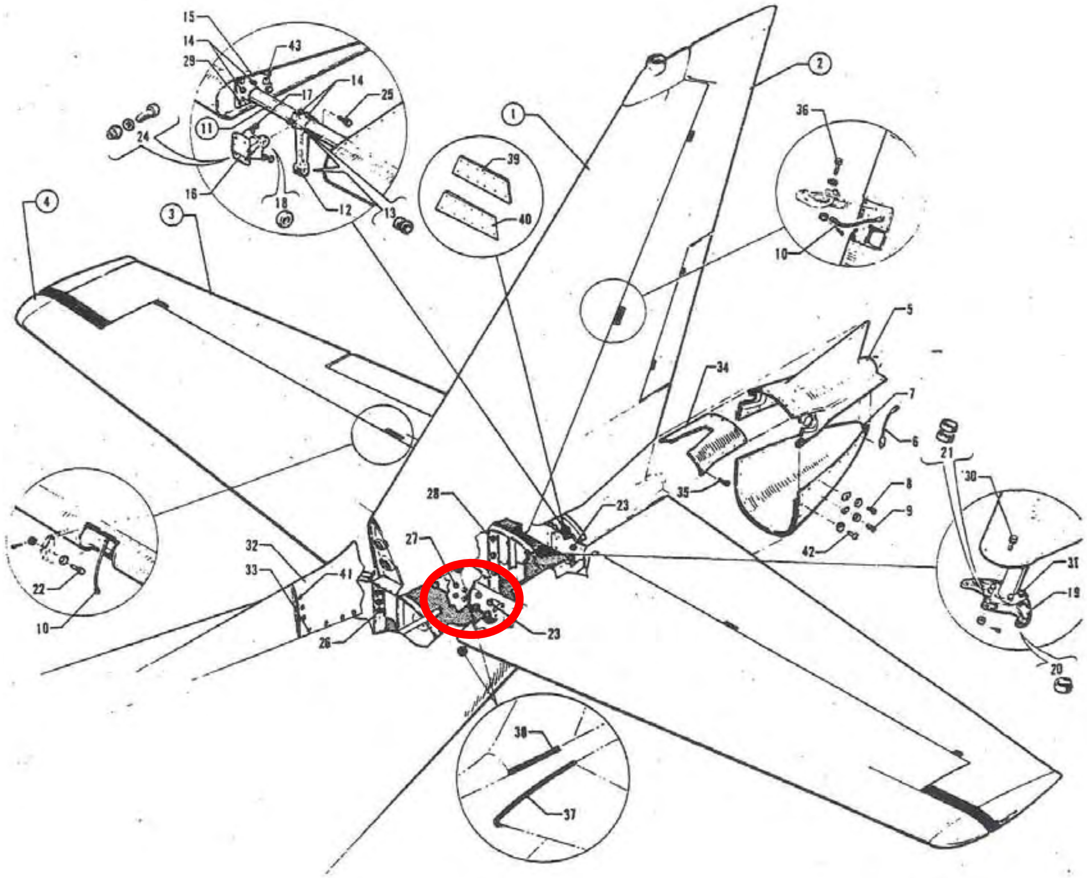
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

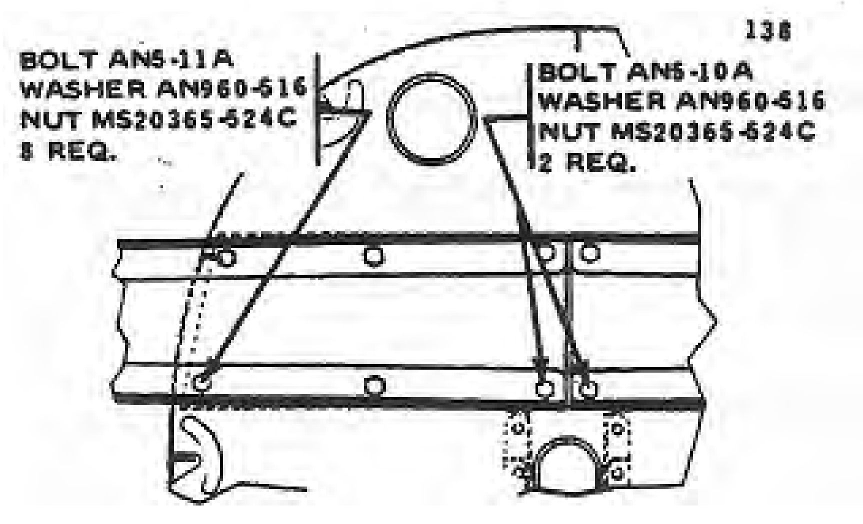
5. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




138
BOLT AN5-11A
WASHER AN960-516
NUT MS20365-524C
8 REQ.
BOLT AN5-10A
WASHER AN960-516
NUT MS20365-524C
2 REQ.

TITLE

JASC 5551, Horizontal Stabilizer Front Spar Attachment Points (P/N 40122-15, 40122-16)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the horizontal stabilizer front spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the left and right elevators.
2. Remove the access plates located on each side of the fuselage under the horizontal stabilizer and the panel located on top of the fuselage aft of the vertical fin.
3. Remove the access panel to the aft inside section of the fuselage.
4. To remove the right side of the stabilizer, locate the elevator trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation, and block the cables at one of the fuselage bulkheads and in the stabilizer to prevent the trim cables from unwinding.
5. Disconnect the trim cables.
6. Through the top access hole, remove the two elevator trim cable pulleys, spacer, and bolt. Draw the cables through the fuselage to this point.
7. Disconnect the elevator trim sender wires and deicer lines.
8. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
9. Remove the mounting bolts that attach the elevator torque tube hinge bracket and rear spar.
10. Pull the stabilizer directly away from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument

Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

<u>Reference Standard Notch Depths Bolthole EC:</u>
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

<u>NOTE:</u>

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

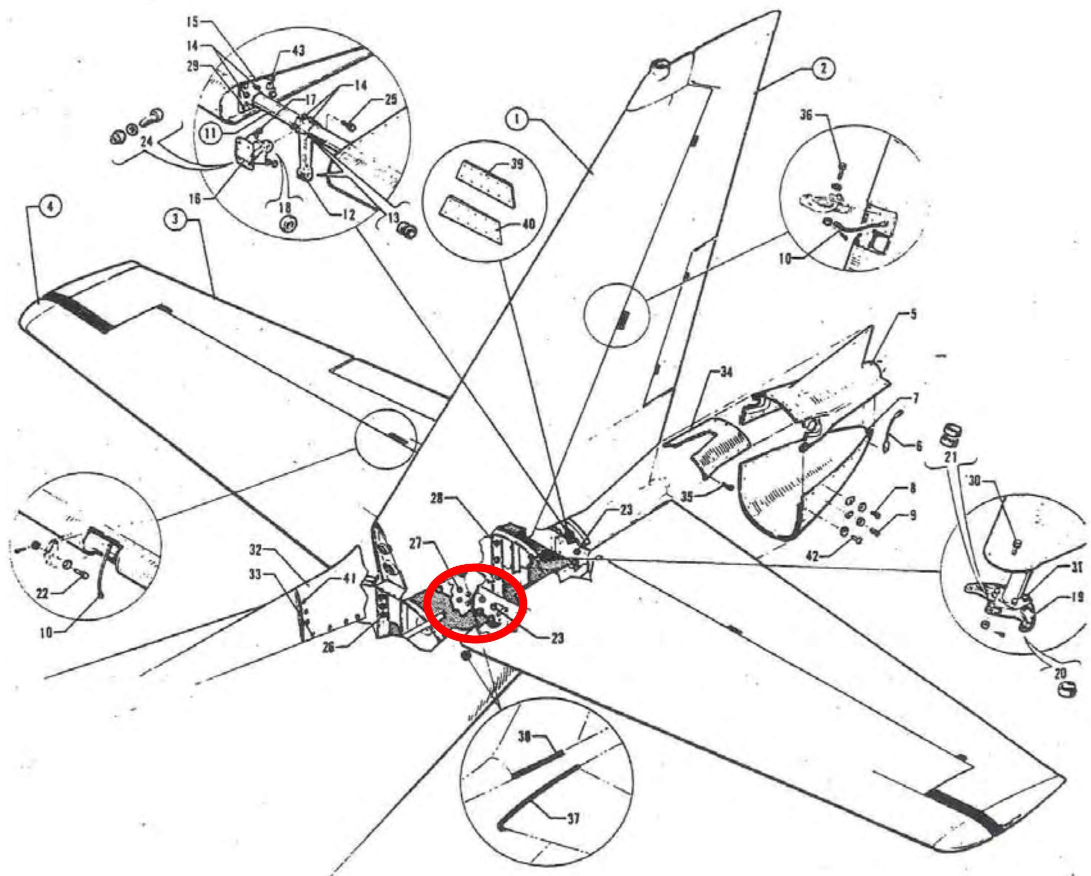
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

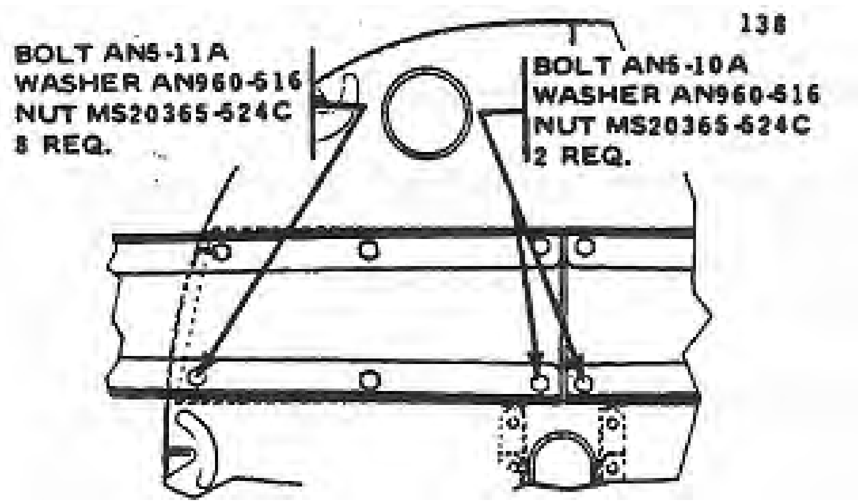
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




138
BOLT AN5-11A
WASHER AN960-516
NUT MS20365-524C
8 REQ.
BOLT AN5-10A
WASHER AN960-516
NUT MS20365-524C
2 REQ.

TITLE

JASC 5551, Horizontal Stabilizer Rear Spar Attachment Points (P/N 40122-02, 40122-03)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the horizontal stabilizer rear spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the left and right elevators.
2. Remove the access plates located on each side of the fuselage under the horizontal stabilizer and the panel located on top of the fuselage aft of the vertical fin.
3. Remove the access panel to the aft inside section of the fuselage.
4. To remove the right side of the stabilizer, locate the elevator trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation, and block the cables at one of the fuselage bulkheads and in the stabilizer to prevent the trim cables from unwinding.
5. Disconnect the trim cables.
6. Through the top access hole, remove the two elevator trim cable pulleys, spacers, and bolts. Draw the cables through the fuselage to this point.
7. Disconnect the elevator trim sender wires and deicer lines.
8. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
9. Remove the mounting bolts that attach the elevator torque tube hinge bracket and rear spar.
10. Pull the stabilizer directly away from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument

Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

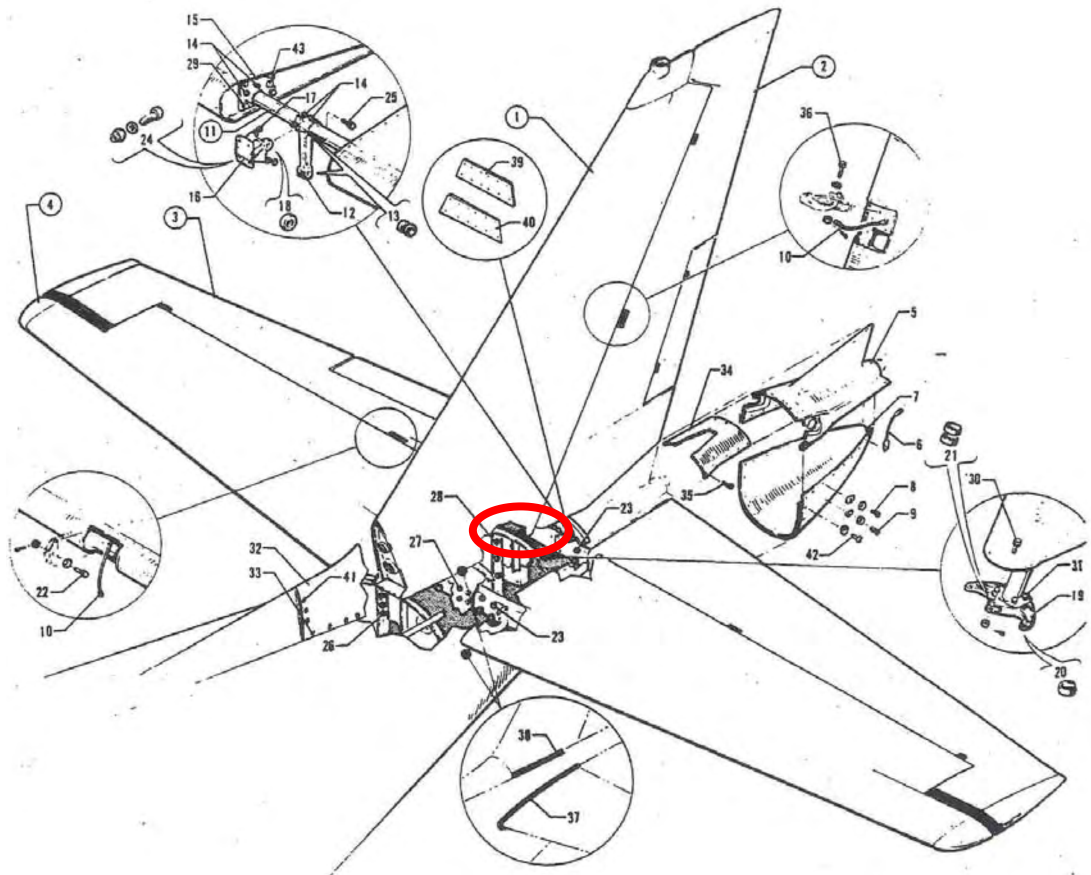
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

