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Teardown Evaluation of a 1969 Cessna 402A Model Airplane

June 2007

Final Report

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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 two aged airplanes (both Cessna 402 models) used in commuter service. The intent of the program was to provide insight into the condition of a typical aged airplane and by determining if a correlation exists between the airplane's maintenance history and current condition from a safety of flight perspective. This document supports this research program by providing the findings of a teardown evaluation of a 1969 Cessna 402A model airplane. The results in this report will provide information for use in future investigations into the aged small airplane fleet and help to determine if additional research is required to address any problems observed. The destructive evaluation of the commuter class airplane was separated into three main tasks: (1) inspection of the airframe and airplane systems, (2) teardown examination of the airframe and airplane systems, and (3) assessment of the airplane wiring. During the inspection phase, three subtasks were performed: a survey of airplane maintenance records, visual inspection of the airframe and airplane systems, and supplemental airframe inspections. The teardown examination involved disassembling the airframe and major airplane sections, inspecting airplane systems' components, inspecting the primary airplane structure using alternative nondestructive inspection techniques, and performing microscopic examinations of critical structural areas. As part of the destructive evaluation, inspections and tests were also performed on airplane wiring 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

A	Ampere
AC	Alternating Current
AD	Airworthiness Directive
BL	Body line
CFR	Code of Federal Regulations
DWV	Dielectric withstand voltage
EDM	Electron discharge machining
FAA	Federal Aviation Administration
FS	Frame station
GA	General aviation
HiPOT	High potentiometer
IVI	Intrusive Visual Inspection
LRT	Loop Resistance Tester
MOI	Magneto-Optic Imager
NDI	Nondestructive Inspection
NIAR	National Institute for Aviation Research
P/N	Part number
SDRs	Service Difficulty Reports
SID	Supplemental Inspection Document
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 established a research program to conduct a destructive evaluation on two aged airplanes (both Cessna 402 models) used in commuter service. The intent of the program was to provide insight into the condition of a typical aged airplane by determining if a correlation exists between the airplane's maintenance history and current condition from a safety of flight perspective. This document provides the findings of a teardown evaluation of a 1969 Cessna 402A model airplane. The results in this report will provide information for use in future investigations into the aged small airplane fleet and determine if additional research is required to address any problems observed. Specific observations are made regarding findings discovered during the teardown evaluation on the particular airplane selected.

The destructive evaluation of the commuter-class airplane was separated into three main tasks: (1) inspection of the airframe and airplane systems, (2) teardown examination of the airframe and airplane systems, and (3) assessment of the airplane wiring. During the inspection phase, three subtasks were performed: a survey of the airplane maintenance records, visual inspection of the airframe and airplane systems, and supplemental airframe inspections. The teardown examination involved disassembling the airframe and major airplane sections, inspecting airplane systems' components, assessing primary airplane structure using alternative nondestructive inspection (NDI) techniques, and performing microscopic examinations of suspect areas and critical structural details. As part of the destructive evaluation, inspections and tests were also performed on airplane wiring to assess the condition and degradation of electrical wiring in small airplanes and to evaluate maintenance procedures.

In the inspection phase, a survey of airplane maintenance records provided information necessary to correlate maintenance practices to airplane condition. Visual inspection of the airframe and airplane systems, along with supplemental airframe inspections, determined the condition of the airplane based on normal maintenance activity. The 1969 Cessna 402A model airplane selected for destructive evaluation accumulated a total of 19,698.9 hours of flight time and was primarily used for tourism flights in the Grand Canyon. Only four findings from the visual airframe and airplane systems inspections were deemed noteworthy by licensed airframe inspectors. These findings included two cracks on the leading edge of the horizontal stabilizer, a loose nut on a wing flap assembly, and broken teeth on the left fuel selector valve. Supplemental NDI, consisting of visual, eddy-current, fluorescent liquid penetrant, and magnetic particle testing, was also performed on the Cessna 402A airplane. Damage was identified in several areas, including the rudder structure, horizontal stabilizer spars, vertical stabilizer spars, wing spar caps, wing tip tank attachments, wing attachment fittings, wing auxiliary spar caps, engine firewall, and the nose gear steering bellcrank.

During the teardown examination phase, all airplane systems' components and wiring were removed, and the disassembly provided full access to all critical structural areas on the airplane. Inspections of the airplane systems' components assisted in determining if any aging effects were related to them. Assessment of primary airplane structure was performed to assess the airframe for surface and subsurface flaws prior to disassembly using alternative NDI techniques with capabilities to detect subsurface flaws. The microscopic examination of critical structural

areas provided verification and detailed quantification of the extent of damage found during the supplemental inspections, assessment of primary airplane structure using alternative NDI techniques, and disassembly of the entire airframe.

Systems' component inspections were accomplished to identify areas where mechanical wear or corrosion had affected either the operation of actuators, linkages, and cables or the ability of hydraulic systems to sustain pressurization. No additional findings were discovered on the systems' components beyond the broken teeth on the left fuel selector valve found during the systems' component visual inspections. A leak test of all oxygen, fuel, and vent lines found nine lines with leaks due to either holes induced by corrosion, loose fittings and couplers, or a recent machining mark. It is possible that the recent machining mark and the leaks around the couplers and fittings were due to the disassembly performed on the airplane and not the actual flight environment. Assessment of the airplane structure with alternative NDI techniques identified potential surface and subsurface defects prior to disassembly of the structure with multiple crack and corrosion indications identified on the front spar of the left wing and horizontal stabilizer. Microscopic examinations of suspect areas and critical structural details located and characterized cracks or areas of corrosion in the following areas: wing attachment fittings, wing tip tank attachment fittings, nose gear steering bellcrank, wing spar caps, wing tip tank baffle, wing auxiliary spars, fuselage channels, stub wing skin lap joint, vertical stabilizer rudder bellcrank, and vertical stabilizer forward attachment fitting.

The inspections and tests performed as part of the airplane wiring assessment were mainly divided into two categories: nondestructive and destructive. The nondestructive inspections and tests were comprised of a general visual inspection, in situ wiring tests, and laboratory tests. The destructive tests were comprised of dielectric withstand voltage (DWV) tests and mandrel bend/wrap back tests. The most dominant issues found in the general visual inspections were chafing, rubbing, cutting, exposed shields, exposed inner conductors, cutting through outer insulation, improper termination of end terminals, improper termination of wires (no cap), unused wires improperly stowed, and corroded terminals. Areas like the instrument panel, side console, wings, and engines displayed the most findings. Looking at the described flaws, it can be inferred that the issues with the wires were caused less by aging and more by maintenance, especially with the wires in the engine compartment. Among all the laboratory tests performed, the circuit breaker test revealed the most important part of the aging issues, showing that nearly 43% of the circuit breakers had defects. The destructive laboratory tests also demonstrated a few issues with wires with respect to age. Since no wires were the required 26 feet or more in length, none of the wires could be reliably tested for DWV. Shorter wires were tested for DWV and 85% of them passed. Only 12.5% of the wires failed the mandrel bend/wrap back test, which indicated that the wires generally had good quality insulation.

1. INTRODUCTION.

The economic and market conditions of present day aviation companies are requiring that airplanes be used far beyond their original design life objectives. This aging airplane concern exists for all types of airplanes including commercial, military, and general aviation (GA). 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 maintenance manuals, Instructions for Continued Airworthiness (ICA), and service bulletins. These practices are not just limited to structural integrity but also include wiring and systems integrity as well. These initiatives have provided timely preventive 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 transport airplanes, where most research funding resources and efforts have been focused, a growing concern also exists in the small airplane fleet. Most small airplanes are generally classified as GA airplanes. When one mentions GA, the traditional image usually involves a four-passenger airplane like a Cessna 172. However, GA covers a wide range of airplanes. In the context of this program, a GA airplane will be defined as a Title 14 Code of Federal Regulations (CFR) Part 23 (or predecessor CAR 3) airplane, which includes normal, utility, acrobatic, and commuter category airplanes. This classification includes airplanes operating commuter, cargo, or taxi service under 14 CFR Part 135. The GA 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. Usage of the GA fleet is categorized as 60% for personal use, 21% for business use, 6% for instruction, 4% for aerial application and observation, 3% for commuter service, 2% for public use, and 4% for other use. The investigations on large transport and military airplanes have focused on the structural integrity as well as wiring and systems-related aging problems. The results of those investigations, however, can benefit similar research programs involving 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 the Aging Transport Systems Rulemaking Advisory Committee have been working to assess the condition of electrical wiring and the effectiveness of wiring maintenance procedures. However, these efforts have been focused on transport category airplanes thus far. Because of the large number of smaller GA airplanes in service, a need for examining the condition of wiring and electrical components and maintenance procedures for these smaller airplanes exist.

Due to the large number of GA airplanes and their wide usage, the aging aspects of these airplanes must be addressed. In September 2002, the FAA Small Airplane Directorate and the Office of Aviation Research and Development Airport and Aircraft Safety R&D Division established a research program to begin addressing the aging concerns regarding small airplanes. The main purpose of this program was to provide insight into the condition of a typical aged

small airplane and determine if a correlation exists between its maintenance and the airplane's apparent condition. The research program was primarily conducted by the Aging Aircraft Research Laboratory at the National Institute for Aviation Research (NIAR), Wichita State University. The program's major objective focused on the integrity and aging aspects of small airplanes.

1.2 RESEARCH OBJECTIVES.

Much of the current concern related to aging effects on airplanes involved calendar age, rather than 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, about 25,000 airplanes presently exist that are older than 50 years 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 types of concerns are what the research program primarily focused on to begin addressing and providing insight into the aging GA fleet.

The research program has a short-term objective, to be achieved over 2 years, and a long-term objective, to be achieved over a 5-year period. The short-term objective is to determine if potential continuing airworthiness problems exist for the GA fleet as a function of the aging process. The long-term objective is to establish guidance to ensure current maintenance programs of small GA 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 issues (corrosion/cracking); electrical systems or wiring; airplane systems, such as fuel, hydraulic, pneumatic, mechanical, and flight control systems; 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 of the initial phase of the program, 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. The finding resulting from this program will be documented in a summary report

for use in future investigations into the aged GA fleet and to determine if additional research is required to address any problems observed. Specific observations made regarding the particular airplane selected and generic recommendations that are applicable to the GA fleet will also be provided.

1.3 TECHNICAL APPROACH.

The destructive evaluation of this commuter-class airplane was separated into three main tasks: (1) inspection of the airframe and airplane systems, (2) teardown examination of the airframe and airplane systems, and (3) assessment of the airplane wiring. Each section below describes the airplane selected for teardown evaluation and the subtasks conducted under each of the main tasks.

1.3.1 Airplane Selection.

The Cessna 402 model was selected because it represents a large portion of the small airplane commuter fleet. It shares many design commonalities with other small twin class airplanes, such as the Piper Navajo. 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 Cessna 402 model are applicable to all GA models, regardless of manufacturer.

Figure 1-1 shows the first airplane selected for destructive evaluation, a 1969 Cessna 402A model (tail number N812BW), which was used in a typical commuter-class role. The twin-engine airplane had almost 20,000 airframe hours with a current registration and was most recently used for commuter service by Sunshine Airlines. Log books, the Airworthiness Directive (AD) compliance list, and FAA 337 forms were included with the purchase of this airplane.

Figure 1-1. 1969 Cessna 402A, Tail #N812BW

1.3.2 Inspection Phase.

In the inspection phase, three subtasks were performed: a survey of the airplane maintenance records, visual inspections of the airframe and airplane systems as prescribed by maintenance manuals, and supplemental inspections per the Supplemental Inspection Document (SID) developed by Cessna Aircraft Company. The maintenance records survey provided information on the airplane maintenance history for correlation of maintenance practices to airplane condition, while the inspections determined the condition of the airplane based on normal maintenance activity.

1.3.3 Teardown Examination Phase.

The research program focused on the destructive evaluation of commuter-class airplanes, although a nondestructive evaluation was also conducted according to recommended practices prior to the destructive evaluation. The teardown examination involved disassembling the airframe and major airplane sections, inspecting the airplane systems' components; inspecting the primary airplane structure using alternative nondestructive inspection (NDI) techniques; and microscopic examination of critical structural areas. All the airplane systems' components and wiring were removed during disassembly, allowing full access to all critical structural areas on the airplane. Inspections of the airplane systems' components also assisted in determining if any aging effects were related to the airplane systems. The alternative NDI was performed to assess the structure prior to disassembly. The microscopic examination of critical structural areas provided verification and detailed quantification of the extent of damage found during supplemental inspections, alternative NDI, and disassembly of the entire airframe.

1.3.4 Airplane Wiring Assessment.

As part of the airplane's destructive evaluation, electrical wiring inspections and tests were performed to assess the condition and degradation of electrical wiring in small airplanes and to evaluate maintenance procedures. The wiring inspections and tests were divided into two categories: nondestructive and destructive. The NDI and tests were comprised of general visual inspection, in situ wiring tests, and laboratory tests, while the destructive tests were comprised of dielectric withstand voltage (DWV), mandrel bend/wrap back, and dynamic cut-through tests.

2. INSPECTION PHASE.

The inspection phase of the Cessna 402A teardown established the current condition of the airplane by a survey of airplane maintenance records, visual inspections per the Cessna 402A Maintenance Manual [1], and supplemental inspections per the 402A SID [2]. These inspections allowed the NIAR team to determine which flaws could be found in the field. The visual inspections were prescribed for specific airframe locations, system components, and wiring locations. The supplemental inspections were composed of NDI techniques targeted at specific airframe locations. These locations were determined based on engineering analysis predictions, structural testing results, or maintenance experience. 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 the airplane maintenance records and a review of Cessna 402A Service Difficulty Reports (SDRs).

2.1 SURVEY OF AIRPLANE MAINTENANCE RECORDS.

Since leaving the production line in 1969, the Cessna 402A airplane was registered under four tail numbers and operated by ten charter services, as shown in table 2-1. The airplane was primarily used for typical charter service from November 5, 1969 to March 1, 1982, and then was used for flying tours in the Grand Canyon, which is considered severe flight profile usage, from March 1, 1982 to July 12, 1999.

Table 2-1. Registration and Usage of Cessna 402A Airplane

Tail Number	Date	Total Airframe Hours	Approved Airplane Inspection Program
N4686Q	11/5/1969 to 2/25/1972	1,186.0	Green Bay Aviation, Green Bay, WI
N4686Q	4/28/1972 to 5/4/1972	1,297.8	Aero Electronics, Memphis, TN
N4686Q	11/1/1974 to 3/4/1975	3,370.6	Airline Services
N4686Q	5/27/1975 to 3/7/1977	5,664.0	Air Midwest, Wichita, KS
N4686Q	3/7/1977 to 7/30/1977	5,925.6	Cruse Aviation, Houston, TX
N3BL	9/19/1977 to 6/8/1979	8,052.4	Bee Line Airlines, Houston, TX
N3BL	6/21/1979 to 3/19/1980	8,567.9	Universal Airways, Gulfport, MS
N3BL	4/25/1980 to 3/1/1982	8,763.8	Clary Aircraft Services, Houston, TX
N300UV	3/1/1982 to 2/13/1998	18,170.9	Air Vegas, Las Vegas, NV
N812BW	2/18/1998 to 7/12/1999	19,698.9	Sunshine Airlines

The service bulletins performed on the airplane are listed in table 2-2, while table 2-3 shows all ADs that the Cessna 402A complied with. The most notable service bulletins and ADs concerning airplane structure were MEB71-2, MEB75-22, and AD 79-10-15. These service bulletins and ADs are briefly described below.

a. MEB71-2 and MEB76-2 address cracking trends in the landing gear wheel wells on high-time 400 series airplane. These service bulletins recommend that the main gear upper side brace support be inspected for looseness and the support attach bolts checked to ensure they are torqued in accordance with the airplane maintenance manual during all future landing gear inspections. These service bulletins also recommend inspection of the wheel wells for cracks. If cracks are found, installation of a reinforcement angle part number (P/N) 5292002-1 is recommended.

b. MEB75-22 advises owners of high-time 400 series airplanes that additional rivets have been added to the horizontal stabilizer front spar on production airplanes and recommends their incorporation on older airplanes.

c. AD 79-10-15 and AD 77-02-02, Amendment 39-2815 requires eddy-current inspections to detect fatigue cracks in critical components of the wing structure. Owners are required to inspect the areas of the wing front spar lower caps and wing front spar root attach fittings, identified as areas A and B in figures 1 and 2 of "Cessna Multi-Engine Service Information Letter MEB 79-16, Revision 3," dated February 8, 1980, for fatigue cracks using eddy-current inspection methods at six locations along the wing front spar lower caps (three locations on the right wing and three identical locations on the left wing) in accordance with Part 2 of the instruction provided in the Service Information Letter. (Note: High-frequency eddy-current inspection is used for areas A and B).

Table 2-2. Service Bulletins the Cessna 402A Airplane Complied With

Date	Service Letter	Description	Total Airframe Hours
1/6/1970	MEB69-35	Improve engine cooling	101
8/17/1970	MEB70-29	Aileron stop bolt inspection	462.5
8/17/1970	MEB70-25	Elevator torque tube inspection	462.5
5/3/1971	MEB71-2	Wing rib inspection	796
7/2/1971	MEB71-8	Elevator torque tube inspection	893
7/2/1971	MEB71-12	Firewall sealant inspection	893
7/2/1971	MEB71-2	Wing rib inspection	893
1/10/1976	MEB75-23	Replacement of two 1/2-inch fork bolts	4296
2/13/1976	MEB75-22	Horizontal stabilizer improvement	4398
3/4/1976	MEB76-4	Gear lock pins	4459
3/18/1983	MEB79-16		9065.1

Table 2-3. Airworthiness Directives Performed on the Cessna 402A Airplane

Date	Airworthiness Directive	Description	Total Airframe Hours
6/29/1970	AD 70-11-2	Exhaust turbo system check	348
7/2/1971	AD 71-9-3	Fuel pump inspection	893
7/2/1971	AD 71-11-4	Oil filter plate modification	893
7/2/1971	AD 70-11-2	Exhaust system check	893
7/2/1971	AD 71-7-4	Emergency exit modification	893
2/25/1972	AD 70-3-4	Turbo housing check	1,186
2/25/1972	AD 72-3-7	N/A	Not Reported
3/7/1977	AD 70-11-2	Engine exhaust inspection	Not Reported
3/7/1977	AD 72-14-8	Fuel and oil leak	2,320
3/7/1977	AD 75-23-8	Exhaust leak	2,320
3/7/1977	AD 76-2-7	Alternator bearings	2,370
3/7/1977	AD 75-223-8	Exhaust clamps left hand	2,490
3/7/1977	AD 75-223-8	Exhaust clamps right hand	2,554
3/7/1977	AD 76-13-7	Fork bolt replacement	6,295
1/3/1979	AD 75-23-8R	Exhaust inspection	7,418.6
7/5/1979	AD 72-14-08	Fuel line inspection	8,148.8
2/11/1981	AD 79-10-15	Wing spar inspections	8,567
12/1/1997	AD 79-10-15	Wing spar inspections	18,170.9
12/1/1997	AMDT 39-3711	Wing spar inspections	18,170.9
11/10/1998	AD 98-23-01	Vacuum pump	18,506.7
12/22/1998	AD 98-25-10	Seat belt inspection	18,547.5
3/7/2000	AD 2000-01-16	Turbocharged twin Cessna exhaust system	19,060.7

All documented major repairs and alterations are listed in table 2-4. The installation of doublers on the right wing lower spar cap is the only major structural repair or alteration. This alteration is discussed briefly below.

The left wing skin from wing station (WS) 75.24 to WS 65.99 was removed for eddy-current and dye penetrant inspection of the spar cap. The 0.040″ wing skin splices, 0.063″ skin splice doubler, and the 0.063″ strap was installed in accordance with engineering drawing #AV SRI at splice zone B. Rivets were replaced in the wing spar cap along the splice zone with AN-470, AN-470-6, CR3243-5, and CR3243-6 fasteners as applicable. This repair was accomplished in accordance with the Cessna Aircraft-recommended repair letter dated November 19, 1990, and was previously approved on FAA Form 337 dated February 11, 1991.

Table 2-4. Major Repairs/Modifications Made on the Cessna 402A Airplane

Date	Description
6/15/1981	Moved battery from wing to nose
6/26/1985	Removed paper induction air filter and installed bracket filters
5/9/1986	Removed Cessna EGT system and installed Alcor EGT system
11/10/1987	Installed trans cal blind encoder
9/8/1989	Removed navigation system, gyro, and associated wiring and circuits installed King Navigation Indicator system
3/6/1991	Installed doublers on right wing lower front spar cap from WS 75.24 to 65.99
5/6/1991	Removed existing passenger tape system and installed cassette tour system

Table 2-5 gives all the maintenance log entries for the Cessna 402A. It should be noted that a 7-year gap exists in the maintenance logs from May 1991 to May 1998, and no airframe hours were recorded from May 1984 to May 1998. Figure 2-1 shows the total airframe hours and dates recorded in the registration and maintenance logs along with the dates that service bulletins, ADs, and major repairs/modifications were performed on the airplane.

Table 2-5. Maintenance Log Entries for the Cessna 402A Airplane

Date	Description	Total Airframe Hours
11/5/1969	Installed autopilot and radio equipment	8
4/24/1970	Installed Cessna service kits SK-402-18 and SK 402-19B	225
6/3/1970	Performed annual inspection and installed SK-402-28	300
7/21/1970	Performed 100-hour inspection and installed altitude gyro	403
9/11/1970	Replaced slave gyro circuit breaker	530
10/27/1970	Installed SK-402-23 and SK-402-25	576
1/19/1971	Replaced turn and bank on right side of panel	686
1/26/1971	Performed 100-hour inspection, installed new lining on right main gear	696
2/10/1971	Hobbs time removed and reinstalled overhauled right engine	698
6/16/1971	Hobbs time removed the horizon and sent it in for an overhaul, horizon has no auto pilot connection	873
6/28/1971	Hobbs time reinstalled the horizon	891
8/17/1971	Replaced right main tire	955
11/3/1971	Performed 100-hour inspection replaced bushings in main gear torque links SK421-23, rebuilt right strut, and installed new O-ring, replaced taxi light	1,092
11/5/1971	Installed all new brake blocks and discs	1,097
4/28/1972	Installed weather radar system	Not Recorded
5/27/1975	Installed two altimeters, repaired leak in static system	3,370.6

Table 2-5. Maintenance Log Entries for the Cessna 402A Airplane (Continued)

Date	Description	Total Airframe Hours
6/5/1975	Installed new tachometer	3,383.5
8/23/1975	Removed KXP750 XPDR and installed Bendix TPR660 ATC XPDR	Not Recorded
9/5/1975	Removed T-25 and installed RT 11-A	Not Recorded
2/13/1976	Reinstalled horizontal stabilizer	4,398
3/7/1977	Airplane removed from Air Midwest AAIP	5,664
5/16/1977	Performed 100-hour inspection, replaced cigarette lighter circuit breaker, repaired lower beacon, removed and resealed emergency exit window, installed new seals in nose shock strut, and repaired pilot's door seal	5,707.8
6/28/1977	Installed new left and right main tires, repaired left nav. light, replaced bulb top beacon, replaced motor and bulb in lower beacon, installed new bolts in left gear upper side brace attach plates	5,824.7
8/17/1977	Registration number changed from N4586Q to N3BL	Not Recorded
9/19/1977	New Hobbs meter installed, placed on the AAIP Beeline Airline	6,027.6
3/22/1978	Removed and replaced with continental 0 time units IAW Cessna Service Manual	6,187.0
5/18/1978	Complied with progress inspection parts 1-6 as required by inspection program	6,398.5
1/17/1979	Installed alternator left engine and right engine	7,529.3
3/8/1979	Installed engine gauge on right side and installed new rudder trim actuator	7,689.6
3/11/1979	Installed new turbo supercharger and turbo relief valve on left engine	7,694.5
4/9/1979	Installed turbo on right engine, installed propeller cable left engine, installed new OHT gauge, tachometer, flap cable, and transmission, installed new side window	Not Recorded
4/20/1979	Rebuilt right master cylinder and right wheel cylinder, installed new O-ring and insulators	Not Recorded
3/19/1980	Removed flap motor and installed new flap motor and re-rigged microswitch	Not Recorded
2/11/1981	Installed left engine, performed 100-hour inspection, replaced all brake discs and housings, replaced left gear fork bolt and trunnions, replaced cracked exhaust stacks, repaired broken wire on landing gear arms, altimeter removed and sent for overhaul, removed outboard flaps to remove corrosion, re-skinned flaps, repaired right skin due to exhaust leak	7,567
6/15/1981	Moved battery from wing to nose	Not Recorded

Table 2-5. Maintenance Log Entries for the Cessna 402A Airplane (Continued)

Date	Description	Total Airframe Hours
7/30/1981	Removed heater for repair and overhaul	Not Recorded
9/1/1981	Right engine installed	8,710
10/1/1981	Installed new starter on right engine, replaced right engine turbo regulator	8,721.4
3/2/1982	Performed 100-hour inspection, installed heater meter	8,763.8
10/21/1982	Performed 100-hour inspection, replaced seals in nose gear strut	8,965.7
11/25/1981	Overhauled engines, installed governors and main pressure relief valve	8,986.7
4/20/1983	Removed propeller for overhaul	Not Recorded
7/6/1983	Installed overhauled propeller on right side	9,169.5
12/1/1983	Installed new battery and replaced exhaust clamp	9,413.8
5/25/1984	Installed overhauled propeller governor and new propeller control cable	9,556.6
6/26/1985	Removed paper induction air filter and installed bracket filters	Not Recorded
5/9/1986	Removed Cessna EGT System and installed Alcor EGT system	Not Recorded
11/10/1987	Installed trans cal blind encoder	Not Recorded
9/8/1989	Removed navigation system, gyro, and associated wiring and circuit breakers, installed King Navigation Indicator System	Not Recorded
3/6/1991	Installed doublers on right wing lower front spar from WS 75.24 to 65.99	Not Recorded
3/7/1991	Installed doublers on right wing lower front spar from WS 75.24 to 65.99	Not Recorded
5/6/1991	Removed existing passenger tape system and installed cassette tour system	Not Recorded
5/18/1998	Removed and replaced all passenger compartment seat tracks with new tracks, installed two each fasten seat belt signs, repaired left nav. light lens, replaced left and right main gear door hinges due to wear, replaced working rivets in right-hand wing locker, inspected repair patches on elevator and horizontal stabilizer, replaced missing fasteners on left and right main tank fairings, installed missing fasteners on nose cone, replaced broken fastener below co-pilot window	18,173.9
5/20/1998	Added onto Sunshine Airline AAIP	Not Recorded
10/15/1998	Removed failed vacuum pump on right engine, installed overhauled vacuum pump	18,453.9
1/24/1999	Removed left engine and installed new engine	18,614.9

Table 2-5. Maintenance Log Entries for the Cessna 402A Airplane (Continued)

Date	Description	Total Airframe Hours
4/22/1999	Painted	Not Recorded
6/10/1999	Installed new seat covers, curtains, carpet, headliner, and side panels	Not Recorded
6/18/1999	Installed TSIO-520-ECEB	18,617.6
7/12/1999	Removed right engine and replaced it	18,617.6
10/15/1999	Removed failed tip tank pump on left engine, installed a new pump	18,673.3
12/12/1999	Installed new heater	Not Recorded
3/15/2002	Performed inspection	19,698.9

AAIP = Approved Airworthiness Inspection Program
OHT = Overheat
IAW = In accordance with
EGT = Exhaust gas temperature

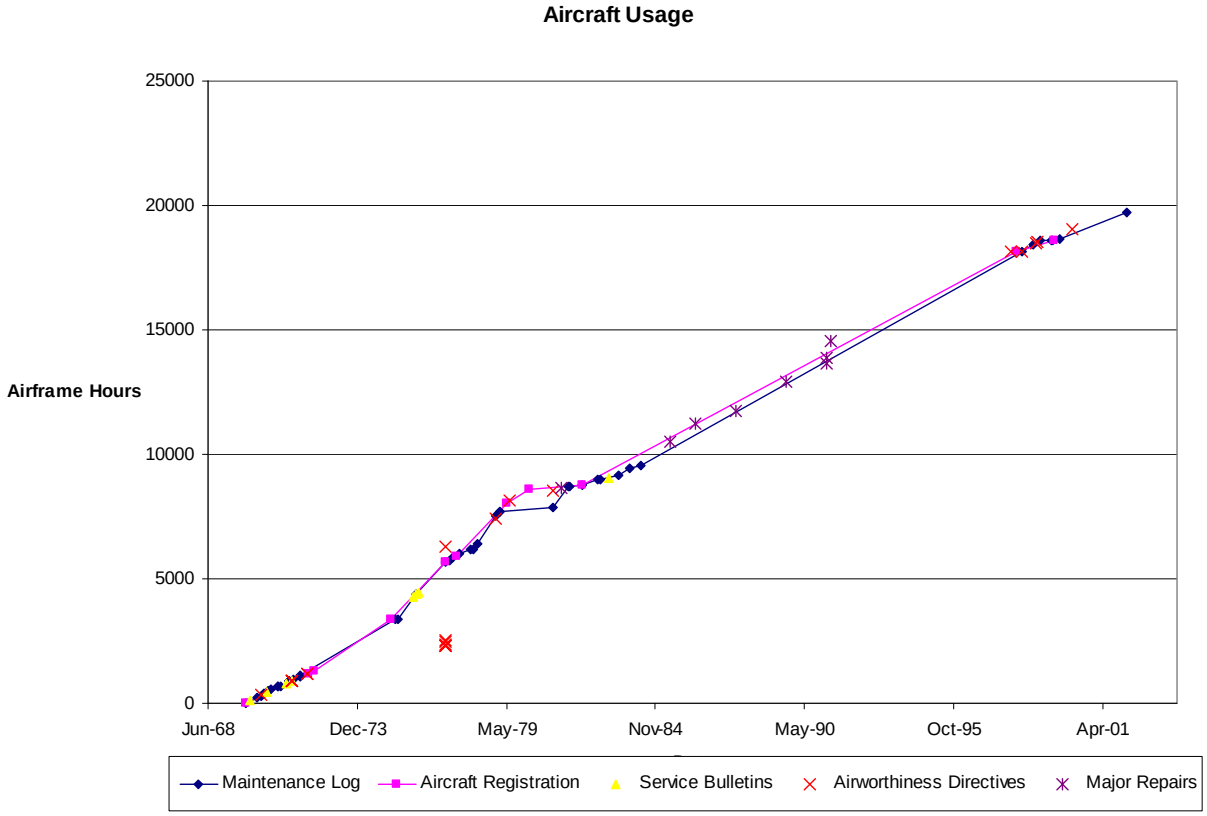

Figure 2-1. Airplane Usage

2.2 SERVICE DIFFICULTY REPORTS DATABASE REVIEW.

The SDR database review of the Cessna 402A consisted of 593 reports from January 1974 to November 2002. Figure 2-2 shows the breakdown of SDRs by airplane section. Almost a third of the reports were related to landing gear and 43% were related to the engines. Only 14% were related to the airframe, which is the area of primary interest for this teardown evaluation. SDRs are not always generated for each issue that arises; therefore, common problems may exist that are not appropriately represented by the SDR database.

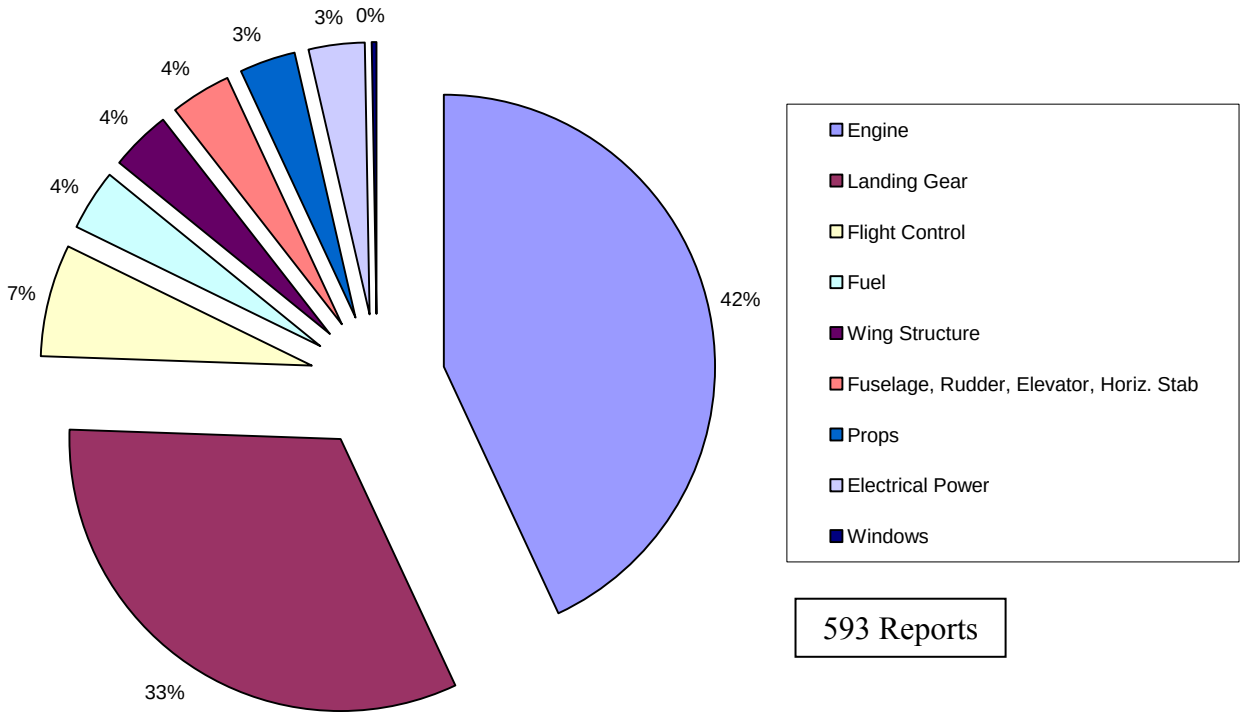

Figure 2-2. Results of the SDR Database Review by Airplane Section

2.3 VISUAL INSPECTION OF AIRFRAME AND AIRPLANE SYSTEMS.

The general condition of the airplane was evaluated through visual inspections prescribed in the Cessna 402A Maintenance Manual [1] supplied by Cessna. These visual inspections are typically required on an annual basis. The intent of these inspections was to discover all possible visible defects and document each defect's severity and location. Each inspection was duplicated by independent inspectors. The results of these inspections were reviewed by licensed airframe mechanics to determine which findings (defects that would require further maintenance action) were noteworthy.

2.3.1 Airplane Systems Inspections.

Forty-seven different systems inspections were performed in accordance with the maintenance manual, which included the following instructions:

- Inspect for basic flaws such as cracks, dents, corrosion, missing fasteners, loose fasteners, and scratches.

- Inspect for security, looseness, wear, proper rigging, bolthole elongation, operation and travel, cable fraying, chafing, cleanliness, tension, turnbuckle security, and freedom of movement.

- Search for evidence of fuel and oil leaks, burned spots, and heat damage.

- Inspect for excessive wear on critical moving parts, leaks or chafing of hoses, and instrument condition.

- Inspect for leaks, kinks, and chafing of hoses, skin and splices for cracks, wrinkles, abnormal stresses, and elongated boltholes.

- Inspect attaching bolts and hinge attaching points for security; check for sheared rivets and buckling of spar web on horizontal stabilizer.

- Check heater fuel system for leaks, secure connections, and condition of seals and gaskets; check seats and belts for condition, security, operation, and fraying of webbing material; examine control quadrant for excessive wear; and ensure that no broken teeth of mixture control racks and ratchet stops exist.

- Inspect miter gears and sprockets for chipped or broken teeth; inspect bolt- and pinholes for elongation; check tire pressure; inspect retainers and snap rings for distortion; check hoses for leaks, bulges, deterioration, and chafing; inspect brake lining for disc warping and wear limits of brake linings.

Each inspection, as shown in figure 2-3, was performed by two inspectors independently. The only finding from the systems visual inspection was broken gear teeth on the left fuel selector valve (P/N 5226005-5). The location of the fuel selector valve is shown in figure 2-4, while the broken gear teeth are shown in figure 2-5.

- Flight Controls
  - Control Column
  - Aileron
  - Aileron Trim Tab and Actuator
  - Cables
  - Elevator
  - Elevator Trim Tab and Actuator
  - Rudder
  - Rudder Trim Tab and Actuator
  - Flaps
  - Flap Motor and Position Indicator
  - Actuator Assembly
  - Autopilot

- Engine
  - Engine Compartments
  - Engine Controls
  - Engine Wire Bundle
  - Engine Mounts
  - Engine Exhaust System

- Vacuum System
  - Vacuum Pump Coupling
  - Hoses
  - Air Filter
  - Relief Valve
  - Regulator and De-ice Flow Valves

- Fuel System
  - Electric Solenoid and Filter
  - Metering Unit Filter Screen
  - Injection System and Manifold
  - Discharge Nozzles
  - Selector Valve
  - Sediment Bowl
  - Selector Valve Filter
  - Main System
  - Tip Tanks
  - Boost Pumps (Auxiliary)
  - Tip Tank Vents
  - Nose Cap Drain
  - Auxiliary Tanks
  - Heater Fuel Filter
  - Main Tank Fuel Transfer Pump and Filter
  - Wing Locker Transfer Pump
  - Auxiliary In-Line Pump and Filter

- Oxygen System
  - Oxygen System
  - Oxygen Regulator
  - Filler Valve
  - Masks and Hoses

- Pitot Static System
  - Sump
  - Pitot Static System
  - Altimeter
  - Alternate Static Drain

Figure 2-3. Airplane Systems Inspections

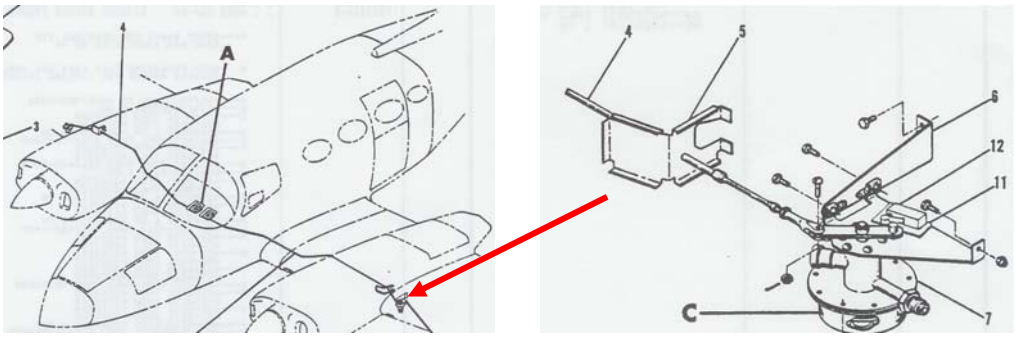

Figure 2-4. Location of the Fuel Selector Valve

Figure 2-5. Broken Teeth on the Left Fuel Selector Valve

2.3.2 Airframe Visual Inspections.

Thirty-six different visual airframe inspections, as shown in figure 2-6, were performed on the Cessna 402A with the intent of finding every detectable flaw on the airplane, which led to a large number of documented flaws. Most of these flaws were deemed minor by licensed airframe mechanics, meaning that no further maintenance action was required. With an aged airplane, flaws such as minor cracks, scratches, dents, paint chips, and slight corrosion were to be expected and posed 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. Also, it was noted that some of the disposable parts such as seals and hoses were due to be replaced.

- Fuselage
  - Exterior Surfaces
  - Internal Structure
  - Windows
  - Doors
  - Nose Ram Air

- Wing
  - Exterior Surfaces
  - Internal Structure
  - Wheel Wells
  - Spar Fittings
  - Front Spar Lower Cap
  - Locker Door
  - Tip Tank Fittings
  - Engine Support Structure
  - Firewall Structure
  - Stub Wing Structure

- Empennage
  - Vertical Stabilizer
  - Horizontal Stabilizer

- Interior/Systems
  - Heater
  - Seats
  - Seat Belts
  - Instruments
  - Control Quadrant

- Landing Gear
  - Main Landing Gear Actuator
  - Emergency Extension System
  - Main and Nose Landing Gear Assemblies
  - Rigging
  - Torque Links
  - Retraction Linkages
  - Dive Tube Seals
  - Shimmy Damper
  - Nose Steering System
  - Tires
  - Wheel Bearings
  - Brake System Plumbing
  - Brake Assembly
  - Master Cylinder

Figure 2-6. Airframe Visual Inspections

Only two areas of the airframe were determined to need corrective maintenance. Figure 2-7 shows the location of two 2.5-inch cracks on the leading edge of the horizontal stabilizer (P/N 5032000-18). Cracks A and B are shown in figures 2-8 and 2-9, respectively. The location of both cracks corresponds to holes for the deice boots. When the deice system was removed from this airplane, the holes were filled with putty. These cracks are in the dried putty, not the aluminum leading edge. The location of one loose nut found during an inspection of the flaps is shown in figure 2-10. This loose nut, shown in figure 2-11, could possibly be attributed to the initial disassembly of the airplane prior to transportation to the laboratory facility.

