NATIONAL TRANSPORTATION SAFETY BOARD
Office of Aviation Safety
Washington, D.C. 20594

Airworthiness Group Chairman's Factual Report of Investigation – Addendum 1

March 24, 2016

A. ACCIDENT CEN13FA196

Location: South Bend, Indiana
Date: March 17, 2013
Time: 1623 Eastern Daylight Time
Aircraft: Hawker Beechcraft Corporation Model 390 Premier 1A, N26DK

B. GROUP

Accident site documentation, March 19 – 22, 2013:

Chairman: Tom Jacky
National Transportation Safety Board
Washington, D.C.

Member: Bob Hendrickson
Federal Aviation Administration
Washington, D.C.

Member: Kris Wetherell
Hawker Beechcraft Corporation
Wichita, Kansas

Member: J. Chris Greene
Williams International
Walled Lake, Michigan

Re-Examination of the Wreckage, South Bend Indiana, May 22, 2013

Chairman: Tom Jacky
National Transportation Safety Board
Washington, D.C.

Member: Brian Weber
Beechcraft Corporation
Wichita, Kansas

Examination of Spoiler Control Unit at East Aurora, New York, June 10, 2013:

Witness:	George Haralampopoulos National Transportation Safety Board Washington, D.C.
Member:	Brian Weber Beechcraft Corporation Wichita, Kansas

Examination of EGPWS Unit at Redmond, Washington, July 2, 2013:

Witness:	Joshua Cawthra National Transportation Safety Board Federal Way, Washington

Examination of Starter-Generators at Wichita, Kansas, September 9, 2013:

Chairman:	Tom Jacky National Transportation Safety Board Washington, D.C.
Member:	Brian Weber Beechcraft Corporation Wichita, Kansas

Examination of Aileron Components at Wichita, Kansas, September 10, 2013:

Chairman:	Tom Jacky National Transportation Safety Board Washington, D.C.
Member:	Brian Weber Beechcraft Corporation Wichita, Kansas

Examination of Throttle Quadrant Assembly at Wichita, KS, September 11, 2013:

Chairman:	Tom Jacky National Transportation Safety Board Washington, D.C.
Member:	Brian Weber Beechcraft Corporation Wichita, Kansas

Wreckage Re-Examination at Wright City, Missouri, November 14, 2013:

Chairman: Tom Jacky
National Transportation Safety Board
Washington, D.C.

Member: Brian Weber
Beechcraft Corporation
Wichita, Kansas

Generator Control Unit Examination at Wichita, Kansas, March 18, 2014:

Chairman: Tom Jacky
National Transportation Safety Board
Washington, D.C.

Member: Brian Weber
Beechcraft Corporation
Wichita, Kansas

Member: John Wills
Ametek Advanced Industries
Wichita, Kansas

Main Battery Examination at West Covina, California, March 24-26, 2014:

Witness: Pocholo Cruz
National Transportation Safety Board
Washington, D.C.

Member: Brian Weber
Beechcraft Corporation
Wichita, Kansas

Maintenance Data Computer, Flight Management Computer, and Air Data Computer Examinations at Cedar Rapids, Iowa, April 7-8, 2014:

Witness: George Haralampopoulos
National Transportation Safety Board
Washington, D.C.

Member: Brian Weber
Beechcraft Corporation
Wichita, Kansas

C. SUMMARY

On March 17, 2013, at 1623 eastern daylight time, a Hawker Beechcraft[1] model 390 (Premier IA), N26DK, serial number RB-226, collided with three residential structures and terrain following an aborted landing attempt on runway 9R located at the South Bend Regional Airport (KSBN), South Bend, Indiana. The private pilot and pilot-rated-passenger occupying the cockpit seats were fatally injured. An additional two passengers and one individual on the ground sustained serious injuries. The airplane was registered to 7700 Enterprises of Montana LLC and operated by Digicut Systems of Tulsa, Oklahoma, under the provisions of 14 Code of Federal Regulations Part 91 while on an instrument flight plan. Day visual meteorological conditions prevailed for the business flight that departed Richard Lloyd Jones Jr. Airport (KRVS), Tulsa, Oklahoma, at 1358 central daylight time.

On March 19, the accident airplane was moved from the accident site to a maintenance hangar on the South Bend Airport. Once placed in the hangar, the airplane was documented from March 19, 2013 to March 22, 2013 for airworthiness aspects (structure, systems, powerplants).

During the on-site examination, the following airplane components were removed from the airplane and retained by the investigation for further examination:

1. Williams Electronic Control Unit (Left)
   Part Number: 117162
   Serial Number: LH2A0603

2. Williams Electronic Control Unit (Right)
   Part Number: 117162
   Serial Number: LH2A0608

3. Rockwell Collins Data Concentrator Unit (Left)
   Part Number: DCU-3000
   Serial Number: 2RKFM

4. Rockwell Collins Data Concentrator Unit (Right)
   Part Number: DCU-3000
   Serial Number: 2RKHP

5. Rockwell Collins Flight Management Computer
   Part Number: FMC-3000
   Serial Number: 28DPH

6. Rockwell Collins Maintenance Data Computer:
   Part Number: MDC
   Serial Number: Unknown

[1] The company is now known as Textron Aviation.

7. Rockwell Collins Air Data Computer (Left)
   Part Number: ADC-3000
   Serial Number: 2NJHP

8. Rockwell Collins Air Data Computer (Right)
   Part Number: ADC-3000
   Serial Number: 2NJLP

9. Ametek Starter-Generator (Left)
   Serial Number: 425AB

10. Ametek Starter-Generator (Right)
    Serial Number: 461AB

11. Moog Spoiler Control Unit
    Part Number: 233700-109
    Serial Number: 0218

12. Premier 1 Pedestal Assembly
    Part Number: 97-2001-11
    Serial Number: 0314

13. Aileron Assembly Components, Right Wing
    a. Sector Assembly, P/N 390-521109-0014,
    b. Aileron Pushrod (P/N 390-381016-0003),
    c. Pivot Fitting Assembly (P/N 390-52115-006)

14. Honeywell Mark V Enhanced Ground Proximity Warning System Unit
    Part Number: 965-0976-040-210-210
    Serial Number: 26547

15. Concorde Main Airplane Battery
    Part Number: RG-380E/44
    Serial Number: 40332918

The airplane wreckage was moved from the South Bend Airport hangar to Wright City, Missouri in November, 2013. On November 14, 2013, the group removed the following components from the airplane wreckage:

16. Ametek Advanced Industries Generator Control Unit (GCU), Left Engine
    Part Number: CG201A-1
    Serial Number: 645

17. Ametek Advanced Industries Generator Control Unit (GCU), Right Engine
    Part Number: CG201A-1
    Serial Number: 643

This report provides additional factual information to the Powerplants Factual Report, dated February 27, 2015, and is and addendum to the Airworthiness Group Chairman's Factual Report of Investigation, dated March 2, 2015.

