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

Airworthiness Group Chairman's Factual Report

February 4, 2016

A. ACCIDENT DCA15FA085

Location:	LaGuardia Airport, Queens, New York
Date:	March 5, 2015
Time:	1102 Local Time
Aircraft:	Delta Air Lines flight 1086, a Boeing MD-88, registration N909DL

B. GROUP

On-Scene Investigation, March 6-12, 2015:

Chairman:	Tom Jacky National Transportation Safety Board Washington, D.C.
Member:	Aziz Ahmed Federal Aviation Administration Westbury, New York
Member:	Chris Heck Air Line Pilots Association Herndon, Virginia
Member:	Rick Hagen The Boeing Company Seal Beach, California
Member:	Brendan O'Driscoll The Boeing Company Seal Beach, California
Member:	Brian Hanson Delta Air Lines Atlanta, Georgia

Examination of Crane Components, Burbank, CA May 5-7, 2015:

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

Member: Simon Leung
Federal Aviation Administration
Long Beach, California

Member: Chris Heck
Air Line Pilots Association
Herndon, Virginia

Member: Brendan O'Driscoll
The Boeing Company
Seal Beach, California

Member: Brian Hanson
Delta Air Lines
Atlanta, Georgia

Examination at Crane Aerospace, Lynwood, WA May 12, 2015:

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

Member: Brian Hanson
Delta Air Lines
Atlanta, Georgia

Examinations at Delta Air Lines Technical Operations, July 16, 2015:

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

Member: Chris Heck
Air Line Pilots Association
Herndon, Virginia

Member: Brian Hanson
Delta Air Lines
Atlanta, Georgia

Examination at Ametek Aerospace & Defense, Irvine, CA, September 30, 2015:

Chairman:	Tom Jacky National Transportation Safety Board Washington, D.C.
Member:	Chris Heck Air Line Pilots Association Herndon, Virginia
Member:	Brendan O'Driscoll The Boeing Company Seal Beach, California
Member:	Brian Hanson Delta Air Lines Atlanta, Georgia

Examination at Aero Fluid Products, November 18, 2015:

Chairman:	Tom Jacky National Transportation Safety Board Washington, D.C.
Member:	Chris Heck Air Line Pilots Association Herndon, Virginia
Member:	Brian Hanson Delta Air Lines Atlanta, Georgia

C. SUMMARY

On March 5, 2015, about 1102 eastern standard time (EST), a Boeing MD-88, N909DL, operating as Delta Air Lines flight 1086, was landing on runway 13 at LaGuardia Airport, New York, New York, and exited the left side of the runway, contacted the airport perimeter fence, and came to rest with the airplane nose on an embankment next to Flushing Bay. The 127 passengers received either minor injuries or were not injured, and the 3 flight attendants and 2 flight crew were not injured. The airplane was substantially damaged. Flight 1086 was a regularly scheduled passenger flight from Hartsfield-Jackson Atlanta International Airport (ATL) operating under the provisions of 14 Code of Federal Regulations (CFR) Part 121. Instrument meteorological conditions (IMC) prevailed, and an instrument flight rules (IFR) flight plan was filed.

The group met at LaGuardia Airport from March 6 – 12, 2015 to document the relevant airplane powerplants, structure and systems. At the time of the group's arrival the airplane had been moved from the airport field (accident site) to a maintenance hangar. The group documented the airplane inside of the hangar.

At the end of the on-scene phase of the investigation, the following airplane components were removed and retained by the National Transportation Safety Board for further examination:

1. Anti-skid control unit
2. Auto brake control unit
3. Anti-skid valves, 4 total
4. Wheel speed transducers, 4 total
5. Auto spoiler switching unit
6. Auto spoiler control box
7. Auto spoiler actuator in pedestal
8. EGPWS (Enhanced Ground Proximity and Warning System)
9. PSEU (Proximity Switch Electronics Unit)
10. Nose wheel steering control valve
11. Brake pressure transducers for flight data recorder (FDR), 2 total
12. Auto brake flight deck panel
13. Digital Flight Data Acquisition Unit (DFDAU)
14. Lateral Control Position Sensor for the FDR
15. MiniQAR

The group met at the Crane Hydro-Aire facility in Burbank, California from May 5-7, 2015 for the examination of the following components removed from the airplane:

1. Anti-Skid Control Box
   Crane Hydro-Aire Part Number: 42-807
   Serial Number: 950

2. Auto Brake Control Box
   Crane Hydro-Aire Part Number: 42-809-3
   Serial Number: 343A

3. Auto Brake Control Panel
   Crane Hydro-Aire Part Number: 42-831
   Serial Number: 540

4. Anti-skid Dual Control Valve, Left Outboard Position
   Crane Hydro-Aire Part Number: 39-249-2
   Serial Number: 4759

5. Anti-skid Dual Control Valve, Left Inboard Position
   Crane Hydro-Aire Part Number: 39-249-2
   Serial Number: 754

6. Anti-skid Dual Control Valve, Right Inboard Position
   Crane Hydro-Aire Part Number: 39-249-2
   Serial Number: 2651ABC

7. Anti-skid Dual Control Valve, Right Outboard Position
   Crane Hydro-Aire Part Number: 39-249-2A
   Serial Number: 4384

8. Wheel Speed Transducer, Main Landing Gear, Wheel #1 (Left Outboard)
   Crane Hydro-Aire Part Number: 40-62575
   Serial Number: 7467A

9. Wheel Speed Transducer, Main Landing Gear, Wheel #2 (Left Inboard)
   Crane Hydro-Aire Part Number: 40-62575
   Serial Number: 7502

10. Wheel Speed Transducer, Main Landing Gear, Wheel #3 (Right Inboard)
    Crane Hydro-Aire Part Number: 40-62575
    Serial Number: 7980A

11. Wheel Speed Transducer, Main Landing Gear, Wheel #4 (Right Outboard)
    Crane Hydro-Aire Part Number: 40-62575
    Serial Number: 7967

12. Spoiler Control Box, Hytrol Mark II
    Crane Hydro-Aire Part Number: 42-091-3
    Serial Number: 1188

The group met at the Crane Aerospace & Electronics facility in Lynwood, Washington on May 12, 2015 for the examination of the Proximity Switch Electronics Unit (PSEU) removed from the airplane, documented as follows:

13. Proximity Switch Electronics Unit (PSEU)
    Crane Part Number: 8-336-05
    Serial Number: 860

The group met at the Delta Technical Operations facility at the Hartsfield-Jackson Atlanta International Airport, Atlanta, Georgia on July 16, 2015 for the examination of the following components removed from the airplane:

14. Auto Spoiler Switching Unit
    McDonnell Douglas Part Number: 5935212-501
    Serial Number: 763

15. Flight Data Acquisition Unit
    Teledyne Part Number: 2222601-6
    Serial Number: 2445

16. Enhanced Ground Proximity Warning System Computer
    Honeywell Part Number: 965-0976-003-212-212
    Serial Number: 1820

17. Nose Wheel Steering Control Valve (Manifold)
    Manufacturer's Part Number: 5914296-5509
    Serial Number: EFS-22315

The group met at the Ametek Aerospace & Defense facility in Irvine, California on September 30, 2015 for the examination of the following components removed from the airplane:

18. Brake Pressure Transducer (Left Position)
    Crane Hydro-Aire Part Number: IBA103G-93
    Serial Number: 90-0107

19. Brake Pressure Transducer (Right Position)
    Crane Hydro-Aire Part Number: IBA103G-93
    Serial Number: 91-1110

The group met at the Aero Fluid Products facility in Painesville, Ohio on November 18, 2015 for the examination of the airplane's Auto Spoiler Actuator Assembly, documented as follows:

20. Auto Spoiler Actuator Assembly
    Part Number: 1040T100-5

Serial Number:	2146B
Actuator Motor Serial Number:	2403

The examination and test plan for each component was agreed to by the group and were conducted in accordance with the component's associated Component Maintenance Manual (CMM) procedure.