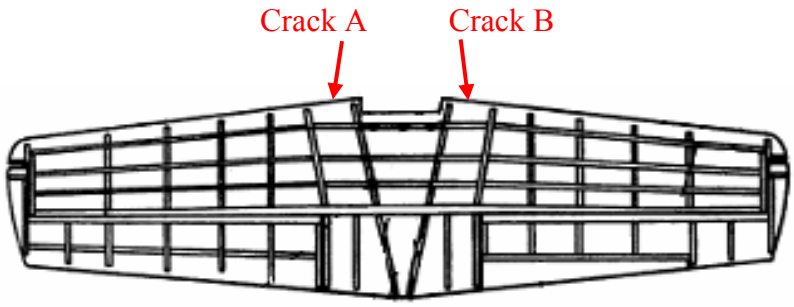

Figure 2-7. Location of Cracks on the Horizontal Stabilizer Leading Edge

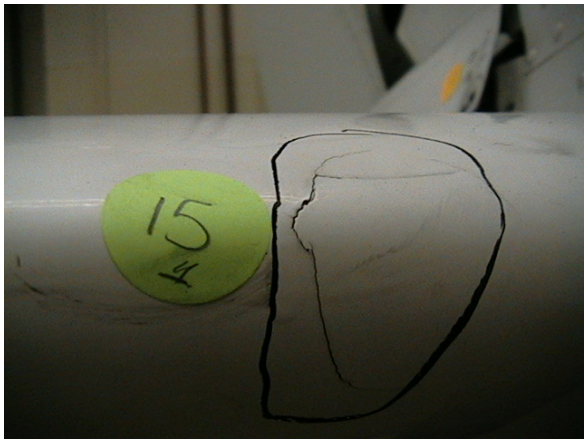

Figure 2-8. Crack A on the Horizontal Stabilizer Leading Edge

Figure 2-9. Crack B on the Horizontal Stabilizer Leading Edge

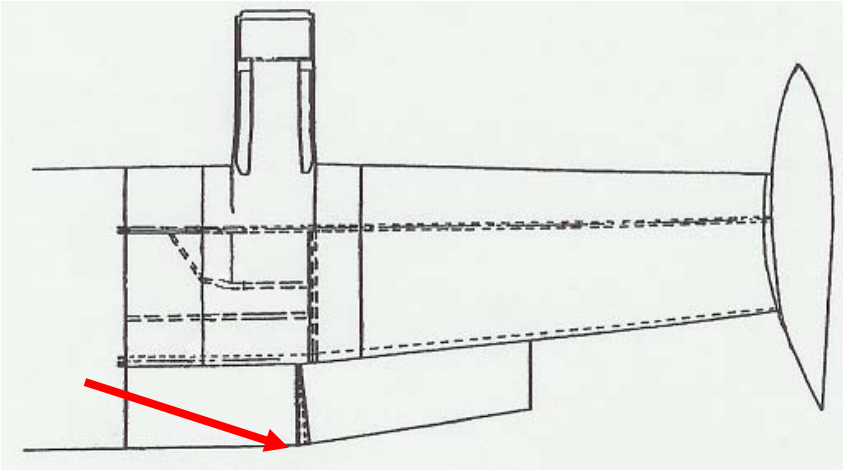

Figure 2-10. Location of Loose Nut

Figure 2-11. Loose Nut on Flap

At the conclusion of the routine visual inspections, 1930 total findings were documented. Only three findings were deemed noteworthy by licensed airframe mechanics. These findings were the broken gear teeth on the left fuel selector valve, the two 2.5-inch cracks on the leading edge of the horizontal stabilizer, and the loose nut found on a flap. A majority of the findings were corrosion-related, as shown in figure 2-12. Most of the findings were located on the fuselage and wings, as shown in figure 2-13.

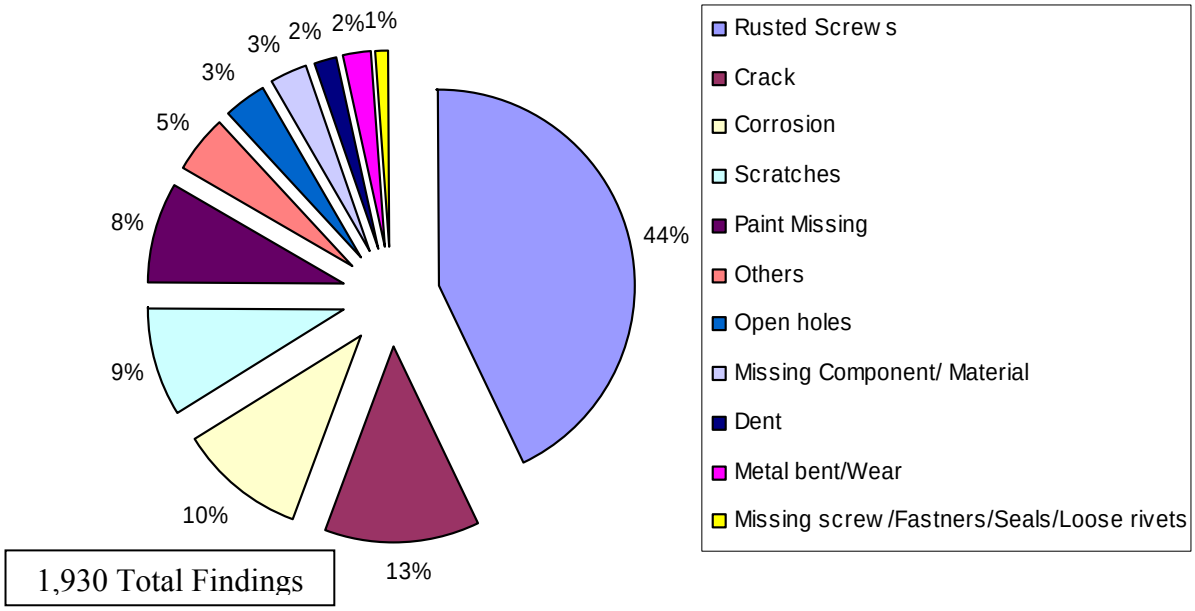

Figure 2-12. Visual Inspection Findings by Type

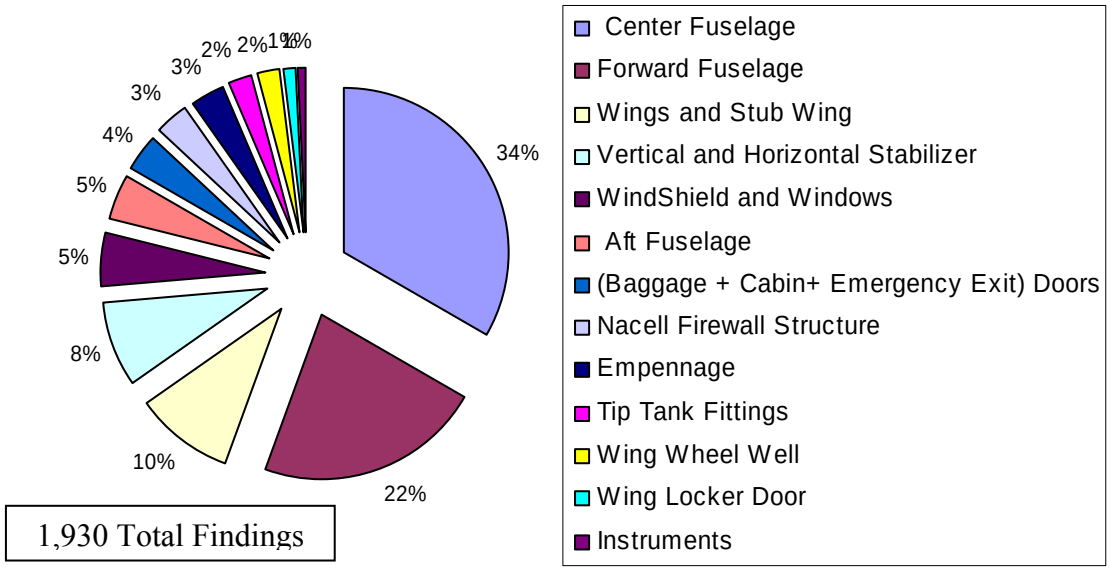

Figure 2-13. Visual Inspection Findings by Airplane Location

2.4 SUPPLEMENTAL INSPECTIONS.

All supplemental inspections were performed in accordance with SID [2]. These supplemental inspections were developed to use NDI techniques to inspect areas that were analytically predicted to have damage, exhibited damage in a structural test, or had a history of maintenance issues. Stress, fatigue, and damage tolerance analyses were used to predict structural locations that were more likely to experience cracking. These analyses were supported with static and fatigue tests of structural components and a full-scale airplane to determine areas where failure or cracks may occur. Three sources of information were used to determine the frequency of maintenance issues: (1) survey of current Cessna model 402A owners, (2) service bulletins and service information letters, and (3) FAA SDRs.

The supplemental inspections 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 and vertical stabilizers, and landing gear. All of the visual supplemental inspections were performed by two licensed airframe mechanics from Cessna and Cape Air. A Cessna-supplied Level III NDI examiner, an NIAR Level II examiner, and three Kansas Air National Guard-qualified examiners performed all the remaining supplemental inspections. Table 2-6 includes the supplemental inspection name, method, and SID number performed on each of the wings. Table 2-7 lists the inspections performed on the fuselage, while tables 2-8 and 2-9 detail the inspections performed on the horizontal and vertical stabilizers and their respective flight control surfaces. Table 2-10 contains the inspections conducted on the main and nose landing gears.

Table 2-6. Supplemental Wing Inspections

Title	Method of Inspection	SID Number
Aileron hinges and fittings	Visual	27-10-05
Aileron hinges and fittings	Liquid penetrant	27-10-05
Engine beams	Visual	54-10-04
Engine beams	Eddy current	54-10-04
Engine beam modification	Eddy current	54-10-05
Wing rib modification-main landing gear side brace	Eddy current	57-10-03
Wing rear spar lower cap at splice, WS 97.87	Eddy current	57-10-06
Rear carry-through spar lower cap at BL 37.60	Eddy current	57-10-07
Front wing spar lower cap inspection and modification	Eddy current	57-10-08
Aft auxiliary spar lower cap at WS 89.65	Eddy current	57-10-09
Front carry-through spar lower cap	Eddy current	57-10-10
Wing front spar lower cap at root fitting attach WS 46.70	Eddy current	57-10-11
Wing front spar lower cap at root fitting attach WS 54.10	Eddy current	57-10-12
Forward auxiliary spar lower cap at WS 86.62	Eddy current	57-10-13
Wing front spar lug inspection	Eddy current	57-10-22
Wing tip tank attachment inspection	Eddy current	57-10-24
Upper wing to carry-through attachment fittings	Visual	57-10-27

Table 2-7. Supplemental Fuselage Inspections

Title	Method of Inspection	SID Number
Emergency exit door lever mechanism inspection	Visual	52-20-01
Fuselage left and right-hand channel assemblies	Surface eddy current	53-10-02

Table 2-8. Supplemental Horizontal Stabilizer and Elevator Inspections

Title	Method of Inspection	SID Number
Elevator torque tube assembly	Visual	27-30-01
Horizontal stabilizer spars and attachments	Visual	55-10-03
Horizontal stabilizer front spar upper cap	Eddy current	55-10-04
Horizontal stabilizer front spar lower cap	Eddy current	55-10-05
Horizontal stabilizer front spar attach, BL 7.69	Eddy current	55-10-06
Horizontal stabilizer rear spar lower cap attach	Eddy current	55-10-07
Horizontal stabilizer rear spar upper cap BL 0.00	Eddy current	55-10-08
Horizontal stabilizer rear spar lower cap BL 0.00	Eddy current	55-10-09
Outboard elevator hinge bracket and attachment	Visual	55-20-01
Elevator hinges and fittings	Visual	55-20-02
Elevator hinges and fittings	Liquid penetrant	55-20-02

Table 2-9. Supplemental Vertical Stabilizer and Rudder Inspections

Title	Method of Inspection	SID Number
Rudder structure	Visual	27-20-03
Rudder torque tube	Visual	27-20-04
Vertical stabilizer spars and attachments	Visual	55-30-01
Rudder hinges and fittings	Visual	55-30-02
Rudder hinges and fittings	Liquid penetrant	55-30-02
Vertical stabilizer rear spar cap attach, WL 108.38	Eddy current	55-30-04

Table 2-10. Supplemental Main and Nose Landing Gear Inspections

Title	Method of Inspection	SID Number
Main landing gear fork bolts (1/2 inch)	Visual	32-10-00
Main landing gear fork bolts (5/8 inch)	Visual	32-10-01
Main landing gear torque tube assembly	Magnetic particle	32-10-02
Main landing gear bellcrank pivot bolt	Visual	32-10-03
Main gear actuator collar	Magnetic particle	32-10-04
Nose gear fork	Surface eddy current	32-20-02
Main/nose gear retraction system	Visual	32-30-05
Nose gear trunnion inspection	Liquid penetrant	32-30-07
Nose gear steering bellcrank	Visual, liquid penetrant	32-50-00

2.4.1 Visual Supplemental Inspection Results.

Visual inspection 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 5 to 10 power hand-held magnifying glass 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 supplemental inspections are listed in table 2-11. The most notable damage observed during the visual supplemental inspections was two cracks on the leading edge of the horizontal stabilizer and a crack in a horizontal stabilizer rib. One of the cracks on the leading edge of the horizontal stabilizer, detected during the routine visual inspections, is shown in figure 2-14. A crack, shown in figure 2-15, was observed in one of the horizontal stabilizer ribs during the visual supplemental inspections.

Table 2-11. Visual Supplemental Inspection Results

SID Inspection Number	Name of Inspection	Discrepancy Reported
27-10-05	Aileron hinges and fittings	1. Need to change two rivets on outboard hinge forward side
27-20-03	Rudder structure	1. Rib with several patches needs to be replaced 2. Elongated holes on lower rudder fittings, P/Ns 5133000-31 and 5133000-32 3. Crack in doubler on top hinge fairing 4. Bonded angle not attached on one end
27-20-04	Rudder torque tube	1. Corrosion on inside and outside of rudder torque tube, P/N 5133011-1
27-30-01	Elevator torque tube assembly	1. Corrosion on inside and outside of elevator torque tube, P/N 5093403-1 2. Crack in skin near inboard attach fitting
55-10-03	Horizontal stabilizer spars and attachments	1. Change cherry max on each end cap to solid rivets 2. Patch crack on leading edge 3. Crack in left-hand inboard rib 4. Remove hi-loks and cherry max, replace with solid rivets
55-20-01	Outboard elevator hinge bracket and attachment	1. Remove cherry max and replace with solid rivets

Figure 2-14. Crack on the Horizontal Stabilizer Leading Edge

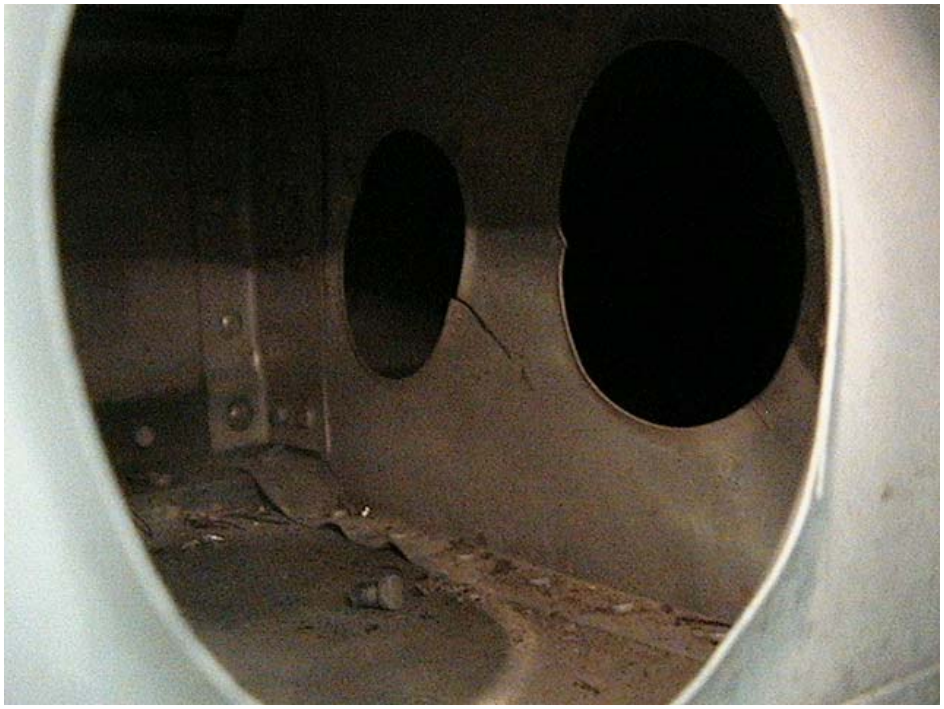

Figure 2-15. Crack in the Horizontal Stabilizer Rib

2.4.2 Fluorescent Liquid Penetrant Supplemental Inspection Results.

The liquid penetrant inspection is a three-step process, which is illustrated in figure 2-16 [3].

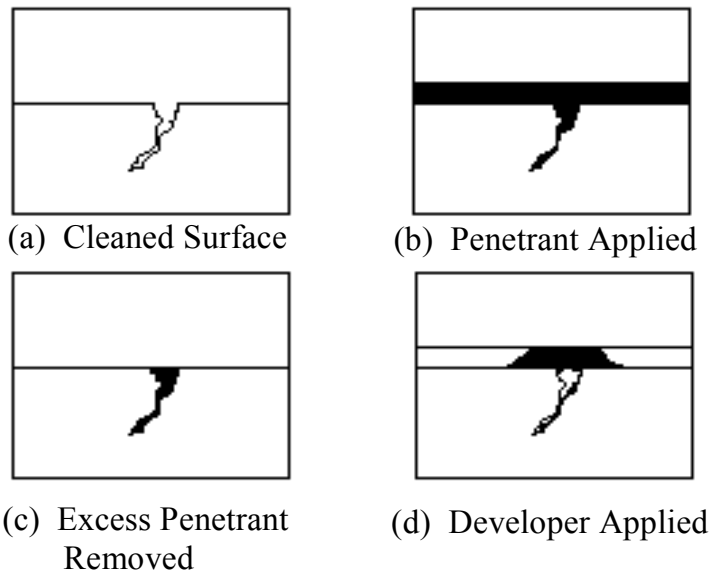

Figure 2-16. Penetrant Inspection Process

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 fluorescent penetrant is used. Fluorescent liquid penetrant is widely used because it is a relatively simple and inexpensive process, but the process is highly dependent on following established procedures to detect discontinuities in the material. Table 2-12 shows the results of the fluorescent liquid penetrant supplemental inspections, which included inspection of the nose gear steering bellcrank. As shown in the table, three small cracks were detected in the bellcrank during the fluorescent liquid penetrant supplemental inspections. The location of the nose gear steering bellcrank (P/N 5042010-1) is illustrated in figure 2-17. Figure 2-18 shows the location of the cracks on the nose wheel steering bellcrank.

Table 2-12. Dye Penetrant Supplemental Inspection Results

SID Inspection Number	Name of Inspection	Discrepancy Reported
32-50-00	Nose gear steering bellcrank	Three small cracks in bellcrank

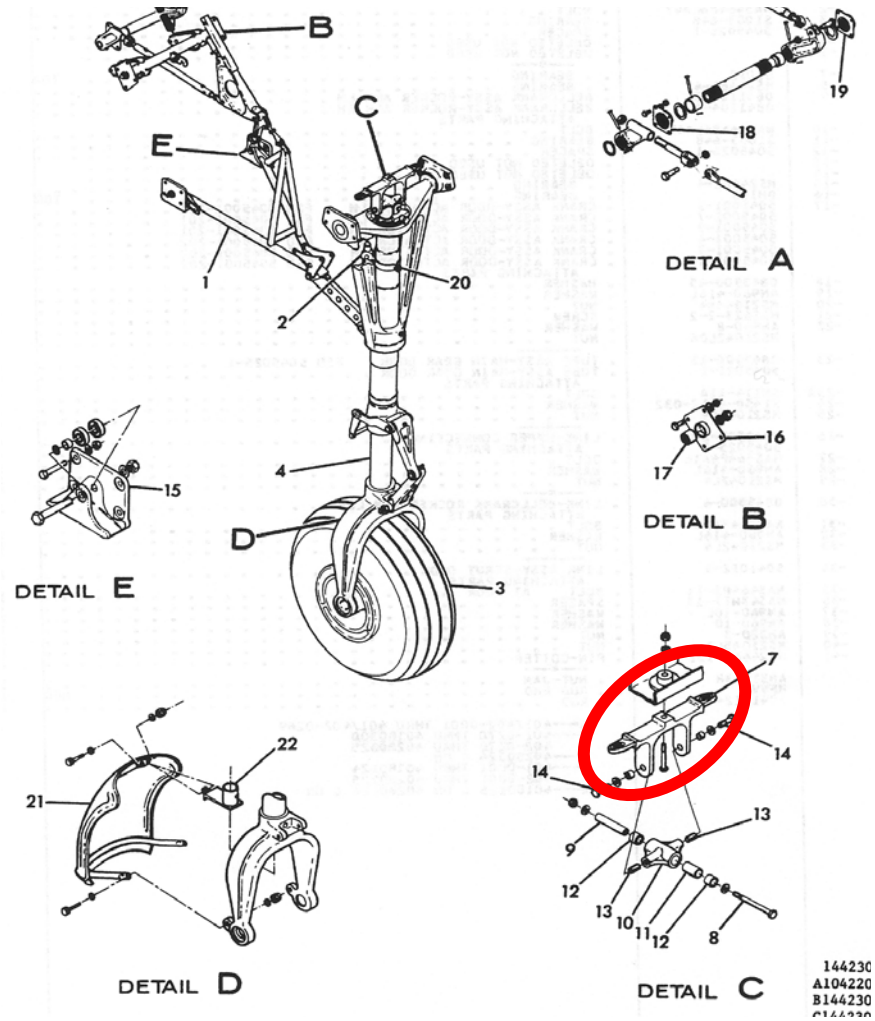

Figure 2-17. Location of Nose Gear Steering Bellcrank

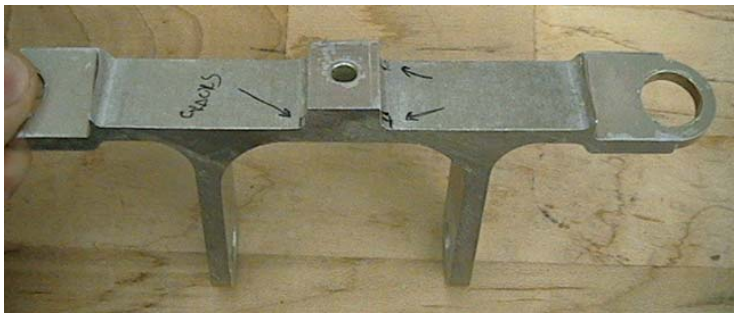

Figure 2-18. Three Small Cracks in the Bellcrank

2.4.3 Magnetic Particle Supplemental Inspection Results.

Magnetic particle inspections consist of three basic steps. The first step is to magnetize the part. Magnetic particles, such as iron fillings, are then sprayed onto the part. These particles are the media through which the flaws become visible and the indications are interpreted by a qualified inspector. Fluorescent magnetic particles are most commonly used for airplane inspections. Figure 2-19 [3] shows a typical indication of a surface crack. Magnetic particle inspection is easy to perform, requires minimal training, and results can be achieved quickly. However, it is limited to relatively small inspection areas comprised of magnetic material. During the inspection of the main landing gear torque tubes, one inspector recorded an indication of both welds being cracked. The location of the main landing gear torque tubes (P/N 5045010-1), is shown in figure 2-20, while one of the welds is shown in figure 2-21.

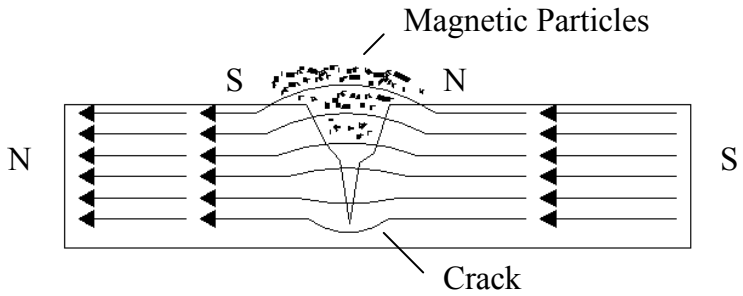

Figure 2-19. Surface Crack Indications

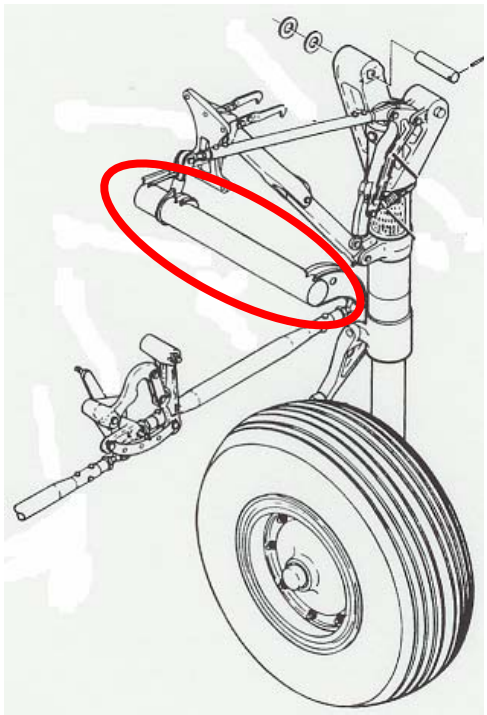

Figure 2-20. Location of Main Landing Gear Torque Tubes

Figure 2-21. Cracked Weld on the Torque Tubes

2.4.4 Eddy-Current Supplemental Inspection Results.

The basic setup for an eddy-current probe is shown in figure 2-22 [3]. A primary magnetic field created by the alternating current causes the current to flow in a circular direction, known as eddy currents. For the current to flow, the test part must be made from 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 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.

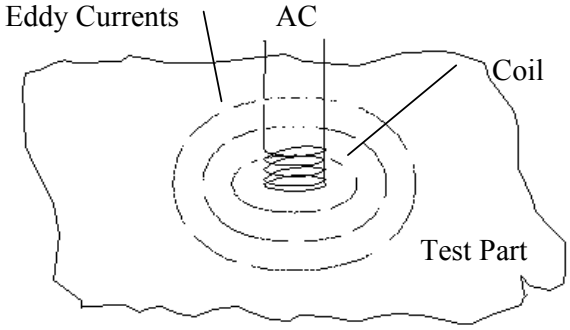

Figure 2-22. Eddy-Current Probe

A majority of the supplemental inspections were performed using eddy current. Table 2-13 lists the results from the SID eddy-current inspections. Figure 2-23 shows the location of a skin crack on the right wing front lower skin at WS 57.5. This crack is shown in figure 2-24. Figure 2-25 shows the location of a skin crack on the left wing aft lower auxiliary spar, while figure 2-26 shows the crack between two fasteners. All crack indications were investigated further in the teardown examination to attempt to confirm the indication.

Table 2-13. Eddy-Current Supplemental Inspection Results

SID Inspection Number	Name of Inspection	Discrepancy Reported
54-10-05	Engine beam modification	Left and right firewalls need to be replaced, failed conductivity test
55-30-04	Vertical stabilizer rear spar cap attach, WL 108.3	Two crack indications
57-10-06	Wing rear spar lower cap at splice, WS 97.87	Crack indication
57-10-07	Wing carry-through spar lower cap at BL 37.60	Crack indications on all inspected holes
57-10-08	Front wing spar lower cap	1. Crack indication at WS 55 on right wing 2. Crack indication at WS 85 on right wing 3. Crack indication at WS 57.5 on right wing
57-10-09	Aft auxiliary spar lower cap at WS 89.65	1. Crack indication at WS 92 on left wing 2. Two crack indications at WS 90 on left wing 3. Two crack indications at WS 92 on right wing
57-10-10	Front carry-through spar lower cap BL 46.89	Crack indications on all inspected holes
57-10-13	Forward auxiliary spar lower cap WS 86.62	Crack indication at WS 89 on left wing
57-10-22	Wing front spar lug inspection WS 46.89	Indications at all inspected holes
57-10-24	Wing tip tank attachment inspection	Indications at all inspected holes

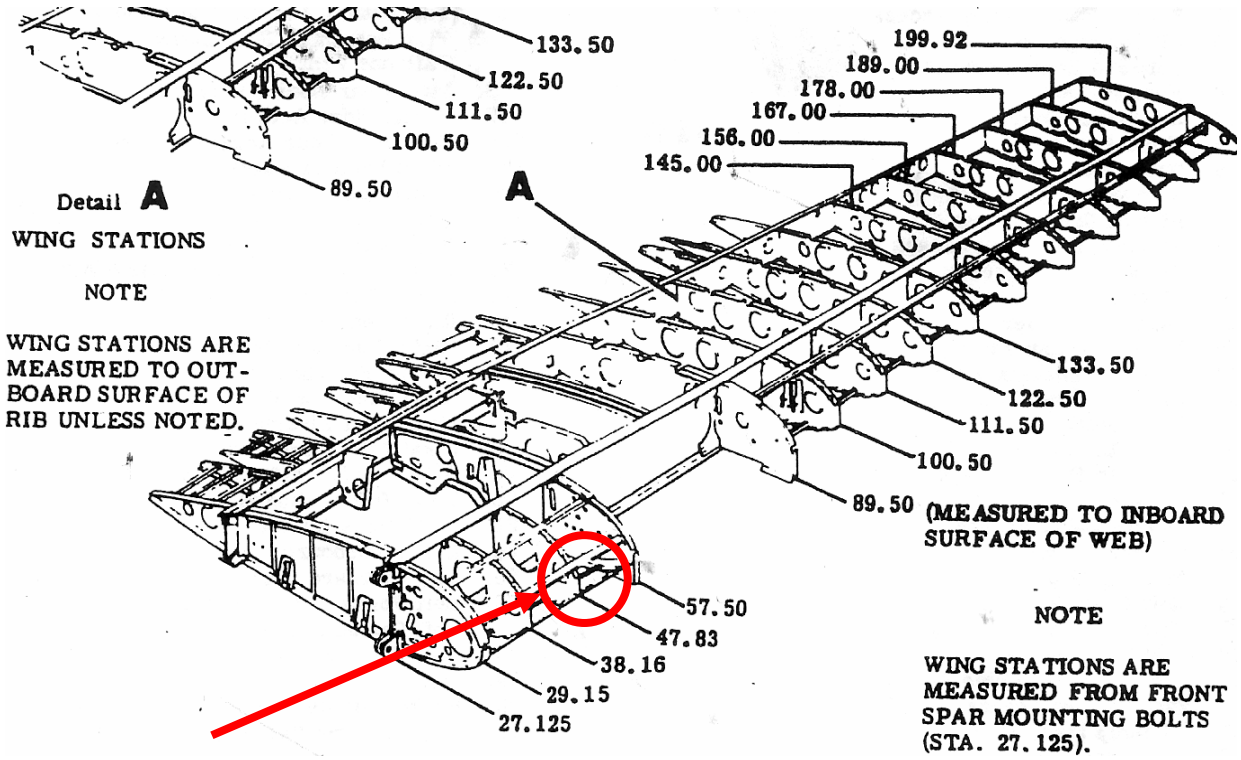

Figure 2-23. Location of Skin Crack on the Right Wing Front Lower Skin

Figure 2-24. Crack on the Right Wing Front Lower Skin

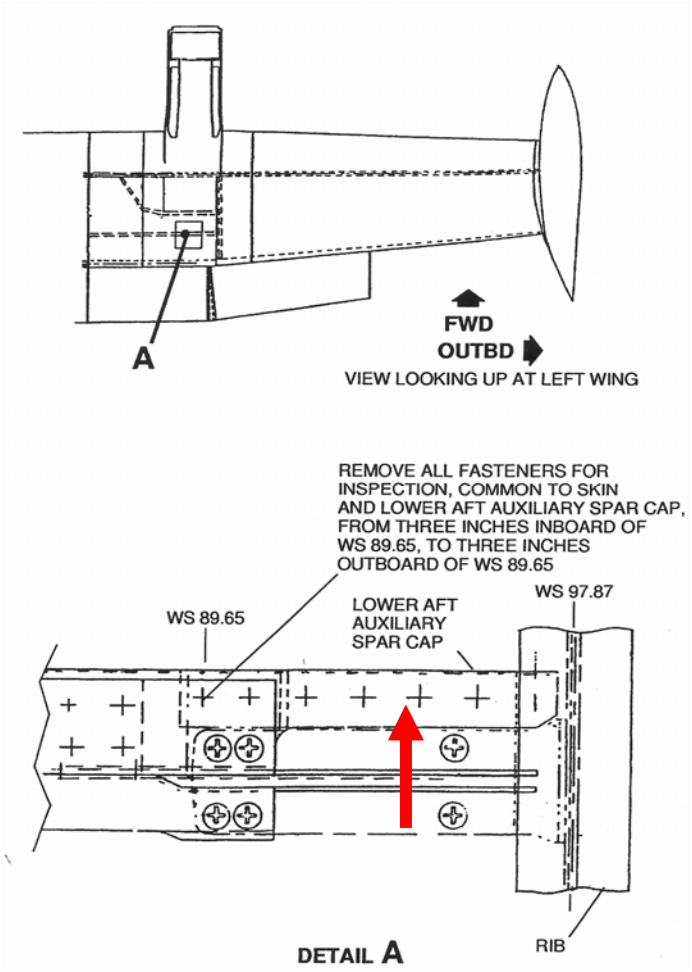

Figure 2-25. Location of Skin Crack on the Left Wing Aft Auxiliary Spar Lower Cap

Figure 2-26. Skin Crack on Left Wing Aft Auxiliary Spar Lower Cap

3. TEARDOWN EXAMINATION PHASE.

Following the completion of the inspection phase, the teardown examination phase was performed to find defects not found by typical maintenance or supplemental inspections. Every defect found in the inspection phase and the teardown examination phase was completely characterized during the microscopic examination portion of this project. The airplane was first disassembled into major airplane sections, including the wings, forward fuselage, aft fuselage, cabin, landing gear, horizontal stabilizer, and vertical stabilizer. The systems' components were then inspected after removal from the airplane. Alternative NDI techniques were used to examine the left wing front spar and the horizontal stabilizer front spar to find additional defects in the airframe prior to disassembly. Following the alternative NDI, 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. Certain cracks and areas of corrosion were sectioned and mounted for fractographic analysis to further determine the extent of the flaw and the mode of failure for cracks (i.e., fatigue, stress corrosion cracking, etc.).

3.1 DISASSEMBLY OF AIRFRAME AND MAJOR AIRPLANE SECTIONS.

To facilitate the alternative NDI and detailed disassembly, the airplane was disassembled per the Cessna 402A Maintenance Manual [1] 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 into sections, the engines were removed and were not investigated as part of this program. The systems' components were also removed for inspection. All disassembly was done as carefully as possible to minimize damage to the wiring, systems components, and the underlying structure of the airplane.

3.2 INSPECTION OF AIRPLANE SYSTEMS' COMPONENTS.

As airplanes age, degradation of mechanical systems may occur, and this degradation must be considered when designing airplane components. All mechanical system parts on the Cessna 402A airplane were visually inspected for signs of degradation during the disassembly. These mechanical systems included flight control cables and linkages, landing gear actuators, combustion heater, flap actuators, and pressurized lines. The only discrepancy noted was the broken gear teeth on the left fuel selector valve.

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 pressurization system for pressurizing the lines in a consistent and reproducible manner. Figure 3-1 illustrates the pressurization system.

Figure 3-1. Apparatus Used to Pressurize Lines

This pressurization system takes air at an unknown pressure from a standard pressure line and regulates it to a known pressure. This regulated pressure is then sent into the pressure line being tested. The pressurization system consists of the following:

- Pressure Control Apparatus
  - Universal quick couplers are used to carry pressurized air into and out of the regulator.
  - The pressure regulator is model number 4ZM08 built by Speedaire. The maximum inlet pressure is 300 psi, while maximum outlet pressure is 125 psi.
  - A pressure gauge is attached to the front of the regulator. The pressure is varied by adjusting the knob at the top of the regulator until the desired pressure is shown on the gauge. Pressures are always approached from the negative direction.
  - A 4-inch metal nipple takes the air out of the universal quick coupler and into the air chuck.
  - All threaded connections are wrapped with Teflon tape before connection.

- Tire Stem
  - Tire stems are used to carry the pressurized air from the air chuck into the tubing.
  - The tire stems have a mushroom-shaped rubber outlet that can form an airtight seal over a variety of tubing sizes.
  - For very small tubing, a cork is inserted into the end of the tire stem. This cork has a small hole drilled into it, which effectively reduces the exit area of the stem and allows a seal to be formed with smaller tubing.
- Corks
  - Corks are inserted in the opposite end of the tubing to seal the system.

The process for testing the lines is described below.

- Debris or fittings that could obscure or block leaks are removed.
- The part number is recorded in the test log, and a visual inspection is performed.
- A cork was firmly placed in one end of the tube, as illustrated in figure 3-2.
- On the other end of the tube, the pressurization apparatus is attached, as shown in figure 3-3. The regulated pressure was 20 psi ±1.
- The entire tube, including the tip connected to the pressurization apparatus, is immersed under water in the dip tank.
- The tube is pressurized for 10 seconds. Air leakage around the tire stem is minimized to reduce disturbances in the water. If air leakage around the stem could not be eliminated, the tire stem is held above the water. After the tube is pressurized one time, the position of the cork and the stem is switched and the test repeated.
- Two inspectors perform the test. One person controls the pressurization system while the other watches for bubbles emitting from the tubing. Any areas of interest are noted in the test log. If small bubbles exists on the surface of the tube, these are brushed off to minimize false indications.

Figure 3-2. Cork Placed in the End of a Tube

Figure 3-3. Pressurization of Tubing

Of the 74 pressurized lines inspected, five were found to have holes induced by corrosion, three had leaks around fittings and couplers, and one had a leak from a recent machining mark. Of the tubing with corrosion, three were oxygen lines and two were unknown. It is possible that the recent machine mark and the leaks around the couplers and fittings were due to disassembly work performed on the airplane and not to the actual flight environment. Table 3-1 summarizes the findings of the leak test due to corrosion or recent machining marks. The damage found during this investigation is shown in figures 3-4 through 3-14.