No failures were found in any of the components examined.

D. DETAILS OF INVESTIGATION

I. AIRPLANE SYSTEMS

During the on-scene investigation and based on the circumstances of the accident, the group removed and subsequently examined components from the following airplane systems:

a. Communications

VHF communications No. 1 antenna is mounted on the lower fuselage (forward centerline of the aft avionics and maintenance compartments) and feeds VHF No. 1. The No. 1 antenna was separated from the fuselage by contact with the runway surface and was recovered from the runway by airport personnel. The No. 1 antenna was separated into 4 pieces. The No. 1 antenna had gouges and scoring on the lower leading edge consistent with contact with the runway surface.

The airplane's cockpit voice recorder (CVR) was removed from the airplane and sent to the National Transportation Safety Board's Vehicle Recorder Laboratory in Washington, D.C. For further information please refer to the Cockpit Voice Recorder Factual Report and the Sound Spectrum Study.

b. Electrical Power

b.1.0 Electrical Power System Description

The airplane's electrical system is designed as a direct current (DC) only system.

The airplane's electrical system includes two 28.5 Volts DC (VDC), 325-amp starter-generators, one main lead-acid battery, one standby battery, a dual parallel bus power distribution system, and an external power receptacle. The distribution of electrical power is accomplished through the power distribution box in the aft section of the fuselage.

The power distribution box consists of power relays, bus bars, fuses, current transformers, load meter shunts, and circuit breakers. It receives input power from the generators and the main battery (and, when on the ground, external power). The power distribution box then distributes the input power to the right main bus, left main bus, essential bus, non-essential bus, standby bus, and the hot battery bus.

Electrical power to the airplane in-flight is normally supplied by the two engine-driven starter/generators. The starter/generator units are the primary source of electrical power to the electrical systems.

Beechcraft considered a loss of one or both starter/generator capability an abnormal power condition. For such abnormal power conditions, a 24-volt, 42-ampere-hour, maintenance free lead-acid battery supplies electrical power for engine starting and emergency requirements. A 24-volt standby battery is also provided. The standby battery supplies voltage to the standby bus and for lighting of selected equipment during abnormal power conditions. In abnormal power conditions, the airplane battery is used to provide airplane power until the starter/generator can provide power (if able). If the starter/generator is inoperative due to a loss of engine power, the battery is designed to power the starter/generator to re-ignite the affected engine.

b.1.1 DC Electrical Load Distribution System

The DC distribution system consists of two linked generator bus systems, one essential bus system, one hot battery bus system and a standby battery bus system. The bussing system is cross-tied together with one bus-tie contactor. See Attachment 1, Power Distribution Schematic.

b.1.2 DC Starter/Generators

The airplane's primary electrical power is generated by two engine-driven starter/generators. The starter/generators are designed to provide up to 30 Volts DC and 350 amperes power to the power distribution box (for distribution). The generators are used as starter motors for engine starting and also for charging the airplane batteries.

Each starter/generator is equipped with a generator control unit (GCU) that regulates, controls, and provides protection for its respective starter/generators. The GCUs are located in the aft section of the airplane but not attached to the engines or starter/generators.

b.1.3 Flight Deck Generator Switches

The L and R GEN toggle-type switches are located on the electrical control panel, mounted on the lower central instrument panel. The GEN switches have three positions:

- OFF – Indicates that the related generator is switched off.
- ON – Indicates that the related generator is switched on.
- RESET - the momentary RESET position allows the related generator to be manually reset to the bus system if the generator had previously been tripped due to a fault.

The RESET switch is spring-loaded; the switch is designed so that it will not stay in the RESET position unless held.

b.1.4 Main Airplane Battery

The airplane is equipped with a 24-volt, maintenance free lead-acid battery with a minimum performance capacity of 42 ampere-hours. The battery provides power for self-contained engine starts and is a backup power source for the essential loads. The battery (BAT) switch is located on the electrical control panel on the lower central instrument panel. During normal flight conditions, the BAT switch is in the ON position. The standby (STBY) position is selected to isolate the standby bus from the essential bus, once the main battery has been exhausted.

Section 3 of the Airplane Flight Manual lists the equipment available on the Essential Bus. The Essential Bus is active when the airplane is operating with the main battery only. The list of available equipment on the Essential Bus follows:

1. Pilot's Primary Flight Display (PFD), including Air Data
2. Oxygen Indicator
3. Engine Instruments (PFD reversionary mode)
4. Hydraulic Pressure Indicator
5. Pilot's Display Control Panel
6. Landing Gear Indicators
7. Annunciator Panel
8. Voltmeter
9. Copilot or Standby Attitude Indicator
10. Fuel Quantity
11. Copilot or Standby Airspeed Indicator*[2]
12. Trim Indicators
13. Copilot or Standby Altimeter*
14. Flap Indicator (flaps inoperative)
15. Standby Compass* DME 1
16. AHRS 1
17. ELT
18. ADC 1
19. Transponder 1
20. ADF 1 (For Serial Numbers RB-32 and after and airplanes modified by Beechcraft Kit Part Number 390-3401)
21. COMM 1
22. Roll Trim System
23. NAV 1
24. Rudder Trim System
25. Standby Pitch Trim System
26. Engine Fire Extinguisher System
27. Spoilers/Speed Brake System
28. Hydraulic Firewall Shutoff Valves

[2] 1. Items denoted with an asterisk (*) are not time limited.