At the conclusion of the examination, all pertinent documentation and photographs were provided to each of the parties.

None of the evidence gathered on-scene or examinations of the removed components revealed a failure of an airplane system. There were no pre-impact discrepancies found in the aileron, elevator, or rudder flight control systems.

D. DETAILS OF INVESTIGATION

The airplane was identified as:

Serial Number:	49540
Line Number:	1395
Delivery Date:	12/29/1987

At the group's arrival, the airplane had been moved into the American Airlines Maintenance Hangar, sitting on both main landing gear wheels, with the fuselage nose sitting on a dolly loaded with wood timbers for support.

Delta Air Lines notified the NTSB that the airplane was damaged during the recovery and movement from the accident site to the maintenance hangar. In addition, a nose gear linkage was cut to facilitate the recovery of the airplane.

The relevant aspects of the airplane's airworthiness were documented as follows:

1.0 AIRPLANE SYSTEMS

The group identified and documented relevant systems of the airplane for the investigation. The relevant airplane systems were documented according to the following categories:

1.1 Communications

In the Electrical/Electronics Compartment the VHF #2 and #3 radio units were noted as crushed. The VHF #1 unit was displaced with its associated radio rack displaced. See Figure 1.

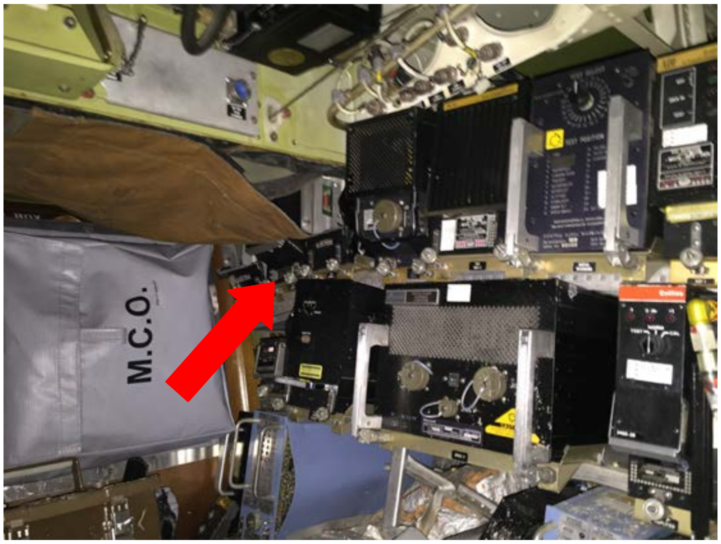

Figure 1- Damage to Electronics Compartment Rack, including VHF Radio Receivers (Red Arrow)

1.2 <u>Electrical Power</u>

Damage was noted in the lower areas of the Electrical/Electronics compartment. Several of the lower avionics racks in the compartment, as well as the avionics units installed on them, were displaced, bent, and damaged. The access door to the Station 110 relay panels was displaced and broken. See Figures 2 and 4.


Figure 2 - Damage to Electronics Compartment, including Station 110 Relay Panel Cover.

Significant damage to the area around the external power receptacle was noted. See Figure 3.

Figure 3 - Damage to area around the External Power Receptacle.

Damaged structure at this location along with displacement of structure located within the Electrical/Electronics Compartment led the group to conclude the application of electrical power on this airplane posed a safety risk. The noted damage included electrical power feeder cables for external power were displaced. This determination limited testing of systems and components to hydraulic testing (using a hydraulic mule) without the application of electrical power.

1.3.0 Equipment & Furnishings

1.3.1 Electronic Shelf/Compartment

Due to damage to the Electrical/Electronics Compartment Door on the underside of the forward fuselage, access to the Electrical/Electronics Compartment was limited to the hatch located behind the Captain's seat in the flight deck. To observe damage to the components located within, one group member climbed down through the opening to document the compartment. The Antiskid Control Unit along with the Autobrake Control Unit was inspected and no evidence of water or other liquid dripping into the boxes were found. Damage was observed including excessive damage to the station 110 relay panel. The left and right main batteries were observed to be displaced from their shelves. See Figure 4. The right battery connection was cracked and twisted from contact with structure. The forward right radio rack was damaged. The number 2 and 3 VHF communication boxes were crushed. The generator rack was also damaged.

Figure 4 - Damage to Airplane Main Batteries and Shelf. The 110 Relay Panel is behind the batteries.

1.3.2 Passenger Cabin Equipment and Furnishings

The on-scene examination of the passenger cabin equipment and furnishings provided the following observations:

- The aft megaphone was located in overhead bin row 35, ABC. It was noted in working condition.
- The forward megaphone was located in overhead bin row 1, AB. It was noted in working condition.
- Except for the tailcone flashlight, all emergency equipment was stowed in their original location.
- The tailcone release mechanism was noted in a position consistent with deployment from the aft cabin door. The tailcone placard above the passenger aft entrance door was punctured and ripped as designed (See Figure 5). The Kevlar lanyard was found intact and still attached to tailcone during inspection.

Figure 5 - Ripped tailcone placard above aft passenger door.

- The tailcone was separated from the airplane. The tailcone passenger inflatable slide and slide pack cover were also separated from the airplane. The aft tailcone slide pack cover was examined with no defects or damage noted.
- The tailcone slide was inspected. The inspection revealed a cut in the slide on the left hand side facing aft of first officer's side of the aircraft. Delta Air Lines indicated that the airport Crash, Fire, and Rescue (CFR) services made the cut to facilitate removal of the slide.
- An examination of on-scene, post-accident photographs and flight attendant statements indicated that the aft tailcone slide deployment system functioned when activated. However, the slide was noted underneath the aft fuselage while inflated. It was noted that the airplane came to rest with a nose up attitude (on the berm) and therefore the airplane tail was estimated to be approximately 3.5 to 4 feet above the ground when the slide was activated.
- Tailcone jettison handle was removed.
- The tailcone manual inflation handle was pulled.
- The tailcone catwalk was noted down and accessible.
- Two passenger service unit panels were dislodged and hanging down. The panels were located 3 rows forward of the 2L flight attendant jumpseat, at passenger rows 28 - 29 ABC.

The panels were dislodged on the inboard (aisle) side, hanging down approximately 8-10 inches, but the window side of the panels was still in their normal position. See Figure 6.

Figure 6 - Dislodged passenger service unit panels.

- The forward right hand overwing exit was noted stowed. Delta indicated that the exit had been re-stowed.
- The aft right hand overwing exit was found on floor and had been removed.
- First class lavatory trash can was noted out of its housing.
- The sidewall panel at row 25 ABC was displaced. The airplane logbook indicated the item was on a non-essential furnishings deferral at the time of the accident.
- No passenger seats were found displaced or exhibited visible damage.