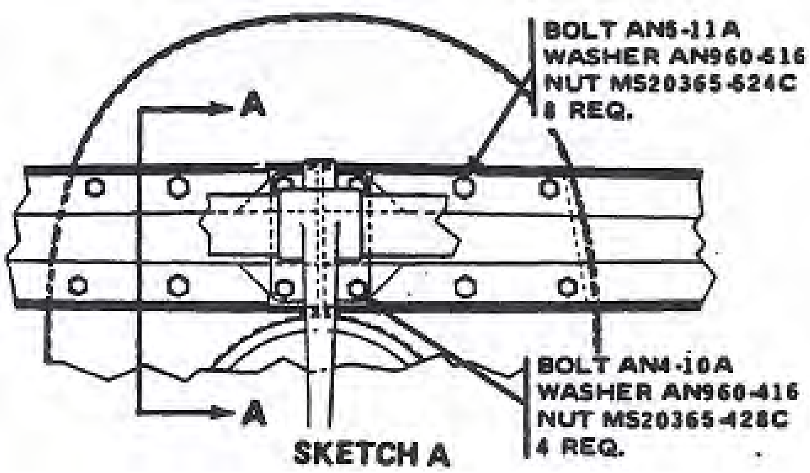
5. Inspect the area around and between the fasteners on the horizontal stabilizer rear spar attachment points. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





BOLT AN5-11A
WASHER AN960-516
NUT MS20365-524C
8 REQ.
A
A
BOLT AN4-10A
WASHER AN960-416
NUT MS20365-428C
4 REQ.

SKETCH A

TITLE

JASC 5551, Horizontal Stabilizer Rear Spar Attachment Points (P/N 40122-02, 40122-03)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for cracks in the horizontal stabilizer rear spar attachment points (BL 0-BL 3).

REQUIRED DISASSEMBLY

1. Remove the left and right elevators.
2. Remove the access plates located on each side of the fuselage under the horizontal stabilizer and the panel located on top of the fuselage aft of the vertical fin.
3. Remove the access panel to the aft inside section of the fuselage.
4. To remove the right side of the stabilizer, locate the elevator trim cable turnbuckles in the aft section of the fuselage, mark the ends of one turnbuckle to facilitate reinstallation, and block the cables at one of the fuselage bulkheads and in the stabilizer to prevent the trim cables from unwinding.
5. Disconnect the trim cables.
6. Through the top access hole, remove the two elevator trim cable pulleys, spacers, and bolts. Draw the cables through the fuselage to this point.
7. Disconnect the elevator trim sender wires and deicer lines.
8. Remove the mounting bolts that attach the front spar to the fuselage bulkhead.
9. Remove the mounting bolts that attach the elevator torque tube hinge bracket and rear spar.
10. Pull the stabilizer directly away from the fuselage.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument

Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

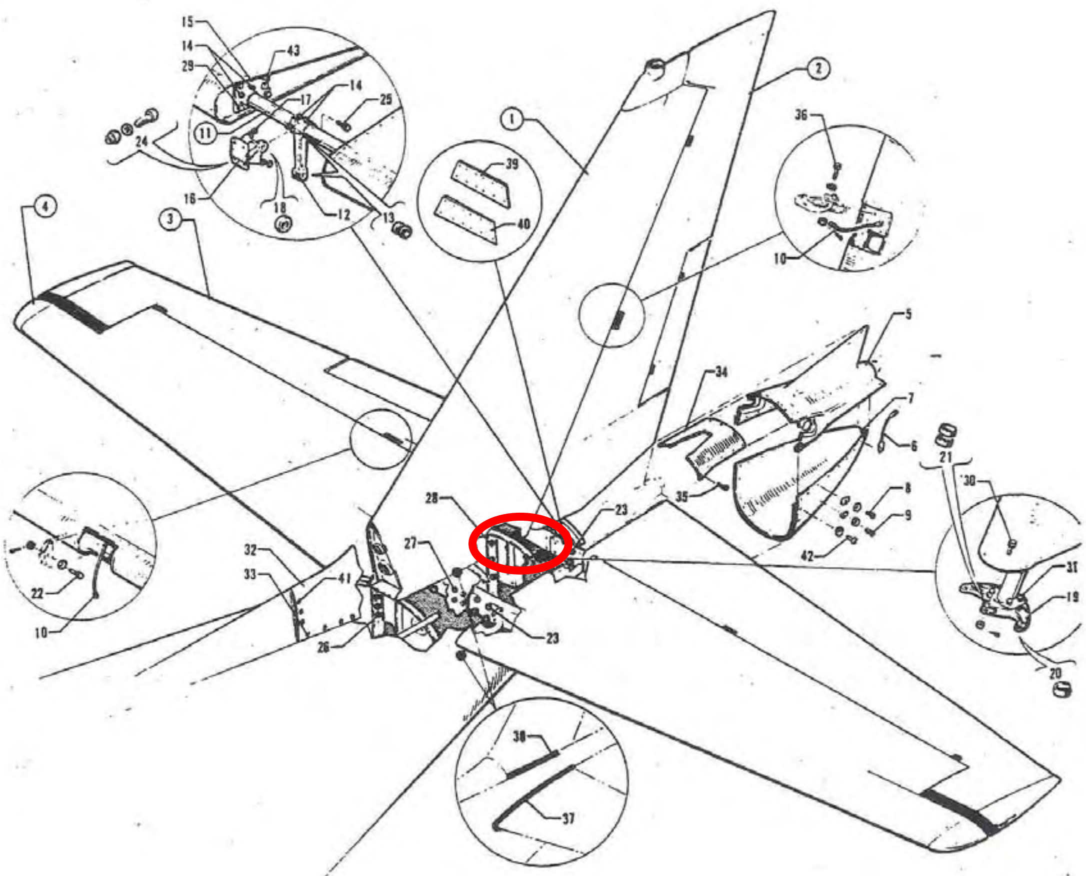
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

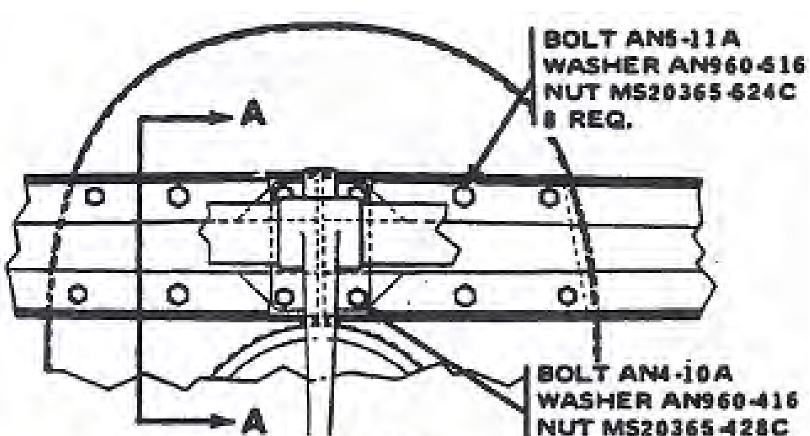
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





SKETCH A

TITLE

JASC 5710, Wing Front Spar Attachment Points (P/N 40487-04, 40487-05)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing front spar attachment points (WS 30).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

   d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at the butt end of the wing and tape cables out of the way at the fuselage.
   e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
   f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.
10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove support blocks and clamps.
13. Disconnect electrical wire connectors.
14. Draw engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay it forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

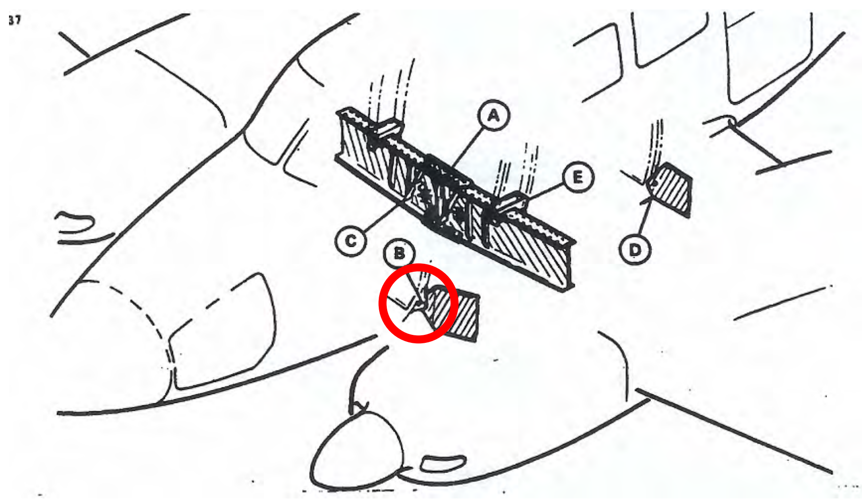
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

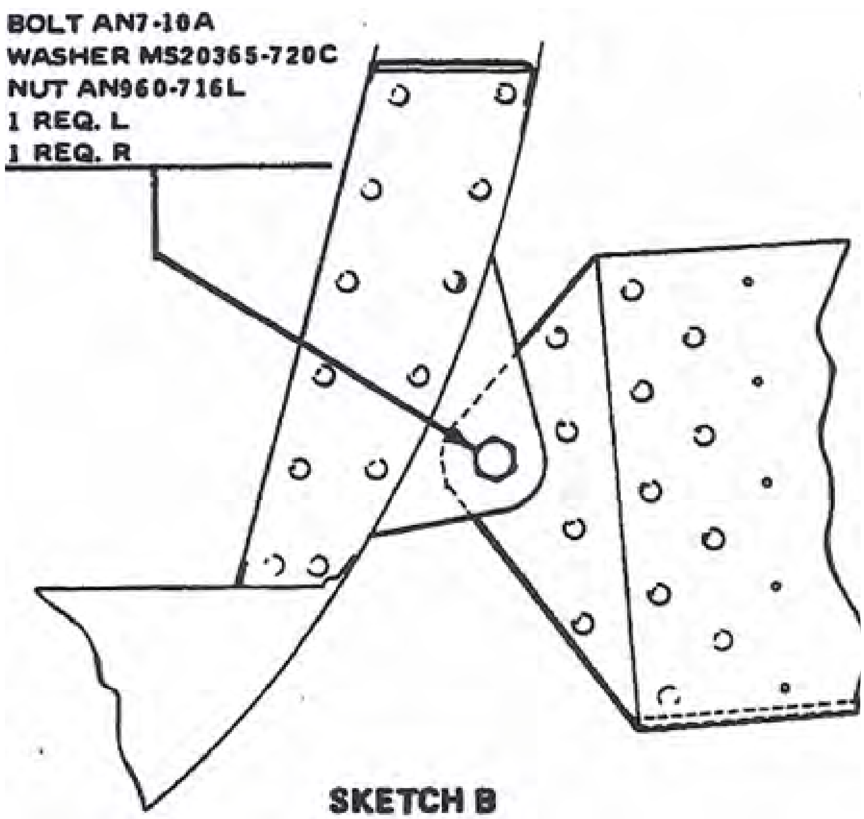
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and length and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedure.





SKETCH B

TITLE

JASC 5710, Wing Front Spar Attachment Points (P/N 40487-04, 40487-05)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing front spar attachment points (WS 30).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove cable guard at the butt end of wing and tape the cables out of the way at the fuselage.
e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.

10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove support blocks and clamps.
13. Disconnect electrical wire connectors.
14. Draw engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

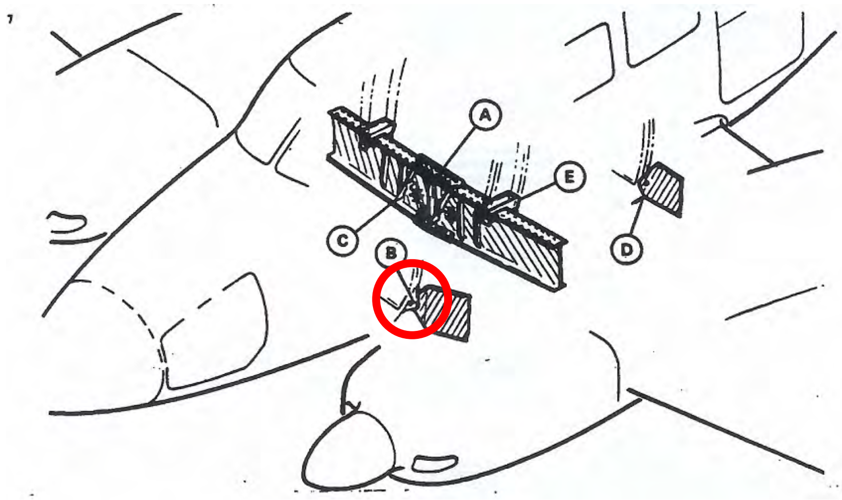
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of there major screen divisions.