29. Left and Right Engine Engine Control Units
30. Fuel Firewall Shutoff Valves
31. Left and Right Thrust Lever
32. Solenoid Engine Fire Detection (fire buttons)
33. Left and Right Ignitions
34. Fuel Transfer System
35. Engine Start Switches Pressurization Controller (dump only)
36. Left and Right Boost Pumps (Auto and ON)
37. Cabin Temperature Control (auto and manual modes)
38. Bleed Air System Cockpit Temperature Control (auto and manual modes)
39. Cabin Door Seal
40. Recognition Lights
41. Crew Oxygen System*
42. Beacon
43. Passenger Oxygen System (automatic and manual cabin mask deploy)
44. Cabin Signs
45. Instrument Panel Indirect Lighting
46. Copilot's Audio
47. Anti-collision Lights
48. Cockpit Voice Recorder
49. Position Lights
50. Left Pitot/Static Heat
51. Aural Warnings
52. Weight-on-Wheels Logic
53. Pilot's Audio
54. Left and Right Wing Anti-ice Systems
55. Anti Skid System Landing Gear System
56. Left and Right Engine Anti-ice Systems (failed on)*
57. Cabin Altitude Gauge

b.1.5 Standby Battery Discussion

The standby battery is a 5-amp-hour lead-acid battery used to supply 24 VDC to the standby bus and 5 VDC for lighting of selected equipment during abnormal power conditions. The standby bus supplies power to dedicated airplane components required to allow operation of the airplane when no other source of power is available. The battery will supply 150 watts of power for a minimum of 30 minutes or until the cutoff voltage of 20 VDC is reached.

Section 3 of the Airplane Flight Manual lists the equipment available when the airplane is operating with the standby battery. The list of available equipment follows:

1. Copilot or Standby Attitude Indicator
2. Bleed Air System (failed on)*[3]
3. Copilot or Standby Airspeed Indicator*

[3] Items denoted with one asterisk (*) are not time limited.

4. Crew Oxygen System*
5. Engine Instruments (display on CDU 1)
6. Copilot or Standby Altimeter*
7. Audio via ALTN and Pilot's Headset
8. Standby Compass*
9. COMM 1 (tune with CDU 1)
10. Transponder 1
11. Left and Right Engine
12. Engine Control Units
13. Fuel Firewall Shutoff Valves**[4]
14. Engine Fire Detection (fire buttons)**
15. Hydraulic Firewall Shutoff Valves**
16. Engine Fire Extinguisher System**
17. Left and Right Data Concentrator Units
18. Engine Anti-ice (failed on)*
19. Passenger Oxygen System (manual mask deploy)*
20. Cabin Altitude Gauge

b.1.6 Flight Deck Electrical System Indicators - GENERATOR OFF ANNUNCIATORS, DC Ammeters, and DC Voltmeter

Two amber annunciators, marked L GEN OFF and R GEN OFF, are located on the annunciator panel. These annunciators illuminate when the respective generator relay opens, isolating the generator from the main bus. The master caution annunciators on the glare shield also illuminate.

Two DC Ammeters (Left and Right Generators) display the load current of their respective generators and are located on the flight deck electrical control panel, mounted on the lower central instrument panel.

A DC voltmeter is a digital meter to display voltage to that part of the electrical system, as selected by the rotary switch. The voltmeter is located on the electrical control panel on the lower central instrument panel.

b.2.0 Examination of Electrical Power Components

b.2.1 Power Distribution Box

The power distribution box external areas were visually examined. The examination did not reveal any evidence of a thermally-deteriorated condition – either localized areas of discoloration in the box material or attached wire bundles or cables. The power distribution box lid was unfastened and partially raised in order to observe the box interior. The examination did not reveal any evidence of thermal deterioration. The immediate area around the power distribution box (rear fuselage just forward of the rear canted bulk head

[4] Items denoted with two asterisks (**) are valid only if the airplane was not modified by Beechcraft Kit 390-3622.

and adjacent to the emergency locator transmitter (ELT) was examined and did not reveal any thermally discolored areas.

b.2.2. Circuit Breaker Panels

The Pilot's Circuit Breaker Panel was examined. No circuit breakers were observed to be OPEN. The LDR GR CTRL (landing gear control) circuit breaker is located on this panel.

The Copilot's Circuit Breaker Panel was examined. The panel and the cockpit sidewall were separated from the cockpit structure, and the circuit breaker panel top cover and fascia was buckled and was partially separated. The fascia panel contains the circuit breaker nomenclature and the majority of the fascia was missing. There were multiple circuit breakers OPEN and/or physically damaged. (See Figure 1)

Figure 1 - Co-Pilot's Circuit Breaker Panel at Accident Site

The airplane flight hour meter read 0457.5 hours.

b.2.3 Engine Accessory Starter/Generators

The two engine accessory mounted starter/generators were removed from their respective engines for examination. Both starter/generator drive shafts were intact. Both starter/generator armatures rotated. The brush inspection covers were removed and the brushes were observed to be in good condition.

The Starter – Generators were examined at their manufacturer, Ametek. For more information related to the examination of the starter-generators, please see the Airworthiness Group Chairman's Factual Report of Investigation, Section D.2.1.

b.2.4 Flight Deck Generator Switches

The L and R GEN switches were found in the wreckage. The L and R GEN switches were found in the ON position. See Figure 2.