1.4.0 Flight Controls

The location of the airplane's flight control surfaces are documented in Figure 7 below.

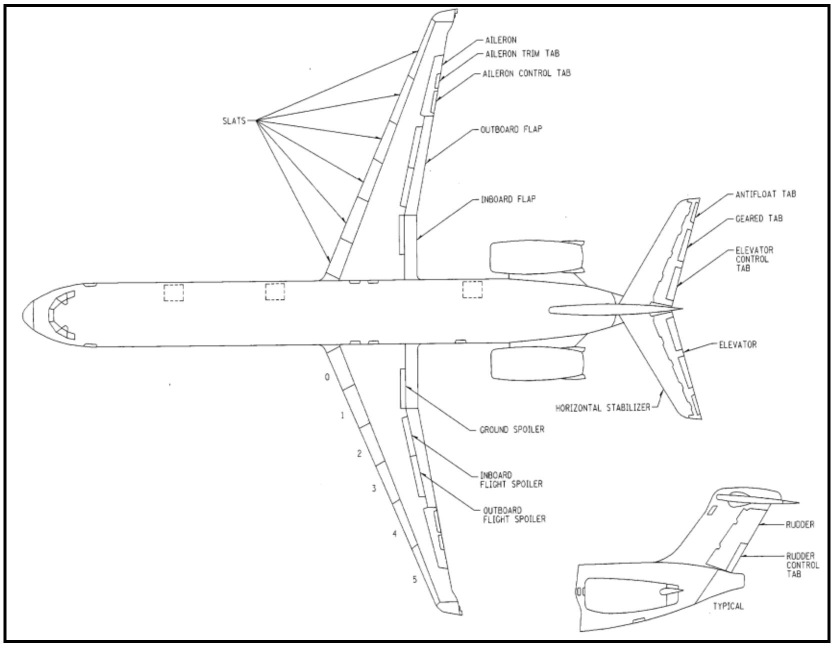

Figure 7 - MD-88 Flight Control Surfaces

1.4.1.0 Primary Flight Control Systems

1.4.1.1 <u>Pitch Control and Trim Systems</u>

A visual inspection was conducted from the ground. Both elevators were noted in a trailing edge up position. Boeing indicated that, without hydraulic power, the elevator trailing edge would float up. No visual anomalies were noted with the elevators or control inputs.

The flight deck indication of the horizontal stabilizer was noted at 7° airplane nose up. (ANU).

1.4.1.2 <u>Roll Control System</u>

<u>Left Wing Observations</u>:

The aileron surface was noted approximately 3-inch Trailing Edge Down (TED) – Bottom surface damaged due to fence.

The aileron trim tab was noted ⅛-¼ inch TED from a faired position.

The aileron control tab was noted 1¼ – 1 ½ inch TEU (Trailing Edge Up) from a faired position.

<u>Right Wing Observations</u>:

The aileron surface was noted approximately 3-inch Trailing Edge Up (TEU) – with no damage noted.

The aileron trim tab was noted ⅛ -¼ inch TEU from faired position, no damage.

The aileron control tab was noted 1¼ – 1½ inch TED (Trailing Edge Down) from faired position, no damage.

The airplane's lateral control position sensor for the FDR was removed from the airplane but, after consultation with the Flight Data Recorder Group Chairman, was not examined further.

1.4.1.3 Yaw Control and Trim Systems

The airplane's yaw control system is a traditional cable and hydraulically-actuated system with two rudder pedals for each flight crew position controlling the rudder surface on the rear of the vertical stabilizer.

The rudder system has a travel restrictor system which, based on a dedicated rudder pitot system, limits rudder travel. However, the rudder is not restricted below 173 knots; the full rudder throw of ±22.5° is available to the flightcrew via the pedals.

The rudder trim control knob is located on the center pedestal in the flight deck. The rudder trim knob moves the rudder trim through a series of cables and pulleys back to the rudder power package.

The rudder control surface and power pack, located in the tail section of the airplane, were visually examined. No visible damage or hydraulic leaks were noted. No evidence of a jam or other malfunction was noted. The rudder trim tab surface was noted at zero or null. The rig pin hole in the cable sector located in the empennage appeared to be aligned corresponding with the cockpit controls (pedal input) and surface positions.

The rudder pedals and associated flight deck rudder control or input components were inspected. No faults or foreign objects in the area of the pedals were found during the inspection.

1.4.2.0 Secondary Flight Control Systems

1.4.2.1 Wing Leading Edge Slats

The airplane has twelve wing leading edge slats, six per wing. The slats are numbered from 0 to 5 from the most inboard slat (0) to the most outboard (5).

The leading edge slats were not exercised by the group during the on-scene investigation.

According to Boeing, the actuation of each leading edge slat was consistent with the fully extended position.

The leading edge slats for both wings were examined and documented as follows:

Left Wing Observations:

The #5 slat was missing with the associated slat cable broken at the outboard pulley.

The #4 slat was damaged, with portions of the slat pressed into the fixed wing leading edge.

The #3 slat major damage and fixed leading edge damage. (Delta indicated this was from post-accident damage). The #3 slat track was displaced and rubbed against structure on the inboard side. (also indicated to be post accident damage)

The #1 slat major damage. (Delta indicated this was post accident damage)

Right Wing Observations

According to Delta, all damage to the right wing slats were as a result of the post-accident recovery of the airplane.

#1 slat - damage 2 ft. from inboard end.

#1 slat to #2 slat metal seal damaged due to the cocking from the #2 slat damage.

#5 slat outboard tip damage bent the number 15 (outboard idler) track.

1.4.2.2 Flight and Ground Spoilers

1.4.2.2.1 MD-88 Spoiler System Description

The MD-88 has two outboard flight spoilers and one inboard ground spoiler on each wing. The ground spoiler actuators are powered by both (left and right) hydraulic systems. Hydraulic power for the inboard flight spoiler actuators is supplied by the left hydraulic system. Similarly, the outboard flight spoiler actuators are supplied by the right hydraulic system.

The flight spoilers are controlled by the aileron control system - by either control wheel in the cockpit, or by the speedbrake control lever on the forward pedestal. When operated by the control wheels to supplement lateral control, the system extends the flight spoilers on one wing to a maximum of 60 degrees from the faired position. When extended via the speedbrake lever, all flight spoilers are extended symmetrically to a maximum of 35 degrees in flight and 60 degrees on the ground. The flight spoilers are mechanically held down by a torsion bar that is spring loaded to the retract position when not extended.

The ground spoilers are dual signal input systems. In order to extend, the spoilers (and their associated control valves) require a mechanical input from the speedbrake control lever and an electrical signal from the proximity switch electronic unit through several relays. The ground spoilers are automatically extended to about 60 degrees during landing (or a rejected takeoff) and are locked down by hydraulic power and an overcenter hold-down link during all other phases of flight.

Operation of the ground and flight spoilers during landing (or a rejected takeoff) may be manual or automatic. Automatic extension of the flight and ground spoilers is through the autospoiler system that mechanically moves the speedbrake control lever full aft. The system consists of the speedbrake control lever, ground spoiler actuator, autospoiler switching unit, wheel spin-up transducers via the ground spoiler control box, and nose oleo switches.