Table 3-1. Inspection Results of Pressurized Lines

Part Number	Visual	Leak	Description
5200106-49	2 Holes	Yes	Corrosion-induced leak
5200106-19	1 Hole	Yes	Machining mark
5200166-21	1 Hole	Yes	Corrosion-induced leak
Unidentified 1	1 Hole	Yes	Corrosion-induced leak
Unidentified 2	1 Hole	Yes	Corrosion-induced leak

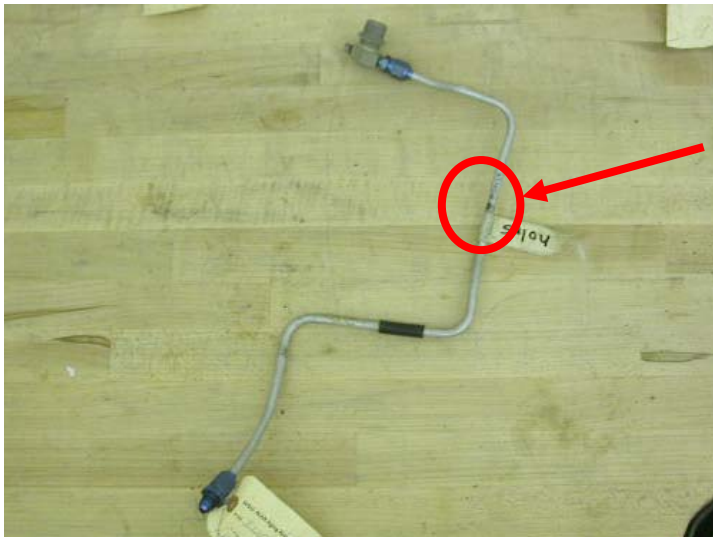

Figure 3-4. Location of Holes in P/N 5200106-49

Figure 3-5. Hole 1 in P/N 5200106-49

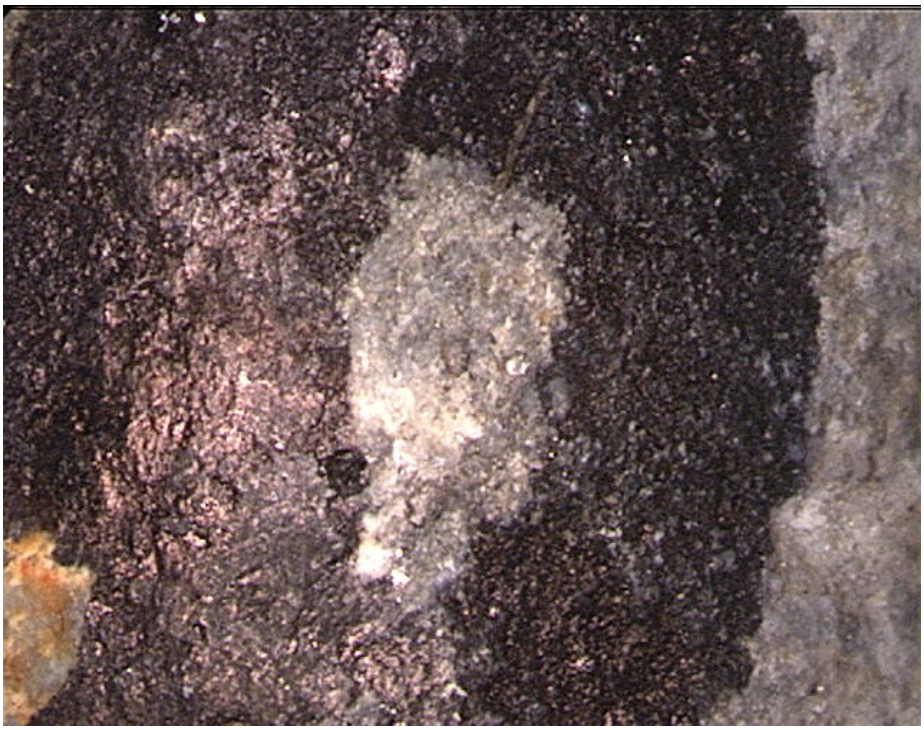

Figure 3-6. Hole 2 in P/N 5200106-49

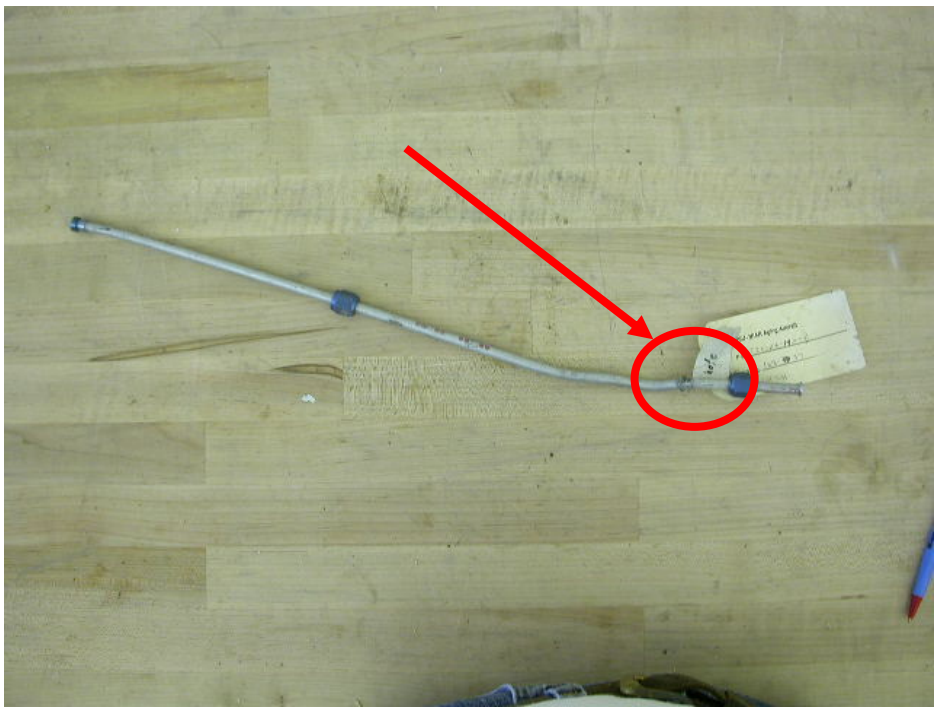

Figure 3-7. Location of Hole in P/N 5200106-19

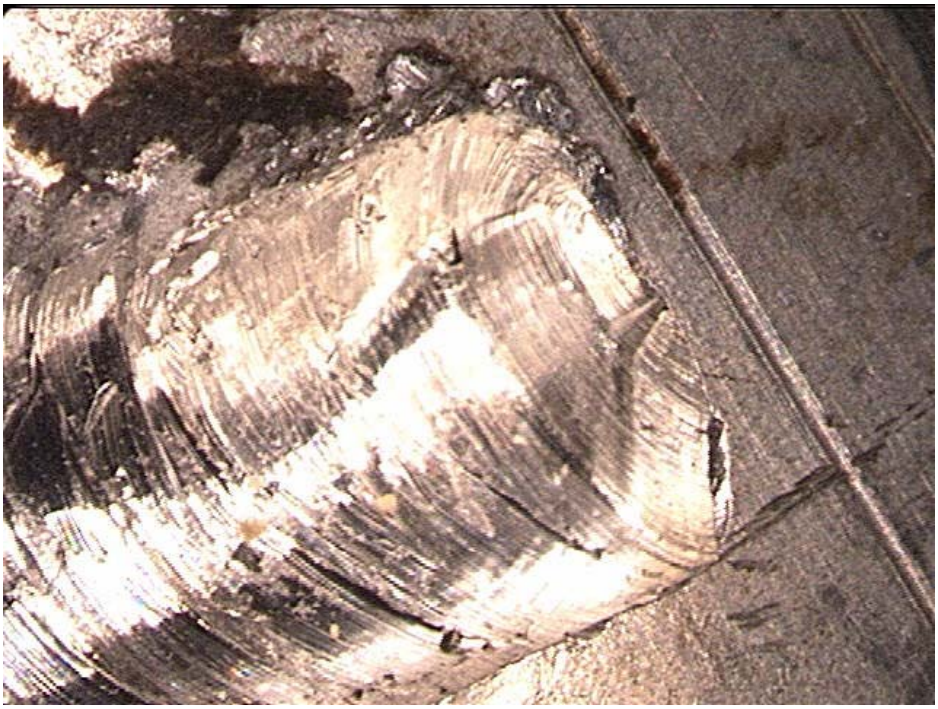

Figure 3-8. Hole in P/N 5200106-19

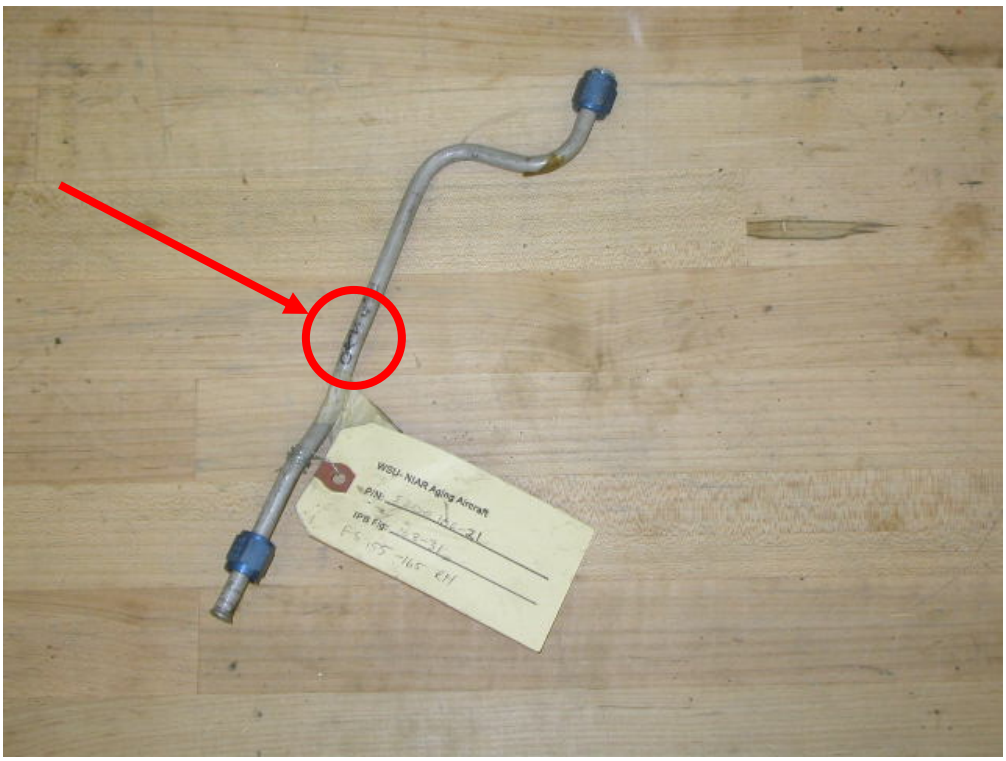

Figure 3-9. Location of Hole in P/N 5200166-21

Figure 3-10. Hole in P/N 5200166-21

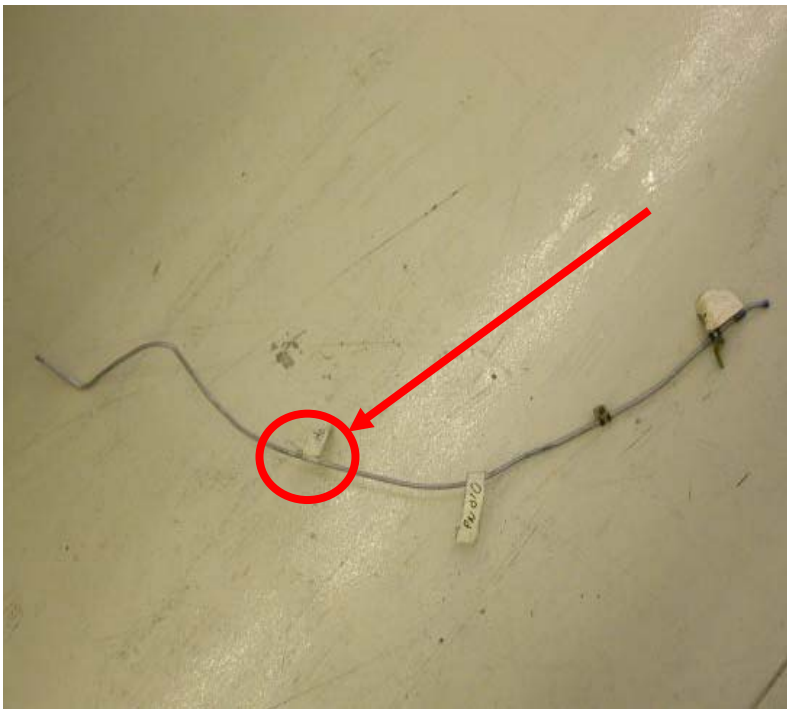

Figure 3-11. Location of Hole in Unidentified Line 1

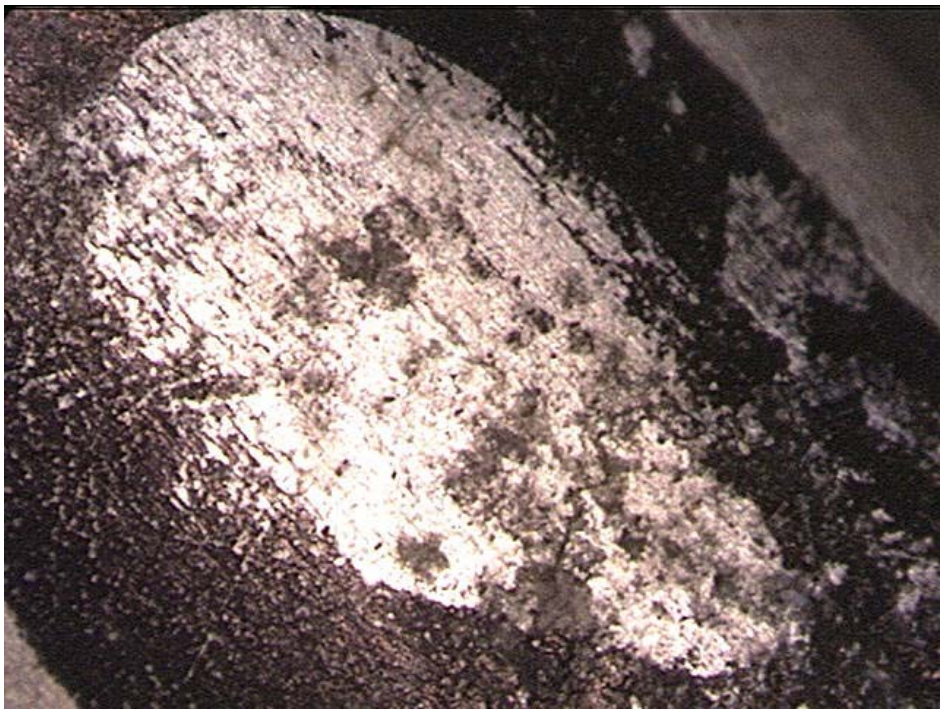

Figure 3-12. Hole in Unidentified Line 1

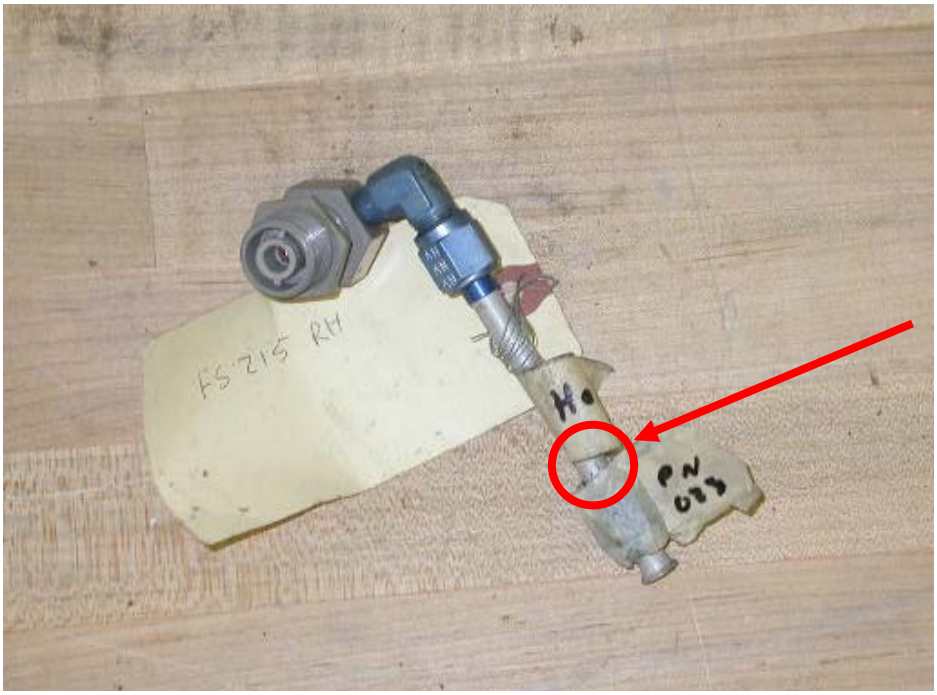

Figure 3-13. Location of Hole in Unidentified Line 2

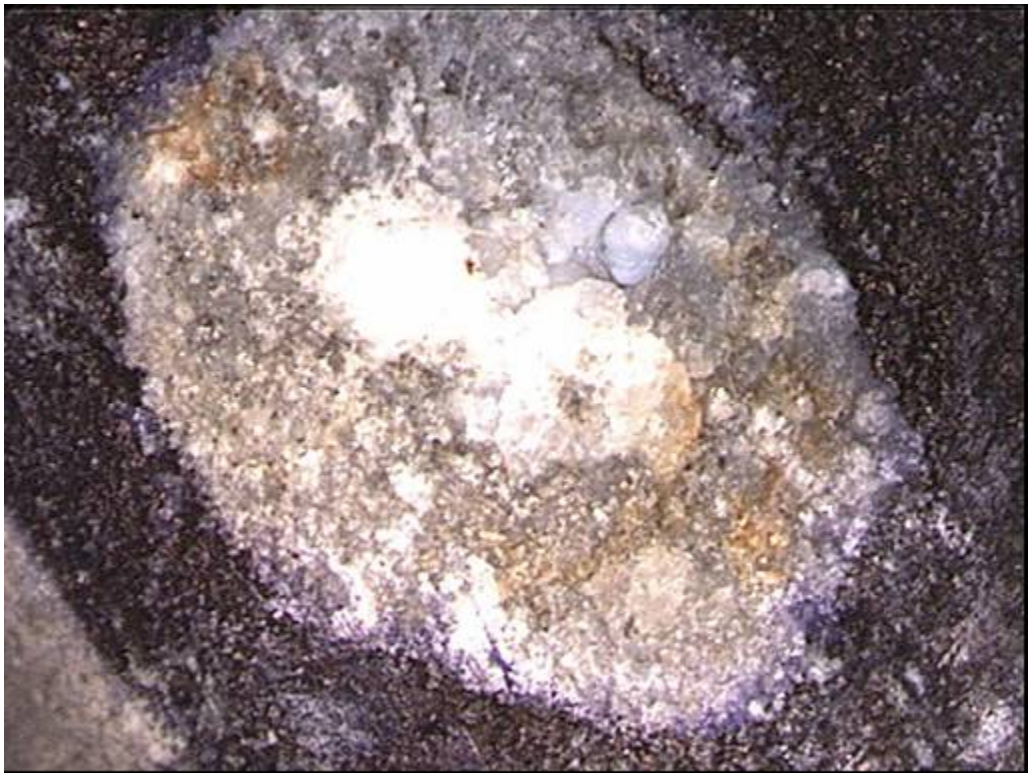

Figure 3-14. Hole in Unidentified Line 2

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 supplemental inspections prior to teardown. The front spars of the left wing and horizontal stabilizer were inspected using three eddy-current-based techniques capable of detecting surface and subsurface cracks and areas of corrosion. The Magneto-Optic Imager (MOI), demonstrated in figure 3-15, is capable of detecting cracks in multiple layers, as well as surface and interlayer corrosion. The sliding probe, shown in figure 3-16, detects cracks in up to three layers of metal, while the spot probe, as shown in figure 3-17, can detect corrosion in multiple layers.

The 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 Cessna 402A SID; therefore, no procedures were established or validated for using these techniques on the Cessna 402A. Using existing structure and locally manufactured calibration standards, 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 MOI, sliding probe, or spot probe should not be made from the results presented in this report.

Figure 3-15. Demonstration of the MOI Instrument

Figure 3-16. Demonstration of the 1- to 100-kHz Sliding Probe

Figure 3-17. Demonstration of the Spot Probe

Initially, the left wing and horizontal stabilizer front spars were inspected using the MOI, sliding probe, and spot probe without any attempt to quantify the indications detected. Baseline signals were determined for each of the different material stackups along both spars. Deviations from these baseline signals were reported as cracks or corrosion. In the attempt to determine the location of flaws on the front spars of the horizontal stabilizer and the left wing, two inspectors conducted the inspections independently using each method. The only information provided to the inspectors was the area to be inspected, the method of inspection, and the appropriate instrument settings. Both inspectors were familiar with, and had prior experience, performing the three inspection methods. To maintain uniformity in the inspections, machine settings were provided, since the quality of eddy-current-based inspections are highly dependent on frequency and gain. Table 3-2 shows the machine settings used for these inspections.

Table 3-2. Nondestructive Inspection Machine Settings

Method	Frequency	Horizontal Gain	Vertical Gain
MOI	5 kHz	–	–
Sliding probe	3 kHz	70	75
Spot probe	3 kHz, 1.5 kHz	–	–

The results for the horizontal stabilizer, which are listed in table 3-3, indicate that no crack indications were detected using the MOI or sliding probe. The indications on the horizontal stabilizer were all found by the spot probe, indicating corrosion. The locations of these indications are shown in figures 3-18 and 3-19 by body line (BL) location.

Table 3-3. Nondestructive Inspection on the Horizontal Stabilizer

Method	Location	Indications
Spot probe	Right front spar upper cap	BL 24-29
Spot probe	Left front spar upper cap	BL 18-20, 22-25, 30-32
Spot probe	Left front spar lower cap	BL 38-41, 57-58
Spot probe	Right front spar lower cap	BL 6, 38-40

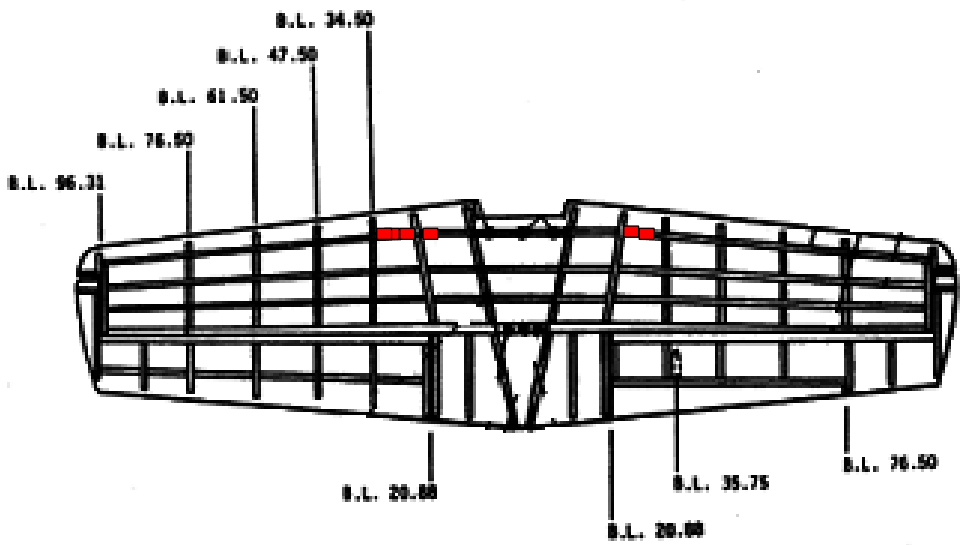

Figure 3-18. Location of Alternative NDI Indications on the Upper Surface of the Horizontal Stabilizer

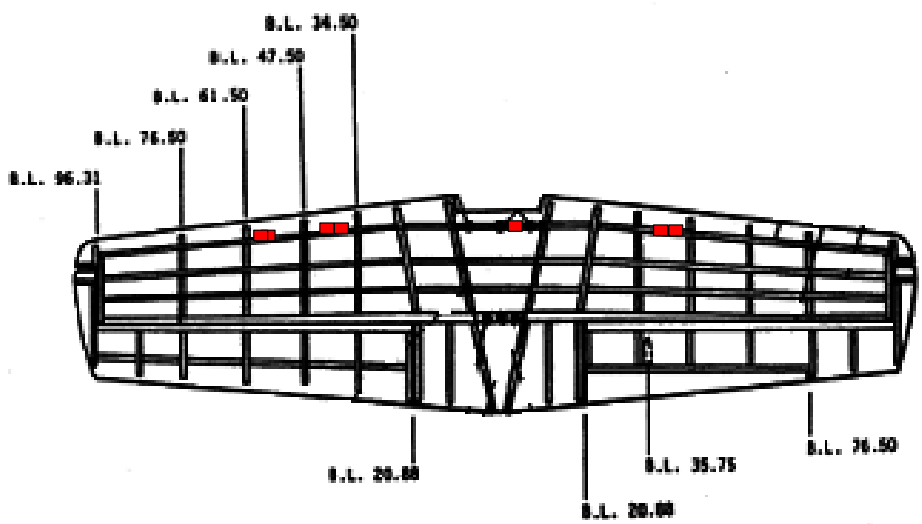

Figure 3-19. Location of Alternative NDI Indications on the Lower Surface of the Horizontal Stabilizer

The inspection results for the left wing, which are listed in table 3-4, showed a large number of crack indications on the front spar upper and lower caps. All indications were found inboard of the outboard engine beams, as shown in figures 3-20 and 3-21, by WS location. These indications were verified visually to determine if the indications were cracks, large areas of corrosion, or inaccurate signals. With eddy-current techniques, large areas of corrosion often mislead inspectors to make a call of a crack instead of corrosion.

Table 3-4. Nondestructive Inspection on the Left Wing

Method	Location	Indications
MOI	Front spar lower cap	WS 78, 79, 81, 94
Sliding probe	Front spar lower cap	WS 50-55, 57-60, 69, 72-80, 81, 87-90, 94
Sliding probe	Front spar upper cap	WS 29-33, 51-59, 69, 73-80, 84-88

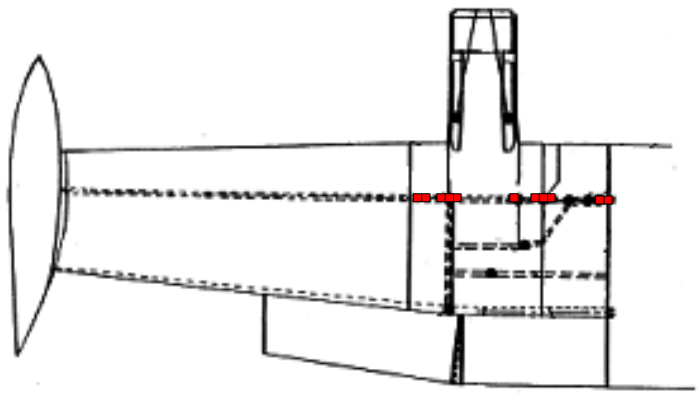

Figure 3-20. Location of Alternative NDI Indications on the Upper Surface of the Left Wing

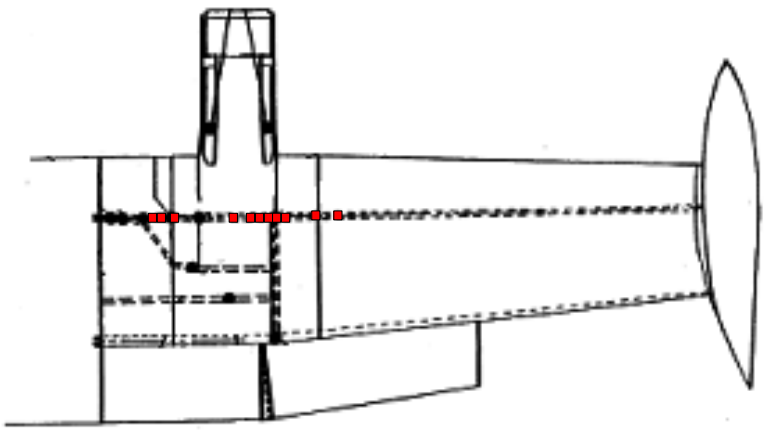

Figure 3-21. Location of Alternative NDI Indications on the Lower Surface of the Left Wing

Since indications vary with material composition and specimen thickness, standards must be created to characterize flaws found in a given structure and to determine the types and extent of flaws that could be found with each technique. Therefore, test specimens were developed for the front spars of the wing and horizontal stabilizer to be used as standards for providing a fundamental understanding of the types and extent of flaws that could be found using the baseline signal inspection methodology prior to disassembly on this specific structure. The standards were built out of actual Cessna 402A structure to completely and accurately represent the front spars of the wing and horizontal stabilizer. Standards were made from the front spar of a horizontal stabilizer salvaged from another Cessna 402A airplane and from the right wing of the 402A airplane used in the teardown evaluation project. Due to the inaccuracy and difficulty in modeling corrosion, no standards were created to quantify indications found with the spot probe. Sections of the spars and skins were removed, and then 0.004-inch-wide electron discharge machining (EDM) notches were placed in the spars to represent cracks. Six different lengths were introduced in each spar ranging in length from 0.050 to 0.250 inch. Two notches of each length were cut into each spar. The skin was then reattached to the spar to form the test specimen. Flush rivets were used on the front spar of the horizontal stabilizer, and the rivets on the wing were an equal mix of flush and button-head fasteners.

The presence of button-head fasteners created additional problems for the inspection team since the MOI and sliding probe have been designed only for flush rivets. Modifications were made to the sliding probe to account for the raised fastener heads. Skis were used to create enough clearance for the raised fastener head, as shown in figure 3-22. The altered sliding probe was only used for the raised fasteners, while the sliding probe with no modifications was used for the flush rivets. No such modifications could be made to the MOI so it was used on both flush and button-head fasteners in its original stock configuration. Ongoing research is investigating the applicability of the MOI in inspecting button-head fasteners without modification. Although this controversy exists, the MOI was used without modification to inspect button-head fasteners in this study.

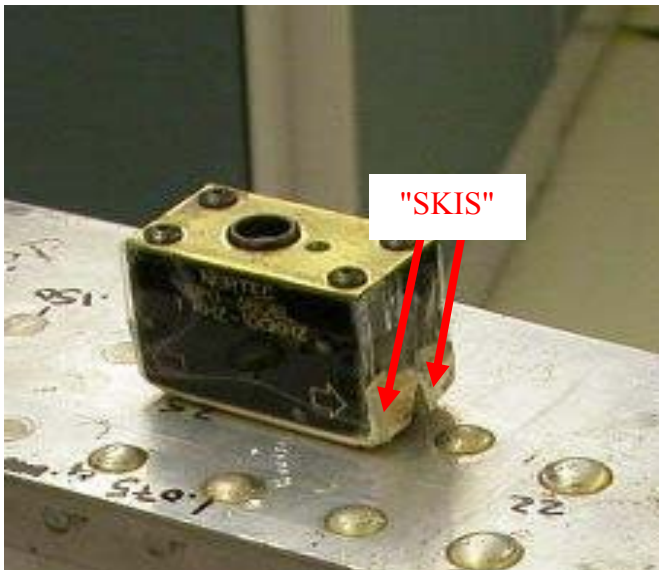

Figure 3-22. Modified Sliding Probe

The wing standard is comprised of a 0.177-inch-thick spar cap and a 0.040-inch layer of skin. The upper and lower surfaces of the standard consisted of two rows of 17 fasteners, with half flush rivets and half button-head rivets, as shown in figure 3-23. Since there was only one layer of skin over the spar, the inspection was a second-layer inspection penetrating through 0.040 inch of aluminum.

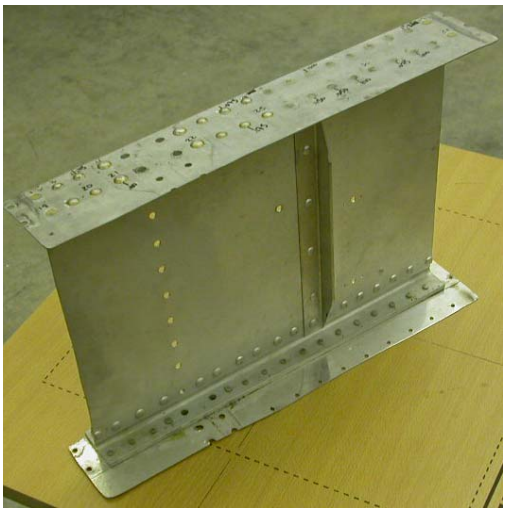

Figure 3-23. Wing Standard

The horizontal stabilizer standard is comprised of a 0.195-inch-thick upper spar cap and two layers of skin, as shown in figure 3-24. The upper layer is 0.030 inch thick, while the lower skin is 0.040 inch thick. The upper and lower surfaces of the standard consist of one row of 26 flush rivets. Six different lengths of EDM notches were cut in the upper and lower surfaces of the standard. Two notches of each length were cut on each surface of the standard; therefore, 12 fastener locations were cracked on each of the top and bottom surfaces. Since two layers of skin cover the spar, the inspection was a third-layer inspection through 0.070 inch of aluminum skin.

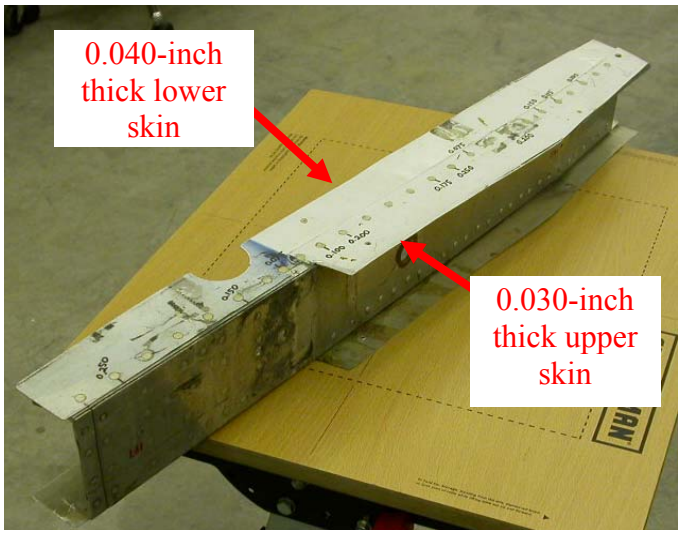

Figure 3-24. Horizontal Stabilizer Standard

Due to the geometry of the parts and the method of EDM notching used, the notches placed in the lower surfaces of both standards were not emanating from the center of the fastener hole. To avoid investigating the effects of off-centered cracks, the inspection results for the lower surface were not presented in this report.

The standards were then given to two inspectors with instructions to determine which holes were cracked using the sliding probe and MOI. The inspectors established baseline responses. Deviations from these responses were noted as indications of cracks. For each standard, the inspector had two opportunities with each method to detect each crack length, which meant that there were four opportunities to detect each crack length with both methods for each standard. The results of the inspections are shown in figures 3-25 and 3-26.

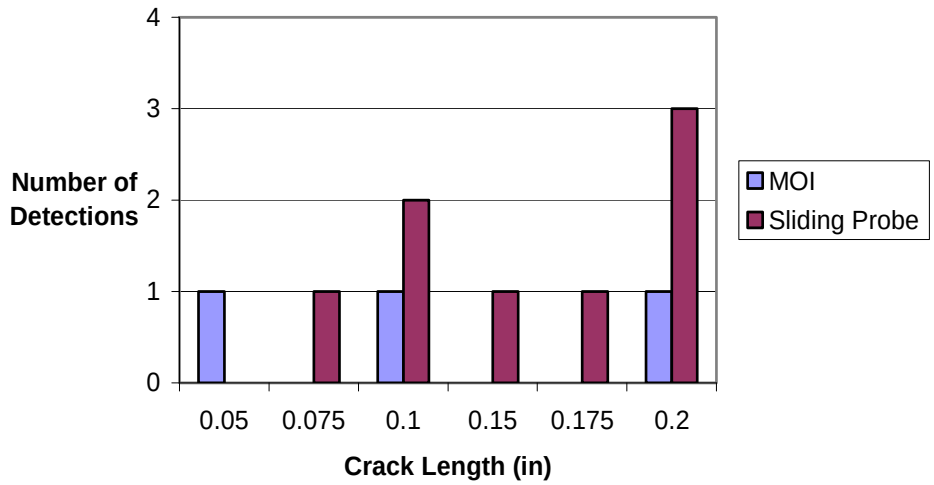

Figure 3-25. Number of Detections for the Wing Based on the Standard Shown in Figure 3-23

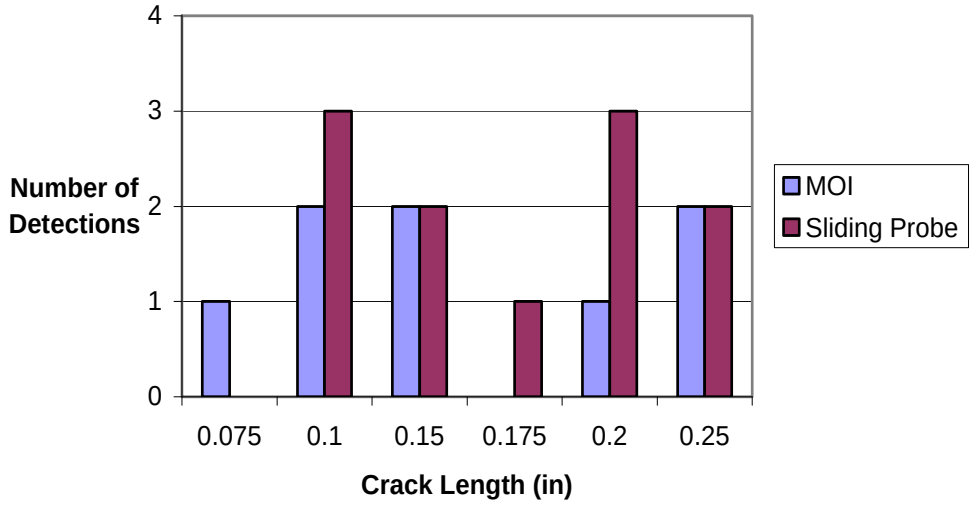

Figure 3-26. Number of Detections for the Horizontal Stabilizer Based on the Standard Shown in Figure 3-24

After the initial inspection of the left wing and horizontal stabilizer front spars using the MOI and sliding probe, all indications were verified using the fluorescent liquid penetrant method. Following the subsurface inspection, the spars were extracted from the surrounding structure, paint stripped, and etched to enhance damage detection. Following the etching process, the spars were again inspected using the Type I, Method A water-soluble fluorescent liquid penetrant method. This inspection method is capable of efficiently inspecting the entire part without damaging it. In this procedure, the part is precleaned, sprayed with fluorescent liquid penetrant, cleaned with water to remove excess penetrant, dried, developed with powder, and then visually inspected under a black light, as shown in figure 3-27. Following the fluorescent liquid penetrant inspection, the spars were examined microscopically to verify NDI indications and to characterize any damage found.