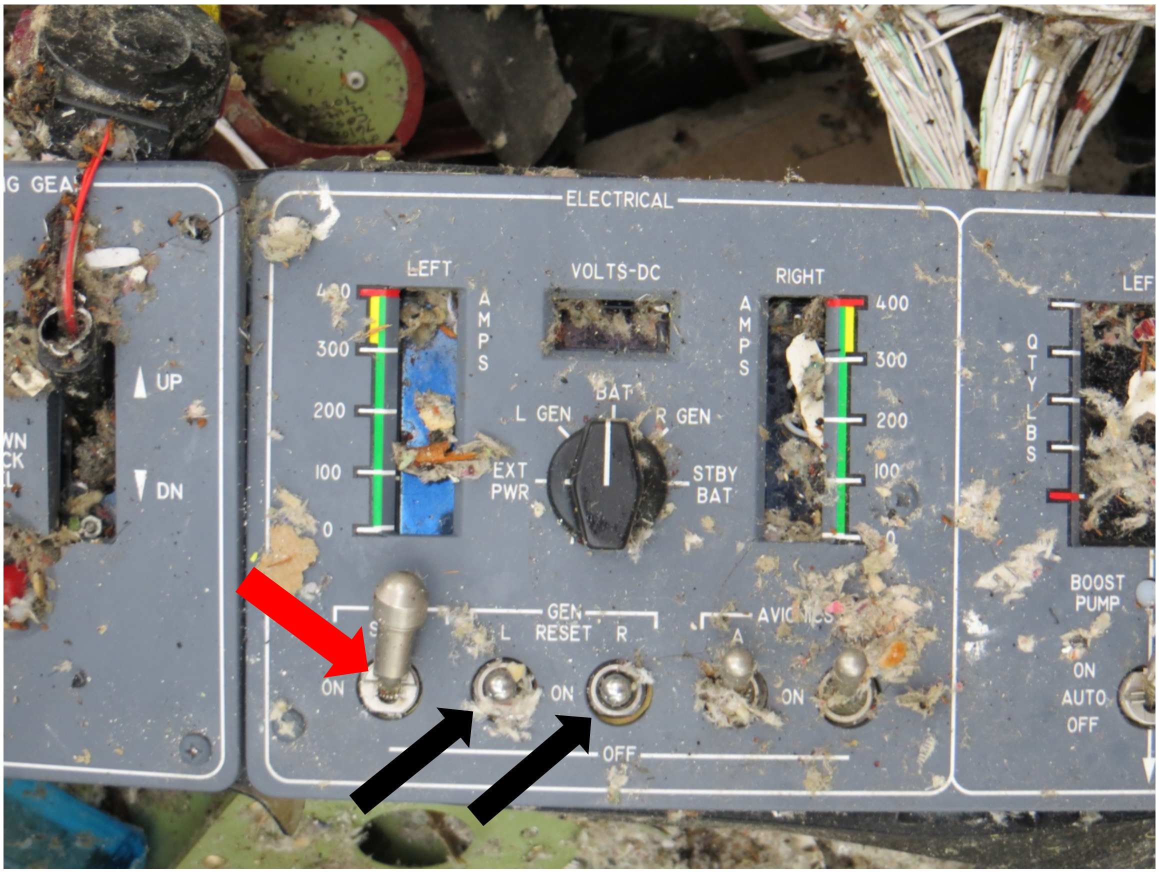

Figure 2 - Electrical Panel, Including L and R GEN Switches (black arrows) and Standby Battery Switch (red arrow). (Note: Photograph taken at accident site)

b.2.5 Main Airplane Battery Examination

The power cable to the airplane battery was first observed to be disconnected from the battery. The connection was not visibly damaged. The battery top cover was separated from the battery base. A digital volt Ohm-meter reading was used to measure the battery voltage during the on-scene phase; the NO LOAD voltage of the battery was measured to be 25.05 volts.

The airplane batter was examined at the manufacturer, Concorde. For more information related to the examination of the battery, please see the Airworthiness Group Chairman's Factual Report of Investigation, Section D.2.3.

b.2.6 Standby Battery

The standby battery was examined. There was no visible physical damage to the outer case of the battery. The NO LOAD voltage was not measured.

The Standby Battery Switch was found in the STANDBY position. See Figures 2 above (taken at the accident site) and Figure 3 (taken after the airplane wreckage was moved to the airplane hangar).

Figure 3 - Standby Battery Switch, After Wreckage Moved to Hangar

c. Equipment & Furnishings

The left crew seat remained intact and attached to the fuselage structure. All four seat feet remained attached to the seat rails. The rotary buckle receiver operated normally. No shoulder or lap belt buckles were observed to be engaged in the rotary buckle assembly. None of left crew seat webbing exhibited any damage or fraying. There were 7 seat track stop holes visible forward of the forward seat foot.

The right crew seat separated from the fuselage. Portions of the seat keel structure remained attached to the right crew seat, seat track assemblies. All four right crew seat feet remained attached to the seat rails. The cross member across the back of the vertical adjustment assembly was broken into two pieces. The rotary buckle receiver operated normally. No shoulder or lap belt buckles were observed to be engaged in the rotary buckle assembly. None of the right crew seat webbing exhibited any damage. There were 7 seat track stop holes visible forward of the forward seat foot.

All the cabin seats remained attached to their respective seat keel and structure assemblies. The left side forward facing cabin seat, seat belt buckle was engaged. The lap belt webbing was separated consistent with being cut and the shoulder harness was not attached to the lap belt buckle assembly. No other cabin seat belt buckle assemblies were observed to be engaged in the lap belt buckle receiver assembly. The two aft, fixed, forward facing seat lap belts had the shoulder harnesses attached to the lap belts.

d. <u>Fire Protection</u>

A fire extinguisher was noted in the airplane wreckage.

e. <u>Flight Controls</u>

The flight control system consists of the primary flight control systems (pitch control, roll control, and yaw control, and secondary flight control systems (pitch trim, yaw trim, speed brake, and trailing edge flaps).

e.1.0 PRIMARY CONTROL SYSTEM DESCRIPTION

Pitch attitude of the airplane is controlled by the elevators and the variable incidence horizontal stabilizer. The elevator control system is operated manually by movement of the control columns. Elevator control surfaces are fully mass balanced.

Roll attitude is controlled through the ailerons, spoilers and roll trim. Roll attitude is controlled by the ailerons and spoilers. The aileron control system is operated manually by movement of the pilot's or copilot's control wheels.

Roll attitude is augmented by the spoilers, three per each wing. The spoilers are hydraulically actuated and electrically controlled. The airplane is equipped with a spoiler control unit (SCU) that commands and monitors the spoiler movement. The SCU receives power from the Essential Bus. The SCU records faults into non-volatile memory that is written upon normal completion of a flight.

Yaw control is accomplished by the rudder and rudder trim tab. Yaw attitude is controlled by the rudder. The rudder control system is operated manually by moving the pilot's or copilot's rudder pedals.

The primary control systems, except the spoilers, are manually operated through control cables, push/pull tubes, and mechanical linkages. The spoilers are electronically controlled and hydraulically actuated. The pitch trim system, roll trim system, and yaw trim system are electrically operated. The speed brake is controlled electrically and operated hydraulically. The flaps are electronically controlled and electrically actuated.

e.1.1 PITCH CONTROL SYSTEM ON-SCENE EXAMINATION

The on-scene examination of the pitch control system revealed the following:

- The Tee Column assembly remained intact;
- The right control wheel shaft was separated from the Tee Column at the universal joint;
- The push rod that connects the base of the Tee Column to the forward bell crank was separated;
- The forward bell crank and forward sector were separated into multiple pieces;
- The pitch control cables remained attached to the forward sector;
- The pitch cables were intact from the forward sector to the aft sector, but were cut just aft of the rear pressure bulkhead to facilitate airplane recovery;
- The pitch cables remained attached to the aft sector;
- The aft sector and idler bell crank remained intact;
- The link that connects the idler bell crank to the elevator torque fitting was separated;

e.1.2 ROLL CONTROL SYSTEM ON-SCENE EXAMINATION

Examination of the roll control system revealed the following:

- The flight control cables remained intact and attached to the aft sector;
- The aileron chains remained engaged with the control shaft sprockets;
- The chains remained attached to their respective flight control cable at the forward sector mounted on the Tee (control) Column;
- Manipulation of the left control wheel moved the right control wheel sprocket;
- Manipulation of the roll flight control cables was met with audible sounds of movement from the aft sector;
- The aft sector and aft sector shaft that connects to the pulley sector pushrod arm were not observed;
- The pushrod arm was partially separated from the aft sector shaft;
- The pushrod remained attached to the pulley sector;
- The two inboard wing sectors remained connected to the pulley sector by their pushrods;
- The right inboard wing sector was separated from its rear wing mount;
- The aileron cables (2 each side) remained attached to their respective inboard sector;
- The aileron control system to the left aileron remained intact and the system was manipulated and appropriate control surface deflection was observed;
- The aileron control cables to the right aileron were separated at approximately the outboard end of the inboard flap at the location the right wing separated;
- An aileron control cable was observed in the separated right wing outboard section;
- At the right aileron outboard wing sector one aileron cable remained attached to the sector. This attached portion of the cable was the correct length to mate to the separated inboard end (a complete cable). The other right aileron control cable outboard portion was not observed;

- The pushrod that connects the sector to the bell crank was bent ~3 inches from the outboard end but remained attached at both ends;
- The right aileron control stops mount to the sector;
- The right aileron controls stops and a piece of the sector edge were separated;
- Manipulation of the sector achieved appropriate control surface movement;

The spoiler control system is electrically controlled by movement of the pilot's or copilot's control wheels and hydraulically actuated. The on-scene examination of the spoiler system revealed the following:

- The left spoiler actuator remained mounted in the wing;
- The spoiler actuator extension was measured and found to be 3.6 inches;
- The left blow down actuator remained mounted in the wing;
- The left blow down actuator extension was measured as 3.2 inches;
- The right spoiler actuator was separated from the wing;
- The spoiler actuator extension was measured as 3.2 inches;
- The right blow down actuator was partially separated from the wing and was deformed;
- The airplane's spoiler control unit was recovered from the wreckage.
- The right blow down actuator extension was not measured.

e.1.3 Examination of Roll Control System Components at Beechcraft Materials Laboratory

As indicated in the <u>Airworthiness Group Chairman's Factual Report of Investigation,</u> dated March 2, 2015, on January 21, 2014 Beechcraft submitted a letter documenting their findings of the three components from the roll control system submitted to the Beechcraft Materials Laboratory. The letter from Beechcraft is provided in Attachment 4 of the March 2, 2015 report.

The purpose of the work at Beechcraft was for further examination of three fractures in the submitted components and a bend in the aileron bellcrank assembly pushrod. In their January 21, 2014 letter, Beechcraft indicated that all of the fractures were considered overload and that all of the components met their material hardness requirements.

e.1.4 YAW CONTROL SYSTEM ON-SCENE EXAMINATION

The yaw flight control system was examined during the on-scene investigation. The examination revealed the following:

- The rudder pedal assemblies and forward sector were separated from the cockpit floor;
- The rudder flight control cables remained intact from the forward sector to the aft sector, but were cut at the rear of the aft pressure bulkhead to facilitate airplane recovery;

- The aft sector remained attached to the (aft) canted bulkhead and the rudder travel spring remained attached to the canted bulkhead and the aft sector.

e.2.0 SECONDARY CONTROL SYSTEMS

e.2.1.0 Pitch Trim and Secondary System Description

Pitch trim is controlled by changing the incidence of the horizontal stabilizer. The pitch trim actuator is electrically operated. The system has two trim rates which are dependent on airplane air speed. A fast trim rate is used for airspeeds less than 210 KIAS and a slow trim rate for airspeeds greater than 210 KIAS. The incidence angle has a range from 1.4° leading-edge up to 7° leading edge down. The effectiveness of the pitch trim system is augmented by an elevator tab system which is driven by horizontal stabilizer movement. When the airplane is trimmed to full nose up (leading edge 7.0° down), the elevator trim tab is driven to 12.6° down. When the airplane is trimmed to nose down (leading edge 1.4° up), the elevator trim tab is driven to 3.06° up.

Normal trim control is accomplished by using the spring-loaded TRIM switch on each control wheel. If a malfunction develops during operation, the entire normal trim system can be electrically rendered inoperative and the standby control system used for pitch trim.

The normal trim and standby trim systems are isolated electrically. Standby trim is activated by using the PITCH TRIM switch located on the center pedestal. Setting the PITCH TRIM mode switch, also located on the center pedestal, to STBY activates the standby system. To trim the airplane nose down or nose up, the spring loaded PITCH TRIM switch is depressed to the ND (nose down) or NU (nose up) position. To disengage the normal pitch trim system, the AP/TRIM MASTER switch is pushed on the control wheel. Setting the PITCH TRIM mode switch to the OFF position will disengage both the normal and standby trim systems.

e.2.1.1 Pitch Trim Position Indication and Pitch Trim Warning

The pitch trim aural warning sounds if the thrust levers are advanced past approximately 80% N1 and the airplane is on the ground with the pitch trim not set in the takeoff trim zone.

The pitch trim position indicator is located on the center pedestal. The pitch trim position and movement direction of the horizontal stabilizer is indicated by the lighting of liquid crystal display (LCD) bars on the indicator. The indicator also has a green bar that indicates the takeoff trim range.

e.2.1.2 On-Scene Examination of Pitch Trim System

The pitch trim actuator was examined on-scene. The examination revealed that the pitch trim actuator remained attached to the mounting location in the vertical stabilizer and attached to the leading edge of the horizontal stabilizer. The actuator extension was measured and found to be extended 17 5/8 inches. The takeoff trim position range was measured as approximately 15 15/16 inches to 16 7/32 inches. The trailing edges of the elevator trim tab surfaces were visually aligned with the trailing edge of the respective elevator.

e.2.2.0 Roll Trim System Description

Roll trim is controlled by electrically activated trim surfaces located on the left and right ailerons. The deflection angle of the tab control surface is 20° up and 20° down.

The same trim switch used for pitch trim is used for roll trim. When the ROLL TRIM switch on the center pedestal is set to NORM and the TRIM switch is depressed and held to LWD (left wing down) or RWD (right wing down), the trim actuator will move the roll trim surface on the left aileron. When additional roll trim authority is required, the ROLL TRIM switch may be set to AUX, causing the trim surfaces on both ailerons to operate. The AUX ROLL TRIM annunciator illuminates any time auxiliary roll trim is active or the right aileron trim tab is out of neutral.