The auto spoiler actuator is retracted when the Auto Ground Spoilers are armed for Takeoff. The actuator remains retracted in preparation for spoiler handle arming prior to the next landing. The pilot "arms" the autospoiler system for landing by raising the speedbrake control lever up to the ARM position, thereby fully revealing the red ARM indicator stripe. This positions the roller on the speedbrake lever in front of the autospoiler crank arm. When the ground spoiler actuator is commanded to drive to the extend position, the actuator arm pushes the speedbrake lever fully aft, extending all the spoilers. The auto spoiler actuator drives the crank arm from the "retract" to "extend" position during each landing. Auto spoiler deployment requires the speedbrake control lever to be armed.

Two methods are available to automatically command extension of the auto spoiler actuator, which is electrically controlled by the autospoiler switching unit (ASU). During landing, the ASU triggers spoiler deployment with main wheel spin-up (both outboard wheels, both inboard wheels, or both wheels on one gear spin at approximately 700 rpm), via the wheel spin-up transducers. Main wheel spin-up is the primary trigger; however, in the absence of main wheel spin-up, spoiler actuator extension occurs automatically with nose strut compression via the nose oleo switches. In addition to the automatic methods, ground spoiler deployment can also be achieved by manually by moving the handle aft.

If a failure is detected in the autospoiler system, the autospoiler switching unit monitors will illuminate the amber "AUTOSPOILER FAIL" annunciator light on the flight deck Electronic Overhead Annunciator Panel. This light will illuminate if the actuator fails to change state within 10 seconds when commanded, if there is an internal short to ground in the spoiler control relay circuit or the ground spoiler control box circuit, or if only one takeoff or land relay channel is energized. In addition, the ground spoiler gear interlock relay will illuminate the annunciator if there is a difference between the two weight-on-wheel sensors and the landing gear handle.

1.4.2.2.2 On-Scene Spoiler System Examination

The flightdeck spoiler control components were examined. The auto spoiler actuator in the forward lower pedestal was found in the extended position, with approximately 3/8" of the red band (signifying armed position) and the handle riding the extend cam within the pedestal. When the ground spoiler actuator was manually retracted, the handle went to the full retracted position (no red band showing).

After the captain's side pedestal panel was removed, the auto spoiler actuator linkages were visually examined, with no faults or damage noted.

The spoiler control system cables were visually examined with no faults found. This airplane did not have the spoiler lockout mechanism installed.

Left Wing Observations

The ground and flight spoiler panels retracted and secured (checked by upward force by hand) with hydraulics off.

The left-hand outboard flight spoiler trailing edge was damaged due to contact with the outboard trailing edge flap vane. The aft outboard section of the spoiler panel was found underneath the left outboard flap vane.

The flight spoiler hold down torsion bars were visually inspected, with no anomalies noted.

Right Wing Observations

The ground and flight spoiler panels were retracted and secured (checked by upward force by hand) with hydraulics off.

No visual damage to any of the ground or flight spoiler panels was observed.

The flight spoiler hold down torsion bars were visually inspected, with no anomalies noted.

1.4.2.2.3 Spoiler Control Box Examination

The spoiler control box was maintained internally by Delta Air Lines. The last Delta shop visit was recorded on 7/9/2014. In addition, the Crane Hydro-Aire database of all customer returns from 1998 until present had no record of the unit returning to Crane.

A visual inspection was performed and the unit appeared in good condition. All connector pins appeared straight.

The Spoiler Control Box was placed onto a Crane Hydro-Aire Hytrol Mark II Spoiler Control Test Set and power applied. A functional test was performed on the unit per the Crane Aerospace Test Procedure TP42-091-3, Revision H, dated December 11, 1997.

With the exception of test elements 7.2.1, 7.3.2, 7.5.1, and 7.6.2, the unit passed all tests with no fault found. For the excepted elements, the unit passed the output volts peak-to-peak recorded value, but the "trigger voltage" (the wheel speed transducer output voltage to trigger/activate the spoilers) exceeded the nominal trigger voltage of 2.2 Volts by 0.05V, 0.06V, 0.02V, and 0.02V for the left inboard, right outboard, left outboard, and right inboard spoilers, respectively.

Despite the noted failures, the control box passed the elements of the test related to spoiler activation. The resultant test data sheet is included in Attachment 1.

1.4.2.2.4 Auto Spoiler Switching Unit Examination

The testing of the auto spoiler switching unit was accomplished in accordance with the Testing and Fault Isolation portion of the Boeing Overhaul Manual, Section 27-62-6, dated May 15, 2013.

A visual examination of the unit indicated that all connector pins appeared straight. A small indention was noted on the lower left edge of the unit's dust cover (when looking at the front panel of the unit). When the dust cover was removed, slight damage was noted to the lower left chassis connection to the front panel. No damage was noted to any of the internal components.

The unit was tested using the Delta Auto Spoiler Switching Unit Test Isolation box.

All of the Test elements in Table 101 and 102 were accomplished on the unit. The tests were accomplished with no faults found.

1.4.2.2.6 Auto Spoiler Actuator Examination

The actuator was originally manufactured by The Talley Corporation but the product manufacturing had since been acquired by Aero Fluid Products, in 2011.

According to Delta Air Lines records, the unit was installed on the accident airplane on May 15, 2013.

Delta Air Lines uses an outside vendor to maintain their auto spoiler actuators. According to Delta, the last shop visit for this unit was about 5/15/2013. In addition, Aero Fluid Products had no record of the unit returning for repair or service in their database, which covered the time period from 2011 until the day of the examination.

The unit was then subjected to an incoming visual examination. A safety wire was noted with excessive number of twists per inch. No other visual damage or inconsistencies were noted.

The group agreed to conduct the acceptance test protocol on the actuator, as described by the Testing and Fault Isolation Section of the actuator CMM (27-60-07). To accomplish the test, the actuator assembly was attached to the Aero Fluid Products test stand. The test results were noted on the Acceptance Test Data Sheet that was part of the test procedure.

All elements of the test procedure were conducted; the actuator assembly passed all elements, with no faults found.

At the request of the group, acceptance test element 1.D.(1), Functional Test, was conducted an additional five times in each direction (clockwise and counterclockwise). These additional results were documented and entered into a data table; all additional functional test runs were satisfactory and met specifications.

The Test Data Sheets for the actuator, including the additional Functional Test element 1.D.(1) documentation table, were included in Attachment 2.

1.4.2.3 Wing Trailing Edge Flaps

According to Boeing, the actuation of the trailing edge flaps was consistent with an extension to 40 degrees. The trailing edge flap system was not activated during the on-scene portion of the investigation.

The wing trailing edge flaps were documented as follows:

Left wing Observations:

Significant damage was observed to the Inboard and Outboard Flap surfaces. For the Outboard Flap, outboard hinge fitting was pulled from the aft spar creating a hole at the bottom of the fitting attach point.

The outboard flap fairings were damaged.

All flap actuators appeared to be in good condition, with no leaks noted, and safetied.

The Flap Bus Cable tension was taut, the control cable nylon cover inside and outside of fuselage noted in good condition, and no anomalies were noted with the pulleys in the LH wheel well.

Right wing observations:

The trailing edge flap surfaces were noted as extended to 40 degrees, and was verified at the inboard flap track mark. No obvious damage to the inboard and outboard flap surfaces.