5. Inspect the area around and between the fasteners on the wing front spar attachment points. Also inspect the structure inside the plane holding the front spar mount. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedure.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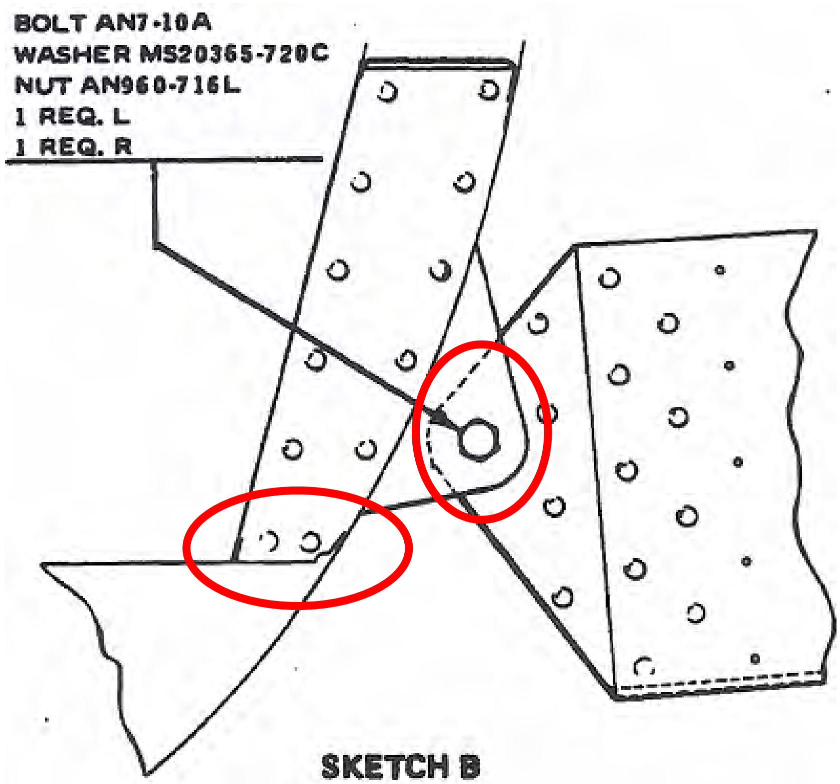
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WASHER MS20365-720C
NUT AN960-716L
1 REQ. L
1 REQ. R

SKETCH B

TITLE

JASC 5711, Engine Rib Lower Cap to Wing Main Spar Attachment (P/N 44677-00, 44677-01)

DESCRIPTION

Inspect for defects in the engine rib lower cap to wing main spar attachment (WS 65-WS 68 and WS 87-WS 91).

REQUIRED DISASSEMBLY

Remove five rivets in each region that are not common to the main spar.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

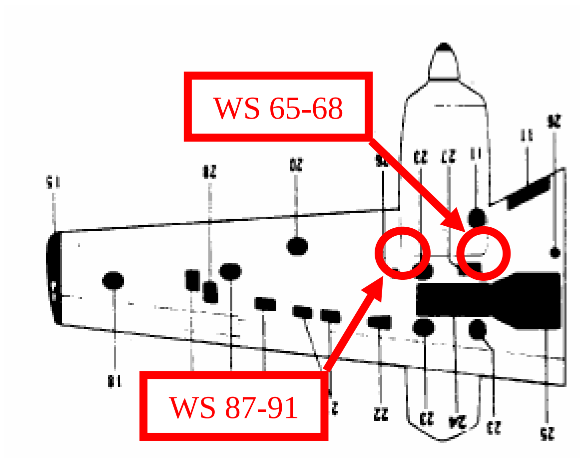
9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

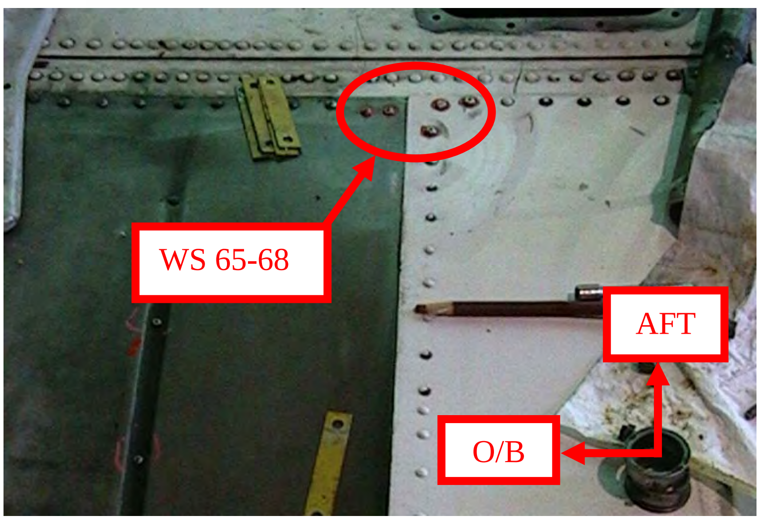
10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and length and direction of the indication.

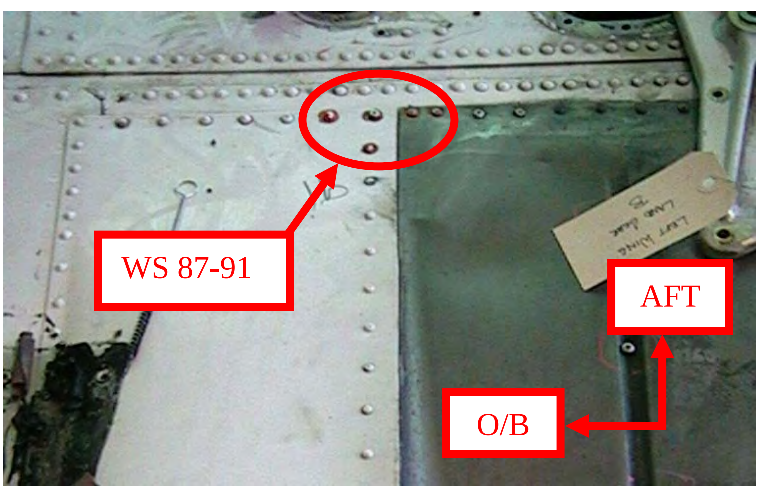
11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.

NOTE:

If holes are damaged and out of round, reaming is allowed as long as edge distance is within specification.







TITLE

JASC 5711, Wing Aft Auxiliary Spar Landing Gear Attachment Points (P/N 45501-00, 45501-01)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing aft auxiliary spar landing gear attachment points (WS 67.5, 85).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

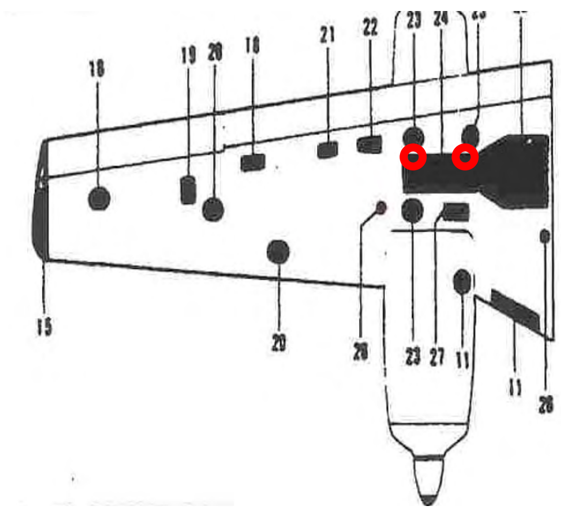
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

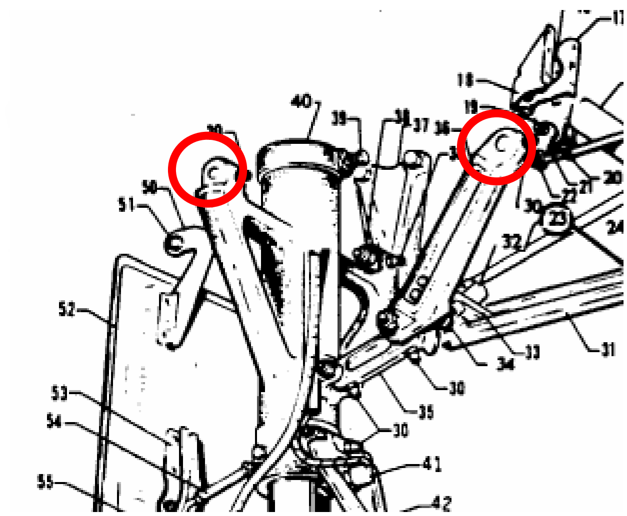
5. Inspect the area around and between the fasteners on the wing aft auxiliary spar landing gear attachment points. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





TITLE

JASC 5711, Wing Aft Auxiliary Spar Landing Gear Attachment Points (P/N 45501-00, 45501-01)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing aft auxiliary spar landing gear attachment points (WS 67.5, 85).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove the down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

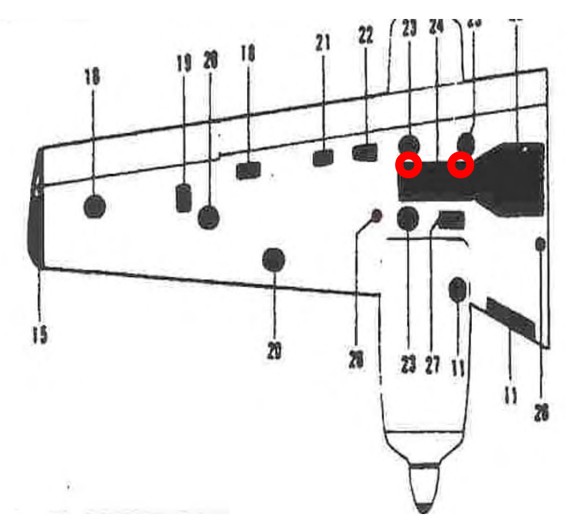
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

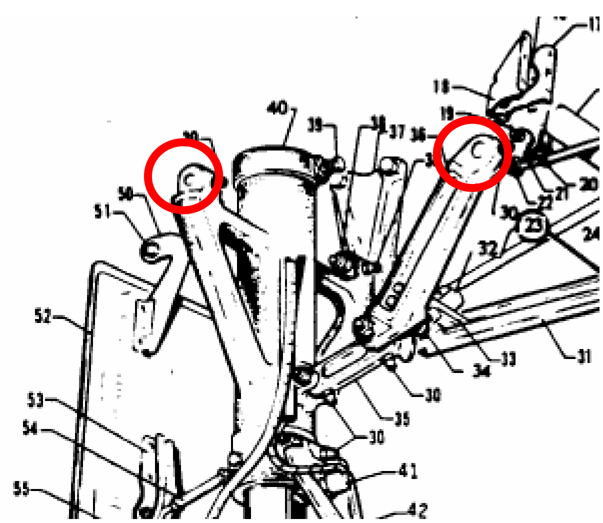
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





TITLE

JASC 5711, Wing Aft Auxiliary Spar Landing Gear Attachment Fittings (P/N 45501-00, 45501-01)

DESCRIPTION

Inspect for defects in the wing aft auxiliary spar landing gear attachment fittings (WS 67.5, 85).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove the down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.
2. Remove paint from the aileron hinge assembly and attachment fittings using an approved chemical paint stripper.

INSPECTION METHOD

Fluorescent Liquid Penetrant

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent liquid penetrant materials may be used providing they meet the minimum of a Type I, Level III sensitivity capable of achieving the requirements listed in the General Section, FLP of the SID.

Manufacturer	Part Number
Met-L-Chek	FP-93A (M) Penetrant
Met-L-Chek	R-504 Cleaner/Remover
Met-L-Chek	D-70 Developer
Spectronics	DSE-100X Light Meter
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Surface Preparation – The landing gear attachment fittings must be clean, dry, free of dirt, grease, oil, paint, or any contaminants which would fill, mask, or close a defect open to the surface.

   a) Remove the paint in the area to be inspected using an approved chemical stripper. The bearing areas around the inspection zone should be masked or protected.
   b) Thoroughly water rinse and dry the area prior to applying cleaner.
   c) Prepare the inspection area by scrubbing the inspection surface with a cloth that is damp with penetrant cleaner to remove any contaminants.
   d) Thoroughly dry the area before penetrant application.

2. Penetrant Application

   Penetrant shall be applied by spraying, dipping, or brushing to provide complete coverage of the aileron hinges and attachment fittings. The penetrant shall completely cover the area of interest for a minimum dwell time of 10 minutes. The penetrant shall not be allowed to dry on the part surface.

CAUTION:

Type II (visible dye) penetrant shall not be used for inspection of aircraft parts.

3. Penetrant Removal

   Remove the excess penetrant by first wiping the part surface with a dry, clean, lint free cloth. The surface of the component shall not be flushed with solvent. Examine the inspection area under a black light to ensure the removal of all surface penetrant. Over removal of the surface penetrant shall require that the components be cleaned and reprocessed. The part surface shall be dried by blotting with a clean, dry towel/cloth, or by evaporation.

4. Application of Developer

   The aileron hinges and attachment fittings shall be completely dry before the application of developer. Nonaqueous developer shall be applied by spraying and allowed to dry at ambient temperature. Apply the developer as a uniform thin coating over the entire surface to be inspected. The minimum dwell time for nonaqueous developers is 10 minutes. The dwell time starts after the developer is dry on the component when using form “d” nonaqueous developers.

NOTE:

The aerosol nonaqueous developer shall be frequently agitated before and during application.

5. Interpretation
   The inspection area shall consist of a darkened booth or an area where the ambient white light does not exceed 2 lumen when measured by a radiometer. Viewing areas for portable FLP inspections shall use a dark canvas, photographer's black cloth, or other methods to reduce the white light background to the lowest level possible during inspection.