In case of trim runaway, the roll trim system can be interrupted by pushing the AP/TRIM MASTER switch on the control wheel or moving the ROLL TRIM switch on the center pedestal to the OFF position.

e.2.2.1 Roll Trim Position Indicator

The roll trim position indicator is located on the center pedestal. LCD bars on the indicator light up, showing position and movement direction of the roll trim tab surface.

e.2.2.2 On-Scene Examination

The roll trim actuators were examined on-scene. The examination revealed the following: The left and right aileron trim tab surfaces remained attached to their respective aileron. The left and right trim actuator extensions measured 1.3 inches and were visually neutral; the tab trailing edge aligned with the aileron trailing edge.

e.2.3.0 Yaw Trim System

e.2.3.1 Yaw Trim System and Indication Description

Yaw trim is controlled by an electrically actuated rudder trim surface on the rudder. The full deflection angle of the rudder trim control surface is 20° left and 20° right.

Yaw trim is controlled by a single action RUDDER TRIM command switch located on the center pedestal. With the RUD TRIM switch in the NORM position, rotating and holding the RUDDER TRIM command switch to the left or right (airplane nose left or right) causes the actuator to move the trim tab surface in the appropriate direction.

In case of trim runaway, the yaw trim system can be interrupted by pressing the AP/TRIM MASTER switch on the control wheel or moving the RUD TRIM switch on the center pedestal to the OFF position.

The rudder trim position indicator is located on the center pedestal. Yaw trim position and movement direction of the rudder trim tab surface is indicated by LCD bars on the indicator.

e.2.3.2 Yaw Trim System On-scene Examination

The yaw trim actuator was not examined. The rudder trim tab trailing edge was observed to be aligned with the rudder trailing edge; i.e., neutral.

e.2.4.0 Flaps System

e.2.4.1 Flap System Description and Indication

The airplane is equipped with two single slotted Fowler-type flaps on each wing. The flap panels are electronically/electrically controlled, monitored, and actuated in a closed-loop positioning system. Flap movement is provided by the Flap Control Unit (FCU) and motor driven actuators, one on each end of each flap.

The flap position is controlled by movement of the flap control lever located on the center pedestal. The flap control lever has four positions: UP (retracted), 10 (degrees), 20 (degrees), and DN (fully extended).

e.2.4.2 Flap System On-Scene Examination

The eight flap actuator extensions were measured, and the extension measurements corresponded to an actuator in the retracted position. Beechcraft indicated that, at the actuator retracted position, the corresponding flap position is flaps up or retracted.

f. Hydraulic Power Description and On-Scene Examination

The airplane is equipped with a hydraulic system to provide hydraulic power to the speed brakes/spoilers, power brake/anti-skid system, and landing gear. The system includes two engine-driven hydraulic pumps, a hydraulic package, ground connectors, and assorted electrical components. Each hydraulic pump is driven by its respective engine accessory gearbox; the pump is connected to the engine accessory gearbox.

The hydraulic reservoir was observed. No fluid remained in the reservoir. The hydraulic system integrity was compromised in more than one location during the impact sequence. The hydraulic filters remained attached to the hydraulic reservoir. Both left and right hydraulic shutoff valves were observed and were found to be in the OPEN position. The engine driven hydraulic pumps were observed and no external discoloration of the pumps was observed.

g. Instrumentents or Indicating/Recording Systems

The airplane's flight instruments are electric CRT displays.

The flight deck was examined for switch positions and indications as follows:

Pilot Audio Panel
XMIT: 1
INPH: 11:30 (O'clock)
COMM 1: 10:00
COMM 2: 12:00
NAV 1: 12:00
NAV 2: 12:00
DME 1: 12:00
DME 2: 2: 00
ADF 1: 12:00
MKR BCN: 5:30
DISPLAY: NORM
AHRS: NORM
ADC: NORM
CDU: NORM
MIC: NORM
COMM: AUTO
SPKR: OFF
DME: VOICE
AUDIO: NORM

COMM/NAV Tuning Unit
Selector Rotary Switch: OFF
Selector: NAV

Pressurization Controller
Knob separated, Shaft flat aligned with OFF

Standby Instruments
Standby Airspeed Indicator: Indicator needle off scale low
Barber Pole: 325 knots
Standby Attitude Indicator: 110° Right bank, 6° Nose up
Standby Altimeter Indicator: 17,500 feet, Kollsman 30.17" Hg.

Copilot Audio Panel
XMIT: 1
MIC: NORM
COMM: AUTO
COMM 1: 2:00 (O'clock)
COMM 2: 2:00
SPKR: ON
NAV 1: 3:00
NAV 2: 1:00
VOICE
DME 1: 9:00
DME 2: 11:00
ADF 1: 4:00
MKR BCN: 2:00
DG: NORM

Overhead Panel
INSTR PNL: 2:00 (O'clock)
INSTR FLOOD: 5:30
SUBPNL & PED: 5:30
IND: 4:00
CDU KEYBD: 3:30
PLT DISPLAYS: 6:00
MASTER TEST: OFF
COPILOT DISPLAYS: 4:00
LIGHTS MASTER: OFF
DIM BRT: OFF
CKPT FLOOD: OFF
CABIN OVHD: OFF
CABIN AISLE: OFF
GALLEY: OFF
NO SMOKE: OFF
SEAT BELTS: ON
LDG: ON
RECOG: ON
WING: OFF
NAV: OFF
BCN: OFF
STROBE: ON

Oxygen

Pull Arm: In
Cabin Deploy Pull: In
Oxy Press: Off Scale Low

Hydraulic
HYDR PRESS: Off Scale Low

Static Source
NORM

Pilot Air
PULL ON: IN
DEFROST PULL ON: IN

Electrical
Voltmeter Loadmeter Bus Selector: BATT
BAT: Standby (Switch Bent)
L GEN: ON
R GEN: ON
AVIONICS PLT: ON
AVIONICS COPLT: ON

Fuel
Transfer: OFF
L Boost Pump: ON
R Boost Pump: Switch Damaged

Environmental
Cabin Alt, Rate of Climb: Both needles 0 (zero)
Bleed Air: Knob Missing, flat aligned at 10:00 position
AUTO TEMP CKPT: 2:00 (O'clock)
AUTO TEMP CABIN: 2:00
BLOWERS CKPT: 11:00
BLOWERS CABIN: 11:00