All flap actuators appeared to be in good condition, with no leaks noted, and safetied.

The Flap Bus Cable tension was taut, the control cable nylon cover inside and outside of fuselage noted in good condition, and no anomalies were noted with the pulleys in the RH wheel well.

1.5 Fuel

Delta Air Lines indicated that 2,120 gallons of fuel were removed from the airplane before the airplane was moved to the hangar.

The left wing fuel tank was noted as breached in the area of the left outboard trailing edge flap idler hinge attachment point. The attachment fitting for the flap was separated from the wing, which opened the fuel tank. During the on-scene investigation, fuel was noted dripping from the area of the breach.

1.6 Hydraulic Power

The hydraulic lines in the area of the forward fuselage/nose wheel well area were damaged and opened. The damaged hydraulic lines in this area were capped prior to the hydraulic exercise (See Section 4.0).

During the hydraulic testing the left and right hydraulic systems were exercised with no anomalies noted. In addition, a visual examination of the hydraulic components was completed with no anomalies noted.

1.7 Ice & Rain Protection

Windscreen wiper blades were noted as attached and exhibited no visible damage.

1.8 Instruments

Flight Deck Indications

The flight deck instrument panels and circuit breakers were examined and documented. The results of the documentation were included as an attachment to the field notes. Included in the documentation of flight deck were the remaining personal items. The flight deck panel documentation is included in Attachment 3.

The flightdeck electrical power center circuit breaker panels were documented for status – either Open/Tripped, Closed, Inoperative, or Deactivated. A table of the circuit breakers that were noted as any status other than "Closed" is included below in Figure 8.

	Circuit Breaker Panel Report	
Location	**Description**	**Status**
B7	RT Radio Bus Flight Recorder	INOP

C14	LT Radio Bus Flight Recorder	INOP
G21	LT DC Bus Flight Recorder	INOP
L7	RT AC Bus Wind Shear Computer	Tripped/Open
F6	Cockpit Voice Recorder, RT Radio Bus	INOP
F21	Flight recorder LT AC bus	INOP
K12	LT Wing Landing Left AC Bus	Tripped/Open
K14	LT Nose Gear Landing and Taxi	Tripped/Open
M26	LT Rain Repellent, Ice Protect LT DC Bus	Permanent Deactivated
N26	Ice Protection RT DC Bus, RT rain repellent	Permanent Deactivated
U21	Air Condition & MISC Bus, Over Heat Wheelwell Sensor	Tripped/Open
Z34	DC Transfer Bus ,FWD Passenger Entry Stairs Carriage.	Permanent Deactivated
Z35	Door Motors	Permanent Deactivated
Z36	Door Controls	Permanent Deactivated
0A	Aft Galley Dash 4 Circuit Breaker	Permanent Deactivated
0B	Aft Galley Dash 4 Circuit Breaker	Permanent Deactivated
0C	Aft Galley Dash 4 Circuit Breaker	Permanent Deactivated
OB	Cabin Outlet	Permanent Deactivated
OC	15V AC Utility Computer 1	Deactivated Placard
A15	Captain and FO White Flood Lights	Deactivated
A16	Control Set	Deactivated
A17	Carriage Motor 1	Deactivated
A18	Carriage Motor 2	Deactivated

Figure 8 - Electrical Center Circuit Breaker Status

Also, deactivated circuit breakers were identified by a black collar or placard.

The airplane's cockpit voice recorder (CVR), flight data recorder (FDR), and flight data acquisition unit (FDAU) were removed from the airplane for further examination. The inoperative (INOP in the table) circuit breakers were identified by orange "INOP" Tags on the breaker. Delta indicated that the B7, C14, G21, F6, and F21 circuit breakers (i.e., all circuit breakers indicated as INOP) were opened to remove the airplane's CVR and FDR.

The Airworthiness Group participated in the examination of the FDAU at the Delta Technical Operations facility at the Hartsfield-Jackson Atlanta International Airport, Atlanta, Georgia on July 16, 2015. For further information regarding the examination, please see the <u>FDR Group Chairman's Factual Report</u>.

1.9.0 Landing Gear

The airplane was fitted with 2 landing gear systems – the main landing gear and nose gear. The gears were examined as follows:

1.9.1 Main Landing Gear System

The gear components were visually inspected with no appreciable impact damage noted. No visible fractures were noted. The gear were covered with mud, grass, and packed snow and ice.

All four main landing gear wheel hubcaps were removed to observe the antiskid wheel speed transducer drive pins. All drive pins were found installed and had proper engagement with the hubcaps.

The main landing gears were lifted from the hangar floor using a hydraulic jack. For each landing gear, when lifted, each tire rotated freely, and when the tire was removed, each of the four wheel brakes were examined. In each case, the brake assembly and AMS 6302 steel brake pads showed no indications of overheating, binding, or leaking. In addition, each brake showed little wear. All rotating brake components, including the gear axle antiskid transducers, were undamaged and rotated freely. Finally, the brake wear indicator pins, two for each wheel, were measured; all were noted within tolerances.

1.9.1.1 Left Main Landing Gear Examination

The scissor link was found in place. The shimmy damper was installed and serviced with hydraulic fluid within the normal range. The left MLG strut had approximately 4 inches of chrome showing. Brake hydraulic tubing was found to be installed in the correct location and showed no evidence of hydraulic leakage. No damage was observed to the brake hydraulic lines or antiskid electrical conduits. The weight on wheels proximity sensor and scissor linkage was found undamaged and the flexible electrical conduit had no evidence of damage. The spray deflector showed no evidence of damage.

<u>Wheel position 1</u>

The wheel assembly for position 1 (serial number 979) was observed to be in serviceable condition. The tire showed no evidence of unusual wear patterns such as flat spotting that would be an indicator of abnormal antiskid activity. Forward and aft flexible hydraulic lines were routed and secured in the normal position. The brake line quick disconnect fittings were in place and secured. The brake was examined and was found to have serviceable brake life remaining as indicated by the forward and aft brake wear pin indicators. No evidence of hydraulic leakage was found. The hubcap was then re-installed.

<u>Wheel position 2</u>

The wheel assembly for position 2 (serial number 852) was found to be in serviceable condition. The tire showed no evidence of unusual wear patterns. Forward and aft brake hydraulic lines were secured in the normal position. The brake quick-disconnect fittings were connected and secured. Brake wear pins showed serviceable brake life remaining. Examination of the brake showed no evidence of hydraulic leakage. The hubcap was then re-installed.

1.9.1.2 Right Main Landing Gear Examination

The antiskid electrical conduit showed no evidence of damage and the brake hydraulic lines show no evidence of damage. Scissor link and shimmy damper undamaged. Shimmy damper serviced within limits and no leakage observed. Weight on wheels sensor and associated mounting brackets undamaged. The strut appeared serviced and showed approximately 4 inches of chrome. No damage was observed to the brake hydraulic lines or antiskid electrical conduits. Number 4 brake temperature sensor convoluted tubing had visible damage, crushed but no cutting or cracking observed. Spray deflector was installed and undamaged.