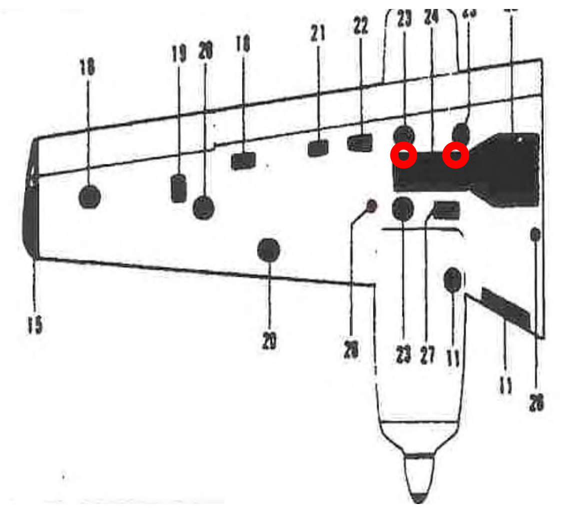
   The inspection area shall be viewed using a black light that provides a minimum of 1000 microwatts per square centimeter at the component surface. Do not position black lights closer than 6 inches from the inspection surface.

   All areas of fluorescence shall be interpreted. Components with excessive background or irrelevant indications which interfere with the detection of relevant indications shall be cleaned and reprocessed. Indications may be evaluated by wiping no more than twice. Ten power magnifiers may be used to interpret or evaluate indications.

6. Cracks detected during this inspection shall be documented and reported.

7. Post Cleaning
   Remove all developer and penetrant material from the part surface using the appropriate penetrant cleaner. Verification of adequate post cleaning shall be conducted using a black light.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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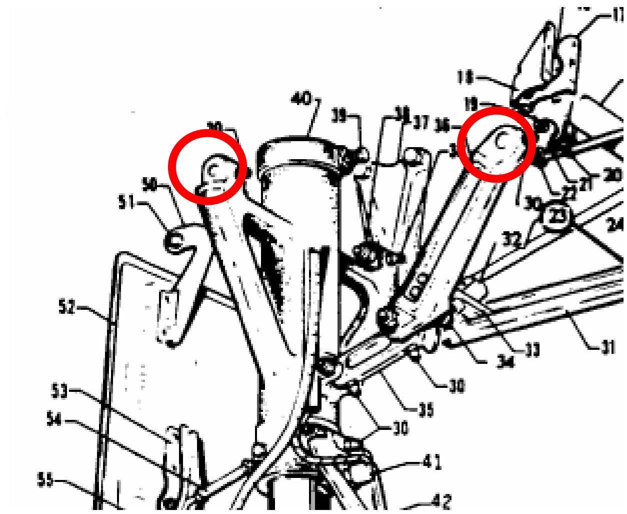
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TITLE

JASC 5711, Wing Forward Auxiliary Spar Landing Gear Attachment Points (P/N 40500-00, 40500-01)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing forward auxiliary spar landing gear attachment points (WS 67.5, 85).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
    a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
    b. Remove down-lock hook and spring by removing the pivot bolt.
    c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
    d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
    e. Remove the bolt that connects the upper and lower side brace links.
    f. Disconnect the aft link from its attachment plate.
    g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
    h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure; equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

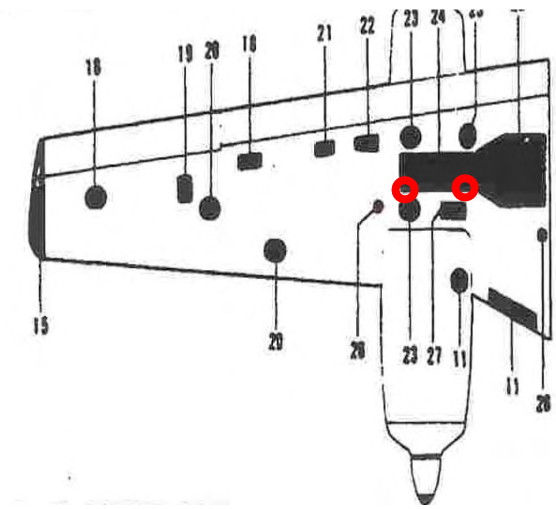
1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.
2. Null the probe on the reference standard away from the calibration notches.
3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

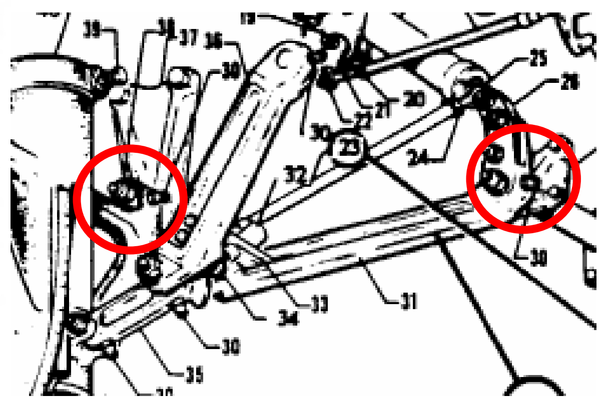
5. Inspect the area around and between the fasteners on the wing forward auxiliary spar landing gear attachment points. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





TITLE

JASC 5711, Wing Forward Auxiliary Spar Landing Gear Attachment Points (P/N 40500-00, 40500-01)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing forward auxiliary spar landing gear attachment points (WS 67.5, 85).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove the down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole, removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

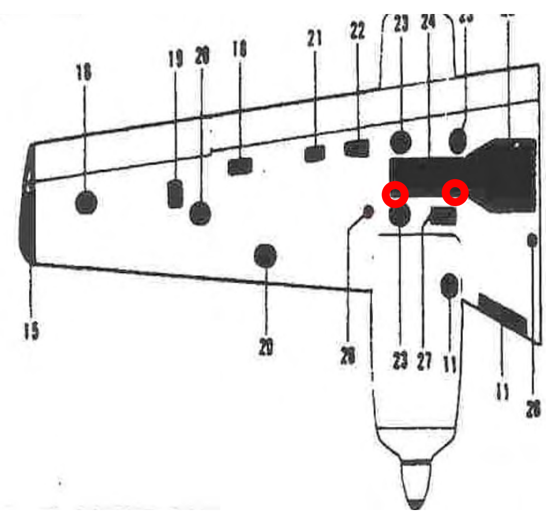
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

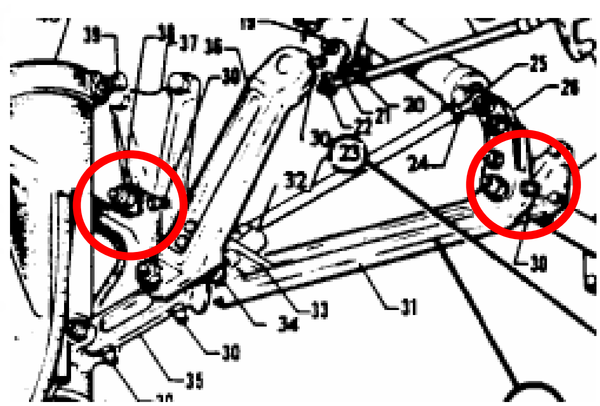
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





TITLE

JASC 5711, Wing Forward Auxiliary Spar Landing Gear Attachment Fittings (P/N 40500-00, 40500-01)

DESCRIPTION

Inspect for defects in the wing forward auxiliary spar landing gear attachment fittings (WS 67.5, 85).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.
2. Remove paint from the aileron hinge assembly and attachment fittings using an approved chemical paint stripper

INSPECTION METHOD

Fluorescent Liquid Penetrant

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent liquid penetrant materials may be used providing they meet the minimum of a Type I, Level III sensitivity capable of achieving the requirements listed in the General Section, FLP of the SID.

Manufacturer	Part Number
Met-L-Chek	FP-93A (M) Penetrant
Met-L-Chek	R-504 Cleaner/Remover
Met-L-Chek	D-70 Developer
Spectronics	DSE-100X Light Meter
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Surface Preparation – The landing gear attachment fittings must be clean, dry, free of dirt, grease, oil, paint, or any contaminants which would fill, mask, or close a defect open to the surface.

   a) Remove the paint in the area to be inspected using an approved chemical stripper. The bearing areas around the inspection zone should be masked or protected.
   b) Thoroughly water rinse and dry the area prior to applying cleaner.
   c) Prepare the inspection area by scrubbing the inspection surface with a cloth that is damp with penetrant cleaner to remove any contaminants.

d) Thoroughly dry the area before penetrant application.

2. Penetrant Application
   Penetrant shall be applied by spraying, dipping, or brushing to provide complete coverage of the aileron hinges and attachment fittings. The penetrant shall completely cover the area of interest for a minimum dwell time of 10 minutes. The penetrant shall not be allowed to dry on the part surface.

CAUTION:

Type II (visible dye) penetrant shall not be used for inspection of aircraft parts.

3. Penetrant Removal
   Remove the excess penetrant by first wiping the part surface with a dry, clean, lint free cloth. The surface of the component shall not be flushed with solvent. Examine the inspection area under a black light to ensure the removal of all surface penetrant. Over removal of the surface penetrant shall require that the components be cleaned and reprocessed. The part surface shall be dried by blotting with a clean, dry towel/cloth or by evaporation.

4. Application of Developer
   The aileron hinges and attachment fittings shall be completely dry before the application of developer. Nonaqueous developer shall be applied by spraying and allowed to dry at ambient temperature. Apply the developer as a uniform thin coating over the entire surface to be inspected. The minimum dwell time for nonaqueous developers is 10 minutes. The dwell time starts after the developer is dry on the component when using form “d” nonaqueous developers.

NOTE:

The aerosol nonaqueous developer shall be frequently agitated before and during application.

5. Interpretation
   The inspection area shall consist of a darkened booth or an area where the ambient white light does not exceed 2 lumen when measured by a radiometer. Viewing areas for portable FLP inspections shall use a dark canvas, photographer's black cloth, or other methods to reduce the white light background to the lowest level possible during inspection.

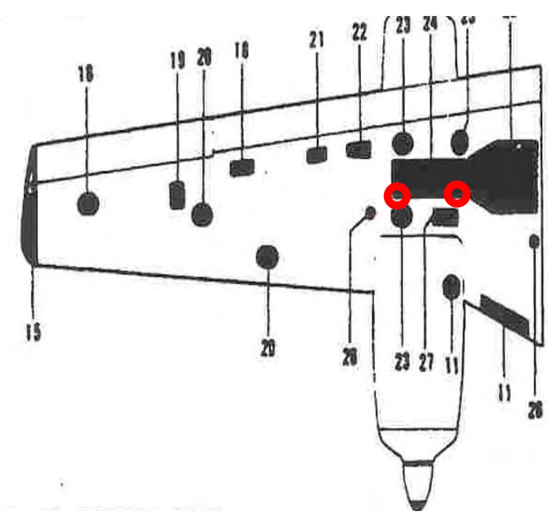
   The inspection area shall be viewed using a black light that provides a minimum of 1000 microwatts per square centimeter at the component surface. Do not position black lights closer than 6 inches from the inspection surface.

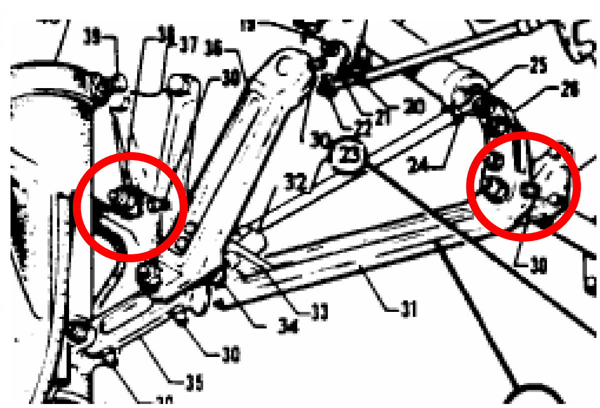
   All areas of fluorescence shall be interpreted. Components with excessive background or irrelevant indications which interfere with the detection of relevant indications shall be cleaned and reprocessed. Indications may be evaluated by wiping no more than twice. Ten power magnifiers may be used to interpret or evaluate indications.

6. Cracks detected during this inspection shall be documented and reported.

7. Post Cleaning
   Remove all developer and penetrant material from the part surface using the appropriate penetrant cleaner. Verification of adequate post cleaning shall be conducted using a black light.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





TITLE

<u>JASC 5711, Wing Main Spar Lower Cap</u> (P/N 44677-00, 44677-01)

DESCRIPTION

Inspect for defects in the wing main spar lower cap (WS 65-WS 68, WS 87-WS 91, BL 0-BL 12).