Copilot Air
Pull On: In

Engine
L IGNITION: ARM

R IGNITION: ARM
L ECU: ON
R ECU: ON
L START, R START: Intact
SYNC: OFF

Pitch Trim
ND/NU Neutral (Rocker)
Selector switch in NORM position

Rudder & Roll Trim
Rudder Trim Switch: Neutral
RUD BOOST: NORM
RUD TRIM: NORM
ROLL TRIM: OFF

Lift Dump
Lock/Unlock: Unlock
Lift Dump Handle: Retracted (Handle end separated)

Engine Power

The left and right power (thrust) levers were noted in the NORM TAKE OFF detent. Both levers were bent to the right and forward approximately 45°. The power levers were exercised through normal flight range from NORM TAKE OFF to IDLE. The levers operated smoothly and stopped at both the NORM TAKE OFF and IDLE detents. The cutoff levers were bent due to power lever deformation and could not be moved.

Parking Emergency Brake

The parking brake handle was stowed and the handle could be operated (actuated) normally.

Flaps

The flightdeck flap handle was located in the wreckage on the center pedestal. The handle was noted in the 20° detent, handle deformed to the right approximately 80°. The flap handle operated smoothly and was firmly in the 20° detent.

h. Landing Gear

h.1.0 LANDING GEAR DESCRIPTION

The airplane was fitted with a retractable, tricycle style landing gear – one nose gear and two main landing gears. The landing gears are electrically controlled and hydraulically actuated. The landing gear is extended/retracted primarily by use of the landing gear control handle. The airplane also has an alternate extension system.

h.1.1 – Primary Extension and Retraction

The primary means of extension and retraction of the landing gear is controlled by the landing gear control handle. The handle is located in the flight deck, on the pilot's right subpanel. The subpanel consists of the landing gear control handle, three green NOSE-L-R indicator lights for down and locked indication, and two red warning lights, located in the knob of the landing gear control handle. The red warning lights indicate that the handle is not in agreement with the gear position or that a landing gear warning condition (in transit or not locked) exists.

The landing gear aural warning is annunciated when any landing gear is not down and locked, one or both thrust levers are retarded to a low power setting, and airspeed below approximately 135 KIAS. The warning is silenced by the HORN CUT switch located on each control wheel. The warning system is reset when the thrust lever is advanced.

The landing gear extension and retraction system uses 28 VDC power from the right main electrical bus.

h.1.2 - Loss Of Power – Alternate System

In the event of a loss of the primary landing gear system (including the loss of airplane electrical power), the airplane is equipped with a landing gear alternate extension system. The main component of the alternate system is the ALTN GEAR RELEASE handle, located under the pilot's subpanel.

The alternate gear extension system is used by manually pulling ALTN GEAR RELEASE handle to release the gear and door uplock hooks via cables connected to quadrant assemblies in the nose and main landing gear assemblies.

The ALTN GEAR RELEASE handle is stowed flush with the pilot's instrument sub-panel. The forward quadrant is on the right side of the forward nose keel, aft of the pressure bulkhead. Three control cables attach to the quadrant: handle, nose gear uplock release actuator, and the quadrant interconnect cable. The quadrant interconnect cable runs aft under the cabin floor, exits the cabin immediately forward of the wing wheel well center beam, and fastens to the aft quadrant. Individual control cables run from the recirculation valve and aft quadrant to the main gear and inboard door uplock hooks and to the dump valve lever.

When the ALTN GEAR RELEASE handle is manually pulled to its full actuation, the gears are released and free-falls to the down and locked position. The alternate release mechanism is sequenced so that the main gear inboard doors are opened before the main gear are released. A recirculation valve connects the main landing gear retract actuator retract line to the extend line to allow a positive free fall during alternate extensions.

According to Hawker Beechcraft Model 390 Maintenance Manual – Maintenance Procedures Section 32-30-00-501, Extension and Retraction – Adjustment/Test, Landing Gear Functional Test, Section 3.B, Landing Gear Alternate Extension System Functional Test, the alternate gear release handle travel will release the landing gear components at the following measurements:

- Nose Landing Gear Extension = 2.25 ± 0.25 inch handle travel
- Main Landing Gear Inboard doors extend = 2.75 ± 0.25 inch handle travel
- Main Landing Gear Extend = 3.25 ± 0.25 inch handle travel
- Alternate Gear Handle full extension = 4.0 inch

h.2.0 ON-SCENE EXAMINATION OF LANDING GEAR

h.2.1 Nose Landing Gear

The nose landing gear was found separated from the airframe trunnion. The nose landing gear drag brace separated from the nose landing gear assembly and the airplane structure. The down lock actuator was separated from the drag brace assembly. The down lock "pawl" assembly also separated from the drag brace assembly. The wheel and tire assembly remained attached to the nose landing gear assembly. The wheel assembly exhibited signs of impact damage to a portion of the bead area of the wheel. The nose landing gear doors separated from the airframe and were found among the main wreckage. The nose landing gear actuator separated from the airframe and separated into two pieces with piston portion remaining attached to the nose landing gear assembly. See Figure 4.

Figure 4 - Nose Landing Gear Components at Accident Site

h.2.2 Main Landing Gear

The landing gear handle in the cockpit was observed to be in the "UP" position.

The left main landing gear assembly was found in the retracted position. The gear was attached to the left wing trunnion and appeared intact. The gear was found in the wheel well, but the uplock was not engaged onto the main landing gear uplock roller. The left main landing gear actuator was found attached to the main landing gear assembly and to the wing assembly.

The left outboard gear door remained attached to the wing structure and the left main landing gear assembly. The left inboard gear door separated from the wing. The left inboard gear door actuator remained attached to the wing. Separation of hydraulic lines and impact damage made it such that usable position information could not be retrieved from the actuator. The inboard gear door was found in pieces among the main wreckage. Approximately 90% of the inboard gear door was recovered and was partially reconstructed. The paint on the exterior portions of the door that were found had been eroded consistent with contact with an abrasive surface.

The right main landing gear remained intact and attached to the wing structure. The right wing was separated between the main landing gear trunnion fitting and the main landing gear actuator wing attach fitting. The main landing gear actuator remained attached to the main landing gear assembly and the wings attach fitting effectively keeping the separated portion of the right wing connected. The right main landing gear actuator was partially extended such that the right main landing gear was not in the fully retracted or the down and locked position.