Figure 3-27. Fluorescent Penetrant Inspection

3.4 DETAILED DISASSEMBLY.

Following the completion of the alternative NDI, the major airplane sections were completely disassembled to access suspect areas and critical structural details. Suspect areas were defined as areas with any NDI indications or areas with noticeable damage found during the visual inspection or disassembly. Critical structural details were identified analytically and experimentally by Cessna in conjunction with the SID program. These suspect areas and critical structural details were the focus of the microscopic examination. Table 3-5 shows the correlation between critical structural details and SID locations, identifying the critical areas that were inspected as part of the SID. It should be noted that the SID identified additional inspections for areas other than the critical areas listed in table 3-5. Thirteen critical structural details, shown in figure 3-28, were determined for the wings and engine beams. Seven details were identified on the front spar, two on the auxiliary spars, one on the rear spar, one on the engine beam, and two on the stub wing forward and aft spar attachment fittings. Two critical details were identified for the fuselage. These details, shown in figure 3-28, were the left-hand longeron and the tailcone angle attachment to the horizontal stabilizer rear spar. Seven critical details were identified for the horizontal stabilizer, as shown in the figure. Three of these critical details were located on the front spar, while three were located on the rear spar. The remaining detail was located on the aft auxiliary spar upper cap. Figure 3-28 also shows the location of two

critical details on the vertical stabilizer, which are the vertical stabilizer rear spar attachment and the rear spar cap. The main landing gear side brace actuator collar and nose landing gear fork were also identified by Cessna as critical structural details. Damage indications from the supplemental inspections were found on the following critical structural details:

- SID #32-50-00: Nose Gear Steering Bellcrank
- SID #57-10-10: Left Stub Wing Forward Upper Attach Fitting
- SID #57-10-11: Left Wing Front Spar Lower Rear Attach Fitting
- SID #57-10-11: Left Wing Front Spar Lower Front Attach Fitting
- SID #57-10-12: Left Wing Front Spar Upper Rear Attach Fitting
- SID #57-10-12: Right Wing Front Spar Upper Rear Attach Fitting
- SID #57-10-24: Right Wing Tip Tank Fitting

Table 3-5. Correlation Between Critical Details and SID Locations

ID	Critical Area Description	Critical Areas Location	SID Number/Title
BW-1	Wing carry-through forward spar lower cap	BL 36.12	57-10-10/front carry-through spar lower cap
BW-2	Wing front spar lower cap root fitting	WS 46.70	57-10-11/wing front lower root fitting attach
BW-3	Wing front spar lower cap root fitting attach	WS 54.10	57-10-12/wing front spar lower cap at root fitting attach
BW-4	Wing front spar lower cap canted rib attach	WS 66.70	No SID inspection
BW-5	Wing front spar lower cap	WS 75.66	57-10-08/wing front spar lower cap inspection and modification
BW-6	Wing front spar lower cap at inboard engine beam attach	WS 83.74	54-10-04 and -05/engine beams and modifications
BW-7	Wing front spar lower cap at outboard engine beam attach	WS 98.74	54-10-04 and -05/engine beams and modifications
BW-8	Wing forward auxiliary spar lower cap	WS 86.62	57-10-13/forward auxiliary spar lower cap
BW-9	Wing aft auxiliary spar lower cap	WS 89.65	57-10-09/aft auxiliary spar lower cap
BW-10	Wing carry-through aft spar lower cap	BL 37.60	57-10-07/aft carry-through spar lower cap
BW-11	Wing rear spar cap at splice	WS 97.87	57-10-06/wing rear spar lower cap at splice
BW-12	Wing front spar upper cap	WS 106.82	No SID inspection
BEB-1	Engine beam at aft engine mount	FS 131.20	54-10-04/model 401-402 engine beams
BF-1	Fuselage left-hand longeron	FS 190.33	53-10-02/fuselage left- and right-hand channel assemblies

Table 3-5. Correlation Between Critical Details and SID Locations (Continued)

ID	Critical Area Description	Critical Areas Location	SID Number/Title
BF-2	Tailcone angle attach to horizontal rear spar	BL 2.90	53-10-03/horizontal stabilizer rear spar angle attachment
BH-1	Horizontal stabilizer front spar upper cap	BL 0.0	55-10-04/horizontal stabilizer front spar upper cap
BH-2	Horizontal stabilizer front spar lower cap	BL 0.0	55-10-05/horizontal stabilizer front spar lower cap
BH-3	Horizontal stabilizer front spar attach bolt through web	BL 7.69	55-10-06/horizontal stabilizer front spar attach
BH-4	Horizontal stabilizer rear spar lower cap at attach bolt	BL 2.90	55-10-07/horizontal stabilizer rear spar lower attach
BH-5	Horizontal stabilizer rear spar upper cap	BL 0.0	55-10-08/Horizontal Stabilizer Rear Spar Upper Cap
BH-6	Horizontal stabilizer rear spar lower cap	BL 0.0	55-10-09/horizontal stabilizer rear spar lower cap
BH-7	Horizontal stabilizer rear auxiliary spar upper cap	BL 8.01	No SID Inspection
BV-1	Vertical stabilizer rear spar at attach	WL 108.38	55-30-04/vertical stabilizer rear spar cap at attach
BV-2	Vertical stabilizer rear spar cap	WL 136.04	No SID inspection
BMLG-1	Main landing gear side brace actuator collar	Not applicable	32-10-04/main gear actuator collar
BNLG-1	Nose landing gear fork	Not applicable	32-20-02/nose gear fork

FS = Frame station

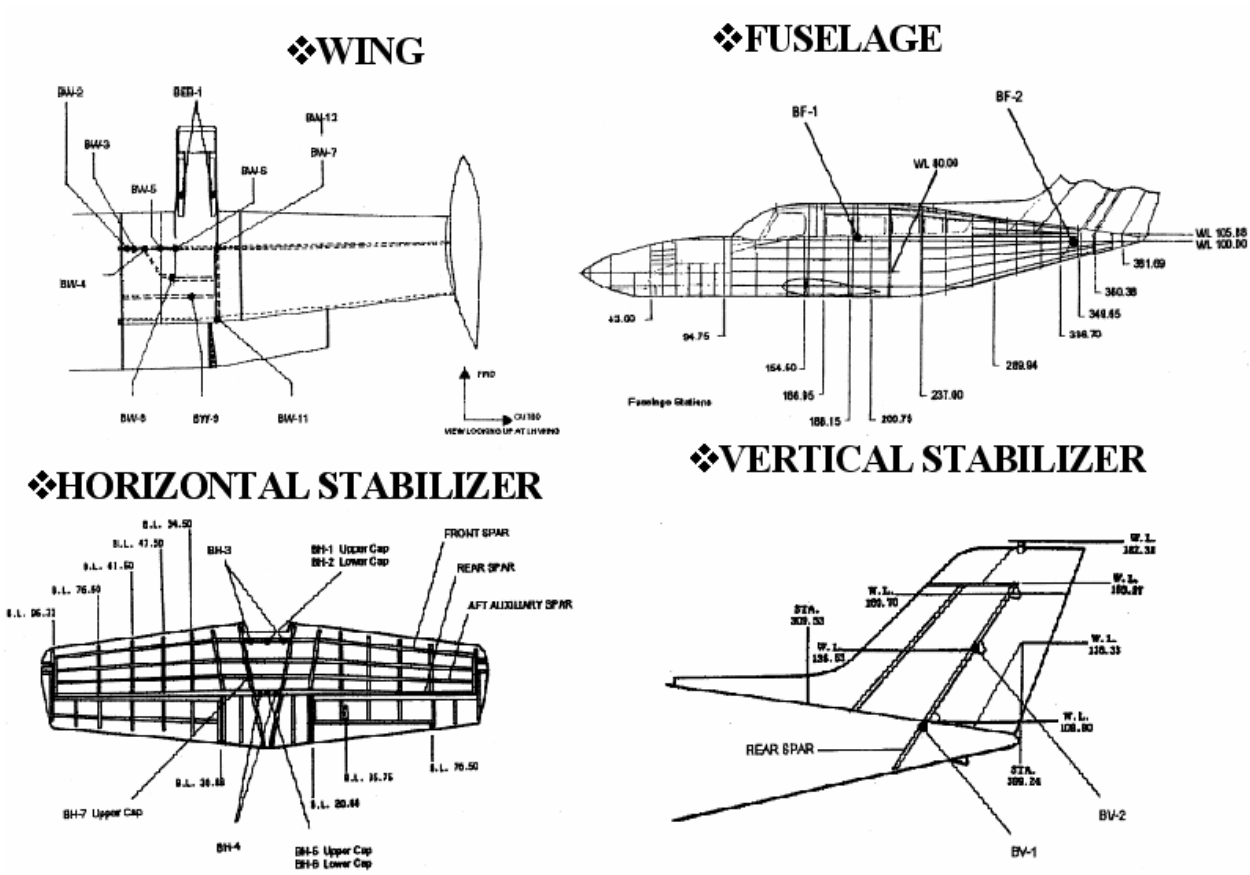

Figure 3-28. Critical Structural Details

3.5 MICROSCOPIC EXAMINATION OF CRITICAL STRUCTURAL AREAS.

Suspect areas from the visual inspections, supplemental inspections, alternative NDI, defects found during disassembly, and critical structural details were examined microscopically to locate cracks and areas of corrosion. A low-power (10 times) magnifying glass and a 7-45-power optical microscope, as shown in figures 3-29 and 3-30, respectively, were used for the examination. All fastener holes were inspected for cracks greater than 0.01-inch, and general part geometry and defect location were noted. Corrosion area and severity were documented by two methods: the maximum pit depth was noted and the percentage of thickness lost due to corrosion was used to compare the NDI results to the microscopic results and to classify the corrosion 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

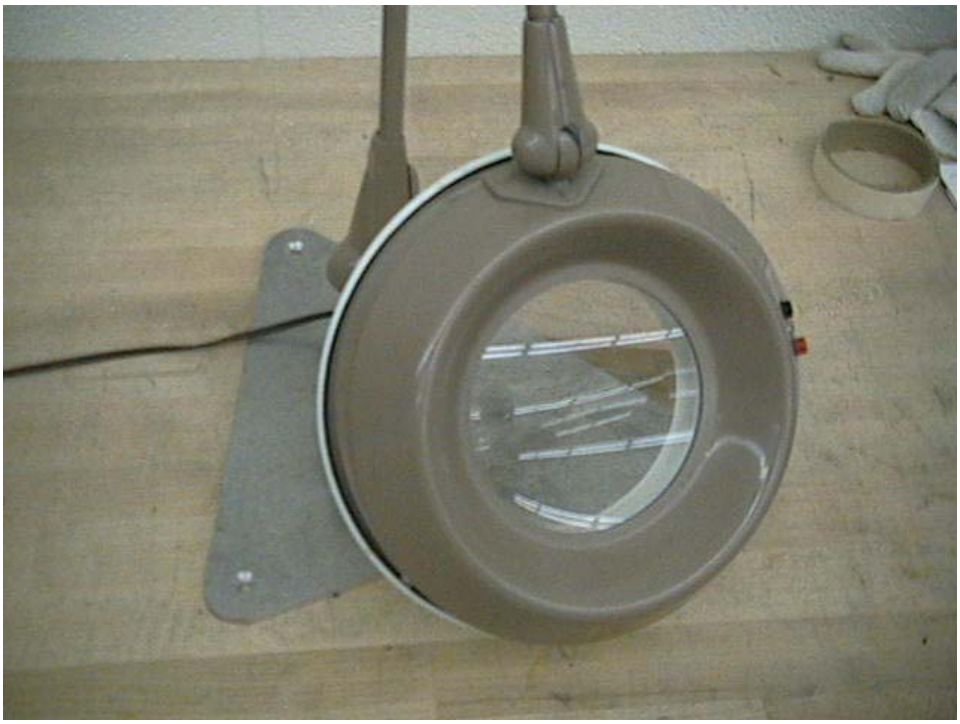

Figure 3-29. A 10-Power Magnifying Glass

Figure 3-30. Microscopic Examination of Parts Using the 7-45-Power Optical Microscope

Corrosion effects were classified by maximum percentage of thickness lost, as it seemed most representative of the actual damage done by the corrosion. In the past, the corrosion was classified by maximum pit depth; however, for the purpose of this study, pit depth would not be 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, but are recommended for usage in the future. It is important to note that corrosion is not assumed to be uniform throughout the entire area, but that it represents the maximum depth of corrosion in that area.

Cracks and areas of corrosion were sectioned out for further investigation. Corrosion depth was quantified using a digital optical micrometer as well as a pin gage, as illustrated in figures 3-31 and 3-32, respectively. Select cracks were broken open in a manner to preserve fracture face detail, and specimens were mounted for further investigation, as shown in figure 3-33. A scanning electron microscope was used to view fracture face features and determine crack growth mode.

Figure 3-31. Corrosion Depth Assessment With Digital Optical Micrometer

Figure 3-32. Pin Gage for Corrosion Depth Assessment

Figure 3-33. Specimens Mounted for Further Investigation

The following sections describe in detail the results of the microscopic examinations for the SID areas, the areas inspected with alternative NDI techniques, defects found during disassembly, and critical structural areas. A summary of all microscopic findings and a comparison between NDI indications and microscopic findings are provided in sections 3.5.6 and 3.5.7.

3.5.1 Microscopic Examination of Supplemental Inspection Areas.

All defect areas found during the SID inspections were examined microscopically. Table 3-6 lists all the areas found during the SID inspections and the corresponding findings of the microscopic examination of these areas. Detailed descriptions of the examinations conducted on the stub wing attach fittings, wing attach fittings, right wing tip tank fittings, and nose gear steering bellcrank are described in the following sections.

Table 3-6. Supplemental Inspection Document Inspection Microscopic Results

Method	Indication	Location	Microscopic
Bolthole eddy current	Crack	Left stub wing upper forward attach fitting SID #57-10-10	0.24″ crack
Bolthole eddy current	Cracks	Right wing tip tank fittings SID #57-10-24	Four cracks 0.045″-1.25″
Liquid penetrant	Cracks	Nose gear steering bellcrank SID #32-50-00	Eight cracks 0.039″-0.093″

3.5.1.1 Wing Attach Fittings.

All the wing attachment fittings were inspected microscopically. Corrosion was found on both front spar lower attachment fittings as well as one of the front spar upper attach fittings on the left wing. The right wing front spar upper attachment fitting also was found to be corroded. Figures 3-36 and 3-37 show the locations of the wing front and rear attach fittings.

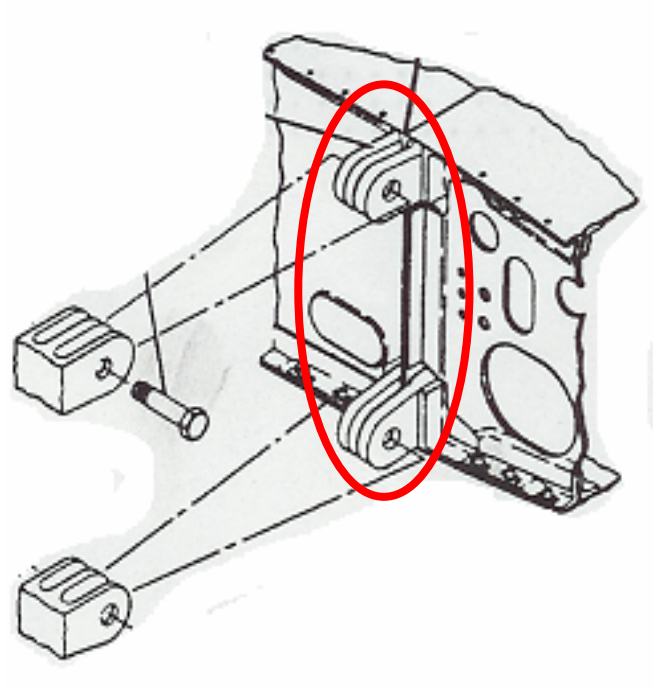

Figure 3-34. Location of Wing Front Spar Attach Fittings

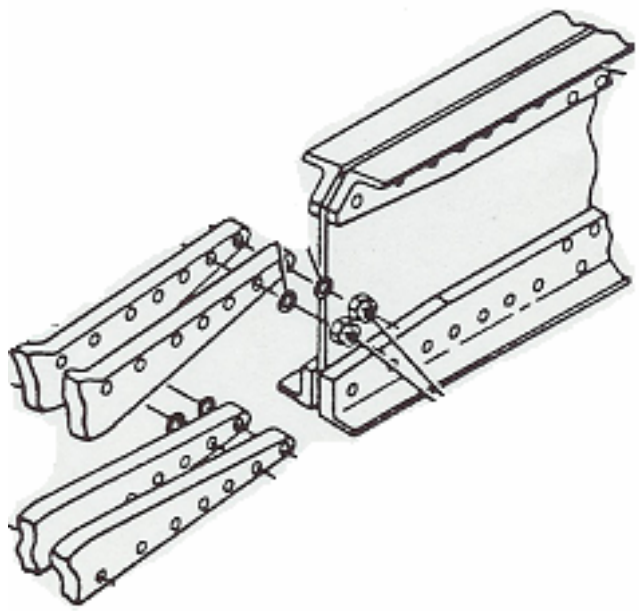

Figure 3-35. Location of Wing Rear Spar Attach Fittings

3.5.1.1.1 Left Wing Front Attach Fittings.

Four front spar root fittings were removed from the left wing, two front and two rear, for examination. An example of the front attach fitting is shown in figure 3-36. Three of the four fittings were found to have areas of corrosion. Light corrosion was detected on the lower rear attach fitting, shown in figure 3-37. The corroded area of this fitting (P/N 0822550-17), encompasses 1.2 square inches. With a maximum pit depth of 0.002 inch and a thickness of 0.204 inch, the total percent thickness loss was 1.0%. One area of moderate-severe corrosion was detected on the lower front attach fitting (P/N 0822550-9), as shown in figure 3-38. The area of the corrosion shown is 0.1875 square inch with a maximum pit depth of 0.015 inch and a thickness of 0.204 inch. The reduction in component thickness due to corrosion was 7.4%. Light-moderate corrosion on the rear spar upper attach fitting (P/N 0822550-7) is shown in figure 3-39. The location of this corrosion was 0.2 inch from the edge of fitting. The corrosion encompasses an area of 0.004 square inch and had a maximum pit depth of 0.006 inch at a thickness of 0.204 inch, which resulted in a 2.9% thickness loss due to corrosion.

Figure 3-36. Front Attach Fitting

Figure 3-37. Corrosion on the Left Wing Front Spar Lower Rear Attach Fitting

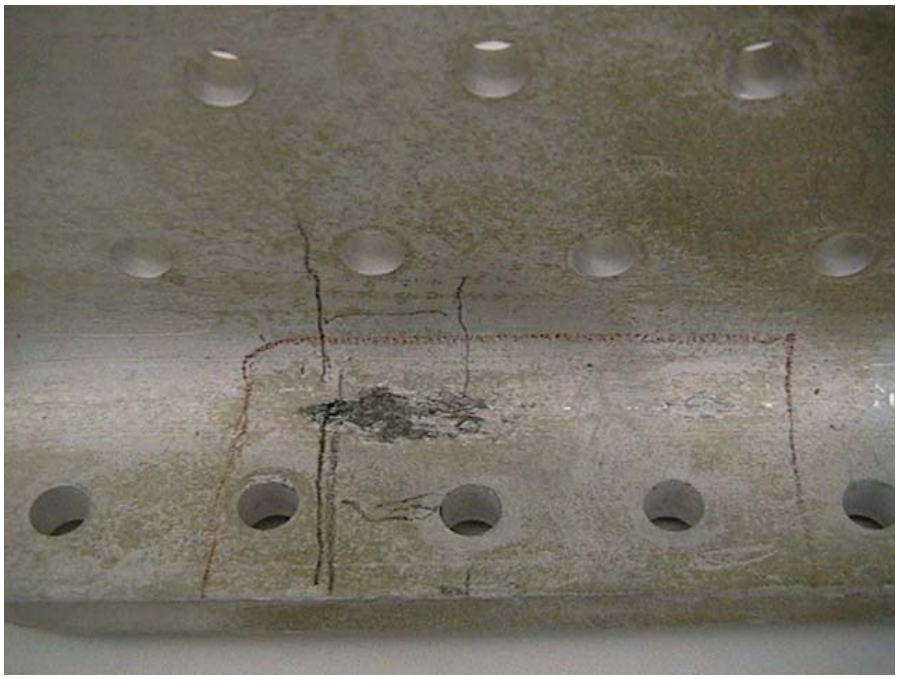

Figure 3-38. Corrosion on the Left Wing Front Spar Lower Front Attach Fitting

Figure 3-39. Corrosion on the Left Wing Front Spar Upper Rear Attach Fitting

3.5.1.1.2 Right Wing Front Attach Fittings.

Four front spar root fittings were removed from the right wing, two forward and two aft, for examination. Figure 3-40 shows the upper rear fitting (P/N 0822550-8) was the only attachment fitting found to have corrosion on the right wing. The corrosion on this fitting was classified as light-moderate. The corroded area of this fitting encompasses 0.25 square inch. The area of corrosion had a maximum pit depth of 0.0045 inch and resulted in a 2.3% reduction in thickness due to the corrosion.

Figure 3-40. Corrosion on the Right Wing Front Spar Upper Rear Attach Fitting

3.5.1.2 Stub Wing Attach Fittings.

All attach fittings on the stub wings were examined microscopically. One crack was found on the left stub wing upper forward attachment fitting. Figures 3-41 and 3-42 show the locations of the stub wing forward and aft attach fittings, respectively. During removal of the aft stub wing fittings, damage to the fitting occurred. Due to limited access to remove bolts common to these fittings, punch marks were induced in the boltholes.

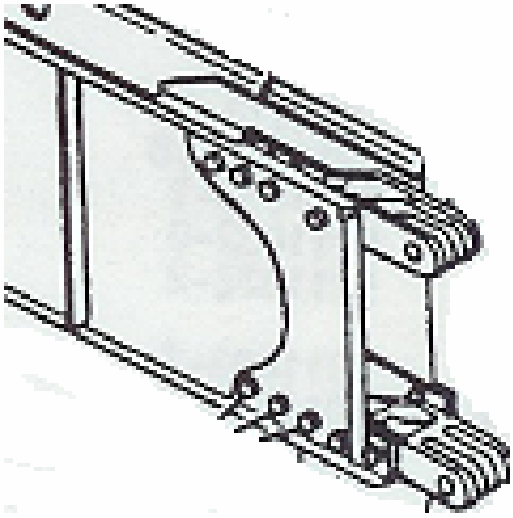

Figure 3-41. Location of Stub Wing Forward Attach Fittings

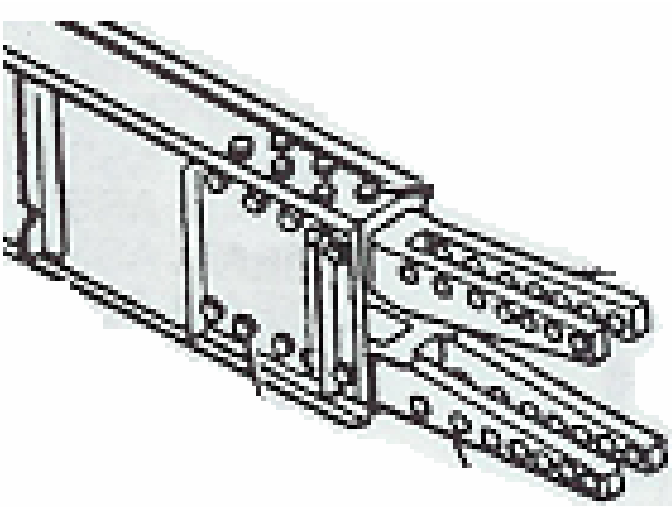

Figure 3-42. Location of Stub Wing Aft Attach Fittings

Only the left stub wing upper forward fitting (P/N 0811350-1) was found to have a crack at a fastener hole. This fitting is shown in figure 3-43. The location of the crack relative to the fastener hole is shown in figure 3-44. Figure 3-45 shows a microscopic view of the crack in the left wing upper forward fitting at 30 power magnification. As shown, the crack extends inboard

towards the fuselage parallel to the load path. The thickness of the left wing upper forward spar fitting in this area is 0.212 inch. The crack is 0.24 inch long and 0.10 inch deep. It can be classified as an elliptical surface crack and was likely caused by stress corrosion. Under high magnification (1250 times) with the scanning electron microscope, predominately blocky and intergranular features, shown in figure 3-46, can be seen. These features are typical of stress corrosion cracking.

Figure 3-43. Left Stub Wing Upper Forward Attach Fitting

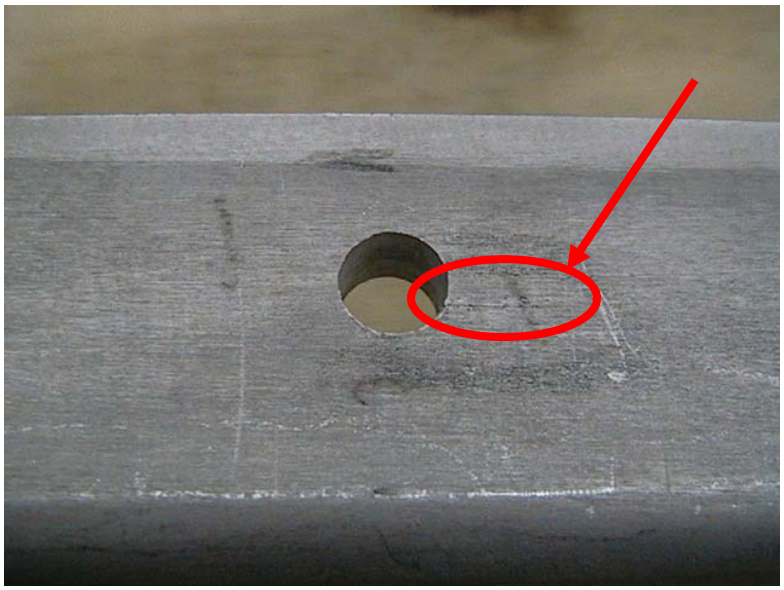

Figure 3-44. Crack in Left Stub Wing Upper Forward Attach Fitting

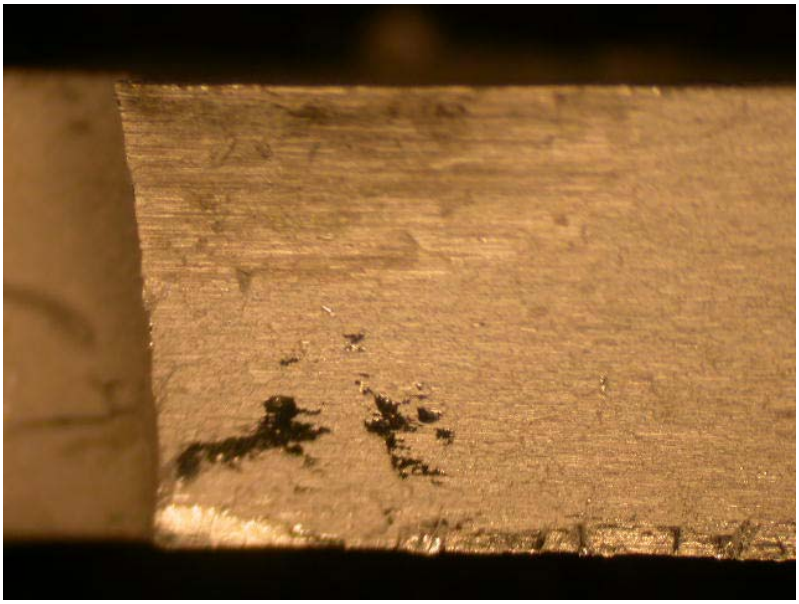

Figure 3-45. Microscopic View of Crack in the Left Stub Wing Upper Forward Attach Fitting

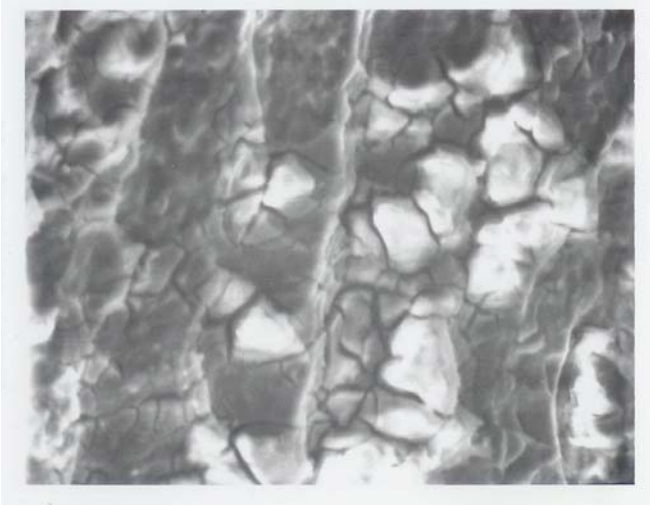

Figure 3-46. Fractograph of Left Stub Wing Upper Forward Fitting

3.5.1.3 Tip Tank Fittings.

During the microscopic examination of the tip tank fitting (P/N 0822500-40), four surface cracks were found on the lower surface of each lower tip tank fitting. The location of these cracks is shown in figure 3-47, while the fittings are shown in figure 3-48. The cracks vary in length from 0.045 to 1.25 inches. A close-up view of the longest of these cracks is shown in figure 3-49. It was determined that these cracks were incurred during the forging process and were not due to fatigue or stress corrosion cracking.

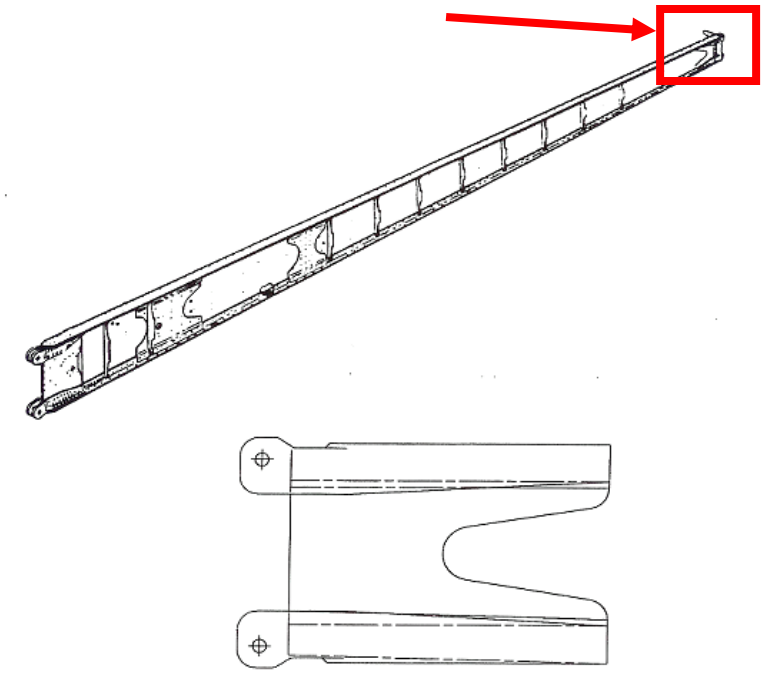

Figure 3-47. Location of Tip Tank Fittings

Figure 3-48. View of Eight Surface Cracks Located on the Tip Tank Fittings

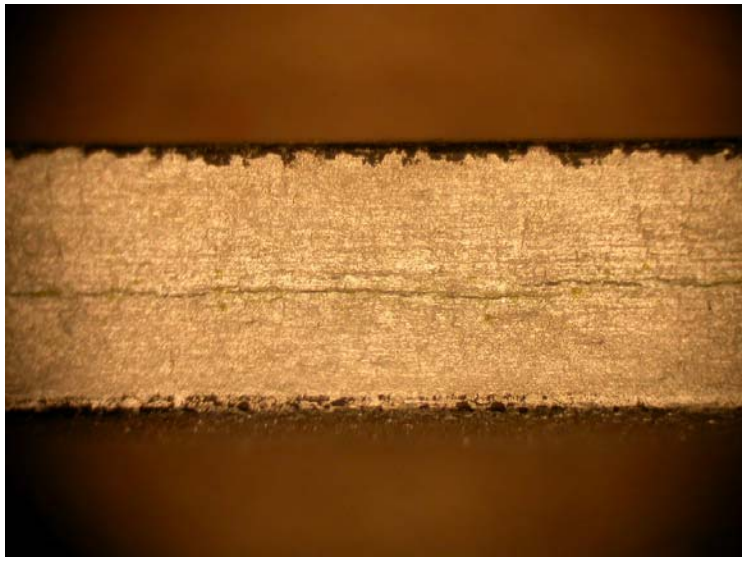

Figure 3-49. Close-Up View of the Longest Crack on the Tip Tank Fittings

3.5.1.4 Nose Wheel Steering Bellcrank.

The location of the nose wheel steering bellcrank relative to the nose gear structure is shown in figure 3-50. Figure 3-51 shows the inspection of the nose gear bellcrank using a 50-300-power microscope. When the nose gear steering bellcrank (P/N 5042010-1) was examined microscopically, eight separate cracks were found. Figure 3-52 shows the four general areas of cracking observed during the microscopic examination. Length and depth measurements of each crack were taken to characterize the individual cracks. The location labeled as rear right had three cracks, shown in figure 3-53, under 30 power magnification. The thickness at this location is 0.215 inch. Crack A is 0.055 inch long and 0.032 inch deep, crack B is 0.065 inch long and 0.036 inch deep, and crack C is 0.058 inch long and 0.027 inch deep. The aft left location contained one crack, as shown under 30 power magnification in figure 3-54. The thickness at this location is 0.215 inch and the crack is 0.093 inch long and 0.018 inch deep. The front right location contained three cracks. Crack A is 0.042 inch long and 0.026 inch deep, crack B is 0.087 inch long and 0.032 inch deep, and crack C is 0.039 inch long and 0.028 inch deep. The thickness at this location is 0.215 inch. A 30 power microscopic view of these cracks is shown in figure 3-55. The front left area, shown in figure 3-56, had one crack measuring 0.086 inch in length with a depth of 0.017 inch. The thickness at this location is 0.022 inch. After the cracks were documented photographically, they were broken open to examine the crack faces. A view of the sectioned pieces of the nose gear bellcrank is shown in figure 3-57. By using a scanning electron microscope, it was determined that all the cracks in the nose gear bellcrank were caused by fatigue. The electron fractograph (1250 power magnification) of the crack in the front left corner shows fatigue striations, as shown in figure 3-58. Cessna has already redesigned the nose wheel steering bellcrank to avoid such fatigue failures in the future.

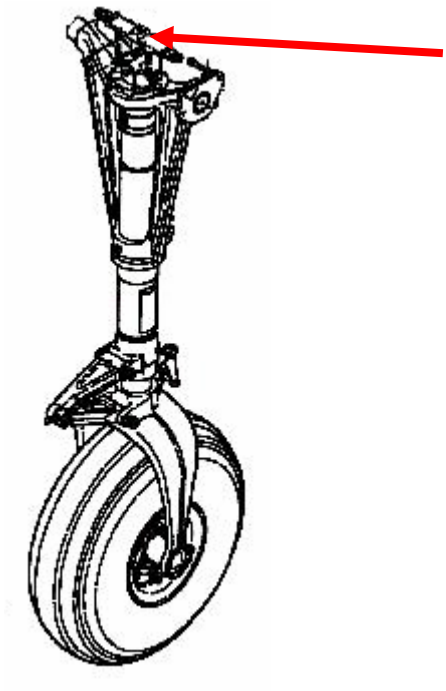

Figure 3-50. Location of Nose Gear Steering Bellcrank

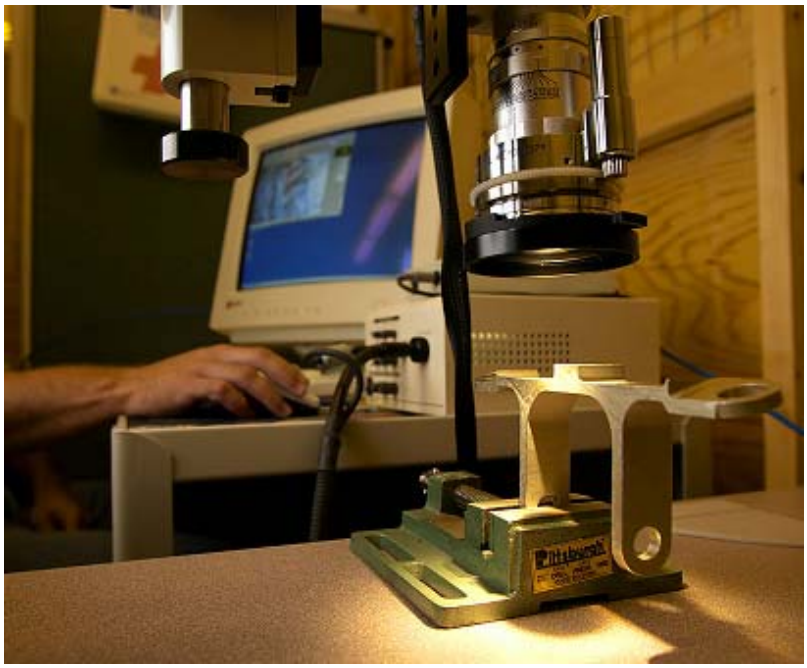

Figure 3-51. Nose Gear Bellcrank Under Inspection With 50-300-Power Microscope

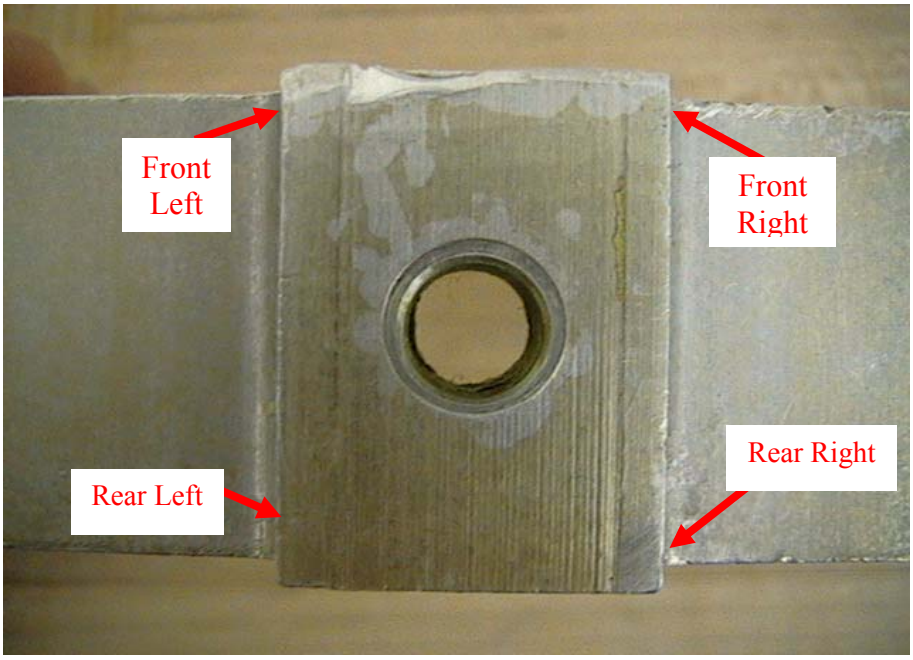

Figure 3-52. Nose Gear Bellcrank Crack Locations

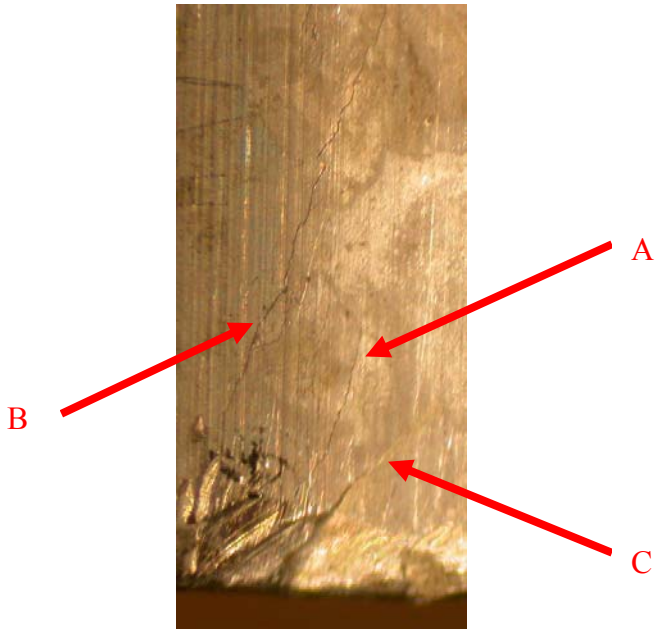

Figure 3-53. Microscopic View of Rear Right Crack Location

Figure 3-54. Microscopic View of Rear Left Crack Location

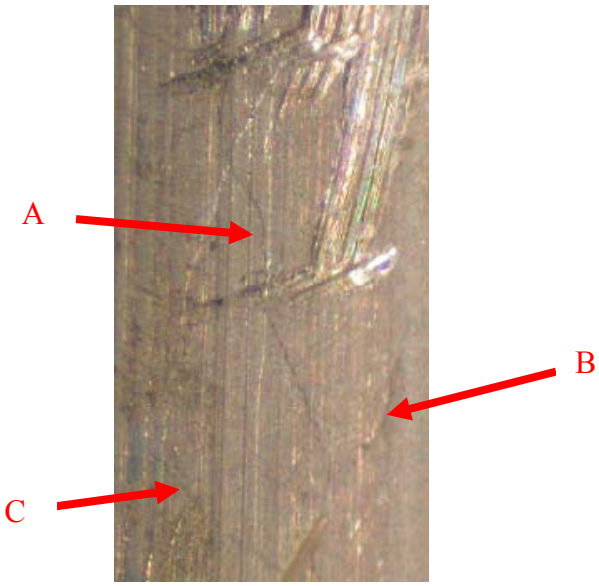

Figure 3-55. Microscopic View of Front Right Crack Location

Figure 3-56. Microscopic View of Front Left Crack Location

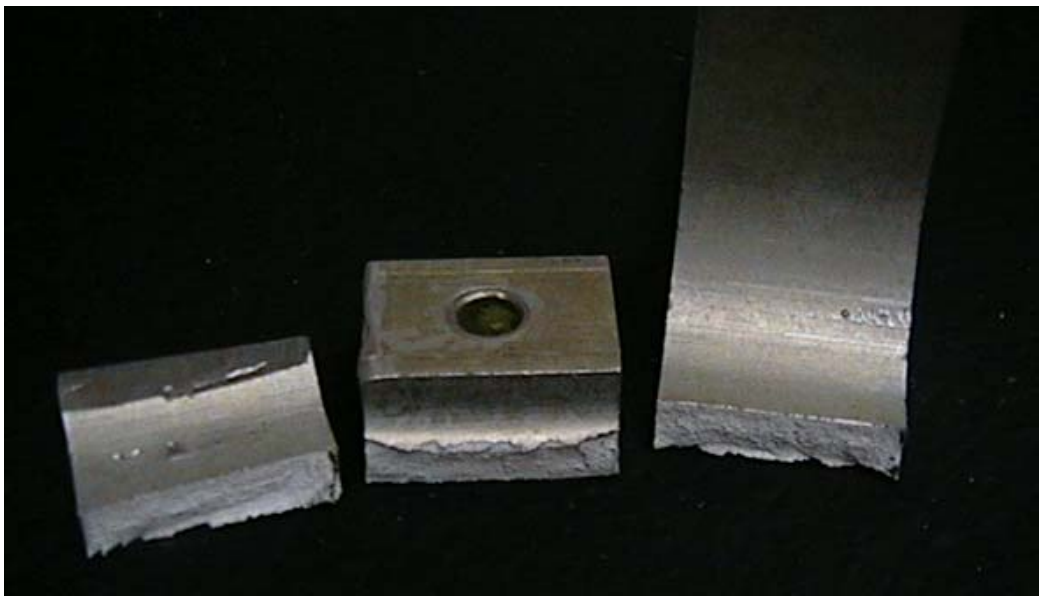

Figure 3-57. Nose Gear Bellcrank Fracture Faces

Figure 3-58. Fractograph of Front Left Corner of Nose Gear Bellcrank

3.5.2 Microscopic Examination of Alternative NDI Indications.

All areas that were inspected during the alternative NDI were also examined microscopically to determine whether the NDI findings could be validated. These areas include the front spars of the left wing and the horizontal stabilizer.