REQUIRED DISASSEMBLY

No required disassembly.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

<u>Reference Standard Notch Depths Surface EC:</u>

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

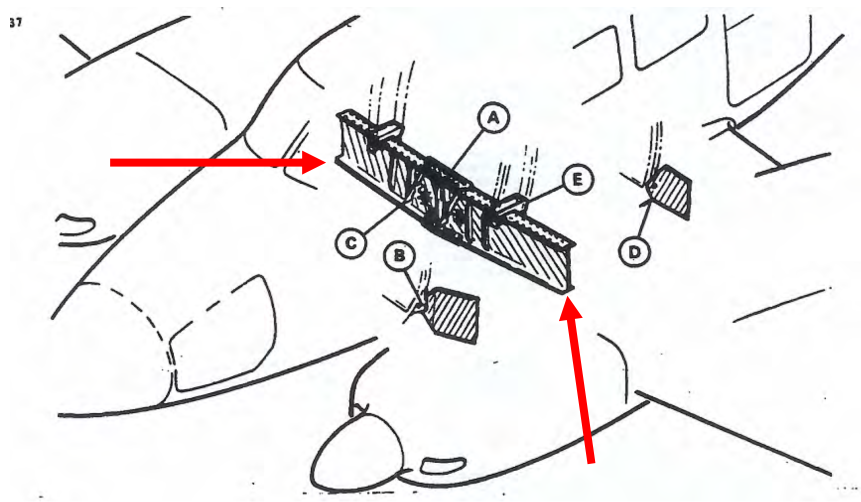
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

5. Inspect the area around and between the fasteners on the wing main spar lower spar cap. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 5711, Wing Rear Spar Attachment Points (P/N 45165-02, 45165-03)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing rear spar attachment points (WS 30).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

   d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at the butt end of wing and tape cables out of the way at the fuselage.
   e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
   f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.
10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove support blocks and clamps.
13. Disconnect the electrical wire connectors.
14. Draw the engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay it forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

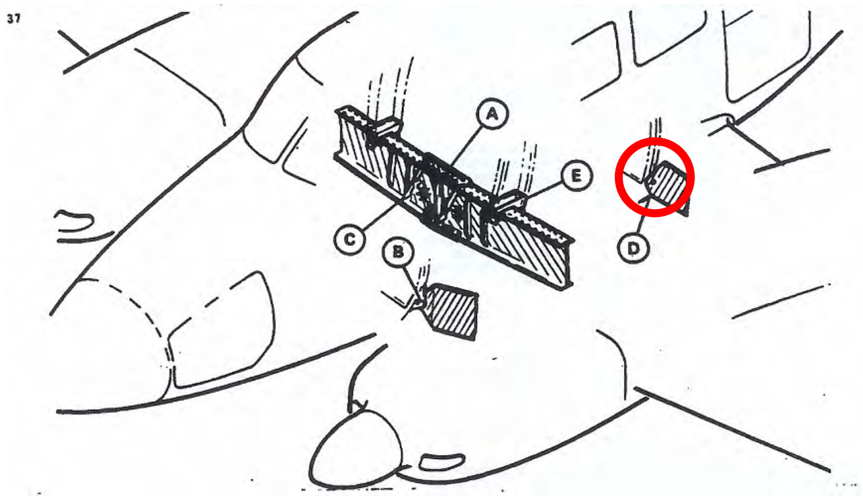
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

5. Inspect the area around and between the fasteners on the wing rear spar attachment point. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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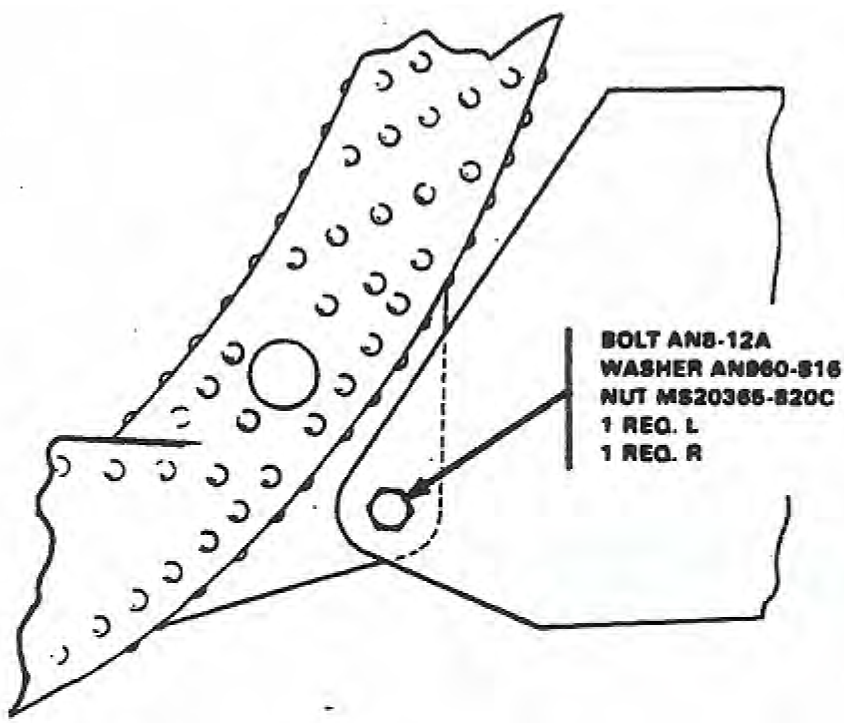
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TITLE

<u>JASC 5711, Wing Rear Spar Attachment Points</u> (P/N 45165-02, 45165-03)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing rear spar attachment points (WS 30).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

   d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at the butt end of wing and tape cables out of the way at the fuselage.
   e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
   f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.

10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove support blocks and clamps.
13. Disconnect the electrical wire connectors.
14. Draw the engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay it forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

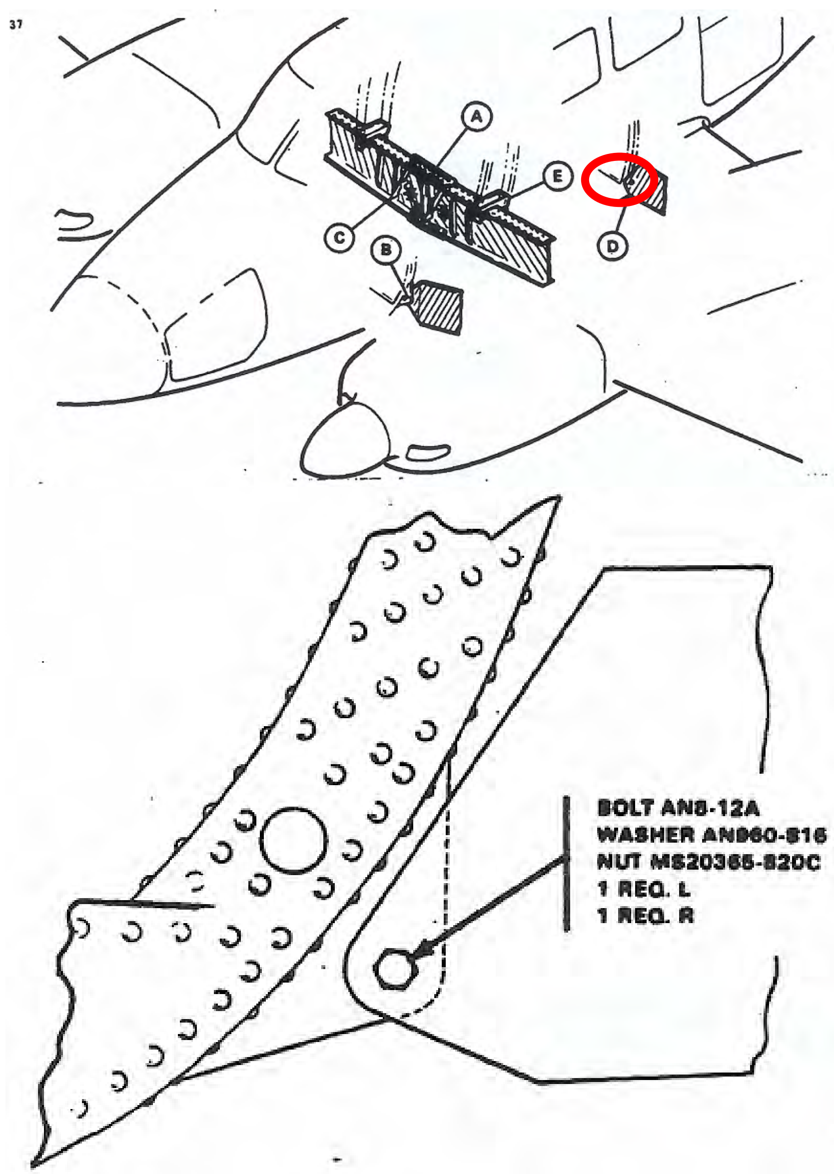
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 5740, Wing Main Spar Splice Attachment Points (P/N 44654-00, 44872-00, 44875-00)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing main spar splice attachment points (BL 0-BL 6).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at the butt end of the wing and tape cables out of the way at the fuselage.
e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.

10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove the support blocks and clamps.
13. Disconnect the electrical wire connectors.
14. Draw the engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay it forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

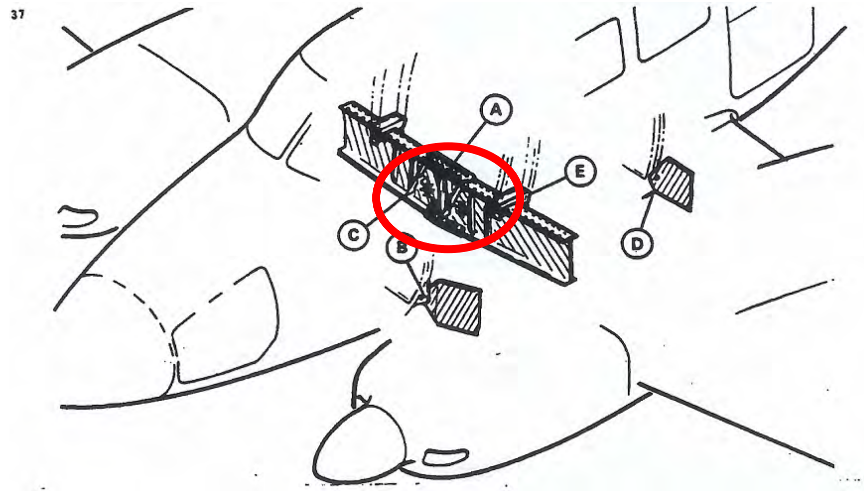
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

5. Inspect the area around and between the fasteners on the wing main spar splice attachment points. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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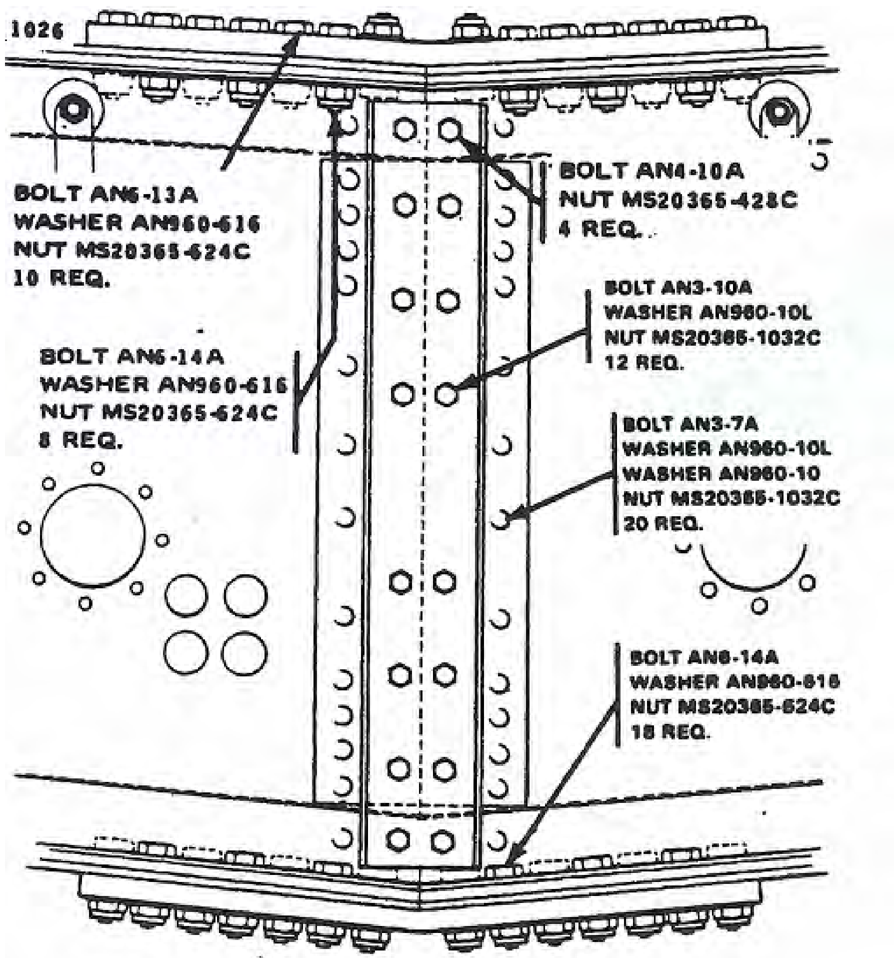
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4 REQ.
BOLT AN3-10A
WASHER AN960-10L
NUT MS20365-1032C
12 REQ.
BOLT AN6-14A
WASHER AN960-616
NUT MS20365-624C
8 REQ.
BOLT AN3-7A
WASHER AN960-10L
WASHER AN960-10
NUT MS20365-1032C
20 REQ.
BOLT AN6-14A
WASHER AN960-616
NUT MS20365-624C
18 REQ.

TITLE

JASC 5740, Wing Main Spar Splice Attachment Points (P/N 44654-00, 44872-00, 44875-00)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing main spar splice attachment points (BL 0-BL 6).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at the butt end of wing and tape cables out of the way at the fuselage.
e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.