Approximately 60% of the right inboard gear door was recovered and reconstructed. The reconstructed portion of the door exhibited exterior paint abrasion that, according to Beechcraft, was consistent with the door in the closed position.

The inboard gear door actuator remained attached the wing. The separation of the hydraulic lines and impact damage made position indications not possible.

The right main landing gear outboard door was separated from the wing and was not located.

h.2.3 Landing Gear Alternate Extension Handle Examination

The landing gear alternate extension handle was located in the wreckage. The actuation of the handle was measured at an extended position of approximately 1 ½ inches. See Figure 5.

Figure 5 - Alternate Landing Gear Release Handle

i. Navigation

i.1 Air Data Computers

The airplane airspeed, mach, and altitude data is computed by use of the two air data computers (ADC). The Rockwell Collins Air Data Computer ADC-3000 measures static and impact pressures and resistance of an external total air temperature probe. The ADC performs the air data computations related to these measurements and outputs the computed air data parameters like air speed, altitude and Mach for use by the aircraft avionics system.

The ADC number 1 receives power from the Essential Avionics Bus and the ADC number 2 receives power from the Right Avionics Bus.

The nonvolatile memory in the ADC contains information related to failures and other events such as weight on wheel transitions and power cycles. The nonvolatile memory is recorded at the normal completion and power cycle of the airplane.

i.2 Enhanced Ground Proximity Warning System

The airplane's enhanced ground proximity warning system (EGPWS) unit provides aural warnings regarding terrain avoidance, too low flight without landing gear, etc. The unit uses position and airplane data to determine and annunciate the warnings.

The warnings are recorded in the EGPWS's non-volatile warning at the time the warning is calculated.

The EGPWS receives power from the left avionics bus.

j. Oxygen

The oxygen system supply bottle was noted in the wreckage. The bottle was damaged with the regulator broken off.

k. Engine Control System

Engine thrust is controlled by the thrust levers, located on the center pedestal in the flight deck. One thrust lever per engine. Angular displacement of the thrust levers is translated, via levers and cables, into displacement/movement of the power control cables attached to the engine fuel control unit in the aft section of the airplane.

The center pedestal is labelled in four areas of the thrust levers – CUT OFF, IDLE, NORM TAKE OFF, and TAKE OFF. In addition, the thrust levers have four stop positions at each of these positions, as well as a single detent at the MCT position. To move the thrust levers forward from CUT OFF to IDLE or backward to CUT OFF, there are finger pull-up locks on the thrust levers. See Figure 6.

Figure 6 - Throttle Quadrant and Engine Controls, including IDLE and CUT OFF positions (picture taken in sister ship).

The engine thrust levers have a physical stop at the flight idle positions which prevents the thrust lever from being accidently pulled into the cutoff position. The finger pull-up lever locks the thrust lever in place when pulled backward into the cutoff position.

If, during flight with engines operating, the thrust lever is moved from the NORM TAKE OFF area back to the IDLE position and then to the CUT OFF position, the finger pull-up locks must be used. By placing the thrust levers into the CUT OFF position, the engine is shut down and fuel is stopped from the engine.

For more information regarding the post-accident examination of the throttle quadrant, see Airworthiness Group Chairman’s Factual Report of Investigation, dated March 2, 2015, Section 7.0 - Engine Controls – Throttle Quadrant Assembly. In addition, see Figure 7 for photograph of the Throttle Quadrant Assembly, as found at the accident site.

Figure 7 - Center Pedestal and Throttle Quadrant (Arrow points to thrust levers)

II. AIRPLANE STRUCTURES

After the airplane wreckage was moved to the airport hangar, the relevant airplane structure was documented according to the following categories:

a. Doors

The main cabin door and the emergency exits were located in the wreckage. The main entry door remained attached at both hinge attach locations and was found open with the latches in the closed position. The main entry door latching mechanism was actuated and operated normally.

b. Fuselage

The radome was separated from the radome (forward) bulkhead and the radome bulkhead separated from the fuselage. The nose baggage and avionics section were separated forward of the forward pressure bulkhead. The nose wheel well was impacted, and the nose landing gear strut assembly was separated. The cabin area exhibited impact damage but remained intact from below the forward windscreens, aft of the forward pressure bulkhead and was contiguous to the aft pressure bulkhead. A portion of the right cabin sidewall from the emergency escape hatch opening forward to approximately the right side galley area had been cut open by first responders to aid in extrication of occupants. The emergency escape door was found among the main wreckage. The aft fuselage separated from the cabin portion at the aft pressure bulkhead. Both engines remained attached to the aft fuselage. The rear fuselage separated from the cabin just aft of the aft pressure bulkhead. The aft fuselage remained attached by flight control cables and other conduits.

c. Stabilizers

The horizontal stabilizer remained attached to the rear fuselage and revealed limited impact damage. The right tip of the stabilizer revealed compression damage. The elevators remained attached to the horizontal stabilizer at all hinges. The right elevator outboard end, including the balance weight separated from the right elevator. The right and left elevator trim tab surfaces remained attached to their respective elevators at their hinges. The rudder remained attached to the rear fuselage at all hinges and revealed no apparent damage. The rudder trim tab remained attached to the rudder at the hinges and did not reveal any damage. The rudder trim tab surface was visually aligned (faired) with the trailing edge of the rudder.

d. Wings

The wing assembly was separated from the airframe at all mounting points. The left wing was deformed by impact forces, but remained intact with all flight control surfaces attached.

The right wing was deformed by impact forces and separated in several locations. The inboard end of the right wing remained attached to the wing. The outboard portion of the right wing, outboard of the inboard flap, revealed impact damage, was deformed and separated into several pieces. The outboard end of the right wing, from the aileron outboard separated as one piece, with the exception of the composite wing tip assembly which also separated from the outboard end and found among the main wreckage.

The lower skin of the outboard portion of right wing and the lower skin of the composite wing tip exhibited gouging/scoring consistent with contact with an abrasive surface (See Figures 8 and 9). The marks made by the gouging/scoring were approximately parallel with the chord of the wing and aligned with the longitudinal axis of the fuselage. Abrasion/gouging consistent with that found on the lower outboard skin of the right wing and wing tip was also observed on the lower aft portion of all right wing flap tracks and the aft approximately one third of the wing center keel structure.

Figure 8 - Abrasions on Lower Wing Surface

Figure 9 - Abrasions on Lower Wing Surface

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