3.5.2.1 Left Wing Front Spar.

A large area of corrosion was found approximately 24.0 inches from the inboard end of the left wing front spar lower cap, as shown in figure 3-59. This area of corrosion starts at the front edge of the spar and goes approximately to 0.5 inch from the rear edge. This moderate area of corrosion encompasses an area of 7.5 square inches. With a maximum pit depth of 0.011 inch, a 5.2% reduction in material thickness results due to the corrosion.

3.5.2.2 Horizontal Stabilizer Front Spar.

Two small areas of corrosion were found on the front spar of the horizontal stabilizer. An area of light corrosion was found on the left forward spar upper cap at BL 30-32. This corrosion had a maximum depth of 0.001 inch. Another area was also found on the right forward spar lower cap at BL 38-40 with a maximum depth of 0.001 inch.

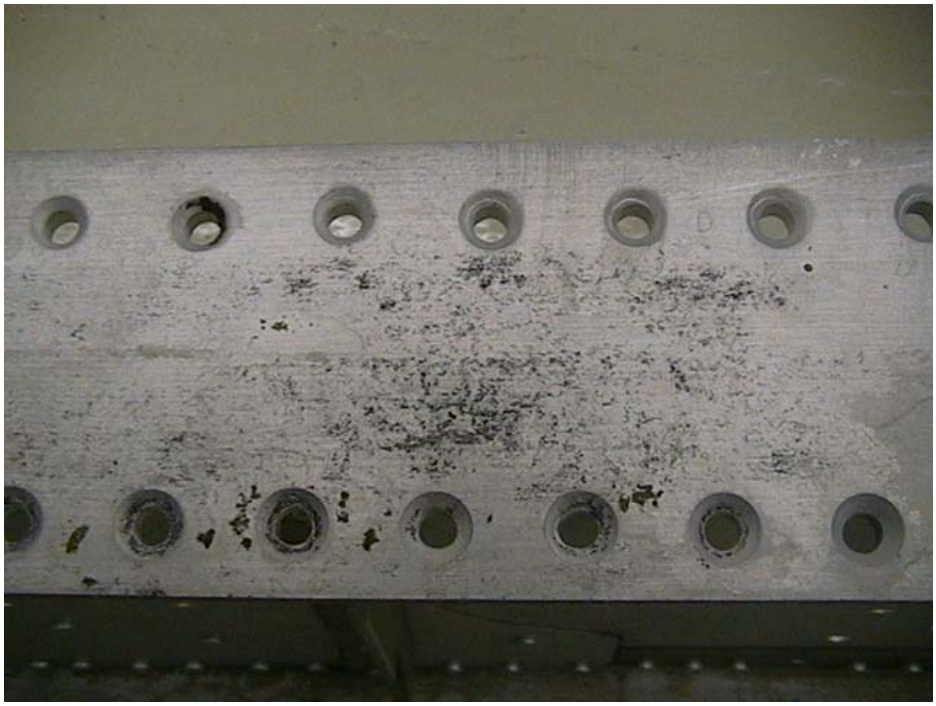

Figure 3-59. Corrosion on the Left Wing Front Spar Lower Cap

3.5.3 Microscopic Examination of Defects Found During and After Disassembly.

After completing the alternative NDI, detailed disassembly of the major airplane sections occurred. During and after disassembly, a number of defects were found. The left wing tip tanks had two cracks, 0.252 and 0.295 inch in length. Five cracks were found in the left wing lower forward auxiliary spar ranging in length from 0.046 to 0.598 inch. Three cracks were found in the left wing lower aft auxiliary spar measuring 0.076 to 0.419 inch in length. The right wing forward auxiliary spar had ten cracks ranging in size from 0.041 to 0.740 inch, while the right wing aft auxiliary spar had two cracks measuring 0.326 and 0.396 inch. The left hand fuselage channel had light-moderate corrosion, while severe areas of corrosion were found on the right stub wing skin lap joints.

3.5.3.1 Left Wing.

Two through cracks measuring 0.252 and 0.295 inch were found on the left wing tip tank baffle. Cracks were also found in the forward and aft auxiliary spars ranging in length from 0.046 to 0.598 inch.

3.5.3.1.1 Left Wing Tip Tank Baffle.

Cracks were found at two baffle holes in the left tip tank next to the lug fitting (P/N 5092300-15). These cracks could only have been found during a destructive evaluation, since they are located in an inaccessible area. The location of the tip tank baffle is shown in figure 3-60. The location of the cracks on the left wing tip tank baffle is shown in figure 3-61. Figure 3-62 provides a closer view of the cracks. After the baffle was paint stripped and etched, the cracks

were sectioned and removed for examination. Once removed, the cracks were broken open for microscopic analysis of the fracture surface. Figure 3-63 is a microscopic view of crack A in the tip tank baffle at 30 power magnification. The thickness of the tip tank baffle in the area of the two cracks is 0.130 inch. The length of crack A is 0.252 inch, while the length of crack B is 0.295 inch. Fracture surfaces on both cracks show evidence of fatigue. Fatigue striations can be seen in figure 3-64, which is a fractograph at 1250 power magnification of the surface of crack A.

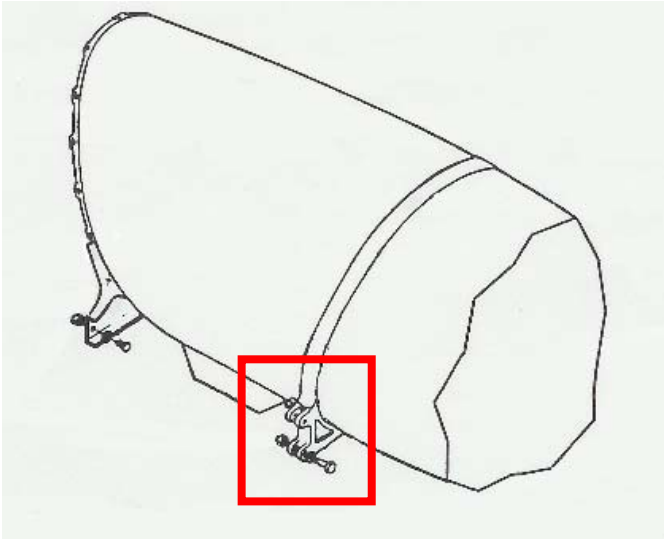

Figure 3-60. Location of Tip Tank Fitting

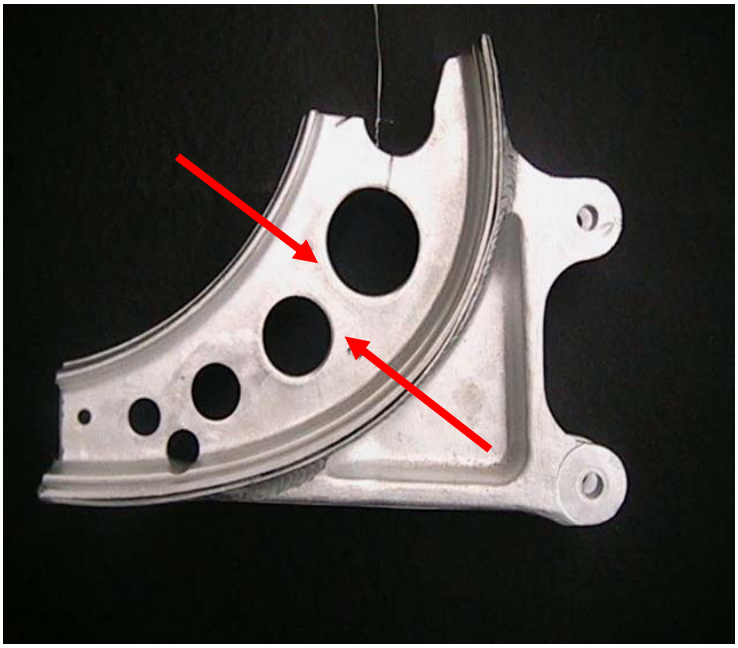

Figure 3-61. Left Wing Tip Tank Baffle

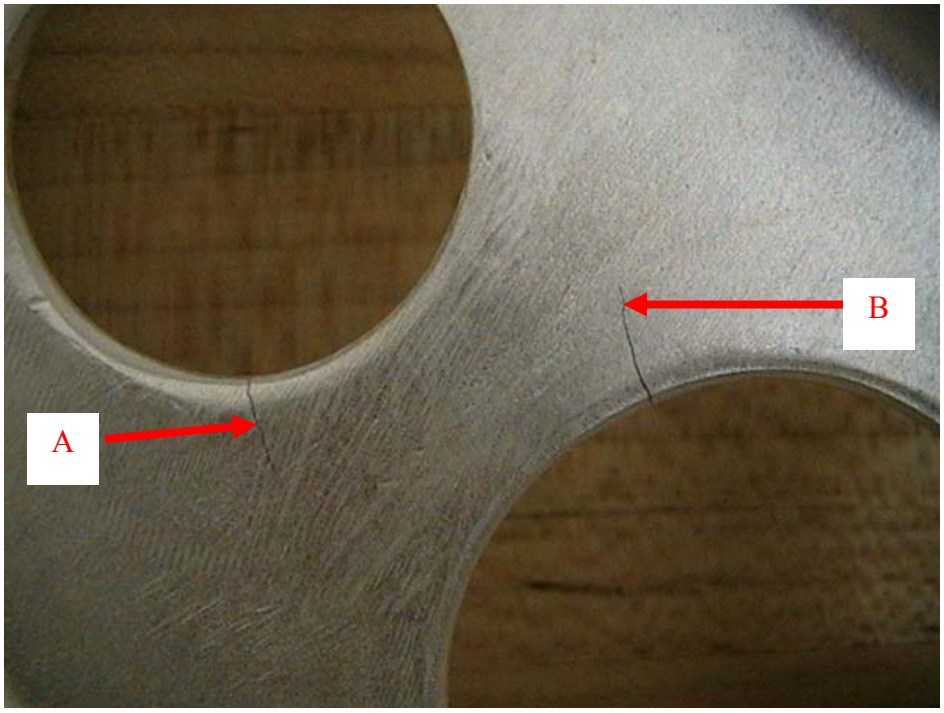

Figure 3-62. Cracks in the Left Wing Tip Tank Baffle

Figure 3-63. Microscopic View of Crack A in the Tip Tank Baffle

Figure 3-64. Electron Fractograph of Crack A in the Tip Tank Baffle

3.5.3.1.2 Left Wing Forward Auxiliary Spar.

The location of the forward auxiliary spar (P/N 5222100-3) is shown in figure 3-65. Three areas of cracking were found on the left wing forward auxiliary spar, as shown in figure 3-66. The lower inboard corner of the forward auxiliary spar contains cracks B and C, as shown in figure 3-67. Crack A is located in the upper inboard corner of the forward auxiliary spar (figure 3-66) and cracks D and E, shown in figure 3-68, are located in the wiring hole above the access panel. The thickness of material at all locations is 0.055 inch and all cracks are through the thickness. Crack A is 0.225 inch long, crack B is 0.046 inch long, crack C is 0.598 inch long, crack D is 0.282 inch long, and crack E is 0.158 inch long. Figure 3-69 illustrates the fracture face of crack D, which was determined to be caused by fatigue.

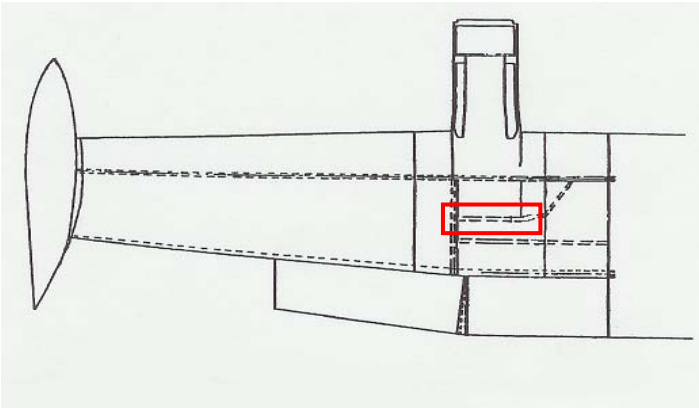

Figure 3-65. Location of Forward Auxiliary Spar

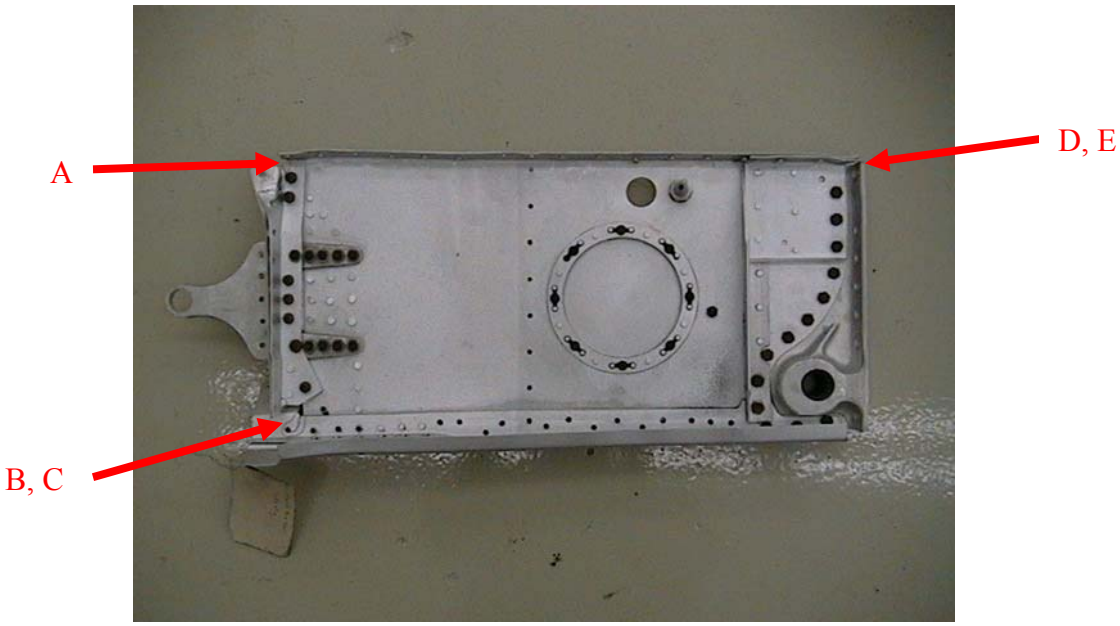

Figure 3-66. Cracks Located in the Left Wing Forward Auxiliary Spar

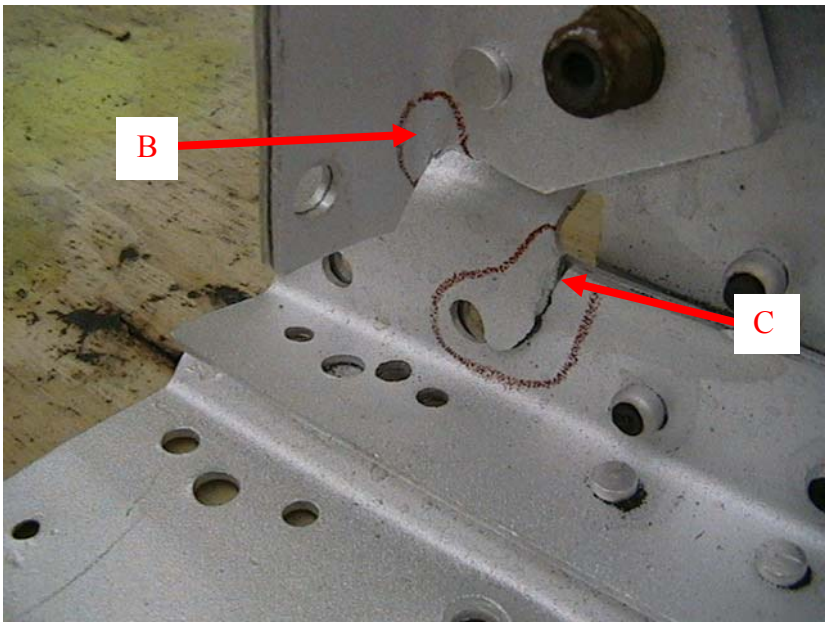

Figure 3-67. Cracks B and C on the Left Wing Forward Auxiliary Spar

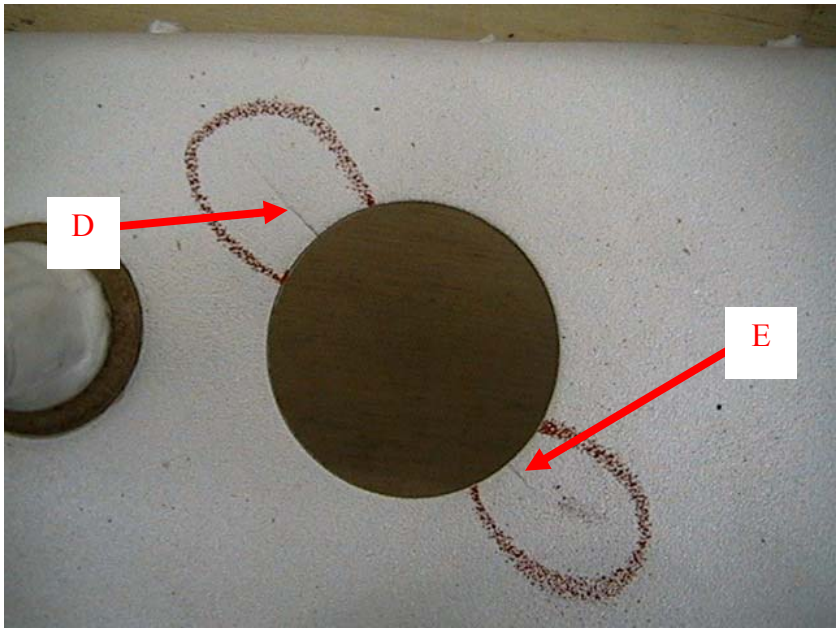

Figure 3-68. Cracks D and E on the Left Wing Forward Auxiliary Spar

Figure 3-69. Fracture Face of Crack D of the Left Wing Forward Auxiliary Spar

3.5.3.1.3 Left Wing Aft Auxiliary Spar.

The location of the left wing aft auxiliary spar (P/N 5221027-13) is shown in figure 3-70. Figure 3-71 shows the location of three cracks, labeled A, B, and C, found in the left wing of the lower aft auxiliary spar in relation to the left wing structure. The thickness of the material at all locations is 0.055 inch, and all cracks are through the thickness. Crack A is located at the lower

inboard corner of the aft auxiliary spar, as shown in figure 3-72. Cracks B and C are located on the upper inboard and outboard spar corners (figure 3-71), respectively. Crack A is 0.242 inch long, crack B is 0.076 inch long, and crack C is 0.419 inch long. Both cracks B and C were broken open; however, due to crack separation, all of the fracture face detail was removed during the etching process. Even though no fractographic information was gathered from these cracks, they were determined to be similar to the cracks on the forward auxiliary spar.

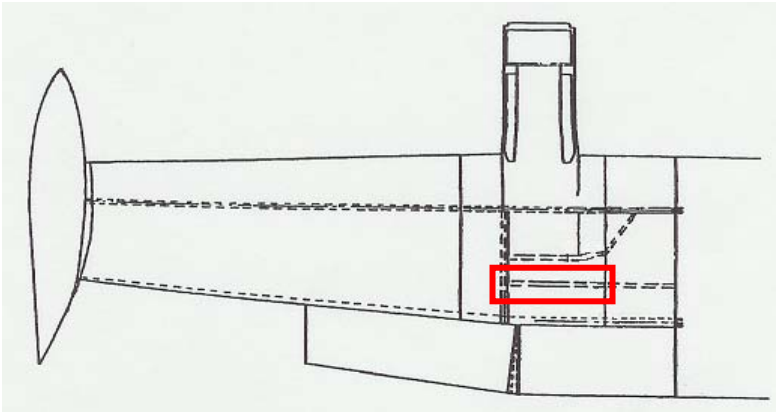

Figure 3-70. Location of the Left Wing Aft Auxiliary Spar

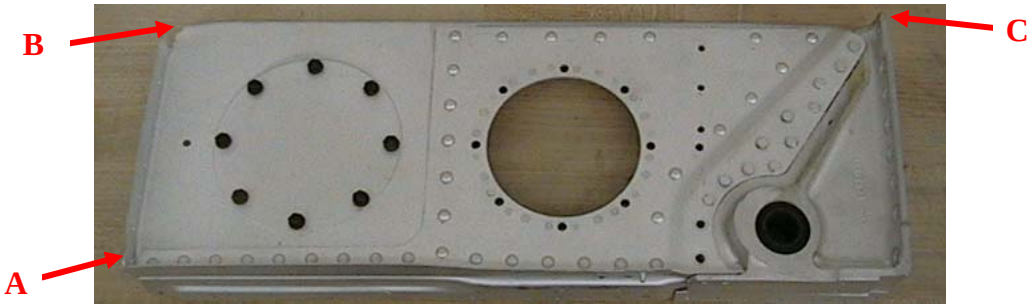

Figure 3-71. Location of Cracks on the Left Wing Aft Auxiliary Spar

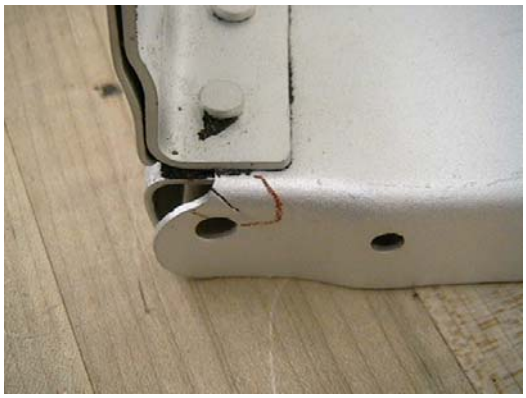

Figure 3-72. Crack A on the Left Wing Aft Auxiliary Spar

3.5.3.2 Right Wing.

The front spar of the right wing was microscopically examined in an attempt to characterize all damage in the structure. A large area of corrosion on the lower front spar cap was found, as well as ten cracks on the forward auxiliary spar. Two additional cracks were also discovered on the aft auxiliary spar.

3.5.3.2.1 Right Wing Front Spar.

A large region of corrosion was found from approximately 4 to 14 inches from the inboard end of the front spar lower cap of the right wing. The most severe area is shown in figure 3-73. This corrosion was scattered throughout this region and was not continuous. The most severe area of corrosion on the spar was classified as light-moderate. The area that encompasses this corrosion is 0.4977 square inch, and the maximum depth found was 0.009 inch, as shown in figure 3-74. The maximum thickness loss in this region due to corrosion was 4.3%.

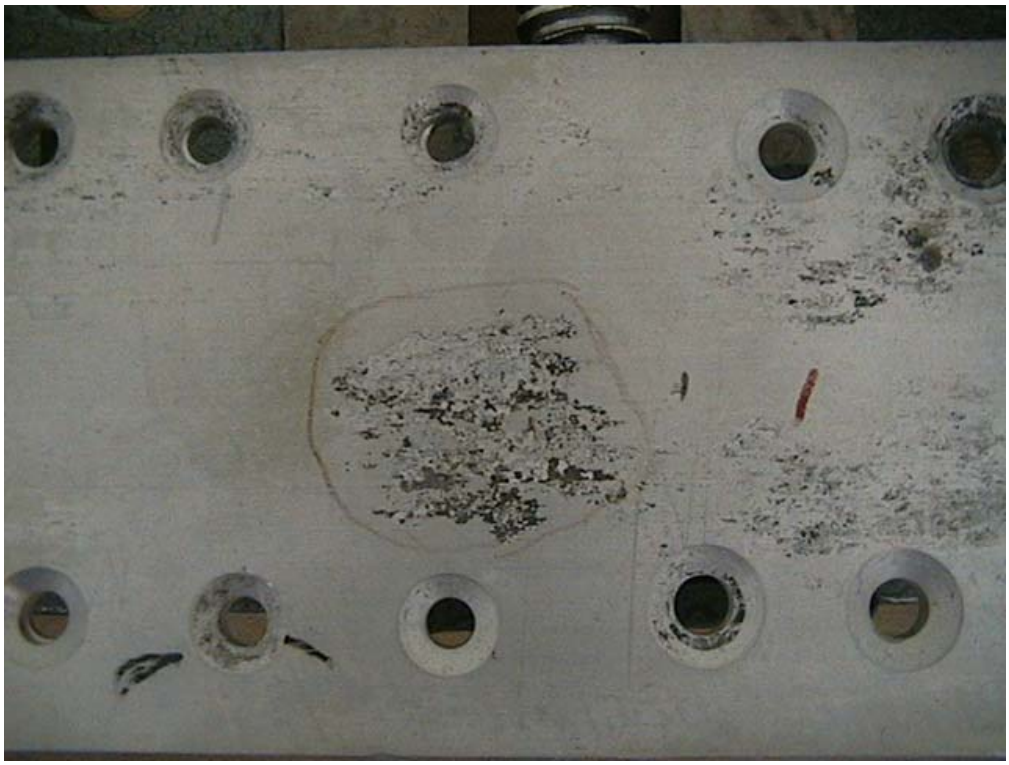

Figure 3-73. Corrosion on the Right Wing Front Spar Lower Cap

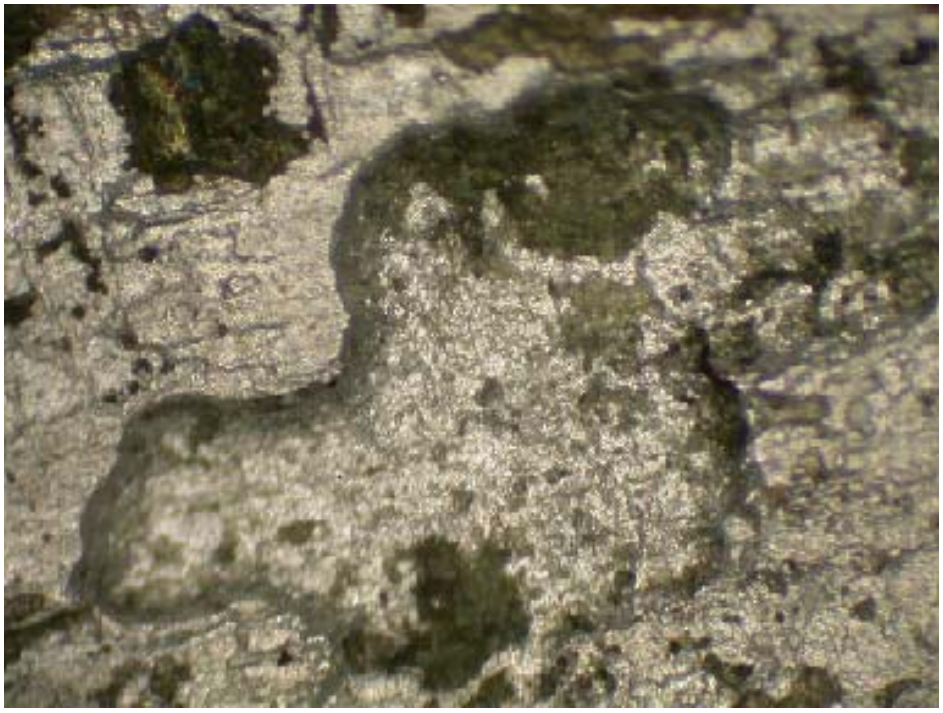

Figure 3-74. Deepest Pit of Corrosion on the Right Wing Front Spar Lower Cap

3.5.3.2.2 Forward Auxiliary Spar.

Two areas of cracking were found on the right wing forward auxiliary spar, similar to the areas found on the left wing, as shown in figure 3-75. The material thickness in the area of these cracked regions is 0.55 inch. All the cracks found on the forward auxiliary spar were through-the-thickness cracks. Crack A, which is depicted in figure 3-76, is 0.167 inch long and is located on the outboard upper corner of the forward auxiliary spar. Crack B is located on the lower inboard corner of the forward auxiliary spar (P/N 5222100-4). The length of crack B, shown in figure 3-77, is 0.133 inch. These cracks are likely fatigue cracks due to their similarity with cracks found in the same area on the left wing. In the area around the access panel and wiring hole, a total of eight cracks were found. These cracks, labeled C-J, are shown in figure 3-78. The lengths of the cracks found are as follows: crack C is 0.589 inch long, crack D is 0.041 inch long, crack E is 0.740 inch long, crack F is 0.126 inch long, crack G is 0.075 inch long, crack H is 0.067 inch long, crack I is 0.271 inch long, and crack J is 0.052 inch long. Crack E was chosen to be broken open for fractographic analysis, and the results of this analysis show that this crack is a fatigue crack. An electron fractograph of the crack face, at 1250 power magnification, is shown in figure 3-79. Due to the similar location and nature of these cracks, it is assumed that all the cracks on the right wing forward auxiliary spar are fatigue cracks.

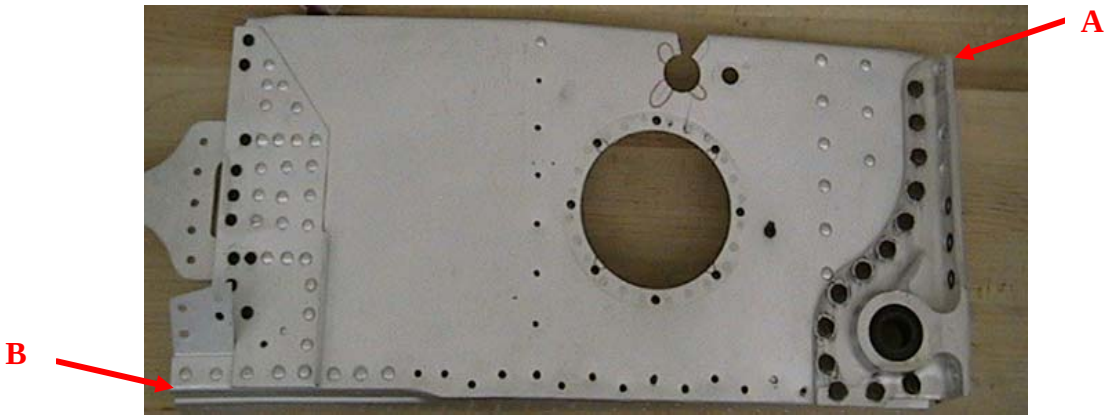

Figure 3-75. Cracks Located on the Right Wing Forward Auxiliary Spar

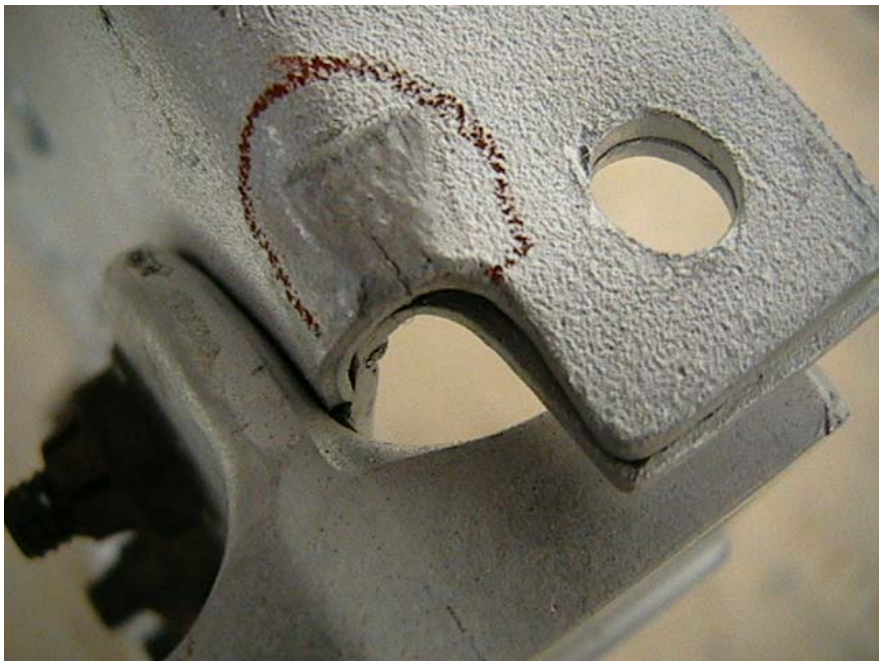

Figure 3-76. Crack A on the Right Wing Forward Auxiliary Spar

Figure 3-77. Crack B on the Right Wing Forward Auxiliary Spar

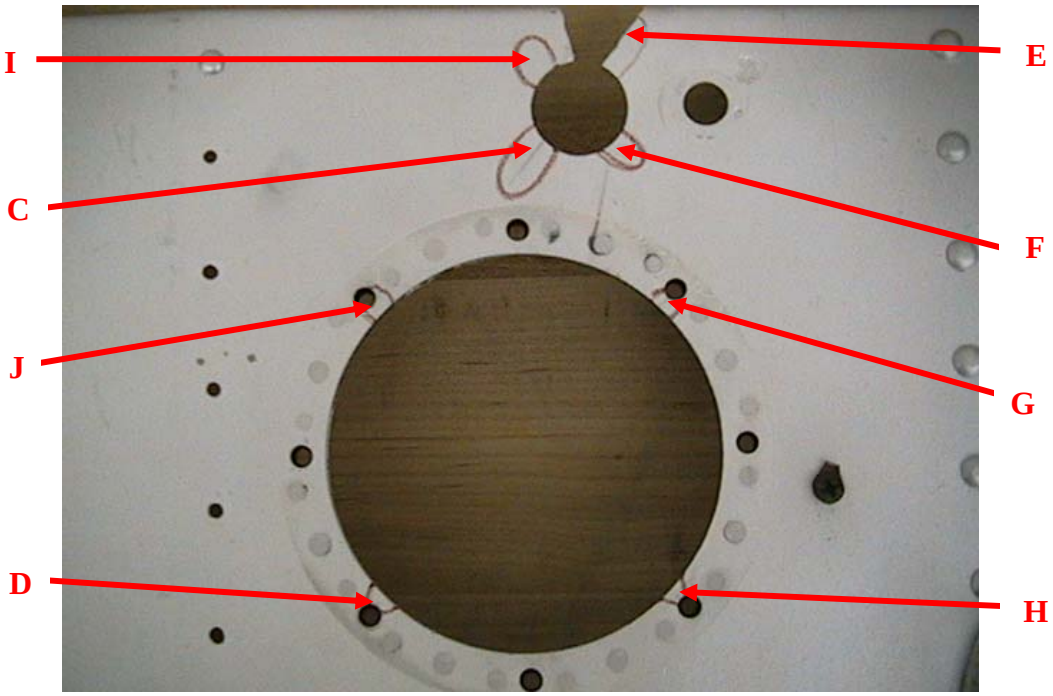

Figure 3-78. Cracks in Access Panel and Wiring Hole of the Right Wing Forward Auxiliary Spar

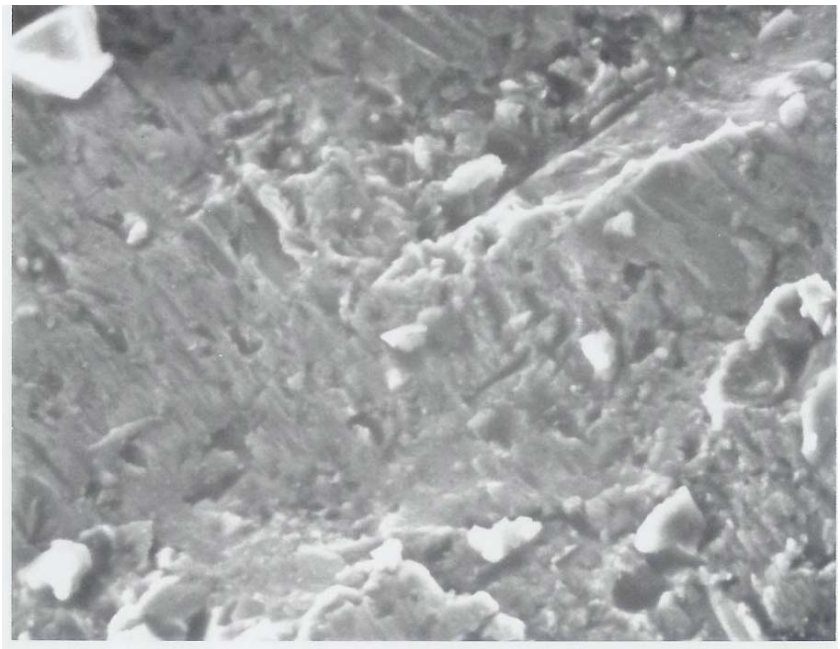

Figure 3-79. Fractograph of Crack E on the Forward Auxiliary Spar

3.5.3.2.3 Aft Auxiliary Spar.

The location of the right wing aft auxiliary spar is shown in figure 3-80. The locations of two cracks found on the aft auxiliary spar during the microscopic examination are shown in figure 3-81. Crack A, shown in figure 3-82, is 0.396 inch long and crack B, shown in figure 3-83, is 0.326 inch long. The material thickness of the aft auxiliary spar is 0.055 inch, and both cracks are through the thickness. Fractographic analysis on crack A could not be completed because of crack face separation. The fracture face details were removed during the etching process due to this separation. However, these cracks are also likely fatigue cracks similar to the ones found in the same area on the left wing.