Wheel position 3

The wheel assembly for position 3 (serial number 520) was examined. Found no evidence of flat spotting. Tire wear was within limits for service. Gouges were observed in the tread of the tire. Brake flexible hydraulic lines were found installed. Quick disconnect fittings were installed and secured. Forward and aft brake wear pins indicated serviceable brake life remaining. There was no evidence of hydraulic leakage from the brake. The hub cap was installed.

Wheel position 4

The wheel assembly for position 4 (serial number 889) was examined. No evidence of tire flat spotting was noted and tire wear was within serviceable limits. Tire had deep gouges in the tread and one shallow gouge in the sidewall. Forward and aft brake flexible hoses were installed. The brake quick-disconnect fittings installed and secured. Brake forward and aft wear pins showed serviceable brake life remaining. No evidence of hydraulic leakage was observed. The hub cap was installed.

1.9.1.3 Proximity Switch Electronics Unit Examination

During the initial visual examination of the unit, it was noted that the lower left edge of the chassis (when looking at the front of the PSEU) exhibited slight impact damage; the edge was lifted up away from the chassis. The damage also pushed the rear top edge of the top of the chassis upwards. When the plastic protective caps were removed from the connectors, all connector pins appeared straight.

The component was placed onto the Crane Aerospace MD88 PSEU Test Fixture (part number 30-058-12). Initially, the PSEU unit would not fit into the MCU rack that was attached to the test stand shelf. Packing tape was used to compress or hold down the rear edge of the unit so that the PSEU would properly mate the rear connectors of the PSEU to the shelf connectors.

The Crane Aerospace Acceptance Test Procedure TP8-336-05, Revision K, dated October 21, 1996, Acceptance Test Procedures Control Unit, DC9-80 Proximity Switch System, was accomplished on the unit. The group determined that the following elements of the ATP were not applicable to the PSEU (as a returned unit) for investigation:

- 4.1.3 Installation and Verification
- 4.1.5 Burn-In Test
- 4.2 Insulation Test

As part of the ATP, the unit passed BITE target far, near, and ground mode.

During the execution of Test 4.5.3, Gear Logic Tests (Nose, Left Main, Right Main, Door), the PSEU initially failed individual steps of Table 4-2, Gear Logic Truth Table (Inputs and Outputs). On step 5 of Table 4-2, Lamp/Diode 24 on the test console (corresponding to Left Main Up Latch) did not illuminate as a result of test inputs that should have illuminated the lamp. The following Table 4-2 steps failed for the same reason: steps 10 through steps 15, steps 17, and step 18. However, during step 18 Diode 24 started illuminating, functioning normally so that the PSEU passed the step. As a result, step 17 was re-tested and Diode 24 illuminated and the PSEU passed. Then the PSEU passed steps 19 and 20. Finally, step 16 was re-tested and Diode 24 did not illuminate, which was the correct state for the PSEU to pass. The group elected not to re-test the earlier steps in which Diode 24 failed to illuminate.

The remainder of the ATP test was accomplished with no faults found.

The ATP test sheets for the PSEU were included in Attachment 4.

1.9.2 Airplane Brake Systems

1.9.2.1 Brake Systems Descriptions

The airplanes brakes are located on the main landing gears. The airplane braking systems included a manual, parking, and auto-brake system.

Manual brakes are applied via the brake pedals in the flight deck. The automatic brake system (ABS) is an electrically-controlled means of automatically applying the brakes on an airplane touchdown and maintaining a constant level of deceleration. The ABS system includes the ABS control panel, ABS control unit, hydraulic land manifold, and hydraulic takeoff manifold. The airplane's parking brake is selected via the flight deck lever.

The ABS has two modes of operation, selected on the ABS control panel: the landing mode and the takeoff mode[1]. The mode is selected on the ABS control panel, located on the aft, right side of the center pedestal. The panel contains two switches; a rotary type and a toggle type. The rotary switch is a 5 position switch used to select the mode of operation and/or deceleration rate. The switch is placarded T.O.(takeoff), OFF, MIN, MED, and MAX. The toggle switch is the DISARM/ARM switch. To arm a mode, the desired deceleration is selected on the Control Panel and the DISARM/ARM switch is placed in the ARM position. This turns on the blue ARM light on the control unit and, if all failure monitors are satisfied (within the ABS control unit), the ARM toggle switch will be magnetically held.

[1] The takeoff mode is used for rejected takeoffs.

For the ABS landing mode, one of the three land positions (MIN, MED, and MAX) is selected on the rotary switch (prior to landing) to determine the airplane deceleration rate during the landing roll[2]. Then, during the landing roll, the ABS operates by using (only) the right brake hydraulic system and compares actual airplane deceleration (derived from wheel speed) with the pilot's selection. Brake system pressure is then modulated to the brakes by the land manifold to maintain the fixed level of deceleration. In the MAX position, full right brake system pressure is applied to the brakes and maximum deceleration is limited to anti-skid system operation.

The landing mode operation automatic braking activity is initiated by spoiler deployment. This occurs with throttles retarded and after the ABS deceleration level is selected and armed. Throttle lever position information is used as a brake inhibit function until spoilers are deployed. If throttle levers are subsequently advanced beyond 20 degrees, ABS operation is prevented. With throttle levers retarded and spoilers deployed, automatic braking is implemented after a delay of one second if MAX is selected or 3 seconds with MIN or MED. With another second allowed for hydraulic delays including brake fill, the total delay to the beginning of braking is approximately 2 seconds and 4 seconds, respectively. These delays are intended to allow for a normal nose touchdown while maintaining a predictable stopping distance.

The ABS Control Unit functions with the Antiskid Control System during landing rollout. The system compares aircraft deceleration derived from wheel speed with the selected deceleration level. Then the ABS control unit modulates brake pressure to achieve the selected deceleration level. The ABS control unit receives wheel-speed information from the Antiskid Control Box to produce a servo control valve signal to maintain the proper brake pressure to achieve the commanded level of deceleration.

By comparison, the ABS takeoff mode utilizes both the right and left brake hydraulic systems. An additional manifold (takeoff manifold) is installed in the left brake system. Upon actuation of the ABS in the event of a rejected takeoff, full system pressure from both brake systems is applied to the brakes. The full system pressure is applied through the corresponding manifold (left or right). The deceleration is regulated by the anti-skid system operation.

The flightcrew can override the ABS at any time and revert to manual brake operation by pressing the brake pedals. Manual braking is regulated via the brake pedals and anti-skid system.

Left and right hydraulic system brake pressure is recorded on the flight data recorder. The FDR values are provided by pressure transducers located within each system, upstream of the anti-skid control valves. The transducer’s output signal is only provided to the flight data acquisition unit.

1.9.2.2 On-Scene Brake System Examinations

[2] The MIN ABS selection produces a deceleration of 4 ft. /sec^2, the MED selection a deceleration of 6.5 ft. /sec^2, and MAX provides the maximum deceleration consistent with anti-skid braking limited by the tire/pavement interface.

Panels were removed to gain access to brake system components located in the left and right main landing gear shelf. In addition, the left and right main landing gear doors were lowered to gain access to components located within. Upon door lowering snow was observed piled and packed on the inside of the left main landing gear door. The right main landing gear door had no snow. The left brake pressure accumulator indicated approximately 1,700 pounds per square inch (psi). The right brake pressure accumulator indicated approximately 600 psi. The left and right ground spoiler bypass valves were found in the ON position. The Hydraulic Power Transfer Unit valve was found in the ON position.