10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove the support blocks and clamps.
13. Disconnect the electrical wire connectors.
14. Draw the engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay it forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

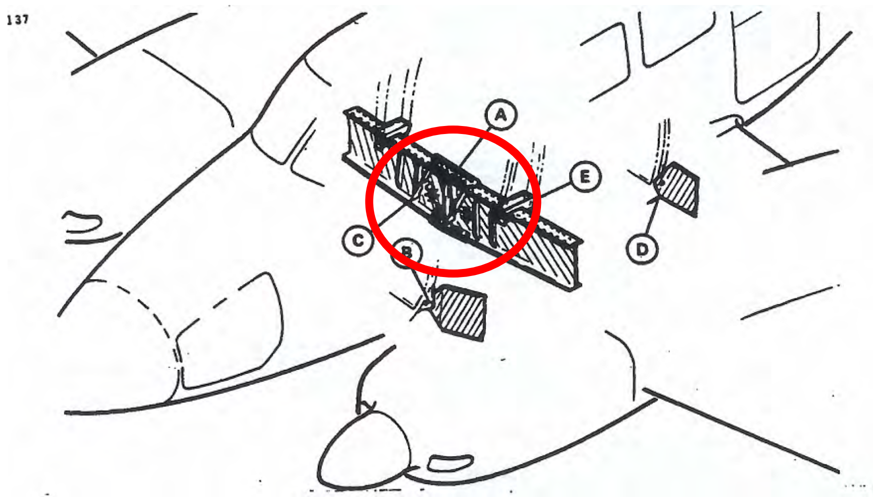
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

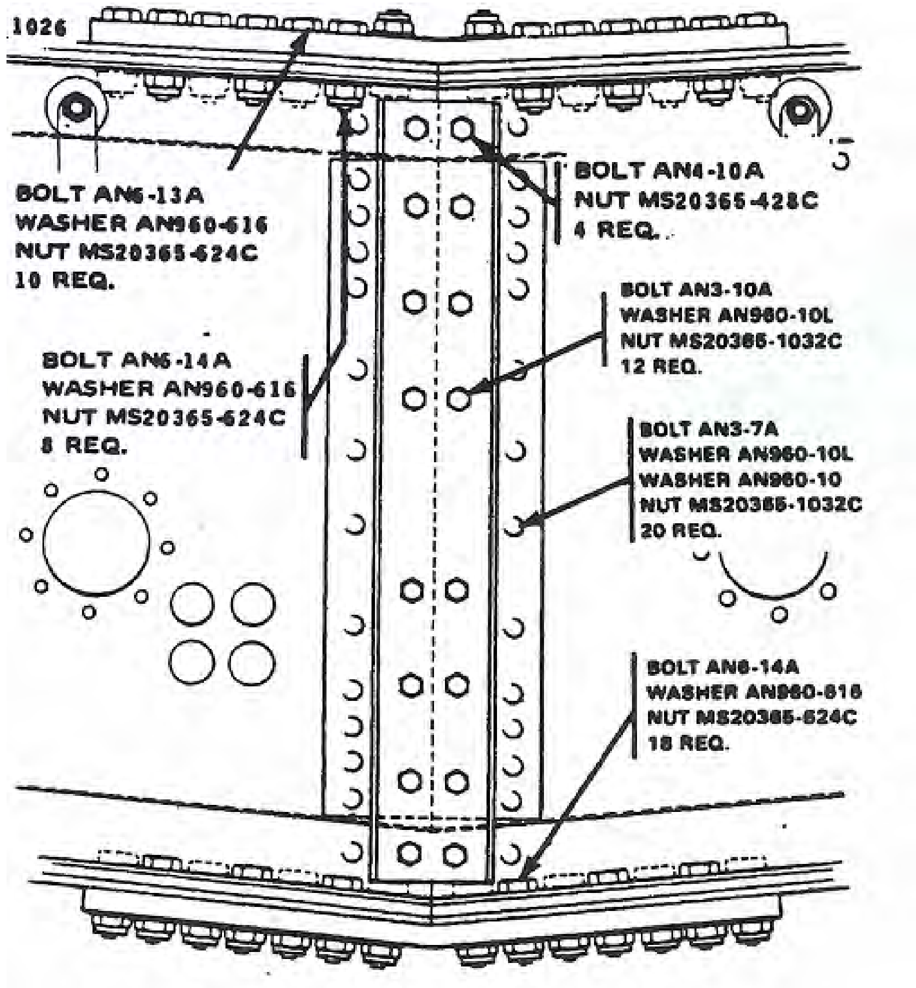
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and length and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.

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12 REQ.
BOLT AN6-14A
WASHER AN960-616
NUT MS20365-624C
8 REQ.
BOLT AN3-7A
WASHER AN960-10L
WASHER AN960-10
NUT MS20365-1032C
20 REQ.
BOLT AN6-14A
WASHER AN960-616
NUT MS20365-624C
18 REQ.

TITLE

JASC 5741, Wing Main Spar to Fuselage Attachment Points (P/N 45155-00, 44964-00, 40319-00)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing main spar to fuselage attachment points (WS 30).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at butt end of wing and tape cables out of the way at the fuselage.
e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.

10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove the support blocks and clamps.
13. Disconnect the electrical wire connectors.
14. Draw the engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay it forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

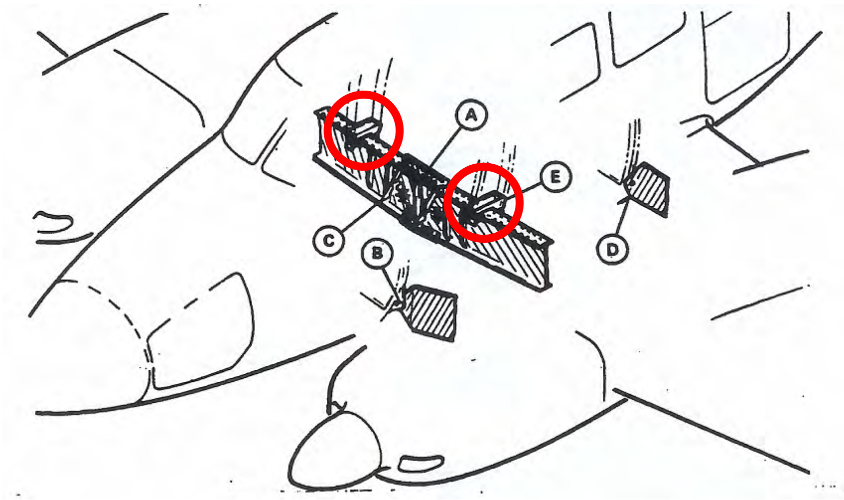
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

5. Inspect the area around and between the fasteners. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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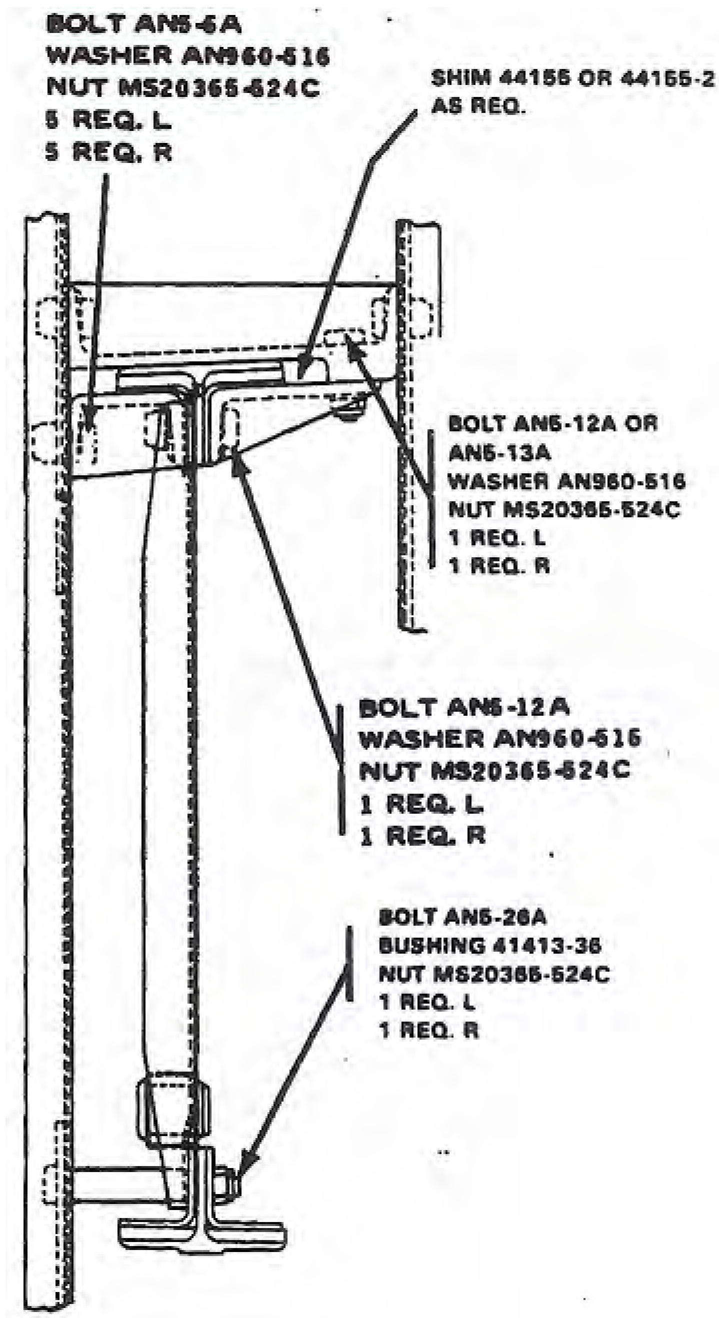
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1 REQ. R
BOLT AN5-12A
WASHER AN960-516
NUT MS20365-524C
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1 REQ. R
BOLT AN5-26A
BUSHING 41413-36
NUT MS20365-524C
1 REQ. L
1 REQ. R

TITLE

JASC 5741, Wing Main Spar to Fuselage Attachment Points (P/N 45155-00, 44964-00, 40319-00)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing main spar to fuselage attachment points (WS 30).

REQUIRED DISASSEMBLY

1. Close the fuel valve and drain the fuel from the wing to be removed.
2. Remove the engine from the wing to be removed.
3. Remove the fairing and access panel from around the leading edge of the wing, located between the fuselage and engine nacelle.
4. At the fillet fairing on top of the wing, between the fuselage and wing, remove the rivets that attach the fairing to the wing.
5. Remove the access plates from the fairing located between the underside of the wing butt and fuselage and the access plate to the spar splice located on the underside of the fuselage.
6. Within the fuselage, remove the spar cover.
7. Remove the forward and aft floor panels adjacent to the main spar and if removing the left wing, remove the left forward floor panel between the fuselage side trim panel and control pedestal.
8. The following items pertain to the removal of the left wing only:
   a. Disconnect the primary control cables at the turnbuckles located at FS 76 and FS 86.5, between the left forward side trim panel and control pedestal. Draw the cables back through the spar. Remove the elevator cable guard pin at FS 110 to allow the cable ends to pass through.
   b. Remove the left aileron cable guard pin at FS 164.
   c. The balance cable to the left wing may be disconnected at the aileron bellcrank, drawn through the wing and taped out of the way at the side of the fuselage. The cable guard pin at the left wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
9. The following items pertain to the removal of the right wing only:
   a. Disconnect the aileron control cable at the aileron bellcrank and draw it out through the wing. The cable guard in the wing near the bellcrank and wing butt will have to be removed to allow the cable end to pass through.
   b. Disconnect the aileron balance cable at WS 171 and draw the cable from the fuselage. Remove the cable pulley to allow the cable to be removed.
   c. Remove the access panel to the aft section of the fuselage. Block the elevator and rudder trim cables ahead of the main spar and in the aft section of the fuselage to prevent the cables from unwrapping at the trim drums. Disconnect the elevator and rudder trim cables between FS 274 and FS 318 and draw the cables forward through the main spar. To allow the cables to be drawn through the fuselage, remove the cable guard at FS 244 and rub blocks at FS 174 and FS 215.

d. Block the aileron trim cable at the side fuselage and within the wing to prevent the trim drum from unwrapping. Disconnect the trim cable turnbuckles at WS 90 and draw the cables inboard through the wing. Remove the cable guard at the butt end of wing and tape cables out of the way at the fuselage.
e. Disconnect the hydraulic lines, which are routed through the spar at FS 90 and FS 137 and slide the lines forward through the wing spar.
f. Disconnect the heater air duct, heat control cable, and antenna cables that lead through the spar.

10. At FS 174, disconnect the flap control cable from the actuating motor and bulkhead and draw the cable out through the fuselage.
11. Through the wing fairing access openings at the underside of the wing, disconnect the fuel line that is routed through the main spar and pull it back through the spar. Disconnect the hydraulic and fuel lines at the exposed fittings and control cables from fuel valves.
12. Through the access openings at the wing leading edge and butt, disconnect the engine instruments, vacuum, fuel, and hydraulic lines. Remove support blocks and clamps.
13. Disconnect the electrical wire connectors.
14. Draw the engine control cables back through the firewall, engine nacelle, and wing.
15. Arrange a suitable fuselage cradle and supports for both wings.
16. Remove the brace assembly that the fuel selector attaches to and lay forward. Unbolt and remove the angle support(s) that extend through the spar.
17. To the side of the fuselage, at the top of the main spar, remove the forward and aft lower support fittings. The upper fitting may remain in place.
18. Also, to the side of the fuselage, at the bottom of the main spar, remove the support bolt assembly and spacer bushing.
19. Unbolt and remove the vertical spar splice channels.
20. Unbolt and remove the upper and lower horizontal spar cap splice plate.
21. Remove the bolt assembly that attaches the front spar and fuselage fitting.
22. Remove the bolt assembly that attaches the rear spar and fuselage fitting.
23. Pull the wing directly and slowly away from the fuselage, allowing lines, cables, etc., to follow.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure; equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

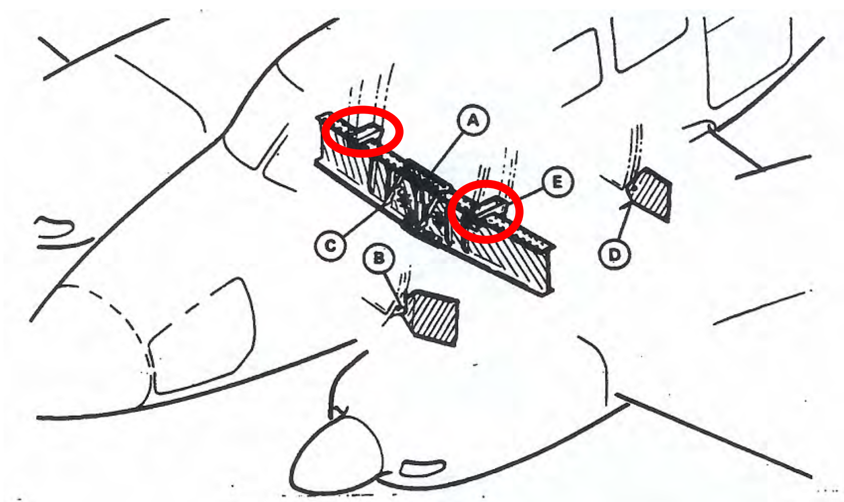
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