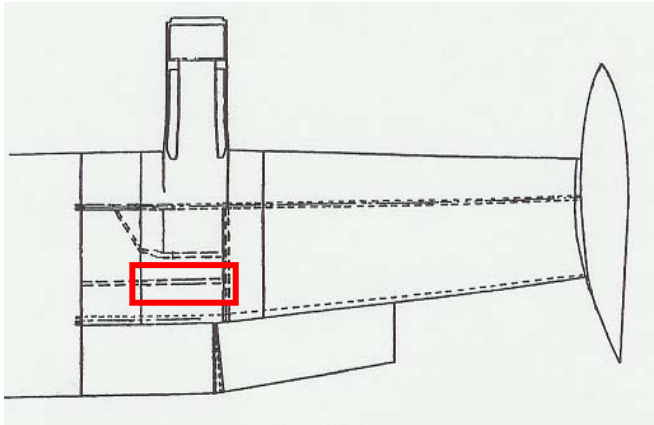

Figure 3-80. Location of the Right Wing Aft Auxiliary Spar

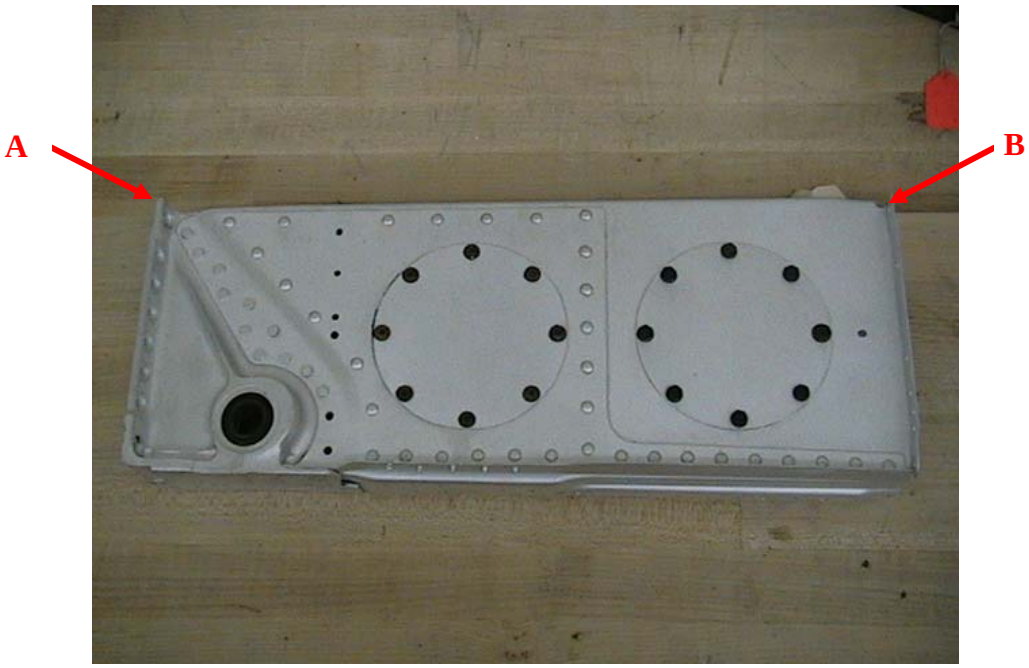

Figure 3-81. Cracks Located on the Aft Auxiliary Spar

Figure 3-82. Crack A on the Right Wing Aft Auxiliary Spar

Figure 3-83. Crack B on the Right Wing Aft Auxiliary Spar

3.5.4 Microscopic Examination of Fuselage and Stub Wings.

During the microscopic examination of the fuselage and stub wings, corrosion was found on the left fuselage channel and on the upper and lower skin lap joints of the right stub wing.

3.5.4.1 Left-Hand Fuselage Channel.

During examination of the left-hand fuselage channel (P/N 5011278-7), an area of light-moderate corrosion was found at the aft end of the channel. The location of the left-hand fuselage channel with respect to the left-hand fuselage structure is shown in figure 3-84. The channel itself is shown in figure 3-85. This area of corrosion is located 0.45 inch forward of the aft end and 0.4 inch from the outboard edge. This region encompasses 0.156 square inch with a maximum depth of 0.004 inch, which resulted in a 5.3% thickness loss. This region is shown in figure 3-86.

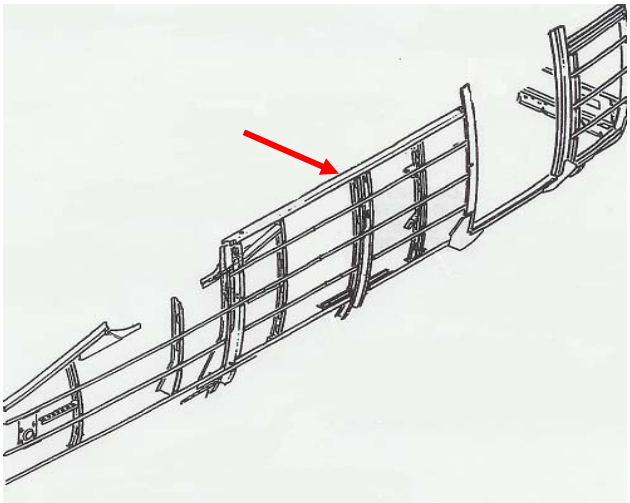

Figure 3-84. Location of Left-Hand Fuselage Channel

Figure 3-85. Left-Hand Fuselage Channel

Figure 3-86. Area of Corrosion on the Left-Hand Fuselage Channel

3.5.4.2 Right Stub Wing Skin Lap.

During the removal of the right stub wing skins several small areas of corrosion were found in the skin lap between the leading-edge and upper skins. The location of this skin lap in relationship to the fuselage can be seen in figure 3-87. The leading-edge skin is shown in figure 3-88. On the lower side of the leading-edge skin, one area of severe corrosion was detected. This area of the corrosion encompasses 0.2 square inch, has a maximum depth of 0.009 inch, and is shown in figure 3-89. The reduction in thickness due to corrosion in this region was 30%. The upper surface of the upper skin had two separate areas of corrosion, as shown in figure 3-90. At location A, severe corrosion was detected that encompasses an area of 0.0625 square inch and a maximum depth of 0.012 inch, as shown in figure 3-91. The reduction in thickness due to this area of corrosion was 40%. At location B, severe corrosion, causing a 30% loss in material

thickness, was detected that encompasses an area of 1.5 square inches with a maximum depth of 0.009 inch, as shown in figure 3-92.

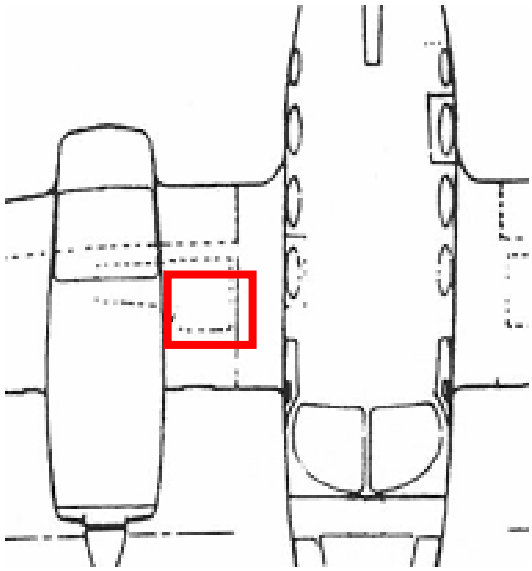

Figure 3-87. Right Stub Wing Skin Lap Location

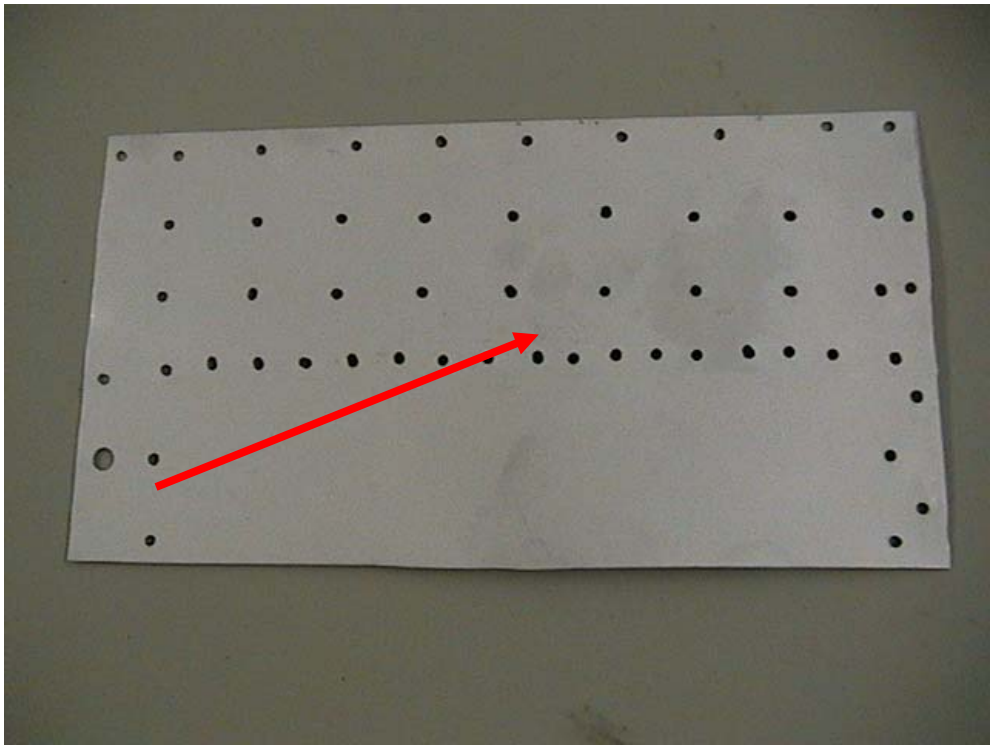

Figure 3-88. Right Stub Wing Leading-Edge Skin

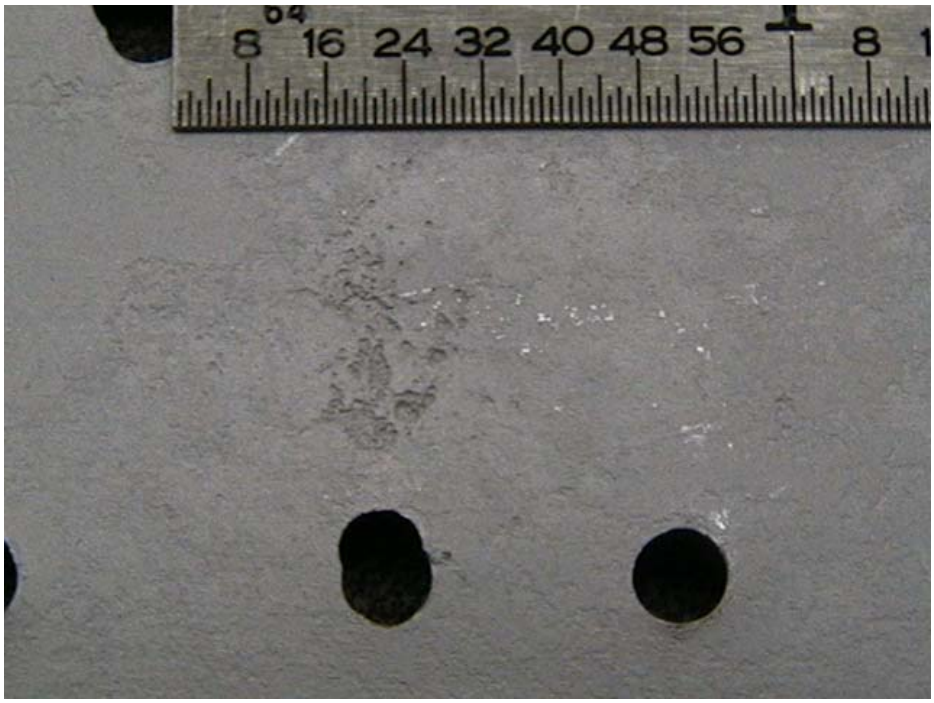

Figure 3-89. Corrosion on the Right Stub Wing Leading-Edge Skin

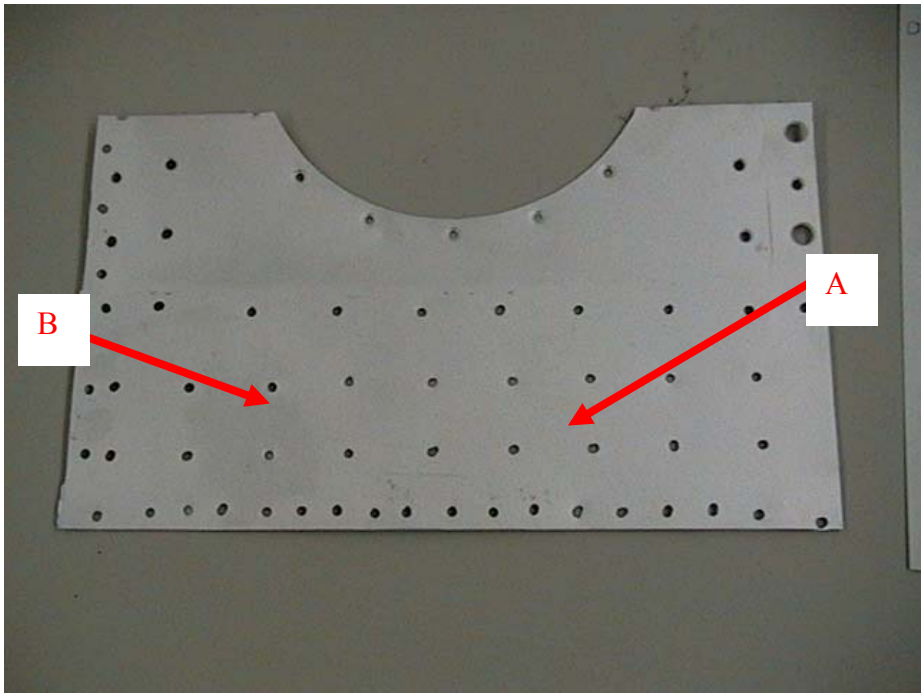

Figure 3-90. Corrosion on the Right Stub Wing Upper Skin

Figure 3-91. Close-Up View of Corrosion in Section A on the Right Wing Upper Skin

Figure 3-92. Close-Up View of Corrosion in Section B on the Right Stub Wing Upper Skin

3.5.5 Microscopic Examination of Vertical Stabilizer.

Figure 3-93 shows the location of two cracks found on the rudder bellcrank (P/N 5115206-1). Figure 3-94 shows a 30-power magnification of these cracks. Crack A was 0.03 inch long, and crack B was 0.045 inch long. The thickness of the part in this area was 0.5 inch.

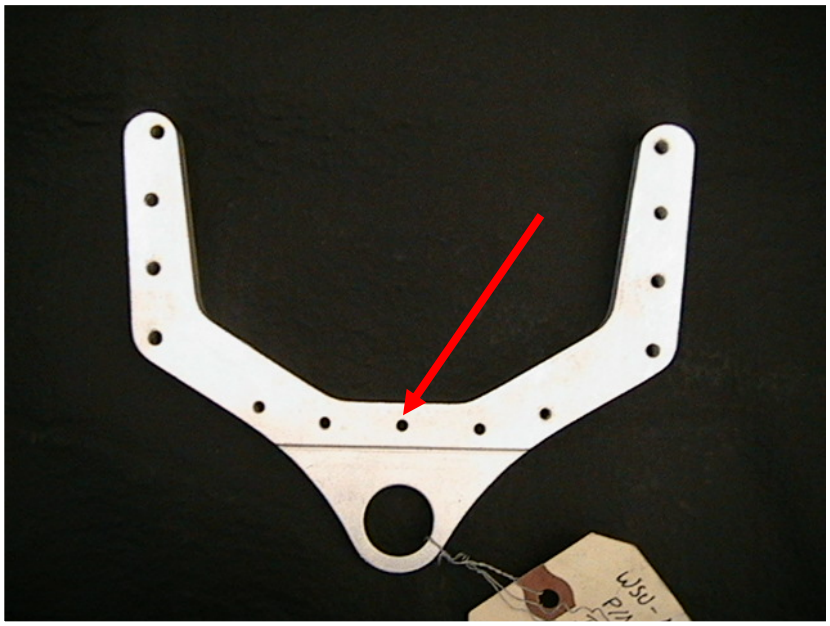

Figure 3-93. Rudder Bellcrank

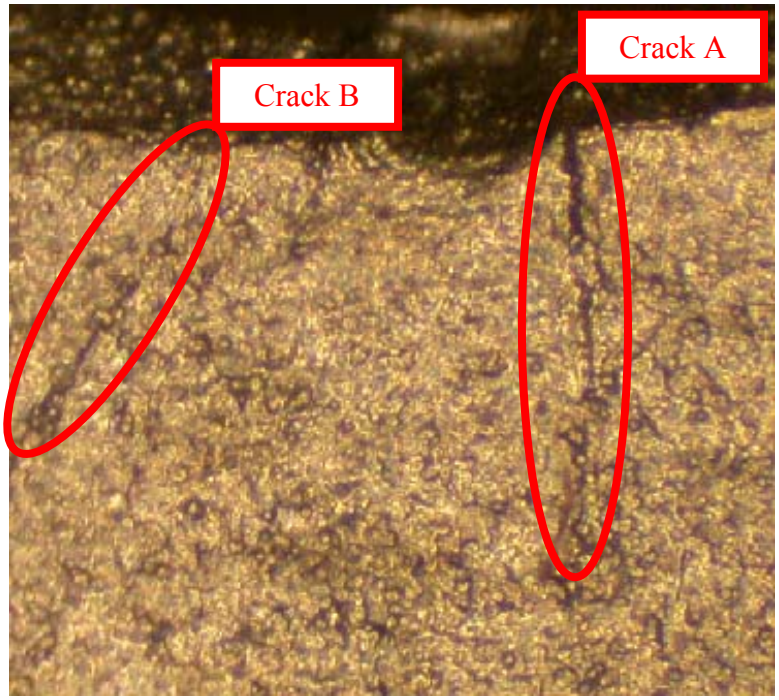

Figure 3-94. Two Cracks on the Rudder Bellcrank

3.5.6 Microscopic Examination of Critical Structural Areas.

Table 3-7 lists the critical structural areas identified in the maintenance manual and their locations, as well as the results found microscopically at each of these locations.

Table 3-7. Critical Structural Areas and Microscopic Results

Critical Area Description	Critical Areas Location	Microscopic Finding
Wing forward carry-through spar lower cap	BL 36.12	0.24-inch crack
Wing front spar lower cap root fitting	WS 46.70	Light-moderate corrosion
Wing front spar lower cap root fitting attach	WS 54.10	Light-moderate corrosion
Wing front spar lower cap canted rib attach	WS 66.70	No defect found
Wing front spar lower cap	WS 75.66	No defect found
Wing front spar lower cap at inboard engine beam attach	WS 83.74	No defect found
Wing front spar lower cap at outboard engine beam attach	WS 98.74	No defect found
Wing forward auxiliary spar lower cap	WS 86.62	No defect found
Wing aft auxiliary spar lower cap	WS 89.65	No defect found
Wing aft carry-through spar lower cap	BL 37.60	No defect found
Wing rear spar cap at splice	WS 97.87	No defect found
Wing front spar upper cap	WS 106.82	No defect found
Engine beam at aft engine mount	FS 131.20	No defect found
Fuselage left-hand longeron	FS 190.33	Moderate corrosion
Tailcone angle attach to horizontal rear spar	BL 2.90	No defect found
Horizontal stabilizer front spar upper cap	BL 0.0	No defect found
Horizontal stabilizer front spar lower cap	BL 0.0	No defect found
Horizontal stabilizer front spar attach bolt through web	BL 7.69	No defect found
Horizontal stabilizer rear spar lower cap at attach bolt	BL 2.90	No defect found
Horizontal stabilizer rear spar upper cap	BL 0.0	No defect found
Horizontal stabilizer rear spar lower cap	BL 0.0	No defect found
Horizontal stabilizer rear auxiliary spar upper cap	BL 8.01	No defect found
Vertical stabilizer rear spar at attach	WL 108.38	No defect found
Vertical stabilizer rear spar cap	WL 136.04	No defect found
Main landing gear side brace actuator collar		No defect found
Nose landing gear fork		No defect found

FS = Frame station

3.5.7 Microscopic Examination Findings.

Table 3-8 lists a summary of the microscopic findings on the Cessna 402A.

Table 3-8. Results of the Microscopic Inspection

Location	Part	Crack/ Corrosion	Length/ Area	Diagnosis
Left wing	Tip tank fitting	Crack (A)	0.252 inch	Fatigue
		Crack (B)	0.295 inch	Fatigue
	Forward auxiliary spar	Crack (A)	0.225 inch	Fatigue
		Crack (B)	0.046 inch	Fatigue
		Crack (C)	0.598 inch	Fatigue
		Crack (D)	0.282 inch	Fatigue
		Crack (E)	0.158 inch	Fatigue
	Aft auxiliary spar	Crack (A)	0.242 inch	Fatigue
		Crack (B)	0.076 inch	Fatigue
		Crack (C)	0.419 inch	Fatigue
	Front spar lower rear root fitting	Corrosion	1.2 square inches	Light
	Front spar lower front root fitting	Corrosion	0.1875 square inch	Moderate-severe
	Front spar upper rear root fitting	Corrosion	0.004 square inch	Light-moderate
	Front spar lower cap	Corrosion	7.5 square inches	Moderate
Right wing	Forward auxiliary spar	Crack (A)	0.167 inch	Fatigue
		Crack (B)	0.133 inch	Fatigue
		Crack (C)	0.589 inch	Fatigue
		Crack (D)	0.041 inch	Fatigue
		Crack (E)	0.740 inch	Fatigue
		Crack (F)	0.126 inch	Fatigue
		Crack (G)	0.075 inch	Fatigue
		Crack (H)	0.067 inch	Fatigue
		Crack (I)	0.271 inch	Fatigue
		Crack (J)	0.052 inch	Fatigue
	Aft auxiliary spar	Crack (A)	0.396 inch	Fatigue
		Crack (B)	0.326 inch	Fatigue
	Front spar upper rear fitting	Corrosion	0.25 square inch	Light-moderate
	Front spar lower cap	Corrosion	0.4977 square inch	Light-moderate
	Tip tank	8 cracks	0.045-1.25 inches	Forging

Table 3-8. Results of the Microscopic Inspection (Continued)

Location	Part	Crack/ Corrosion	Length/ Area	Diagnosis
Fuselage	LH channel	Corrosion	0.0156 square inch	Light-moderate
	LH upper forward spar fitting	Crack	0.240 inch	Stress corrosion
Stub wing	RH upper lap	Corrosion	0.2 square inch	Severe
	RH upper lap	Corrosion (A)	0.0625 square inch	Severe
		Corrosion (B)	1.5 square inches	Severe
Landing gear	Bellcrank (rear right)	Crack (A)	0.055 inch	Fatigue
		Crack (B)	0.065 inch	Fatigue
		Crack (C)	0.058 inch	Fatigue
	Bellcrank (front right)	Crack (A)	0.042 inch	Fatigue
		Crack (B)	0.087 inch	Fatigue
		Crack (C)	0.039 inch	Fatigue
	Bellcrank (rear left)	Crack	0.093 inch	Fatigue
	Bellcrank (front left)	Crack	0.086 inch	Fatigue
Vertical stabilizer	Rudder bellcrank	Crack (A)	0.03 inch	Fatigue
		Crack (B)	0.045 inch	Fatigue

LH = Left hand
RH = Right hand

All indications from the alternative NDI techniques were microscopically examined to determine if actual flaws were present in these target areas. Table 3-9 shows a comparison of the alternative NDI indications and the flaws found with the optical microscope on the horizontal stabilizer. Of the eight areas indicated by the spot probe, only two were found to have mild corrosion during the microscopic examination. Corrosion with pits less than 0.01 inch depth were classified as mild corrosion, while pits with depths greater than 0.01 inch is denoted as moderate corrosion. Figures 3-95 and 3-96 show the locations of cracks and areas of corrosion detected during the microscopic examination and indications from the alternative NDI assessment of primary structure for the lower and upper surfaces of the horizontal stabilizer front spar, respectively. Even though some of the alternative NDI indications were verified microscopically, not all of the alternative NDI indications were verified during the microscopic examination.

Table 3-9. Comparison of Alternative NDI Indications and Microscopic Examination on the Horizontal Stabilizer

NDI Method	Location	NDI Indications	Microscopic Results
Spot probe	Left forward spar lower cap	BL 57-58	No flaw found
Spot probe	Left forward spar lower cap	BL 38-41	No flaw found
Spot probe	Left forward spar upper cap	BL 18-20	No flaw found
Spot probe	Left forward spar upper cap	BL 22-25	No flaw found
Spot probe	Left forward spar upper cap	BL 30-32	Mild corrosion
Spot probe	Right forward spar lower cap	BL 6	No flaw found
Spot probe	Right forward spar lower cap	BL 38-40	Mild corrosion
Spot probe	Right forward spar upper cap	BL 24-29	No flaw found

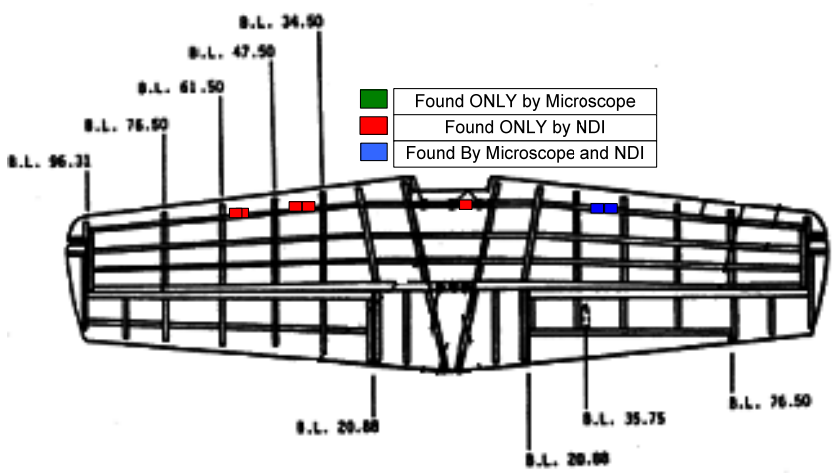

Figure 3-95. Location of Findings From Microscopic Examination and Alternative NDI on the Horizontal Stabilizer Lower Surface

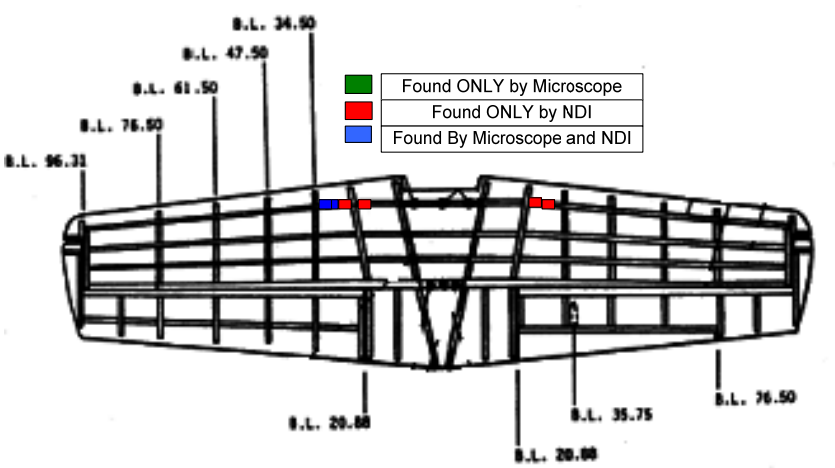

Figure 3-96. Location of Findings From Microscopic Examination and Alternative NDI on the Horizontal Stabilizer Upper Surface

Table 3-10 shows a comparison of the alternative NDI indications and the flaws found with the optical microscope on the left wing. Six of the fourteen indications were corrosion. Some of the corroded areas were called crack indications during NDI. One skin crack was found as well, but it did not penetrate into the underlying structure. Also, a nutplate was detected as a crack during the NDI phase. Since the nutplate caused a thickness change, or discontinuity, in the eddy-current field, the NDI inspectors recorded a crack indication. A large area of corrosion was found during the microscopic examination on the lower surface of the front spar from WS 21 to 32; however, this area was not detected using the alternative NDI techniques. Figures 3-97 and 3-98 show the locations of the microscopic examination findings and alternative NDI indications on the left wing upper and lower surfaces. As with the horizontal stabilizer front spar, not all of the alternative NDI indications were verified during the microscopic examination.

Table 3-10. Comparison of Alternative NDI Indications and Microscopic Examination on the Left Wing

NDI Method	Location	NDI Indications	Microscopic Results
Sliding probe	Front spar upper cap	WS 29-33	Corrosion
Sliding probe	Front spar upper cap	WS 51-53	No flaws found
Sliding probe	Front spar upper cap	WS 54-59	No flaws found
Sliding probe	Front spar upper cap	WS 69	No flaws found
Sliding probe	Front spar upper cap	WS 73-80	Mild corrosion
Sliding probe	Front spar upper cap	WS 84-88	Mild corrosion
Sliding probe	Front spar lower cap	WS 50-55	Nutplate
Sliding probe	Front spar lower cap	WS 57-60	Corrosion
Sliding probe	Front spar lower cap	WS 69	No flaws found
Sliding probe	Front spar lower cap	WS 72-80	No flaws found
Sliding probe	Front spar lower cap	WS 87-90	Mild corrosion
MOI/spot probe	Front spar lower cap	WS 78-79	No flaws found
MOI/spot probe	Front spar lower cap	WS 81	Skin crack
MOI/spot probe	Front spar lower cap	WS 94	Corrosion

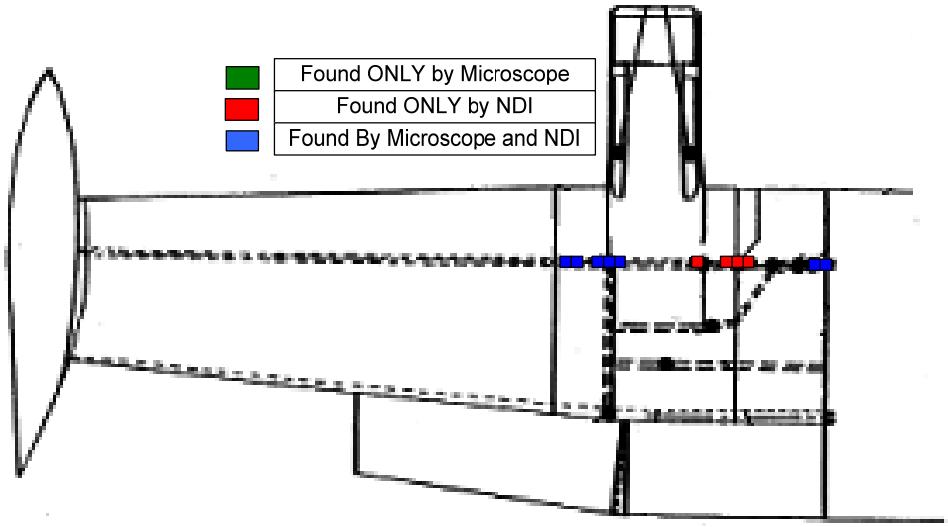

Figure 3-97. Location of Findings From Microscopic Examination and Alternative NDI on the Left Wing Upper Surface

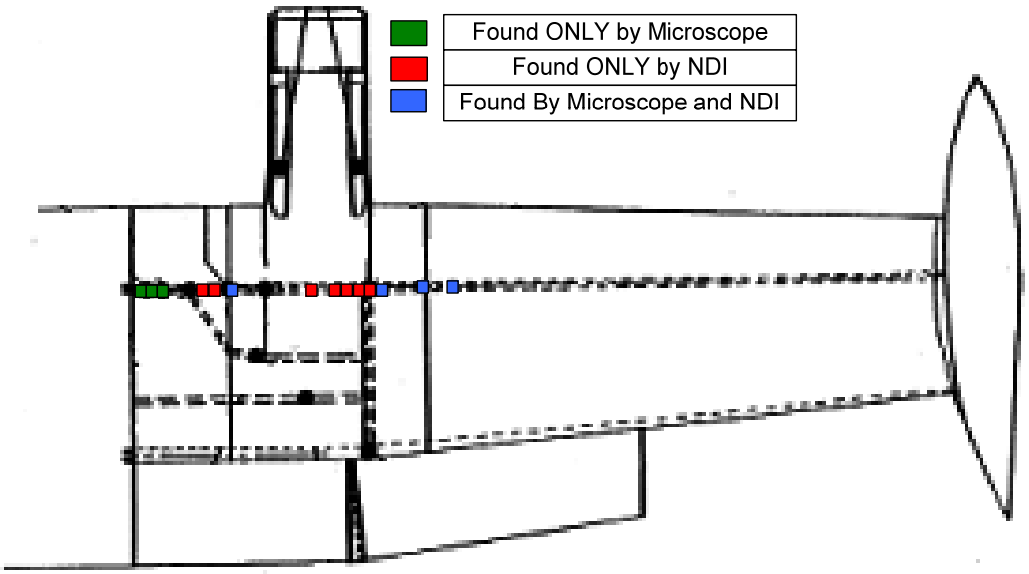

Figure 3-98. Location of Findings From Microscopic Examination and Alternative NDI on the Left Wing Lower Surface

4. AIRPLANE WIRING ASSESSMENT.

The wiring assessment for the Cessna 402A was composed of two major phases: the NDI and test phase and the destructive test phase. The NDI and test portion of the wiring assessment was composed of general visual inspections, bond resistance measurements, intrusive visual inspections, wiring insulation microscopic inspections, the Del™ Test, the excited dielectric test, insulation resistance test, circuit breaker test, and relay inspection. The destructive test portion was composed of the wet DWV test, the mandrel/wrap back test, and the dynamic cut-through test.

4.1 NONDESTRUCTIVE INSPECTION AND TESTS.

The NDI and test phase of the airplane wiring assessment included the following areas:

- General visual inspection
- In situ wiring tests
- Laboratory inspection and tests

4.1.1 General Visual Inspections.

The purpose of performing the general 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 nondestructive tests that is done to assess the maintenance activities performed on the airplane.

General visual inspections were 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.

4.1.1.1 Inspection Areas.

The airplane was divided into 16 different zones to perform the general visual inspections on the wiring components. A location code was given to each zone for recording convenience, as shown in table 4-1. Each zone then represented a wiring system or part of a wiring system of an airplane. Each zone was thoroughly observed to record the initial conditions.

Table 4-1. Location Numbers and Names

Location Number	Location Name
300-301	Right and left engine
302	Side console
303	Forward bulkhead
304	Baggage compartment
305-306	Right and left wing tip

Table 4-1. Location Numbers and Names (Continued)

Location Number	Location Name
307	Upper bulkhead
308	Tail
309-310	Right and left wing
311	Left landing gear
312	Right landing gear
313	Nose landing gear
314	Cockpit floor
315	Instrument panel

4.1.1.2 Inspection Codes.

The inspection codes were designed in accordance with the wiring inspection and practice training presentation provided by the Aging System Transportation Rule and Community. The coding was adopted so that it would be less complicated when tagging various defective locations on a particular wire. The defect codes and their description are given in table 4-2. In each zone, the defective wires and wire bundles were tagged separately for identification purposes. For example, the first wire inspected under zone 309 is tagged 309-1. Each defect or flaw observed on a wire is labeled and tagged in this manner. For example, 309-1-3 would have an area of chafing of the outer insulation, which is represented by the code 01A.

Table 4-2. Inspection Codes

Type	Serial	Defect Description	Defect Code
Wiring conditions	1	Rubbing and chafing of outer insulation	01A
	2	Cutting through outer insulation	01B
	3	Exposed shield	01C
	4	Damaged shield	01D
	5	Chafing and cutting of inner insulation	01E
	6	Exposed inner conductor	01F
	7	Damaged inner conductor	01G
	8	Heat damage	01H
	9	Fluid, chemical, and dust contamination	01J
	10	Corroded shield and conductors	01K
	11	Illegible labels	01L
	12	Others	01X
Installations	13	Inadequate clearance to structure	02A
	14	Improper wire riding on other wire bundle	02B
	15	Improper bend radius (10 times wire and bundle diameter)	02C

Table 4-2. Inspection Codes (Continued)

Type	Serial	Defect Description	Defect Code
Installations	16	Missing and deteriorated ties	02D
	17	Missing and deteriorated grommets	02E
	18	Improper clamp condition and size	02F
	19	Excessive slack and sag between clamps	02G
	20	Excessive strain on wires	02H
	21	Improper T or Y breakout	02J
	22	Repaired wires	02K
	23	Improperly labeled	02L
	24	Unused wires improperly stowed	02M
	25	Improper termination of wire (no cap)	02N
	26	Others	02X
Terminations	27	Loose and broken terminals	03A
	28	Corroded terminals	03B
	29	Improper grounding condition	03C
	30	High bonding resistance	03D
	31	Others	03X
Connectors	32	Insert damage and deterioration	04A
	33	Contact arcing and fretting	04B
	34	Missing, damaged, and loose back shells	04C
	35	Missing hardware	04D
	36	Others	04X

4.1.1.3 Recording the Wiring Conditions.

The wiring conditions of each zone, after labeling and tagging, were recorded in a relational database using Microsoft® Access®. Photographs were taken of each defective location during the inspections. All photographs were linked to the database to support the wiring conditions. The following are the important parameters recorded for clearer understanding of the conditions:

- The zone name and number
- The wire number
- The type of defects which are associated with the wiring codes
- Type of wire, for example, shielded or unshielded, threaded or nonthreaded, etc.
- Additional comments describing the wiring faults

4.1.1.4 General Visual Inspection Results.

The general visual inspections were performed on undisturbed wires. The purpose of these inspections was to determine the general condition of the wires in the right and left engine

compartment, right and left console, forward bulkhead, baggage compartment, upper bulkhead, tail, right and left wings, landing gear, cockpit floor, and instrument panel. The inspectors were looking for wiring defects such as rubbing and chafing of the outer insulation, exposed inner conductor, damaged shield, repaired wires, contamination, cracked wires, corroded terminals, improper termination, and heat damage.

4.1.1.4.1 Engine Compartments.

The wires in the engine compartments were generally found to be worn and dirty, as shown in figure 4-1. Basic flaws such as chafing and rubbing of the outer insulation, exposed inner conductor, and repaired wires were also observed. Some wires were observed to have deep cuts, as shown in figure 4-2. Some of the thread-coated insulated wires had exposed shields. Improperly stowed wires were observed in the engine compartments as well. Many of the unshielded wires were found to exhibit the rubbing and chafing condition. The inner conductors of some of the wires were cut. A cut inner conductor would result in an inoperative system. A few wires also had inadequate clearance to surrounding structure, which is a precursor to the rubbing and chafing defect. Figure 4-3 shows a ground wire exposed through the outer insulation with a corroded terminal. Some repaired wires had damaged outer covers, while some were observed to be improperly clamped or routed.

Figure 4-1. Dirty Wires in the Engine Compartment

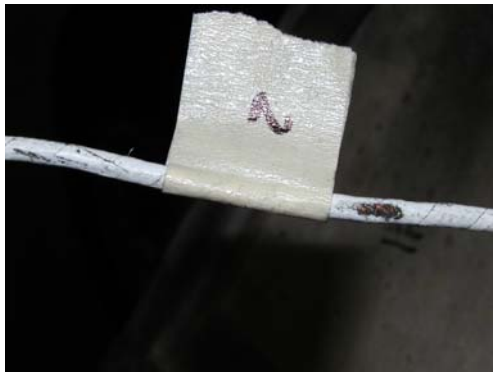

Figure 4-2. Exposed Inner Conductor

Figure 4-3. Corroded Ground Wire With Exposed Shield

4.1.1.4.2 Side Console.

Chafing and rubbing and cutting of the outer conductor occurred on a few wires in the side console. Wire routing, as shown in figure 4-4, was unorganized on the Cessna 402A. A large number of repaired wires, as shown in figure 4-5, were observed in the side console.