The Left and Right hydraulic system components were inspected. No evidence of hydraulic leakage was found. All component installations appeared normal and secure. Inputs to both dual brake control valves intact and brake cables appeared to be in a serviceable condition. The forward accessory compartment was accessed and the brake input rods were inspected for condition and security. No issues were noted.

Personnel were positioned to observe all 4 brakes in preparation for brake application. Brakes were applied using brake accumulator pressure from the left brake accumulator. All brakes were observed to apply normally. Following several brake applications, the accumulator pre-charge pressure was observed at approximately 600 psi in both the left and right systems.

1.9.2.3 Examination of Brake Pressure Transducers

Per the Flight Data Recorder Group Chairman's Factual Report, the recorded right brake pressure was consistently offset to about 900 psi when the brake pressure was expected to be 0 psi. Therefore, the left and right brake pressure transducers were removed from the airplane for further examination.

The transducers were manufactured by MagneTek (Simi Valley, California) but the product manufacturing had since been acquired by Ametek Aerospace & Defense.

Delta Air Lines considered the transducers as expendable and are replaced with new units rather than repaired. Therefore, Ametek had no record of the units returning for repair or service.

The Ametek Aerospace & Defense Acceptance Test Procedure for the Model IBA103G-93 Transducer, Drawing 66-176, Revision K, Dated December 8, 1994 was accomplished on the two sensors. The group decided not to accomplish ATP Test Element 3.3.1.3, Thermal Effects at +165° Fahrenheit (F).

An initial visual examination of the unit indicated that all connector pins appeared straight.

Both transducers were placed into the Ametek High Pressure Compensation and Temperature Chamber Test Stand. The transducers were allowed to adjust to room temperature. Pneumatic pressure from 0 – 5,000 psi was applied to the transducers at 1,000 psi intervals. The resultant voltage readings from the transducers were recorded.

The temperature chamber was then reduced to -65° F and allowed to stabilize (cold soak) for 1 hour. After the cold soak, pneumatic pressure from 0 – 5,000 psi was applied to the transducers at 1,000 psi intervals; note that additional test points from the ATP procedure (Test element 3.3.1.2 only requires measurements at zero and 100% full scale) were taken at the request of the group. The resultant voltage readings from the transducers were recorded, in Table 1:

Serial Number: 90-0107	
Pressure (PSI)	**Output (VDC)**
0	(-0.054)
1,000	0 .971
2,000	1.993
3,000	3.010
4,000	4.025
5,000	5.038
Serial Number: 90-110	
Pressure (PSI)	**Output (VDC)**
1,000	1.900
2,000	2.912
3,000	3.921
4,000	4.926
5,000	5.928

Table 1 - Additional Test Points at -65 Degrees F.

The results of the ATP for each unit were provided in Attachment 5. Both transducers passed the internal resistance test.

A summary of the results of the ATP for each transducer were as follows:

<u>Left Position Transducer (Serial Number 90-0107) ATP Results</u>

The transducer passed the pressure accuracy (at room temperature) and thermal effects testing (temperature chamber at -65° F).

Since the group declined the thermal effects testing at +165° F, the unit was considered to have failed the ATP and is so noted on the ATP results sheet.

<u>Right Position Transducer (Serial Number 91-1110) ATP Results</u>

For each test point at each temperature (room or -65° F), the returned voltage from the transducer was ~.8 volts above the expected value. The unit was considered to have failed the ATP and is noted on the ATP results sheet.

In addition, since the group declined the thermal effects testing at +165° F, the unit was considered to have failed the ATP and is so noted on the ATP results sheet.

1.9.2.4 Examination of Auto Brake Control Box

Delta Air Lines maintains the Auto Brake Control Box internally. Delta indicated that the unit's last shop visit was 1/19/10. Additionally, Crane Hydro-Aire had no record of the unit returning to Crane for service. The Crane Hydro-Aire records covered the time period from 1998 until the day of the examination.

A visual inspection was performed and the unit appeared in good condition. All connector pins appeared straight.

The Auto Brake Control Unit was placed onto a Crane Hydro-Aire Auto Brake Test Stand and power applied. The Crane Aerospace Test Procedure TP-42-809-3, Revision C, dated December 12, 1989 was performed on the unit.

The unit passed all tests with no faults found.

1.9.2.5 Examination of Auto Brake Control Panel

Delta Air Lines maintains the Auto Brake Control Panel internally. Delta indicated that the unit's last shop visit was 1/30/08. Additionally, Crane Hydro-Aire had no record of the unit returning to Crane for service. The Crane Hydro-Aire records covered the time period from 1998 until the day of the examination.

The panel was visually examined as part of the initial examination procedure; aside from scratches and other minor physical defects, no damage associated with shipping was noted. All connector pins appeared straight.

The test procedure TP42-831, Revision B, released January 22, 1996 was performed on the panel using the Crane Hydro-Aire Auto Brake Control Panel Test Box 42-399. The unit passed all the test elements with no faults found.

1.9.3 Brake Anti-Skid System

1.9.3.1 Anti-Skid System Description

The anti-skid system monitors and modulates the hydraulic pressure to the main landing gear brakes to stop the main landing gear wheels from skidding. The system consists of the anti-skid control box, “dual” anti-skid hydraulic control valves (one per each wheel) and wheel speed transducers (also one per wheel, located on the axle hub of each main landing gear wheel).

The anti-skid control box receives and monitors the wheel speeds and modulates hydraulic fluid to the four brakes. The commands are provided to the four anti-skid control valves which execute the electrical commands from the anti-skid control box into hydraulic fluid to the brakes.

The anti-skid control valves are segregated, two per each main landing gear; the valves are referenced as inboard and outboard for each gear. In addition, for each landing gear, one anti-skid control valve is powered by the left hydraulic system and one by the right hydraulic system. Each anti-skid control provides dual control as it regulates hydraulic power to the brakes on both wheels on the associated landing gear.

1.9.3.2 On-Scene Anti-Skid System Examinations

Access panels were removed to gain access to examine all four Dual Antiskid Control Valves. No physical damage was noted and no evidence of hydraulic leakage was noted. The hydraulic fuses were not tripped and no hydraulic leaks were observed.

All four of the anti-skid control valves were removed. See section C. SUMMARY above for the control valve serial number and position.

1.9.3.3 Examination of Anti-Skid Control Box

Delta Air Lines maintains the Anti-Skid Control Box internally. Delta indicated that the unit’s last shop visit was 4/25/08. Additionally, Crane Hydro-Aire had no record of the unit returning to Crane for service. The Crane Hydro-Aire records covered the time period from 1998 until the day of the examination.

A visual examination of the unit indicated that all connector pins appeared straight.

The component was placed onto the Crane Hydro-Air Hytrol Mark II Control Shield Test Set and the Test Procedure TP42-807, Revision K, Dated September 25, 1997 was accomplished on the unit. When power was applied to the unit the “Wheel 1” fault lamp was lit. The fault was cleared and the test procedure accomplished.

The test was performed twice on the component. The first test the unit passed all the portions of the test related to anti-skid function but failed test items (Sections 8.1, 8.2, and 9.1) related to fault identification/clearing, and Section 9.3 elements related to meter movement on the front of the box.

For the Section 8.1, 8.2, and 9.3 elements, the box would correctly identify the simulated faults but fail to clear the faults.