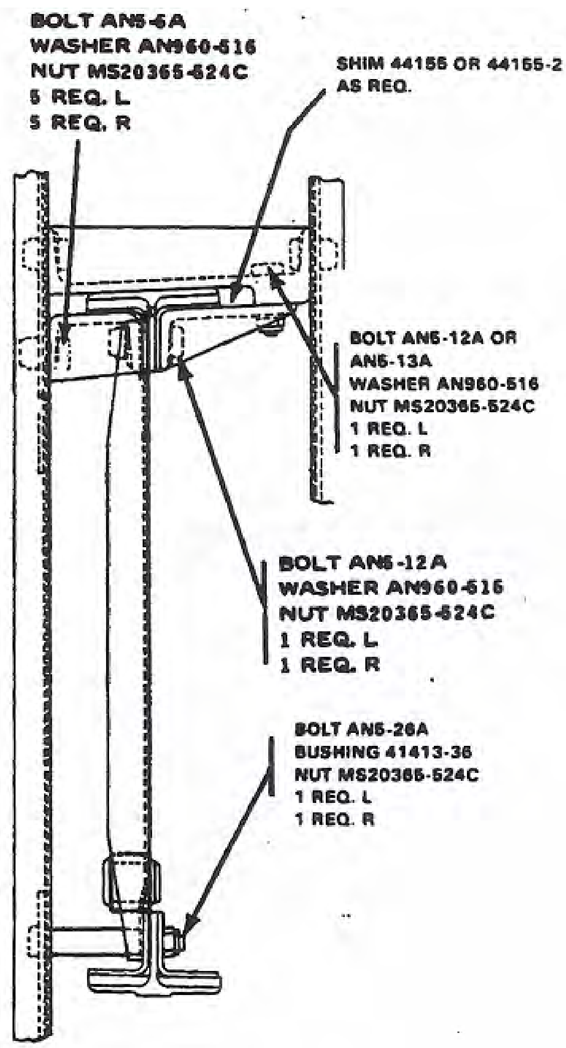
8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and length and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




BOLT AN5-6A
WASHER AN960-516
NUT MS20365-524C
5 REQ. L
5 REQ. R
SHIM 44155 OR 44155-2
AS REQ.
BOLT AN5-12A OR
AN5-13A
WASHER AN960-516
NUT MS20365-524C
1 REQ. L
1 REQ. R
BOLT AN5-12A
WASHER AN960-516
NUT MS20365-524C
1 REQ. L
1 REQ. R
BOLT AN5-26A
BUSHING 41413-36
NUT MS20365-524C
1 REQ. L
1 REQ. R

TITLE

JASC 5743, Wing Main Spar Landing Gear Attachment Points (P/N 44677-00, 44677-01)—Surface Scan Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing main spar landing gear attachment points (WS 67.5).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole, removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Scan Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	ML/100kHz-500kHz/A/90.5/6 EC Probe
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Surface EC:

0.005″
0.010″
0.020″
0.050″

INSPECTION INSTRUCTIONS

1. Connect the surface probe to the EC instrument and adjust the frequency to 200 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. Adjust liftoff on the impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from the right to left as the probe is lifted from the part surface.

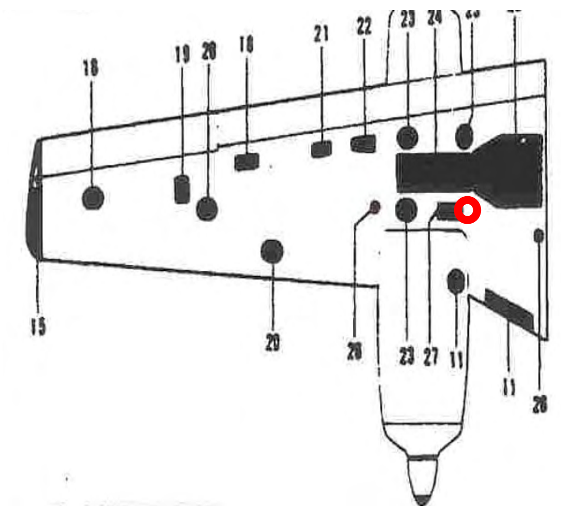
4. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.020-inch depth calibration notch that is a minimum of three major screen divisions.

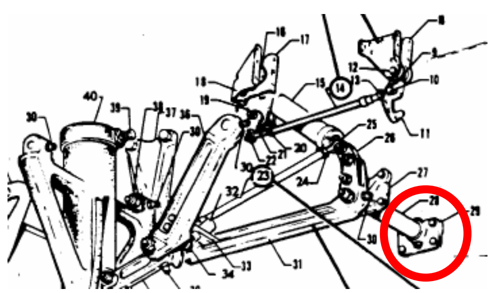
5. Inspect the area around and between the fasteners on the wing main spar landing gear attachment points. Observe the phase and amplitude changes on the EC instrument.

6. If an indication is noted, carefully repeat the inspection in the opposite direction to verify the indication.

7. Cracks detected during this inspection shall be documented and reported.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.





TITLE

JASC 5743, Wing Main Spar Landing Gear Attachment Points (P/N 44677-00, 44677-01)—Bolthole Eddy-Current Inspection

DESCRIPTION

Inspect for defects in the wing main spar landing gear attachment points (WS 67.5).

REQUIRED DISASSEMBLY

1. Place the airplane on jacks in accordance with the Piper Navajo Chieftain Service Manual.
2. Remove the two access plates forward and two access plates aft of the outboard wheel door.
3. With the hand pump, retract the main gear slightly to relieve the gear from its down-locked position and to lower the inboard gear door out of the way.
4. Disconnect the brake line.
5. To remove the side brace link assembly, the following procedure may be used:
   a. Disconnect the actuating cylinder and down-lock rod or cable from the upper side brace link arm by removing the clevis bolt. Disconnect the other end of the down-lock rod or cable at the down-lock hook.
   b. Remove down-lock hook and spring by removing the pivot bolt.
   c. Remove the down-lock switch bracket with switch by removing the four screws that attach the bracket between the forward and aft side brace links. Remove the clamps that secure the electrical wiring to the side brace link.
   d. Disconnect the lower side brace link from the gear oleo housing and let the link assembly swing down.
   e. Remove the bolt that connects the upper and lower side brace links.
   f. Disconnect the aft link from its attachment plate.
   g. To remove the forward link, remove the nut with washer that is holding the link on its pivot shaft. Slide the link from the pivot shaft.
   h. The pivot shaft may be removed by reaching through the pivot shaft bracket access hole and removing the bolt securing the shaft to the shaft fitting. Slide the tube through the attachment bracket. The shaft fitting is attached with cap bolts, washers, and anchor nuts.
6. Disconnect the outboard gear door retraction rods at the gear housing. With the lower side brace link disconnected from the housing, the gear may be removed by removing the attachment bolt assemblies at the attachment plates on each side of the gear housing.
7. The uplock hook and spring may be removed by disconnecting the uplock rod or cable from the hook and then the hook pivot bolt.
8. The uplock rod or cable may be removed by disconnecting the rod or cable end at the lock crank.

9. The landing gear and upper drag link attachment plates may be removed by reaching through the access holes to the nuts that secure the plates. While holding the nuts, wrench the attachment bolts.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolthole Eddy Current

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent EC test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

Manufacturer	Part Number
Nortec	Model 2000S EC Instrument
Nortec	SPO-5965 Bolthole EC Probe
Staveley	Mini-Mite Bolthole Scanner
PH Tool	7947479-10 7075-T6 EC Standard

Reference Standard Notch Depths Bolthole EC:
0.030″ by 0.030″ notch, first layer bottom corner notch at the interface with the second layer

INSPECTION INSTRUCTIONS

1. Connect the bolthole probe to the EC instrument and adjust the frequency to 500 kHz.

2. Null the probe on the reference standard away from the calibration notches.

3. With the scanner turned off, insert the probe into the appropriate size hole in the reference standard. The probe should fit snug in the hole without binding. Verify the coil is 90 degrees away from the corner EDM notch and away from hole edges and press NULL.

4. Remove the probe from the hole.

5. Hold the scanner so that the probe is parallel to and rotates against the reference standard surface. Place the coil in contact with a flat area of the reference standard at least 0.250 inch away from any edge or notch. With the Sweep **OFF,** turn the scanner on and rotate the probe making coil contact with the reference standard to generate a liftoff signal.

NOTE:

If the liftoff signal is not linear, reduce the Probe Drive to LOW and repeat step 4. Gain levels may be increased to obtain an acceptable signal.

6. Adjust the phase rotation by pressing the ANGLE key and rotating the SmartKnob to achieve a substantially horizontal liftoff signal.

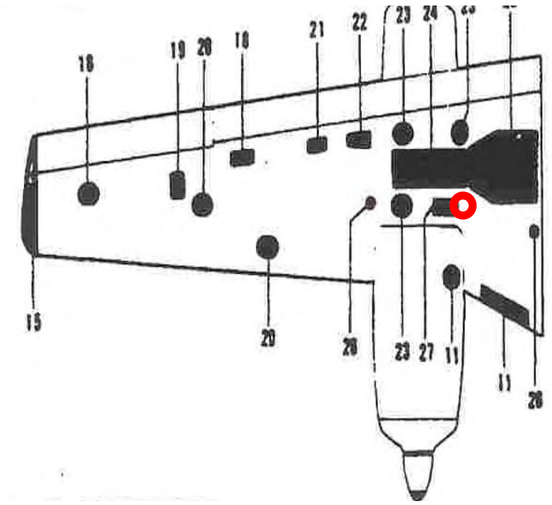
7. With the Sweep **OFF**, turn the scanner on and insert the probe into the appropriate hole.

8. Adjust the instrument gain controls to obtain a signal amplitude response from the 0.030-inch depth calibration notch that is a minimum of 80 percent full screen height. Observe the phase and amplitude changes on the EC instrument.

9. If an indication is noted, carefully repeat the calibration and perform the inspection again to verify the indication.

10. Cracks detected during this inspection shall be documented with hole diameter, location of hole, and direction of the indication.

11. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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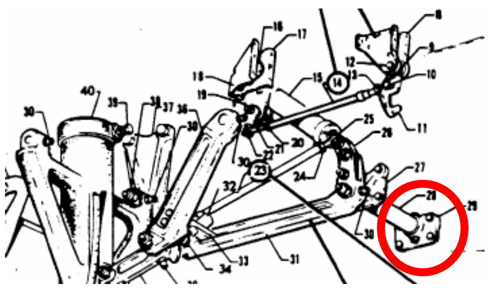


TITLE

JASC 5753, Flap Attachment Fittings (P/N 45379-07, 45809-10, 45381-14, 45381-15)

DESCRIPTION

Inspect for defects in the flap attachment fittings (WS 41, 88.5, 147.5).

REQUIRED DISASSEMBLY

1. Lower flap to within a few degrees of full extension.
2. At the left flap, disconnect the position sender rod by removing the cotter pin from the forward end of the rod.
3. Disconnect the flap control tube at the flap. Do not rotate the control tube unless it is intended to adjust the flap.
4. Remove the upper roller assemblies from the flap brackets.
5. Remove the lower roller assemblies and remove flap.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.
2. Remove paint from the flap attachment fittings using an approved chemical paint stripper.

INSPECTION METHOD

Fluorescent Liquid Penetrant

EQUIPMENT

The following equipment was used to develop this procedure. Equivalent liquid penetrant materials may be used providing they meet the minimum of a Type I, Level III sensitivity capable of achieving the requirements listed in the FLP General Section.

Manufacturer	Part Number
Met-L-Chek	FP-93A (M) Penetrant
Met-L-Chek	R-504 Cleaner/Remover
Met-L-Chek	D-70 Developer
Spectronics	DSE-100X Light Meter
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Surface Preparation – The flap attachment fittings must be clean, dry, free of dirt, grease, oil, paint, or any contaminants that would fill, mask, or close a defect open to the surface.

   a) Remove the paint in the area to be inspected using an approved chemical stripper. The bearing areas around the inspection zone should be masked or protected.
   b) Thoroughly water rinse and dry the area prior to applying cleaner.
   c) Prepare the inspection area by scrubbing the inspection surface with a cloth that is damp with penetrant cleaner to remove any contaminants.
   d) Thoroughly dry the area before penetrant application.

2. Penetrant Application
   Penetrant shall be applied by spraying, dipping, or brushing to provide complete coverage of the flap attachment fittings. The penetrant shall completely cover the area of interest for a minimum dwell time of 10 minutes. The penetrant shall not be allowed to dry on the part surface.

CAUTION:

Type II (visible dye) penetrant shall not be used for inspection of aircraft parts.

3. Penetrant Removal
   Remove the excess penetrant by first wiping the part surface with a dry, clean, lint free cloth. The surface of the component shall not be flushed with solvent. Examine the inspection area under a black light to ensure the removal of all surface penetrant. Over removal of the surface penetrant shall require that the components be cleaned and reprocessed. The part surface shall be dried by blotting with a clean, dry towel/cloth or by evaporation.