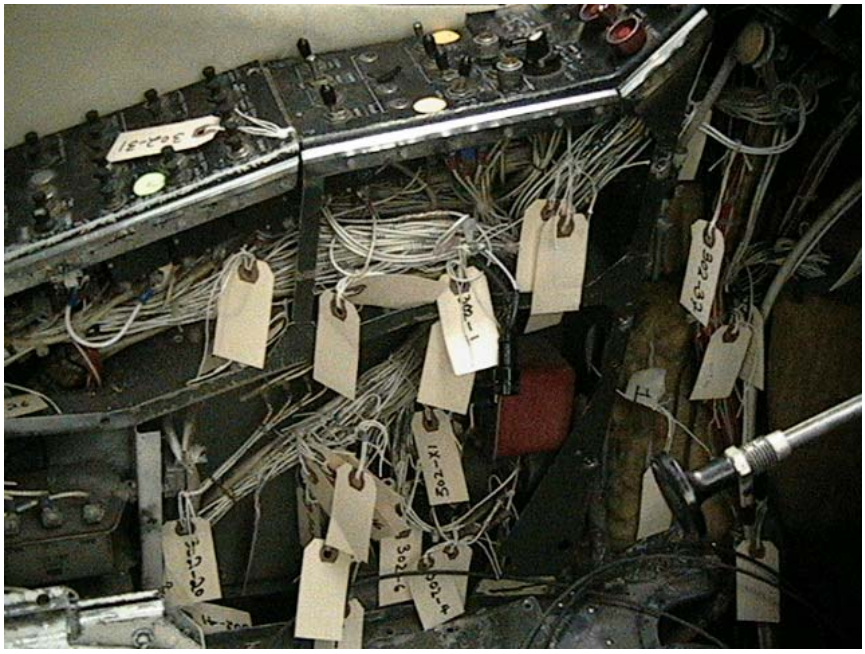

Figure 4-4. Side Console and Switch Circuit Breaker Panel

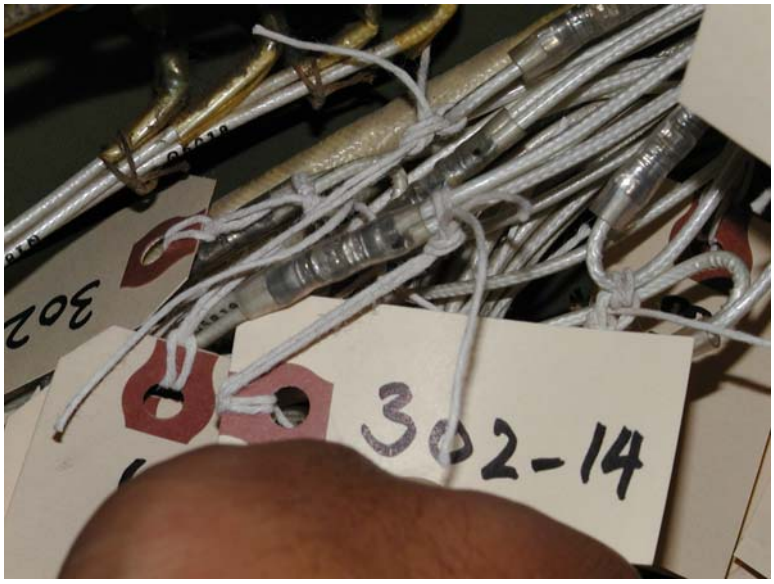

Figure 4-5. Repaired Wires Found in the Side Console

4.1.1.4.3 Nose Area.

The wires in the nose area of the airplane were generally in better condition that the areas mentioned previously. The wire routing was significantly better than in the side console. There were only a few instances of chafing/rubbing and cutting of the outer insulation on these wires. Coaxial cables, installed since manufacturing, had inadequate clearance to surrounding structure. Improper clamping and excessive slack in wires in the nose area were also observed. End terminals of wires connected to voltage converters were observed to have heat damage, as shown in figure 4-6.

Figure 4-6. Heat-Damaged Terminals

4.1.1.4.4 Baggage Compartment.

The wires in this zone were observed to be in good condition. There were very few cases of conductor exposure of wires. Very few defects were observed during the visual inspections. The terminations of some of the wires were not up to code, as shown in figure 4-7. Some wire bundles had inadequate clearance to structure and had improper clamp conditions or clamp size. Many repaired wires were also observed.

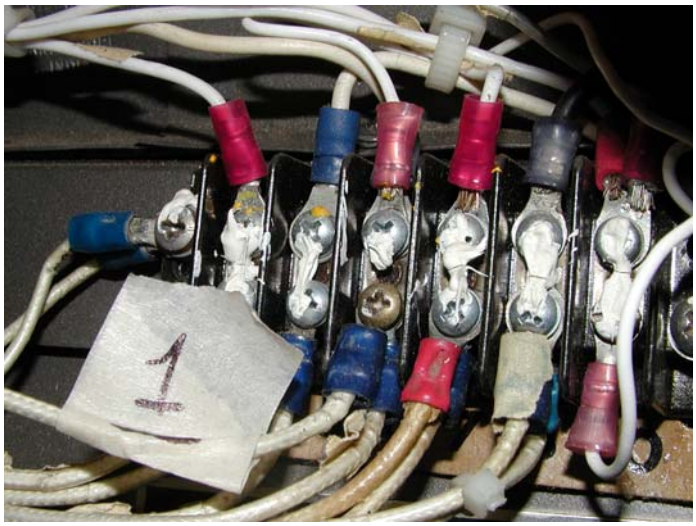

Figure 4-7. Improper Terminal Conditions

4.1.1.4.5 Left and Right Wing Tip.

A few wires in the left wing tip showed chemical contamination. A wire running from the left wing fuel pump was heat damaged. Wires having exposed inner conductors were also observed at some locations. One of the terminals was broken, as shown in figure 4-8. Rubbing and chafing of wires and shield damage was observed in the right wing tip. Some of the wires in the right wing tip were repaired with the wrong type and size of wire. A heat-damaged wire was also observed in the right wing tip.

Figure 4-8. Broken Terminal

4.1.1.4.6 Upper Forward Bulkhead.

Scattered cases of rubbing and chafing of insulation were observed around the upper forward bulkhead. Defects like missing grommets, shown in figure 4-9, and wire bundles having inadequate clearance to structure were detected. Coaxial cables, installed since manufacturing, were observed riding on other wire bundles. A broken terminal on a ground wire was also noted near the upper forward bulkhead.

Figure 4-9. Missing Grommet

4.1.1.4.7 Tail.

This zone was found to have unused wires, which were improperly stowed and terminated. The shielding of some of the wires was damaged due to corrosion and cutting of the outer insulation. A ground wire had a damaged lug and loose or broken terminal. A few wire bundles had inadequate clearance to structure and improper bend radii.

4.1.1.4.8 Wing (Right and Left).

The wire bundles in this zone were found to be improperly installed. A few wire bundles were found to be overriding other wire bundles. Repaired wires were observed to have inadequate clearance to structure. Missing grommets, loose terminals, and contamination of the shield or conductor due to chemicals or dirt were also observed. Many unused wires were improperly stowed, as shown in figure 4-10. A few of the wire bundles had deteriorated ties and broken terminals.

Figure 4-10. Unused Wires Improperly Stowed

4.1.1.4.9 Landing Gear (Left, Right, and Nose).

Only a few wires were accessible for visual inspection in the landing gear. In the left main landing gear, cutting and rubbing of the outer insulation was seen on some of the wires. Repaired wires were observed in this zone as well as wire bundles improperly riding on other wire bundles. Cracked wires, as shown in figures 4-11 and 4-12, were observed in the landing gear area as well. The wires in the right main landing gear had some instances of rubbing and chafing as well as fluid and chemical contamination. The inner conductors were also found exposed in some locations. There were also some repaired wires and missing grommets. Installation problems such as improper clamp condition and inadequate clearance to structure were observed for only one wire bundle in the landing gear. The wires in the nose landing gear area had minor flaws like missing or broken ties and rubbing and chafing of the outer insulation. There was one damaged wire due to overheating.

Figure 4-11. Cracked Wires in the Left Main Landing Gear

Figure 4-12. Cracked Wires in the Landing Gear Zone

4.1.1.4.10 Cockpit Floor.

The wires under the cockpit floor were not installed with modern procedures. The wire bundles had brittle ties due to aging as well as inadequate clearance to the structure. There was also a wire that was damaged due to overheating and chemical contamination. Missing grommets, wires improperly riding on other wires, chafing, and exposed shields were observed. A few wires were improperly clamped, as shown in figure 4-13.

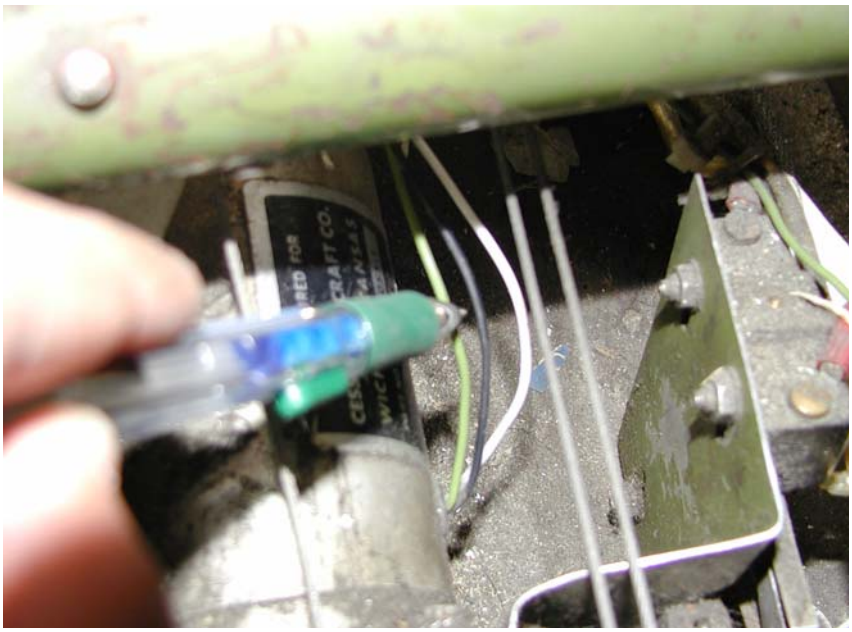

Figure 4-13. Loose Wires, Improperly Clamped

4.1.1.4.11 Instrument Panel.

The routing of wires in the instrument panel was improper, as shown in figure 4-14. In many places, wires or wire bundles were riding on the control cables and there was not enough clearance from the structure, as shown in figures 4-15, 4-16, and 4-17. A connector was found loose in the instrument panel, as shown in figure 4-18. There were many loose and repaired wires, as well as a wire that was found partly stripped, as seen in figure 4-19.

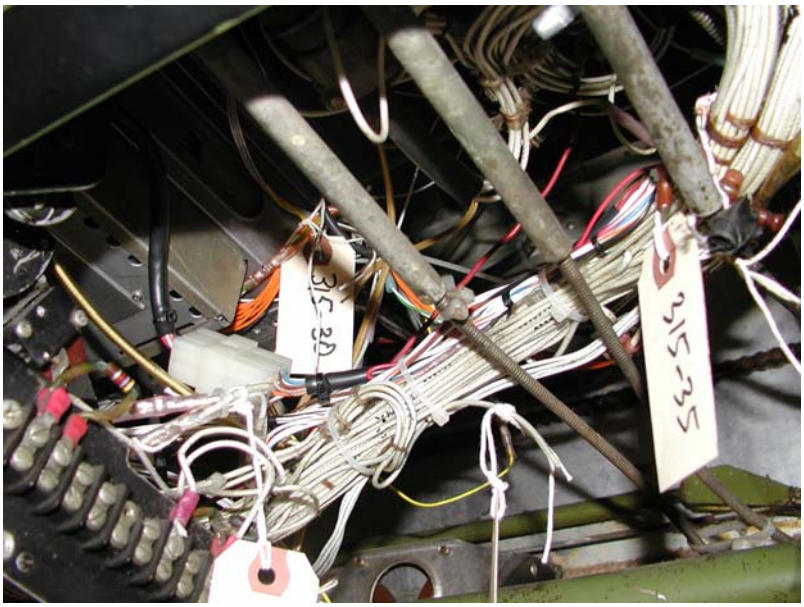

Figure 4-14. Overview of the Front Instrument Panel

Figure 4-15. Wire Bundles Riding on Control Cables

Figure 4-16. Wire Riding on Other Wires

Figure 4-17. Chafing Due to Wire Riding on Other Wires

Figure 4-18. Connector Loose in the Front Instrument Panel

Figure 4-19. A Partially Stripped Wire Found in the Instrument Panel

4.1.1.5 Analysis of General Visual Inspection Results.

Following the general visual inspections, the wiring condition defects, installation defects, and termination defects were analyzed to determine which flaws occurred most frequently.

4.1.1.5.1 Wiring Condition Defects.

Table 4-3 shows the wiring condition defects and the number of times each was encountered during the wiring general visual inspection of the Cessna 402A. One hundred and three wiring condition defects were found, with over half of them being either rubbing and chafing or cutting of the outer insulation. The most potentially harmful wiring condition defect is the exposed inner conductor, which could lead to arcing. Figure 4-20 shows the breakdown of wiring condition defects encountered on this airplane.

Table 4-3. Wiring Condition Defects

Defects	Number of Occurrences
Rubbing and chafing of outer insulation	25
Cutting through outer insulation	32
Exposed shield	11
Damaged shield	5
Exposed inner conductor	14
Heat damage	5
Fluid, chemical, and dust contamination	6
Corroded shield and conductors	5

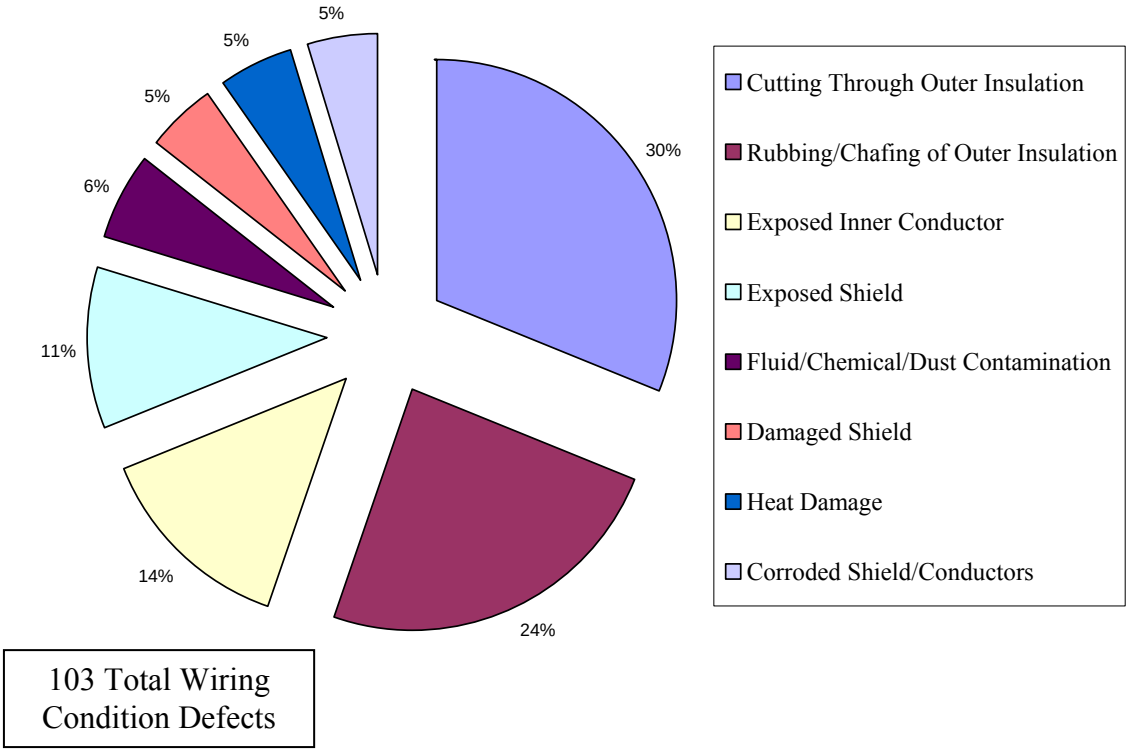

Figure 4-20. Wiring Condition Defects

4.1.1.5.2 Installation Defects.

Table 4-4 shows the wiring installation defects and the number of times each was encountered during the wiring general visual inspection of the Cessna 402A. Two hundred forty-five installation defects were found with almost a third being repaired wires and another fifth being inadequate clearance to the structure. Figure 4-21 shows the breakdown of the installation defects encountered on this airplane.

Table 4-4. Installation Defects

Defects	Number of Occurrences
Inadequate clearance to structure	47
Improper wire riding on other wires and bundles	13
Improper bend radius	11
Missing and deteriorated ties	23
Missing and deteriorated grommets	18
Improper clamp condition and size	6
Excessive slack and sag between clamps	12
Improper T or Y breakout	12
Repaired wires	75
Unused wires improperly stowed	28

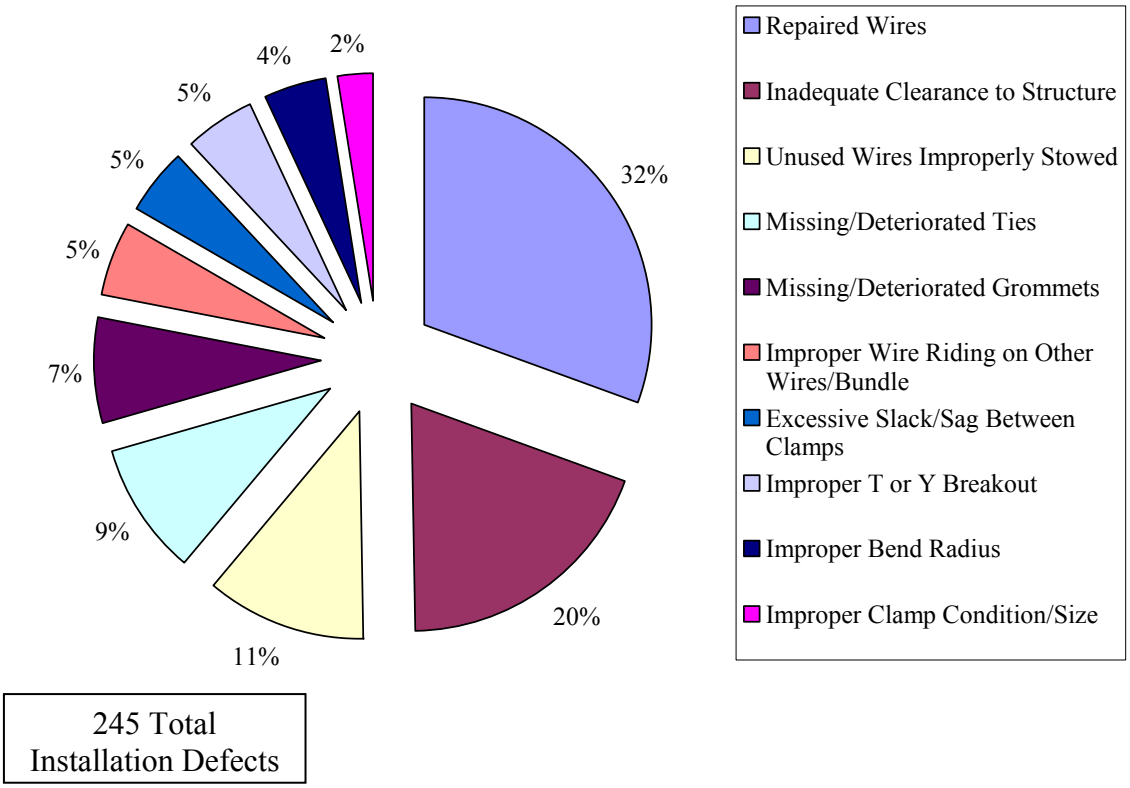

Figure 4-21. Installation Defects

4.1.1.5.3 Termination Defects.

Table 4-5 shows the termination defects observed on this airplane. Almost a third of all the termination defects were loose or broken terminals with almost another third being terminations with high bonding resistance.

Table 4-5. Termination Defects

Defects	No. of Occurrences
Loose and broken terminals	4
Corroded terminals	3
Improper grounding condition	3
High bonding resistance	4

4.1.2 In Situ Wiring Tests.

Three nondestructive tests were performed with the wires still in their service configuration in the airplane. (1) The first of these tests is the loop resistance measurement test. This test is used to verify the electrical shield integrity of the wires and can be performed without demating the connectors. The bond resistance measurement test measures the electrical resistance of closed loops by induction of currents at known voltages. (2) The Del™ Test was developed by Lectromec, which determines the level of insulation breach of a wire. (3) The excited dielectric

test was developed by CM Technologies. This test detects and locates defects that are not visible. It works on the functionality of a combination of time domain reflectometry and dissipation factor.

4.1.2.1 Loop Resistance Test.

The purpose of the loop resistance test is to verify the integrity of the electrical cable shielding without disconnecting the connectors/cables. The Loop Resistance Tester (LRT) measures the electrical resistance of the closed loops by inducing an alternating current (ac) at known voltages into the loops.

Current is induced into the loop under test by the generator in the current injection source, via the clip-on transformer (CT1), and follows through the specimen, its bonding connection with the structure, and the structure itself. Loop-induced current is detected by the clip-on transformer (CT2) and measured by the multimeter. The generator voltage present at CT1, which gives rise to loop-induced current, is displayed on the panel meter of the current injection source.

Loop Resistance (R_{loop}) is calculated by using the formula:

$$R_{loop} = \frac{\textit{Voltage (displayed on current injection source)}}{\textit{Current (measured by multimeter)}}$$

In the normal operating mode, a current of 1 ampere (A) is made to flow in the loop under test; the panel meter then displays loop resistance directly in milliohms. The equipment used to perform this test is shown in figure 4-22. The results of this test are shown in table 4-6.

Figure 4-22. Loop Resistance Tester

Table 4-6. Loop Resistance Test Results

Number	Wire No.	Wire Type	LRT Value (milliohms)	Location
1	302-X1	Wire bundle	11.2	Side console
2	303-13	Coaxial	80	Nose
3	307-6	Coaxial	80	Nose
4	307-X1	Bundle	68	Baggage compartment
5	315-X1	Wire bundle	43.2	Instrument panel
6	315-X2	Wire bundle	72.1	Instrument panel
7	315-X3	Coaxial	135.4	Instrument panel
8	315-20	Wire bundle	23.4	Instrument panel
9	315-35	Wire bundle	82.5	Instrument panel
10	315-41	Shielded wire	59.4	Instrument panel

4.1.2.2 Del™ Test.

The Del™ Test was developed by Lectromec. The test determines the level of insulation breach of a wire. To perform this test, water is sprayed onto the selected wire bundle; thereafter, a small voltage is applied to wire bundle. A meter was used to measure any leakage current through the water.

This test was performed by the personnel from Lectromec. No formal test results or conclusions were provided for analysis.

4.1.2.3 Excited Dielectric Test.

The excited dielectric test was developed by CM Technologies. This test detects and locates defects that are not visible. This is a nondestructive test and works on the functionality of a combination of time domain reflectometry and dissipation factor. Through this test, wire condition can be determined through a single set of tests (no baseline).

This test was performed by the personnel from CM Technologies. No formal results or conclusions were provided for analysis.

4.1.3 Laboratory Inspections and Tests.

The laboratory tests were performed after the teardown of the airplane. During the teardown phase, all the wires were removed from the airplane and were subjected to various tests as well as inspections. Five types of tests were performed as follows.

- Intrusive Visual Inspection (IVI)
- Wiring Insulation Microscopic Inspection
- Insulation Resistance Test
- Circuit Breaker Test
- Relay Inspection

4.1.3.1 Intrusive Visual Inspection.

The wires that were inaccessible for simple visual inspection were subjected to IVI. These tests were performed on selected segments of wire bundles after they had been removed from the airplane. These segments were:

- Cockpit floor
- Instrument panel
- Right wing
- Side console

After carefully performing the IVI, it was observed that many wires had cracks. Some wires, especially the bent wires, were found to have chafing and cracking. A few wires, especially the ones that pass through the landing gear area, were found to be dirty, as shown in figures 4-23 and 4-24. An example of a wire with cracked insulation is shown in figure 4-25, while an example of a cut wire is shown in figure 4-26.

Figure 4-23. Dirty Wires

Figure 4-24. Dirty Wires From Landing Gear Area

Figure 4-25. Cracks in Insulation

Figure 4-26. Cut Wires

Although general visual inspection showed several defects in wiring conditions, IVI also provided enhanced information of wiring defects from the areas that were inaccessible.

4.1.3.2 Wiring Insulation Microscopic Inspection.

Wiring insulation microscopic inspections were performed on selected segments of wire bundles after they had undergone IVI. Any wires that appeared to have some defect that was not visually clear on type or amount of degradation were subjected to microscopic inspection. Wire samples were chosen from different sections. Figures 4-27 through 4-32 are examples of defects found.

Microscopic inspection provided more information on the wire defects, which were otherwise visually not clear.

Figure 4-27. Wire Specimen With Exposed Shield

Figure 4-28. Wire Specimen With Broken Outer Insulation

Figure 4-29. Overheated Wire

Figure 4-30. Overheated Wire Exposing Inner Insulation

Figure 4-31. Wire With Exposed Inner Conductor

Figure 4-32. Cracks in Wires

4.1.3.3 Insulation Resistance Measurement Test.

The insulation resistance measurement test is used to determine the insulation resistance of a wire specimen. Insulation resistance is of interest in high-impedance circuits and as a measure of quality control. Changes in insulation resistance may indicate deterioration of other properties.

An insulated vessel, approximately an 8-gallon capacity, was filled with water containing 0.1% Triton X-100, a wetting agent. In this experiment, the water bath serves as one electrode. A megohmmeter was used to measure the insulation resistance. All wire specimens were immersed to within 152 mm (6 inches) of the twisted ends in the water bath, which was maintained at 23° ±5°C (73° ±9°F). Figure 4-33 illustrates the setup for the insulation resistance measurement test.

Figure 4-33. Insulation Resistance Measurement Test Setup

The insulation resistance measurement procedure is defined in the following steps:

- Wire specimen should ideally be 8.3 m (26 ft) in length. For a distance of 25 mm (1 inch), the insulation of the wire specimen is removed and the ends are twisted together.
- The wire specimens were then immersed to within 152 mm (6 inches) of the twisted ends in the water bath, which was maintained at 23° ±5°C (73° ±9°F).
- Since a measuring apparatus was required that can measure at least 40,000 MΩ or as little as 12.5 pA (for current measurements), a megohmmeter was used. The megohmmeter was setup to provide 500 Vdc with an electrification time of 60 seconds (1 minute).
- An initial resistance measurement was performed between the conductor and the water bath for the purpose of detecting nontypical values.

- The wire specimens were then soaked in the water bath (containing 0.1% Triton-X as wetting agent) for 4 hours.

- After 4 hours of soaking, the resistance between the conductor and the water bath was remeasured.

- These measurements were performed again at 500 Vdc with an electrification time of 1 minute.

- The resistance measurements were tabulated and the insulation resistance of wire specimens was calculated according to the formula explained below.

In the test procedure described above, the wire specimen length and the measured resistance is tabulated. Insulation resistance is expressed in terms of Ω-1000 ft.

This is calculated as follows:

$$\Omega\text{-}1000\ \text{ft} = (R \times L)/1000$$

where

R = measured resistance by the megohmmeter
L = length of the immersed wire specimen

Table 4-7 shows multiple wire specimens and their insulation resistance measurement readings. All wire specimens whose insulation resistance was less than 40 MΩ-1000 ft are indicated in the comments section as inappropriate.

Table 4-7. Insulation Resistance Measurement Readings

Wire Specimen	Wire Type	Length of Wire (ft)	Measured Resistance (GΩ)	Calculated Insulation Resistance (MΩ-1000 ft)	Comments
1	M5086/2	13.00	4.8	62.40	OK
2	M5086/2	11.00	3.065	33.72	Inappropriate
3	M5086/2	7.00	1.605	11.24	Inappropriate
4	M5086/2	7.33	19.195	140.70	OK
5	M5086/2	14.67	3.388	49.69	OK
6	M5086/2	7.33	8.405	61.61	OK
7	M5086/2	14.33	2.001	28.67	Inappropriate
8	M22759/8	4.50	64.9	292.05	OK
9	M5086/2	17.67	8.71	153.87	OK
10	M5086/2	7.50	11.87	89.03	OK
11	M22759/8	14.00	34.62	484.68	OK

Table 4-7. Insulation Resistance Measurement Readings (Continued)

Wire Specimen	Wire Type	Length of Wire (ft)	Measured Resistance (GΩ)	Calculated Insulation Resistance (MΩ-1000 ft)	Comments
12	M22759/8	13.33	23.50	313.26	OK
13	M5086/2	10.00	9.925	99.25	OK
14	M22759/8	6.50	38.38	249.47	OK
15	M22759/8	6.42	48.95	314.26	OK
16	M5086/2	13.33	32.46	432.69	OK
17	M22759/8	7.50	1.73	12.98	Inappropriate
18	M5086/2	14.00	37.335	522.69	OK
19	M5086/2	2.30	43.15	99.25	OK
20	M5086/2	5.00	1.305	6.53	Inappropriate
21	M5086/2	5.33	0.978	5.21	Inappropriate
22	M5086/2	3.25	4.75	15.44	Inappropriate
23	M5086/2	13.66	7.54	103.00	OK
24	M5086/2	6.00	24.655	147.93	OK
25	M5086/2	16.166	12.325	199.25	OK
26	M5086/2	16.333	22.695	370.68	OK
27	M5086/2	14.05	19.23	270.18	OK
28	M5086/2	14.667	11.765	172.56	OK

Insulation resistance measurements specify some requirements, which are not always practical for small airplanes. As per the specifications mentioned in ASTM D 3032, the wire specimen required for this test must be 26 feet. However, there is not a single wire in the Cessna 420A which is 26 ft in length. The average wire length found was 7 or 8 feet. This shorter wire specimen renders the test unreliable. The interpolation of these wire specimens (length less than 26 ft) is not appropriate and does not produce accurate results. As can be seen from table 4-7, most of the wires that show inappropriate values are the wires whose length is very small compared to the requirements.

Apart from the wire specimen requirements, there are other significant factors that were also found to be important. Some of these factors include soak time and temperature. A study was performed on the insulation resistance measurement test. It was found that the insulation resistance of a wire increases at lower soak times and low temperatures. The temperature required as per the ASTM D 3032 specification varies from 73° ±9°F, which means the temperature range is from 64° to 82°F. This range is relatively wide. When the insulation resistance measurement test is performed at lower temperatures, the readings have more instability than when the test is performed at higher temperatures. Also at higher temperatures, the insulation resistance tests are more robust to changes in wire lengths as well as soak times.

4.1.3.4 Circuit Breaker Test.

Circuit breakers are specifically designed to protect airplane wiring, where a circuit breaker will trip off when the current through it exceeds its rating. The circuit breakers were removed from the airplane and tested in the laboratory to verify that current trip times met specifications. The circuit breakers were not cycled before laboratory testing. The procedure is described in the following steps:

- The rating of the circuit breaker was noted per the wiring service manual [1].
- The rating of the circuit breaker, which was actually present in the plane, was also noted.
- Any discrepancies found between the actual existing circuit breaker and the wiring manual circuit breaker was noted.
- The circuit breakers were tested at two levels: 200% and 300% rating of the circuit breaker was calculated as per the wiring service manual circuit breaker rating.
- 200% rated current was passed through the circuit breaker and the time required for the breaker to trip off was again recorded. The circuit breaker was then reset and allowed to cool.
- 300% rated current was passed through the circuit breaker and the time required for the breaker to trip off was recorded.
- All the measurements were tabulated and compared with the circuit breaker specifications.

Figures 4-34 and 4-35 shows the circuit breaker in closed and open positions, respectively. The circuit breaker is in the open position after the rated current is passed and is closed under normal conditions. The circuit breakers were manufactured by Wood Electric Co. Inc. As per the specification sheet from the manufacturer, at 200% of the rated current, the circuit breaker should trip within 22 seconds, and at 300% of the rated current, the circuit breaker should trip within 6 seconds.

Figure 4-34. Circuit Breaker in the Closed Position

Figure 4-35. Circuit Breaker in the Open Position

Table 4-8 lists the circuit breaker test results. Trip times listed under test 1 are the results from the circuit breakers after the first time they were tested. The test was repeated two more times, with the results listed under tests 2 and 3. The purpose of the second and third test was to find if the cycling of the circuit breakers has any impact on circuit breaker trip times.

Table 4-8. Circuit Breaker Test Results

Circuit Breaker	Model No.	Specified Current	Applied Current 200%, 300%	Trip Time, Test 1 (sec)	Trip Time, Test 2 (sec)	Trip Time, Test 3 (sec)	Remarks
1	S1232-507	7.5 A	15 A, 22.5 A	14, 6	14, 6	--	Passed
2	S1232-510	10 A	20 A, 30 A	10, 3	10, 3	11, 4	Passed
3	S1232-507	7.5 A	15 A, 22.5 A	22, 6	23, 7	22, 6	Passed
4	S1232-507	7.5 A	15 A, 22.5 A	11, 5	11, 5	--	Passed
5	S1232-507	7.5 A	15 A, 22.5 A	40, 7	38, 7	17, 5	Failed 1 and 2 Passed 3
6	S1232-507	7.5 A	15 A, 22.5 A	58, 8	54, 8	16, 4	Failed 1 and 2 Passed 3
7	S1232-507	7.5 A	15 A, 22.5 A	11, 4	11, 4	--	Passed
8	S1232-505	5 A	15 A, 22.5 A	160, 8	148, 8	7, 3	Failed 1 and 2 Passed 3
9	S1232-507	7.5 A	15 A, 22.5 A	2, 1.5	2, 1.5	--	Passed
10	S1232-510	10 A	20 A, 30 A	16, 4	16, 4	15, 5	Passed
11	S1232-510	10 A	20 A, 30 A	10, 4	10, 4	12, 4	Passed
12	S1232-507	7.5 A	15 A, 22.5 A	12, 4	12, 4	12, 4	Passed
13	S1232-507	7.5 A	15 A, 22.5 A	11, 4	11, 4	11, 4	Passed
14	7277-2-3	3 A	6 A, 9 A	11, 2	11, 2	--	Passed
15	S1232-510	10 A	20 A, 30 A	187, 9	10, 4	10, 4	Failed 1 Passed 2 and 3
16	S1232-510	10 A	20 A, 30 A	228, 11	9, 4	11, 4	Failed 1 Passed 2 and 3
17	S1232-507	7.5 A	15 A, 22.5 A	17, 6	17, 6	--	Passed
18	S1232-507	7.5 A	15 A, 22.5 A	17, 6	17, 6	--	Passed
19	P/N 111-201-101	--	--	--	--	--	Open Circuit
20	7277-2-3	3 A	6 A, 9 A	10, 2	10, 2	--	Passed
21	S1232-507	7.5 A	15 A, 22.5 A	15, 5	15, 5	15, 4	Passed

There were a total of 21 circuit breakers tested. As can be seen from table 4-8, one breaker (4.8%) did not function and would not conduct any current, five circuit breakers (23.8%) did not meet trip time specifications after the first test, three (14.3%) did not meet trip time specifications after the second test, and all functioning circuit breakers met specifications by the

third test. As seen in the table, the trip times of the functioning circuit breakers, that first failed, changed drastically as the breakers were cycled. The trip times of the circuit breakers that were functioning barely changed, if at all. This evidence suggests that circuit breakers should be mechanically cycled periodically to maintain proper functionality.

4.1.3.5 Relay Inspection.

The relays available were opened for inspection and were also observed under a microscope. Microscopic pictures were also taken. The relays were found to have pitting. Occurrences of pitting in relays are a normal phenomenon due to the nature of the function of relays to open and close.

4.2 DESTRUCTIVE LABORATORY WIRING TESTS.

Destructive laboratory wiring tests were performed after the teardown of the airplane when all the wires had been removed and separated for laboratory testing. The destructive laboratory tests of the wires were performed on the same selected wire bundle on which nondestructive tests were performed. These tests include:

- Wet DWV test
- Mandrel bend/wrap back test
- Dynamic cut-through test

4.2.1 Wet DWV Test.

The DWV test provides a method to determine insulation integrity following any type of performance test. The DWV test was performed on the same wire segments used in the insulation resistance measurement test. This test is also used to determine whether exposure to environmental test conditions has reduced the breakdown strength below some prescribed level.

The test specimen consists of an insulated wire of the length required for environmental exposure. The insulation was removed for a distance of 25 mm at each end of the wire before the wires were twisted together. All specimens were then subjected to insulation resistance measurement test. Any specimen whose insulation resistance was less than 1 MΩ was discarded and was replaced with another specimen.

The DWV procedure is explained below.

- The test specimen was immersed to within 51 mm (2 inches) of the twisted ends in the water (containing 5% NaCl) solution, as shown in figure 4-36.

- The resistance between the conductor and the water solution was measured at 500 (±10%) Vdc to detect gross flaws. The apparatus (megohmmeter) described in section 4.1.3.3 was used for these measurements.

- All specimens that passed the insulation resistance test were subjected to 4 hours of soaking in the water bath.

- After 4 hours, a High Potentiometer (HiPOT), shown in figure 4-37, was used, which was setup to ramp up in 1 minute up to 2500 Vac and then remain at that level for 300 seconds.

- The current rating of the HiPOT was kept at 20 mA to detect the flaw current in case it goes high. Although the rating renders any current above 10 mA as a failure, it was intentionally kept high to spot the exact amount of current flow more than 10 mA.

- If the wire was found faulty, the HiPOT drew current in excess and then shuts off automatically, indicating a failure in the wire. See figure 4-38.

- The voltage and the time of the failure were recorded and tabulated.

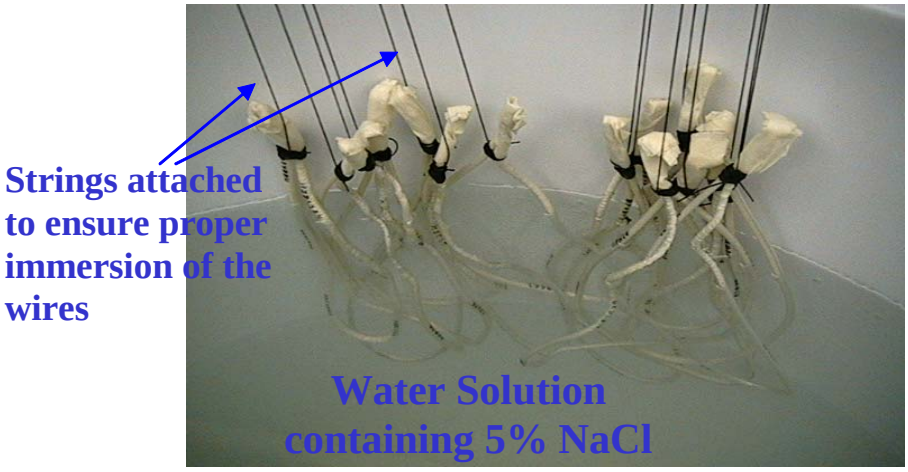

Figure 4-36. Wire Specimen Immersed to Within 2 Inches of Twisted Ends

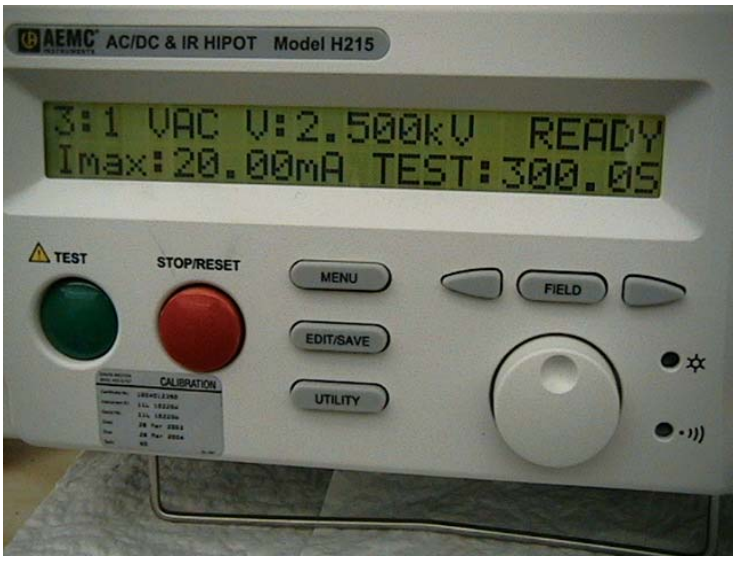

Figure 4-37. HiPOT Set to Ramp Up to 2500 Vac in 1 Minute and Electrification Time of 300 Seconds

Figure 4-38. HiPOT Indicating a Failure of a Specimen Tested

Table 4-9 lists the recorded readings of the DWV test.

Table 4-9. Wet DWV Test Results

Wire Specimen	Wire Type	Length of Wire (ft)	Calculated Insulation Resistance (MΩ-1000 ft)	DWV (Measured Current in mA)	Comments
1	M5086/2	13.00	62.40	1.56	Passed
2	M5086/2	11.00	33.72	1.57	Passed
3	M5086/2	7.00	11.24	1.02	Passed
4	M5086/2	7.33	140.70	1.00	Passed
5	M5086/2	14.67	49.69	1.60	Passed
6	M5086/2	7.33	61.61	0.98	Passed
7	M5086/2	14.33	28.67	>20	Failed
8	M22759/8	4.50	292.05	0.58	Passed
9	M5086/2	17.67	153.87	2.01	Passed
10	M5086/2	7.50	89.03	0.96	Passed
11	M22759/8	14.00	484.68	1.37	Passed
12	M22759/8	13.33	313.26	1.55	Passed
13	M5086/2	10.00	99.25	1.41	Passed
14	M22759/8	6.50	249.47	1.04	Passed
15	M22759/8	6.42	314.26	1.13	Passed
16	M5086/2	13.33	432.69	1.56	Passed

Table 4-9. Wet DWV Test Results (Continued)

Wire Specimen	Wire Type	Length of Wire (ft)	Calculated Insulation Resistance (MΩ-1000 ft)	DWV (Measured Current in mA)	Comments
17	M22759/8	7.50	12.98	>20	Failed
18	M5086/2	14.00	522.69	2.02	Passed
19	M5086/2	2.30	99.25	0.57	Passed
20	M5086/2	5.00	6.53	0.78	Passed
21	M5086/2	5.33	5.21	1.09	Passed
22	M5086/2	3.25	15.44	>20	Failed
23	M5086/2	13.66	103.00	>20	Failed
24	M5086/2	6.00	147.93	1.61	Passed
25	M5086/2	16.166	199.25	2.86	Passed
26	M5086/2	16.333	370.68	2.91	Passed
27	M5086/2	14.05	270.18	2.13	Passed
28	M5086/2	14.667	172.56	2.44	Passed

The wet DWV test was performed successfully on the wire specimens. The test results indicated 15% of the specimens failed the test. Some other aspects were observed about the test. Since this is a high potential test, if the insulation of any wire specimen is not properly removed (for twisting the ends together), probable arcing takes place at that location. This also causes the test to indicate a failure condition, albeit the test results displayed do not contain any such cases. Since no wires were 26 feet or more in length, as required by the standard, they could not be reliably tested for DWV. Although the test indicated failures, the specimen failed only the test, but it did not mean it failed to function. It might have been a functioning wire but may have failed the test since the DWV test provides a method to determine insulation integrity and does not determine the functionality of the wires.

4.2.2 Mandrel Bend/Wrap Back Test.

The mandrel bend/wrap back test is used to determine whether a specimen will crack when wrapped upon itself or around a mandrel. This test has been used for process control purposes to determine the degree of sintering of the insulation.

Twelve-inch wire specimens were used for the wrap back test. All wires were 10 through 30 gage. The specimen was bent back on itself at the mid-portion on a radius not less than the radius of the wire. One end of the specimen was wound tightly around the other end as a mandrel for a total of four close turns, see figure 4-39.

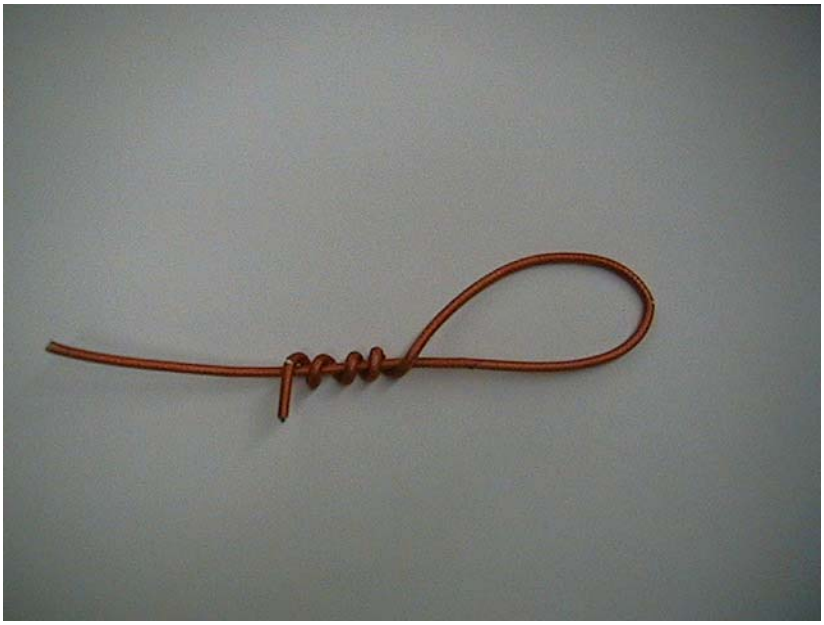

Figure 4-39. Mandrel Bend/Wrap Back Test Specimen

Many of these specimens were prepared for the test. All specimens were placed in a 203° ±3.6°F temperature oven for 24 hours. Figures 4-40 and 4-41 show the temperature chamber and gauge.

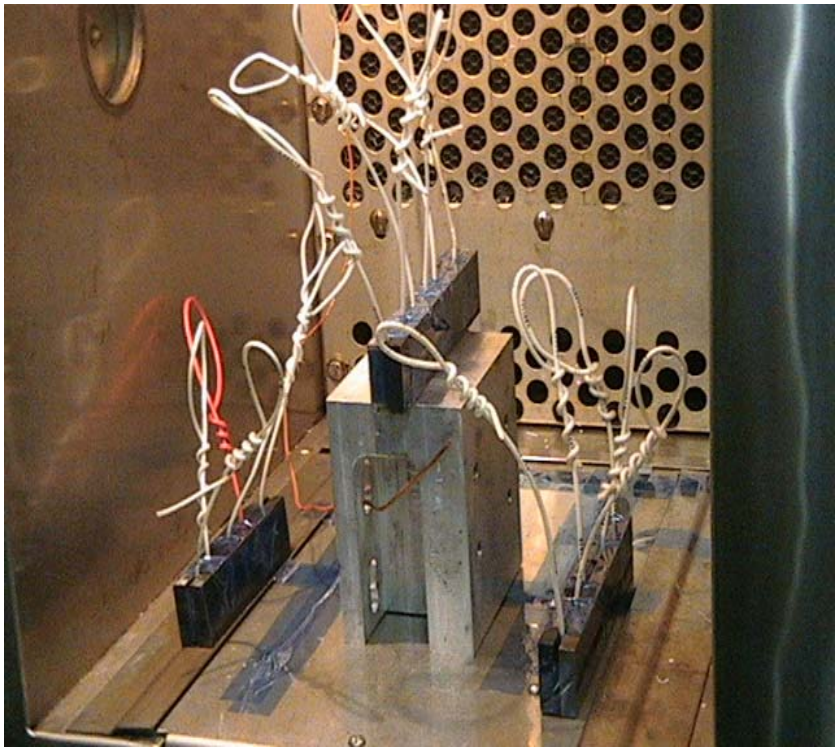

Figure 4-40. Multiple Wire Specimens Placed in a Temperature Chamber

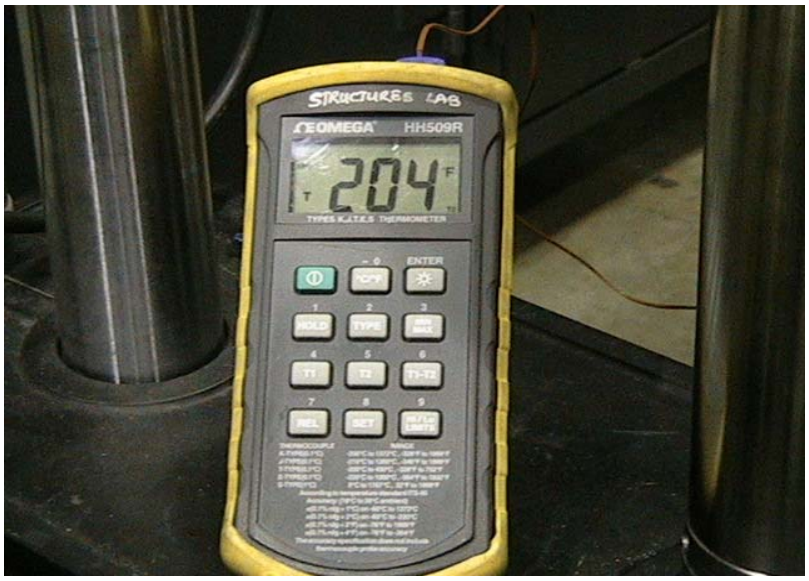

Figure 4-41. Temperature Gauge

After removal from the oven, the specimens were kept in a desiccator to cool. The desiccator absorbs the humidity from the atmosphere, allowing drier cooling conditions. Figure 4-42 shows a wire specimen placed in a desiccator. After 2 hours of cooling, the specimen was examined visually, without the aid of magnification, for cracks. The entire specimen, shown in figure 4-43, with and without cracks, was subjected to the DWV test. The procedures for the DWV test remain the same as described in section 4.2.1.

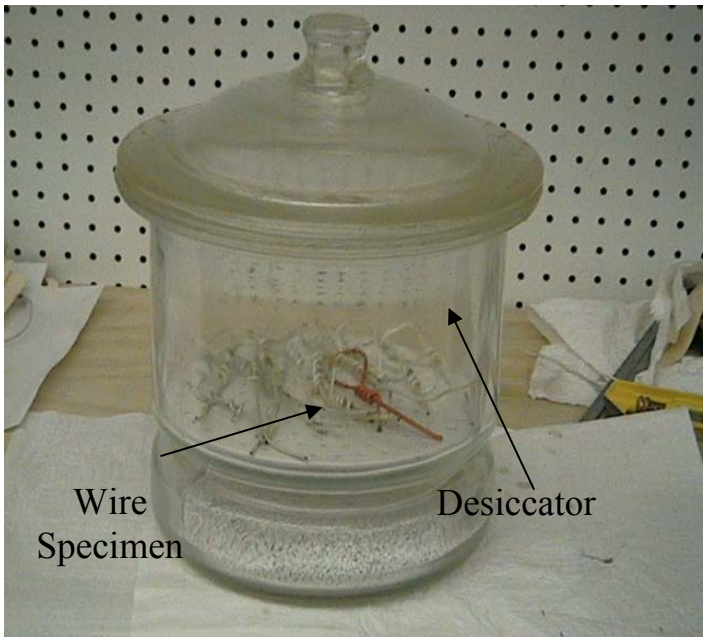

Figure 4-42. Desiccator Cooling Wire Specimens in a Dry Atmospheric Condition

Figure 4-43. Wire Specimen Showing Cracks After Unwinding

Table 4-10 lists the results of the mandrel bend/wrap back test performed on the wire specimens.

Table 4-10. Mandrel Bend/Wrap Back Test Results

Wire Specimen	Visual Inspection After Cooling	Insulation Resistance (GΩ)	DWV Test Results
A	No cracks	287.9	Passed (I_{max} = 0.25 mA)
B	No cracks	387.3	Passed (I_{max} = 0.29 mA)
C	Slight cracks	94.4	Passed (I_{max} = 0.29 mA)
D	No cracks	574	Passed (I_{max} = 0.25 mA)
E	Slight cracks	244	Failed (after electrification for 147 seconds at 2.5 KV) (location of the wire specimen: side console)
F	Slight cracks	111.6	Passed (I_{max} = 0.24 mA)
G	No cracks	65	Passed (I_{max} = 0.31 mA)
H	No cracks	805	Passed (I_{max} = 0.31 mA)

Only 12.5% of the wire specimens failed the mandrel bend/wrap back test. The results also indicate that when the wire is unwound after being subjected to temperature conditions, many wires do not produce cracks, thereby indicating good insulation characteristics. Around 62.5% of the wires showed no cracks after unwinding. The wires that did show cracks only showed minor cracks, which indicate good insulation robustness.

4.2.3 Dynamic Cut-Through Test.

The dynamic cut-through test is designed to measure the ability of wire insulation to resist being cut by a sharp edge, such as the edge of a structure or a tool. A cut in the insulation may lead to potential shorting due to an exposed conductor. The force required to cut the insulation is dependent on several factors, including the modulus of the insulation material and the physical toughness. Hard materials tend to require more force than soft materials to cut through.

- Dynamic cut-through blade: For the dynamic cut-through test, a standard cutting edge is required. The standard cutting edge used was a 0.508-mm-diameter sewing needle backed up by a 90° support with a half cylindrical groove machined to conform to the needle. Figure 4-44 shows the actual cutting edge and the cutting blade.

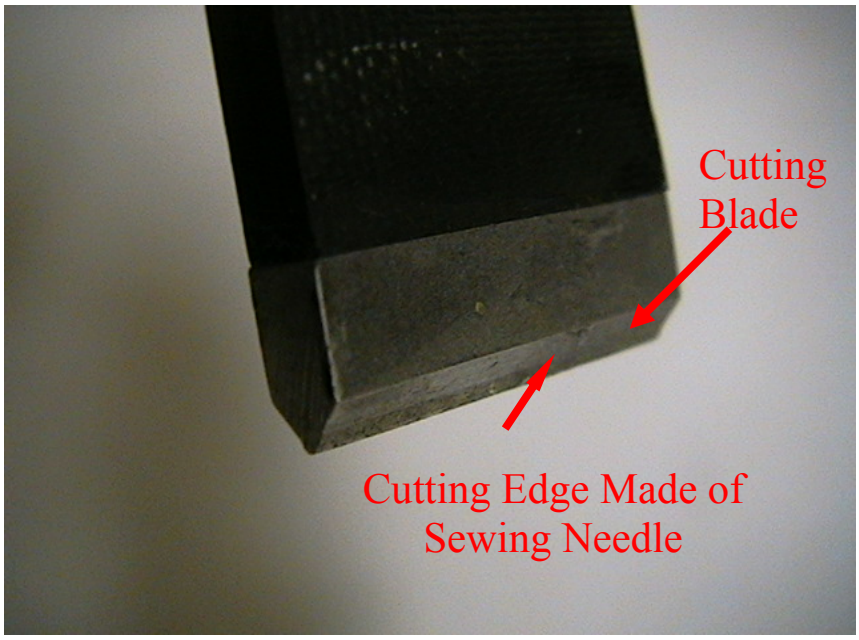

Figure 4-44. The Cutting Blade and the Sewing Needle Edge

- Dynamic cut-through wire specimen: The wire specimen used for the test was 450 mm of insulated wire. The outer insulation was removed since the test is determining the cut-through force of the inner insulation only. Sufficient insulation from one end of the specimen was removed for connecting it to the detection circuit. Figure 4-45 shows a wire specimen that was used for the test.

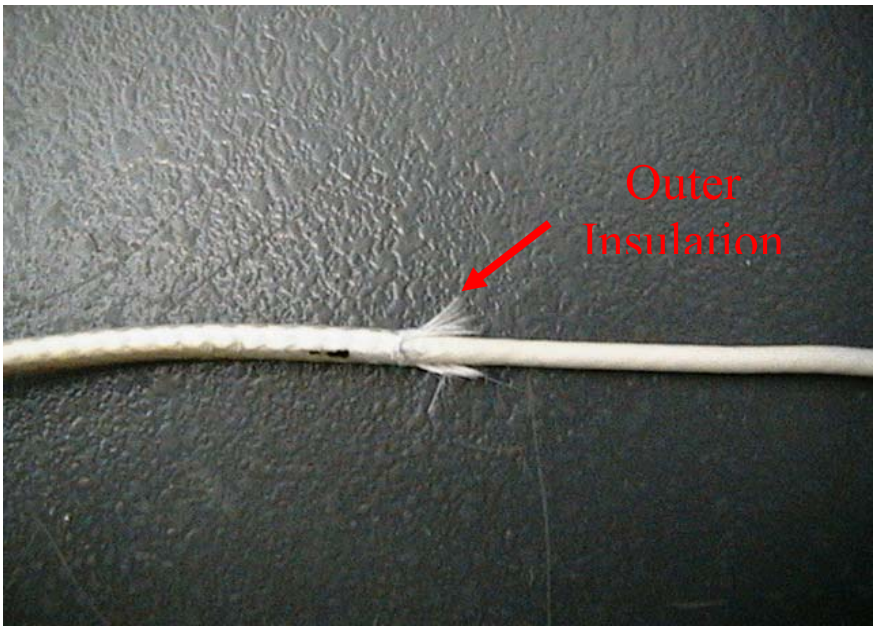

Figure 4-45. Wire Specimen Used for Dynamic Cut-Through Test

- Detection Circuit: Since the test was performed only to cut through the insulation and not the conductor, a detection circuit was designed. This circuit detects the contact of the cutting edge blade with the inner conductor and gives a visual light indication signaling the test personnel to stop the cutting procedure and record the force required to cut through the insulation. Figure 4-46 shows the detection circuit.

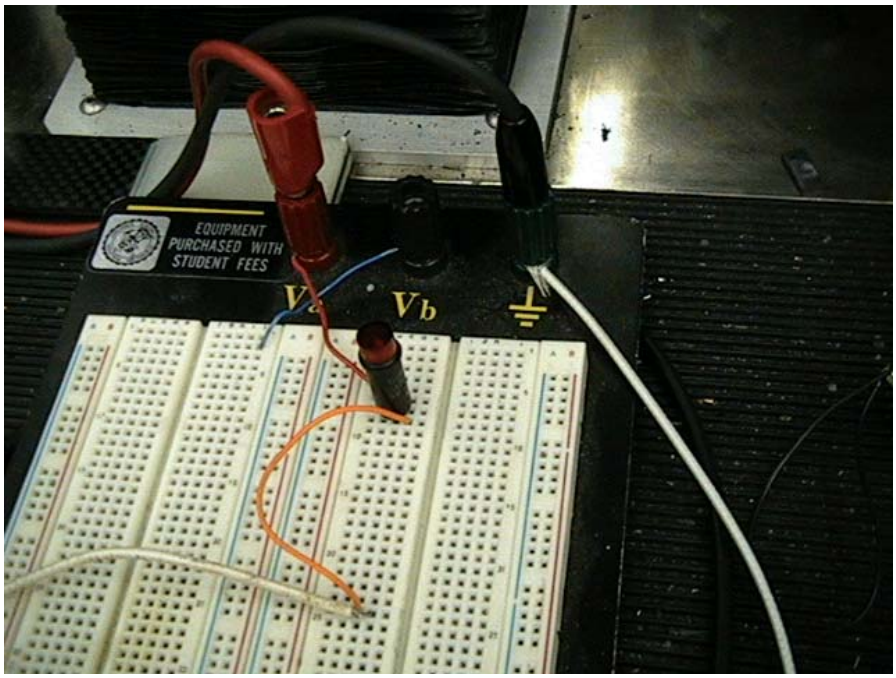

Figure 4-46. Detection Circuit

Numerous specimens were prepared for the dynamic cut-through test with their outer insulation removed. The specimens were individually placed on the support anvil and the cutting edge was oriented perpendicular to the axis of the specimen. Figures 4-47 and 4-48 show the support anvil and wire specimen, respectively.

Figure 4-47. Dynamic Cut-Through Test Machine Support Anvil

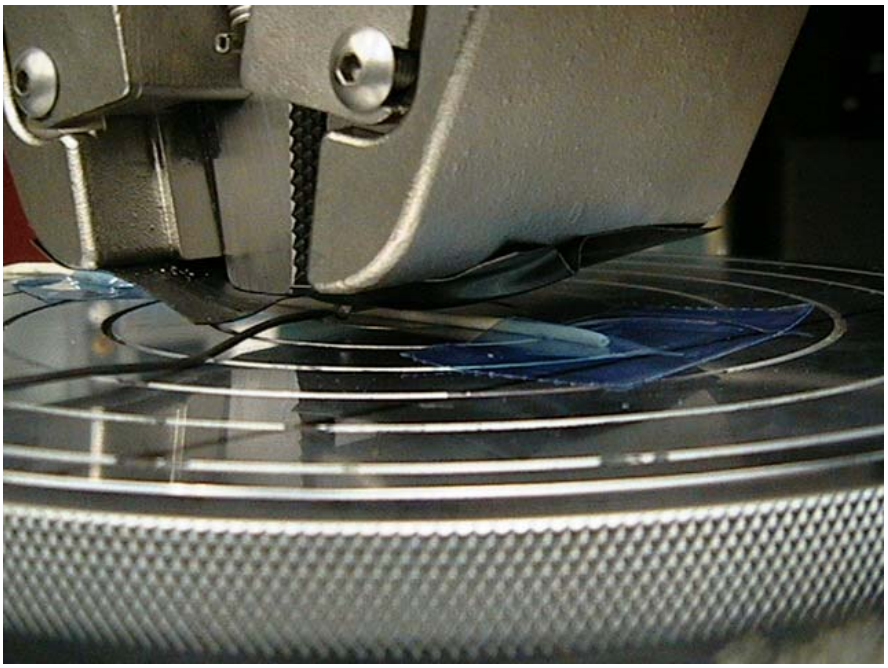

Figure 4-48. Dynamic Cut-Through Test Machine Blade and Wire Specimen Oriented Perpendicular to Each Other

The cutting edge was forced at a constant rate of 5 mm/min until contact with the conductor was achieved. The detection circuit signaled blade contact with the conductor. The test specimen and the test apparatus, particularly the support anvil and the cutting fixture, were maintained at a controlled temperature of 73° ±4°F at the time of the test. At least eight tests were performed on each wire specimen, with the specimen being moved forward 25 mm minimum and rotated 90° always in the same direction before subsequent tests. The cut-through force is calculated to be the mean average of all eight readings. Eight sample specimens were tested. Figure 4-49 shows the entire assembly of the dynamic cut-through test.

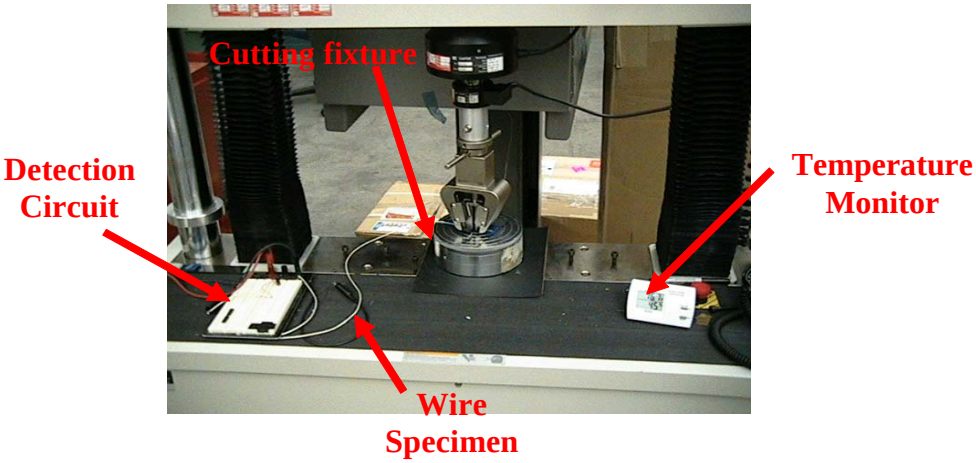

Figure 4-49. Dynamic Cut-Through Test Assembly

Each wire specimen requires a different amount of force to cut through the insulation. Even each specimen shows a variation in the force required to cut through its insulation. A graph of the variation of the force required on each specimen is plotted in figure 4-50.

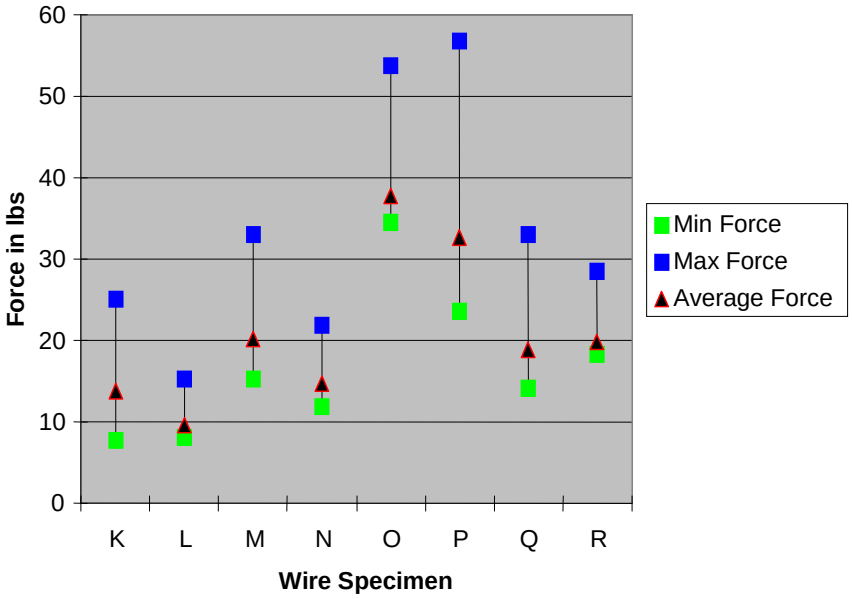

Figure 4-50. Dynamic Cut-Through Force Variations

Figure 4-50 reflects the minimum, maximum, and average force that was applied to each specimen to cut through the insulation. As shown in the figure, there is a variation of force required for each specimen. Table 4-11 shows the force required to cut through the insulation of a wire. Multiple specimens were used and each specimen was subjected to the force applied multiple times.

Table 4-11. Dynamic Cut-Through Test Results

Wire Sample	K	L	M	N	O	P	Q	R
Force in pounds	8.898	15.375	27.208	18.699	47.060	23.589	14.313	25.597
	14.536	13.017	15.204	14.394	43.775	48.231	16.877	22.207
	7.769	8.180	19.377	19.648	43.342	56.796	26.963	28.577
	20.382	10.983	28.219	17.396	53.772	33.929	33.105	22.379
	17.207	8.912	18.76	21.900	38.810	30.834	14.058	19.461
	25.024	10.215	32.98	14.378	34.618	40.360	23.024	18.350
	16.501	9.916	19.243	11.849	39.997	26.929	22.053	21.603
Average force	13.789	9.564	20.123	14.783	37.671	32.583	18.799	19.771

5. 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 on two aged airplanes (both Cessna 402 models) used in commuter service. The intent of the program was 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 document provides findings in a summary report on the teardown evaluation of a 1969 Cessna 402A model airplane in support of the research program. 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). Specific observations are made regarding findings discovered during the teardown evaluation on the particular airplane selected.

The destructive evaluation of the commuter-class airplane was separated into three main tasks: (1) inspection of the airframe and airplane systems, (2) teardown examination of the airframe and airplane systems, and (3) assessment of the airplane wiring. In the inspection phase, three subtasks were performed: a survey of the airplane maintenance records, visual inspections of the airframe and airplane systems, and supplemental airframe inspections. The teardown examination involved disassembly of the airframe and major airplane sections, inspection of airplane systems' components, an inspection of primary airplane structure using alternative NDI techniques, and microscopic examination of critical structural areas. As part of the destructive evaluation, inspections and testing were also performed on the airplane wiring to assess the condition and degradation of electrical wiring in small airplanes and to evaluate maintenance procedures.

5.1 SUMMARY OF INSPECTION PHASE.

During 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 visual inspection of the airframe and airplane systems along with the supplemental airframe inspections determined the condition of the airplane based on normal maintenance activity.

5.1.1 Maintenance Record Review.

The first airplane selected for destructive evaluation was a 1969 Cessna 402A model (tail number N812BW). This airplane was registered under four different tail numbers (N4686Q, N3BL, N300UV, and N812BW) and was operated by ten different charter services (Green Bay Aviation, Green Bay, WI; Aero Electronics, Memphis, TN; Airline Services; Air Midwest, Wichita, KS; Cruse Aviation, Houston, TX; Bee Line Airlines, Houston, TX; Universal Airways, Gulfport, MS; Clary Aircraft Services, Houston, TX; Air Vegas, Las Vegas, NV; and Sunshine Airlines). Primarily used for touring the Grand Canyon from Las Vegas, the airplane accumulated 19,698.9 hours. A 7-year gap existed in the maintenance logs from May 1991 to May 1998, and no airframe hours were recorded from May 1984 to May 1998. The service bulletin performed on the airplane that was of concern was MEB71-2, Wing Rib Inspection, on July 2, 1971. The AD with the most concern was AD 79-10-15, Wing Spar Inspections, on

February 11, 1981 and December 1, 1997. The major repair and alteration that was of major concern was installed doublers on right wing lower front spar cap from stations 75.24 to 65.99 on March 6, 1991.

5.1.2 Visual Inspections.

The visual airframe inspections were performed with the intent of finding every detectable flaw on the airplane, which led to a large number of documented flaws. The results of these inspections were reviewed by licensed airframe mechanics to determine which findings were noteworthy. Most of these flaws were deemed minor, meaning that no further maintenance action was required. With an aged airplanes, flaws such as minor cracks, scratches, dents, paint chips, and slight corrosion were to be expected and posed no immediate threat to the safety of the airplane. These flaws were recorded, but were not determined to be serious. In addition to dents and areas of missing paint, slight to moderate corrosion was noted on a majority of the screws of the airplane. Also, it was noted that some of the disposable parts, such as seals and hoses, were due to be replaced.

A few of the defects noted by the inspectors were determined to be more serious in nature. Two 2.5-inch cracks on the leading edge of each horizontal stabilizer were discovered during these inspections. The cracks were further investigated and found to be body filler that cracked where the deicing boots were removed. One loose nut was found during inspection of the flaps. The loose nut could possibly be attributed to the initial disassembly of the airplane prior to transportation. Broken gear teeth were observed on the left fuel selector valve.

5.1.3 Supplemental Inspections.

The inspections called out in the Supplemental Inspection Document (SID) for the Cessna 402A airplane under investigation were performed at the Aging Aircraft Research Laboratory. The SID inspections consisted of the visual test, eddy-current test, fluorescent liquid penetrant test, and magnetic particle test methods. The Cessna 402A landing gear, wings, stabilizers, and fuselage were each inspected using one or more of these inspection methods. Two certified airframe inspectors from Cessna and Cape Air performed the visual supplemental inspections. All the other SID inspections were performed by a Cessna Level III NDI examiner, a National Institute for Aviation Research (NIAR) Level II inspector, and three Kansas Air National Guard inspectors.

During SID visual inspections, the following indications were identified:

- Two rivets needed replacing on an aileron hinge
- A patched rib in the rudder structure needed replacing
- Elongated holes on a lower rudder fitting
- A cracked doubler located at the rudder top hinge fairing
- A bonded angle in the rudder structure not attached at one end
- Corrosion on the inside and outside of the rudder torque tube
- Corrosion on the inside and outside of the elevator torque tube
- Several locations of cherry max and hi-loks needed to be replaced by solid rivets

- A patched crack on the horizontal stabilizer leading edge
- A crack in the left side inboard rib of the horizontal stabilizer
- Crack in skin near inboard elevator attach fitting

During SID liquid penetrant inspections, the following indication was detected:

- Three cracks in the nose gear steering bellcrank

During SID magnetic particle inspections, the following indication was detected:

- Two cracked welds, one on each main landing gear torque tube

During SID bolthole eddy-current inspections, the following indications were detected:

- Damage to two upper cap left side holes on the vertical stabilizer rear spar attach at WL 108.38
- One crack indication on the lower cap of the left wing rear spar at WS 97
- Damage to all holes on the lower cap of the right and left wing carry-through spar at BL 37.60
- One crack indication on the front row of the lower cap of the right wing front spar at WS 55
- One-inch crack between two fasteners on the lower cap of the right wing front spar at WS 57.5
- One crack indication on the lower cap of the aft left wing auxiliary spar at WS 92
- Two crack indications on the aft row of the lower cap of the aft left wing auxiliary spar at WS 90
- Two crack indications on the aft row of the lower cap of the aft right wing auxiliary spar at WS 92
- Crack indications at all holes on the right and left side lower cap of the carry-through forward spar at WS 46.9
- One crack indication on the aft row of the lower cap of the left wing forward auxiliary spar at WS 89
- Severe damage to most lug holes on the right and left side wing front spar lugs at WS 46.89
- Crack indications at all holes of the right and left side wing tip tank attachments

- One crack indication on the right wing front spar lower cap at WS 85

Eddy-current SID inspections also showed both engine beam firewalls failed conductivity tests.

5.2 SUMMARY OF TEARDOWN EXAMINATION PHASE.

The teardown examination involved disassembly of the airframe and major airplane sections, inspection of airplane systems' components, inspections of primary airplane structure using alternative NDI techniques, and microscopic examination of critical structural areas. All airplane systems' components and wiring were removed during disassembly, and the disassembly provided full access to all critical structural areas on the airplane. Inspection of the airplane systems' components assisted in determining if any aging effects were related to the airplane systems. The inspections of primary airplane structure using alternative NDI techniques were used for evaluating the structure prior to disassembly. The microscopic examination of critical structural areas provided verification and detailed quantification of the extent of damage found during supplemental inspection, alternative NDI, and disassembly of the entire airframe.

5.2.1 Inspection of Systems' Components.

Systems' components visual inspections were accomplished to identify areas where mechanical wear or chemical attack had caused material damage, loss of joint integrity, or corrosion on Cessna 402A airplane mechanical components. The only finding during system component visual inspections consisted of broken teeth on a fuel selector valve gear.

Leak tests were performed on 74 pressurized system lines from the Cessna 402A airplane. Nine lines were found to have leaks. Each of these leaks was caused by material damage, loss of fitting joint integrity, or corrosion. None of the leaks appeared to be caused by any two or three of those three defective conditions combined.

5.2.2 Structural Assessment Using Alternative NDI Techniques.

The structural assessment performed with alternative NDI techniques on the Cessna 402A airplane was accomplished using the Magneto-Optic Imager (MOI), the sliding probe, and the spot probe eddy-current techniques. Surface and subsurface cracks and corrosion were the conditions of interest. These techniques were used to assess the condition of the primary structural members in the airplane prior to disassembly.

During inspections with alternative NDI techniques, the following indications were identified:

- Four indications on the horizontal stabilizer front spar upper cap
- Four indications on the horizontal stabilizer front spar lower cap
- Eleven crack indications on the left wing front spar lower cap
- Five indications on the left wing front spar upper cap

The alternative NDI indications were compared to the microscopic examination results. Below is a comparison of the results, showing that not all of the NDI indications were verified during the microscopic examination.

- Two of the eight spot probe indications on the horizontal stabilizer were verified as mild corrosion.

- Six of the sixteen alternative NDI indication locations on the left wing were verified to have corrosion.

- One skin crack found using MOI was verified microscopically.

5.2.3 Microscopic Examination.

Microscopic examinations were conducted on suspect areas from all inspections, on defects found during disassembly, and on critical structural details to locate and characterize cracks and areas of corrosion. The following areas had at least one or more defects: the stub wing attach fittings, stub wing skins, wing attach fittings, front spars of the wings, tip tank fittings on both wings, forward and aft auxiliary spars on both wings, rudder bellcrank, landing gear bellcrank, and the left-hand fuselage channel.

- One of eight stub wing attach fittings was found to have a 0.24-inch crack. This crack was found to be caused by stress corrosion. This was the only stub wing attach fitting that had any defects.

- Two of the right stub wing upper skins had areas of corrosion. These corrosion areas were classified as severe and only occurred on the right wing.

- Four of the eight front wing attach fittings were found to have corrosion. Three of these fittings were located on the left wing. The left wing front spar lower front root fitting had the most corrosion, which was classified as moderate-severe.

- The front spars on both wings had several areas of corrosion, and both were located on the lower cap of the front spars. The left wing was the most corroded, with an area of moderate corrosion.

- Both tip tank fittings had areas of cracking. The left wing tip tank fitting had two cracks ranging from 0.252 to 0.295 inch, which were both caused by fatigue. The right wing tip tank fitting had several cracks, but they were found to be caused by the manufacturing process.

- All four auxiliary spars had multiple cracks. Eight total cracks were found on the left wing auxiliary spars, ranging from 0.046 to 0.598 inch. Twelve total cracks were found on the right wing auxiliary spars, ranging from 0.041 to 0.74 inch. All these cracks show signs of typical fatigue cracks.

- Two cracks, ranging from 0.03 to 0.045 inch, were detected on the rudder bellcrank. Both cracks exhibited fatigue-like characteristics.

- Eight total cracks were found on the landing gear bellcrank, ranging from 0.039 to 0.093 inch. These cracks were located at the corners of the bellcrank and were caused by fatigue.

- The left-hand fuselage channel exhibited several small areas of corrosion, classified as light-moderate.

5.3 SUMMARY OF WIRING ASSESSMENT.

As part of the airplane destructive evaluation, electrical wiring inspections and tests were performed to assess the condition and degradation of electrical wiring in small airplanes and to evaluate maintenance procedures. The wiring inspections and tests were mainly divided into two categories: nondestructive and destructive. The nondestructive inspections and tests were comprised of a general visual inspection, in situ wiring tests, and laboratory testing. The destructive tests were comprised of wet dielectric withstand voltage tests, mandrel bend/wrap back tests, and dynamic cut-through tests.

5.3.1 Nondestructive Inspection and Tests.

The purpose of performing the general visual inspections on the wiring systems of the Cessna 402A airplane was to check the condition of the wires for any defects or flaws that could be hazardous to an airplane in flight. Visual inspections revealed extensive information about the wiring condition. Based on the data obtained from visual observations, it could be concluded that wires in general had issues related to aging. The most dominant issues found were chafing, rubbing, cutting, exposed shield, exposed inner conductor, cutting through outer insulation, improper terminations of end terminals, improper termination of wires (no cap), unused wires improperly stowed, corroded terminals, and several others. Looking at the described flaws, it can be inferred that the issue with the wires were less of aging and more on maintenance, especially with the wires in the engine compartment.

A loop resistance test, Del™ Test, and an excited dielectric test were performed on the airplane prior to conducting laboratory tests. The loop resistance measurement shows how well the airplane structure is electrically grounded to demonstrate the capability of the airplane to discharge static that is generated during flight. The lower the value measured, the better the dissipation of the static charges accumulated. The other tests did not yield any reportable results.

The laboratory tests were performed after the teardown of the airplane. During the teardown phase, all the wires were removed from the airplane and were subjected to various tests as well as inspections. The laboratory tests consisted of the following:

- Intrusive Visual Inspection
- Wiring Insulation Microscopic Inspection
- Insulation Resistance Test
- Circuit Breaker Test
- Relay Inspection

All the flaws found in the general visual inspection were very conspicuous, and the intrusive visual inspection as well as the microscopic inspection confirmed these results. Intrusive visual inspection data provided more information of the wiring condition of the specific location inspected. The insulation resistance test was used to determine the insulation resistance of a finished wire specimen. Insulation resistance is of interest in high-impedance circuits and as a measure of quality control. Since circuit breakers are designed to limit the current flow through the wires in different sections of the airplane, they were tested for various current ratings, e.g., 135%, 200%, and 300%. Five percent of the breakers did not function and would not conduct any current, 24 percent of the breakers did not meet trip time specifications after the first test, and 14 percent of the breakers did not meet trip time specifications after the second test. The relays were opened for inspection, and they all were found to have normal pitting. Among all the laboratory tests, the circuit breaker test revealed the most important part of the aging issues.

5.3.2 Destructive Tests.

Three destructive laboratory tests were performed on the wires.

- Wet Dielectric Withstand Voltage (DWV) Test
- Mandrel Bend/Wrap Back Test
- Dynamic Cut-Through Test

The DWV test provides a method to determine insulation integrity following any type of performance test. This test was performed soon after the insulation resistance measurement test. The test was used to determine whether exposure to environmental test conditions had reduced the breakdown strength of the wires below some prescribed level. The mandrel bend/wrap back test was used to determine whether a wire specimen would crack when wrapped around itself or around a mandrel. This test was used to determine the degree of sintering of wire insulations. The dynamic cut-through test is used to determine the amount of force required to cut through the outer insulation of a wire.

The destructive laboratory tests also demonstrated few issues with wires with respect to aging airplane. Since the wires were not 26 feet or more in length, as required by the standard, they could not be reliably tested for DWV. Only 12.5% of the wires failed the mandrel bend/wrap back test, which indicated that the wires generally had good quality insulation. The dynamic cut-through test gave a wide variability in the force required to cut each specimen; therefore, no solid conclusion could be drawn about the wire's ability to resist cutting of the outer insulation.

6. REFERENCES.

1. "Cessna 402A Maintenance Manual," Cessna Aircraft Company, Wichita, Kansas.

2. "Cessna 402A Supplemental Inspection Document," Cessna Aircraft Company, Wichita, Kansas.

3. http://www.staveleyndt.com/help/fixedslideprobes.pdf