4. Application of Developer
   The flap attachment fittings shall be completely dry before the application of developer. Nonaqueous developer shall be applied by spraying and allowed to dry at ambient temperature. Apply the developer as a uniform thin coating over the entire surface to be inspected. The minimum dwell time for nonaqueous developers is 10 minutes, with the maximum time of 1 hour. The dwell time starts after the developer is dry on the component when using form “d” nonaqueous developers.

NOTE:

The aerosol nonaqueous developer shall be frequently agitated before and during application.

5. Interpretation
   The inspection area shall consist of a darkened booth or an area where the ambient white light does not exceed 2 lumen when measured by a radiometer. Viewing areas for portable FLP inspections shall use a dark canvas, photographer's black cloth, or other methods to reduce the white light background to the lowest level possible during inspection.

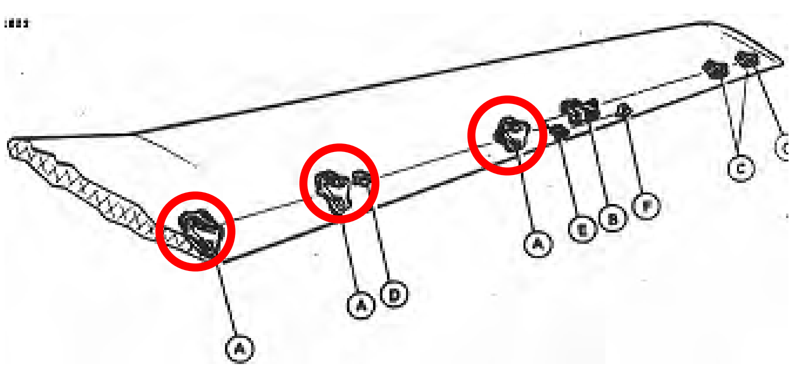
   The inspection area shall be viewed using a black light that provides a minimum of 1000 microwatts per square centimeter at the component surface. Do not position black lights closer than 6 inches from the inspection surface.

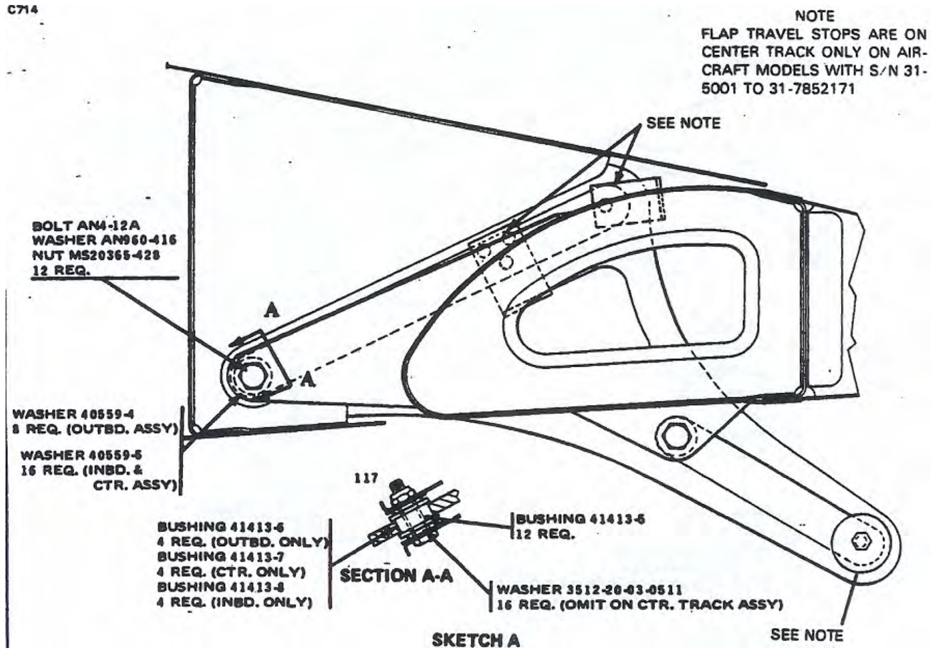
   All areas of fluorescence shall be interpreted. Components with excessive background or irrelevant indications which interfere with the detection of relevant indications shall be cleaned and reprocessed. Indications may be evaluated by wiping no more than twice. Ten power magnifiers may be used to interpret or evaluate indications.

6. Cracks detected during this inspection shall be documented and reported

7. Post Cleaning
   Remove all developer and penetrant material from the part surface using the appropriate penetrant cleaner. Verification of adequate post cleaning shall be conducted using a black light.

8. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.




C714
NOTE
FLAP TRAVEL STOPS ARE ON CENTER TRACK ONLY ON AIRCRAFT MODELS WITH S/N 31-5001 TO 31-7852171
SEE NOTE
BOLT AN4-12A
WASHER AN960-416
NUT MS20365-428
12 REQ.
A
A
WASHER 40559-4
8 REQ. (OUTBD. ASSY)
WASHER 40559-5
16 REQ. (INBD. & CTR. ASSY)
117
BUSHING 41413-5
4 REQ. (OUTBD. ONLY)
BUSHING 41413-7
4 REQ. (CTR. ONLY)
BUSHING 41413-8
4 REQ. (INBD. ONLY)
SECTION A-A
BUSHING 41413-5
12 REQ.
WASHER 3512-20-03-0511
16 REQ. (OMIT ON CTR. TRACK ASSY)
SEE NOTE

SKETCH A

TITLE

JASC 7120, Engine Structure (P/N 40007-07)—Visual Inspection

DESCRIPTION

Inspect for defects in the right and left wing engine structures (BL 11.81 – Engine Coordinates).

REQUIRED DISASSEMBLY

1. Turn off all cockpit switches and then disconnect the battery ground wire at the battery.
2. Move the fuel valve control lever located on the outboard side of the fuel selector panel, labeled “Emergency Fuel Shut-off,” to the OFF position.
3. Remove the engine cowling.
4. Remove the access panels on the top, sides, and inboard bottom of the nacelle, just aft of the firewall.
5. Drain the engine oil and reinstall drain plug.
6. Remove the propeller.
7. Disconnect the starter cable at the starter, remove the cable clamps at the left side of the engine and engine mount, and draw the cable aft through the engine baffle to the firewall.
8. Disconnect the alternator primary cable that leads from the firewall at the filter box located on the right lower side of the engine mount. Disconnect the field wire.
9. Disconnect the electrical leads to the oil temperature sender at the accessory housing, the cylinder head temperature probe at the number 6 cylinder, and the exhaust temperature at the aft side of the exhaust manifold.
10. Disconnect the magneto ground leads and the retard spark lead of the left magneto at the magneto.
11. Disconnect the propeller deicer electrical wires.
12. Disconnect the pressure pump hose at the upper left side of the firewall.
13. Disconnect the tachometer drive cable at the engine accessory housing.
14. Disconnect the throttle and mixture control cables at the injector, the governor control cable at the governor, and the alternate air door control cable at the left side of the air filter plenum. Disconnect the cables from their attachment clamps.
15. Disconnect the hydraulic pressure line at the hydraulic oil filter on the firewall.
16. Disconnect the hydraulic suction, fuel supply, fuel flow pressure, fuel pressure, air deck pressure, oil pressure, manifold pressure, and deicer lines at the firewall.
17. Attach a 1/2-ton (minimum) hoist to the hoisting hooks and relieve the tension on the engine mount.
18. Remove the nuts and washers from the bolts that attach the engine mount to the firewall.
19. Remove the engine mount mounting bolts and swing the engine a few inches from the firewall. Check the engine for any attachments remaining to obstruct its removal.
20. Swing the engine clear and place on a suitable support.

PREPARATION

Clean the inspection area with solvent to remove dirt, grease, oil, and other substances that may interfere with the inspection.

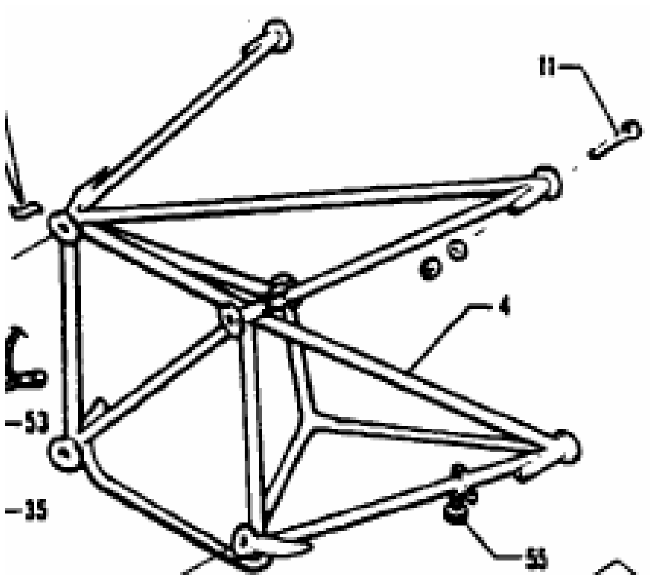
INSPECTION METHOD

Visual

INSPECTION INSTRUCTIONS

1. Visually inspect engine structures for condition and defects.

2. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.



TITLE

JASC 7120, Engine Structure (P/N 40007-07)—Magnetic Particle Inspection

DESCRIPTION

Inspect for defects in the welds on the right and left wing engine structures (BL 11.81 – Engine Coordinates).

REQUIRED DISASSEMBLY

1. Turn off all cockpit switches and then disconnect the battery ground wire at the battery.
2. Move the fuel valve control lever located on the outboard side of the fuel selector panel, labeled “Emergency Fuel Shut-off,” to the OFF position.
3. Remove the engine cowling.
4. Remove the access panels on the top, sides, and inboard bottom of the nacelle, just aft of the firewall.
5. Drain the engine oil and reinstall drain plug.
6. Remove the propeller.
7. Disconnect the starter cable at the starter, remove the cable clamps at the left side of the engine and engine mount, and draw the cable aft through the engine baffle to the firewall.
8. Disconnect the alternator primary cable that leads from the firewall at the filter box located on the right lower side of the engine mount. Disconnect the field wire.
9. Disconnect the electrical leads to the oil temperature sender at the accessory housing, the cylinder head temperature probe at the number 6 cylinder, and the exhaust temperature at the aft side of the exhaust manifold.
10. Disconnect the magneto ground leads and the retard spark lead of the left magneto at the magneto.
11. Disconnect the propeller deicer electrical wires.
12. Disconnect the pressure pump hose at the upper left side of the firewall.
13. Disconnect the tachometer drive cable at the engine accessory housing.
14. Disconnect the throttle and mixture control cables at the injector, the governor control cable at the governor, and the alternate air door control cable at the left side of the air filter plenum. Disconnect the cables from their attachment clamps.
15. Disconnect the hydraulic pressure line at the hydraulic oil filter on the firewall.
16. Disconnect the hydraulic suction, fuel supply, fuel flow pressure, fuel pressure, air deck pressure, oil pressure, manifold pressure, and deicer lines at the firewall.
17. Attach a 1/2-ton (minimum) hoist to the hoisting hooks and relieve the tension on the engine mount.
18. Remove the nuts and washers from the bolts that attach the engine mount to the firewall.
19. Remove the engine mount mounting bolts and swing the engine a few inches from the firewall. Check the engine for any attachments remaining to obstruct its removal.
20. Swing the engine clear and place on a suitable support.

PREPARATION

Remove paint from the inspection area using an approved chemical paint stripper.

INSPECTION METHOD

Magnetic Particle

EQUIPMENT

The following types of magnetic particle yokes may be used to accomplish this inspection. Equivalent substitutes may be used for the listed equipment.

1. Direct current electromagnetic yokes with a dead-weight lifting capacity of at least 50 pounds with 4- to 6-inch spacing.

2. Alternating current electromagnetic yokes with a dead-weight capacity of at least 10 pounds with leg spacing of 2 to 4 inches.

Manufacturer	Part Number
Magnaflux	Magnaglo 14AM Magnetic Particle Bath
Magnaflux	ZB-32A Black Light

INSPECTION INSTRUCTIONS

1. Remove all dirt, oil, grease, and paint from the inspection area.

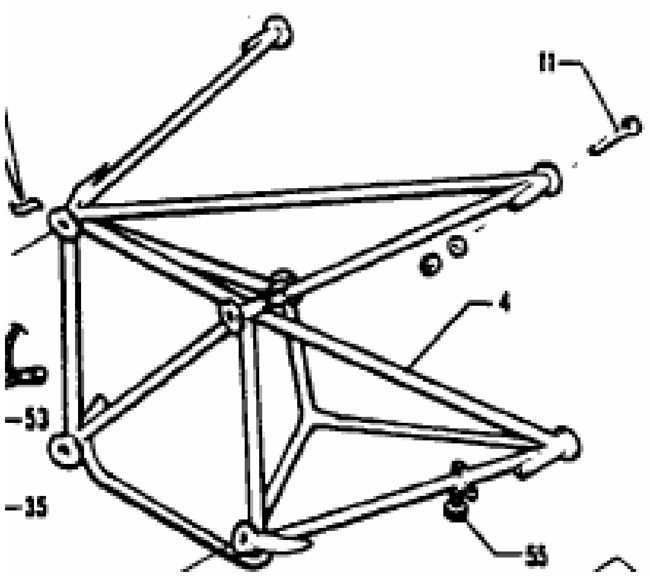
2. Position one leg on each side of the weld.

3. Apply the fluorescent magnetic particle bath to the inspection area. Stop the bath application, then immediately energize the yoke for approximately 1 second.

4. Inspect the welds for defects using a black light that has a minimum intensity of 1200 microwatts per square centimeter. The ambient light in the inspection area shall not exceed 2 lumen.

5. After completing the inspection, demagnetize the engine structure using the maximum alternating current. The residual magnetic field shall not exceed 3 gauss.

6. Reinstall and rerig all components that were removed in support of this inspection. Refer to the Piper Navajo Chieftain Service Manual for reinstallation and rerigging procedures.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