After discussion of the test results, the group decided to accomplish the test a second time. When the unit was connected to the test set the second time, the fault lamp that was previously lit did not light again.

The portions of the test procedure that the unit previously failed in Sections 8.1, 8.2, and 9.1 were accomplished again. The second test verified that the unit was not properly clearing the identified faults. The differences between the results of the first and second test were noted on the test data sheets.

Despite the noted faults, the box passed the sections of the test related to anti-skid performance.

The acceptance testing of the unit did not identify any failures of the anti-skid performance of the unit.

The results of the tests were included in Attachment 6.

1.9.3.4 Examination of the Four Anti-Skid Dual Control Valves

Delta Air Lines maintains the Anti-Skid Dual Control Valves internally. Delta indicated that the units' last shop visits were as follows:

Left Outboard:	11/15/11
Left Inboard:	10/4/2006
Right Inboard:	7/12/2011
Right Outboard:	05/14/1998

Additionally, Crane Hydro-Aire had no record of the units returning to Crane for service. The Crane Hydro-Aire records covered the time period from 1998 until the day of the examinations.

Each of the four anti-skid dual control valves were visually examined as part of the initial examination procedure; aside from scratches and other minor physical defects, no damage was noted to any of the units; no damage associated with shipping was noted.

After dielectric and resistance testing, each of the units was placed into the Crane Hydro-Aire Hydraulic Test Stand 102 for hydraulic testing. The Crane Aerospace test procedure TP39-249, Revision P, released September 13, 1990 was performed on each of the units.

All four units passed all the test procedure paragraphs except for elements related to Hysteresis B1 and B2, Sections 15.3.1 and 16.1.

In addition, for Serial Number 2651ABC, the unit was measured above the maximum pressure value for test paragraph 9.1, Differential Pressure to Port B2.

Despite the noted failures, all four units passed the paragraphs of the test related to anti-skid performance.

The examination results for all four valves were included in Attachment 7.

1.9.3.5 Examination of the Main Gear Wheel Speed Transducers

Delta Air Lines maintains the Main Gear Wheel Speed Transducers internally. Delta indicated that the units' last shop visits were as follows:

Wheel #1:	7/24/2001
Wheel #2:	10/27/2003
Wheel #3:	6/30/1999
Wheel #4:	January 1996

Additionally, Crane Hydro-Aire had no record of the units returning to Crane for service. The Crane Hydro-Aire records covered the time period from 1998 until the day of the examinations.

Each of the four wheel speed transducers were visually examined as part of the initial examination procedure; aside from scratches and other minor physical defects, no damage was noted to any of the units.

Each wheel speed transducer was placed onto the Crane Hydro-Aire Wheel Speed Transducer Tester Stand and the Crane Aerospace TP40-62575 Test Procedure, Revision E, Released March 23, 1995 was performed.

Each transducer passed the TP with no faults found, with the exception of Section 4.5, Operating Characteristics, the 525 RPM (revolutions per minutes) output voltage measurement.
For each transducer the voltage was noted higher than the maximum-allowed 3.0 Volts.

In addition, for wheel speed transducer serial number 7467A, the unit's pin B to pin E resistance was measured above the maximum-allowed 22.5 Ohms, per Test section 4.1.1. The other 3 wheel speed transducers passed this test element.

Despite the noted failures, all four units passed the paragraphs of the test related to anti-skid and spoiler activation.

The test data sheets for all four valves were included in Attachment 8.

1.9.4.0 Nose Landing Gear and Nose Wheel Steering Description

The nose wheel landing gear system provides nosewheel steering and ground-shift information for the airplane. The steering system is mechanically controlled and hydraulically actuated via the steering wheel (on the left side of the flightdeck) and both flightcrew member's rudder pedals. The steering wheel controls ±82 degrees of nose wheel turning while the rudder pedals control the steering ±17 degrees.

The nose wheel steering is hydraulically controlled by the nose gear steering control valve. The valve consists of two independent control valves and cylinders that are mechanically linked to the steering control cable system. Both hydraulic systems, left and right, supply pressure to the valve, the left and right control valve, respectively. As the steering wheel or rudder pedals provide input to the steering control valve, the valve reacts and ports hydraulic fluid to the corresponding nose gear steering cylinders.

The ground sensing control mechanism positions the steering mechanism to actuate whenever the nosegear strut is compressed, and locks out the steering system when the strut is extended. The ground sensing is comprised of two oleo switches connected by cables to the nose gear strut and the landing gear handle interlock.

1.9.4.1 On-scene observations

The nose landing gear (NLG) was attached to the aircraft after the accident but was damaged and partially separated during the accident sequence. See Figure 9. Delta Air Lines indicated that the drag link brace and landing light wiring were cut to remove the NLG prior to moving the airplane into the hangar.

Figure 9 - Airplane on berm following the accident, prior to removal of the nose landing gear.

The Captain-side trunnion structure was damaged and the first officer's side trunnion pin was found sheared.

The ground shift mechanism in the NLG cheek area was fractured and was connected to the airplane only by the mechanism's control cable. No damage was noted to the ground switch oleo switches.

The nose wheel steering actuator on the captain's side was broken and the first officer's side nose wheel steering actuator was bent.

The right NLG tire assembly was still inflated but the left NLG tire assembly was deflated, with damage to the wheel and tire. See Figure 10.


Figure 10 - Nose Landing Gear in hangar, after removal from the airplane

The nose wheel steering control valve was removed from the airplane for further examination.

1.9.4.2 Examination of the Nose Wheel Steering Control Valve

Delta Air Lines had no record of the unit returning to Delta Tech Ops for service. The Delta records covered the four years prior to the day of the accident.

A visual inspection was performed and the unit appeared in good condition, with no noted damage or defects. The unit was dirty and covered with residual hydraulic fluid. The valve was visually examined as part of the initial examination procedure; aside from scratches, dirt, hydraulic fluid, and other minor physical defects, no damage was noted.

The test was conducted using the procedure described in the Boeing Overhaul Manual, Section 32-52-1, dated September 15, 2011. The valve was installed onto the Delta Test Stand #2, a Seaton Wilson Flow Bench, part number B-1104-4.

As otherwise noted, the tests described in Table 103, Valve Assembly Test Procedures, were accomplished with the following results:

Test 1: Due to a test stand limitation, Test 1 was conducted at 40 psi (instead of 25 psi). No external leakage was noted.

Test 2: No external leakage noted.

Test 3: Not accomplished, per group decision, to preserve "as is" status of the unit.

Test 3a: Pass – no external leakage noted.

Test 4a: Left direction – 240 cc
Right direction – ~103 cc

Test 4b: Valve 1, Cylinder 1 – 200 cc

Test 4c: Valve 1, Cylinder 2 – 250 cc

Test 4d: Valve 2, Cylinder 2 – 260 cc

Test 4e: Valve 2, Cylinder 1 – 300 cc

Test 5 3a: OK

Test 5 3b, 3c, 3d – The results were recorded onto a copy of the Dual Valve Pressure Gain Curve as noted in Figure 105 in the Overhaul Manual section. The plot was included in Attachment 9.

In general, the valve results did not fit within the boundaries specified in the Figure 105 graph. However, it was noted that the valve had been in service for several years and would not be expected to meet the test qualifications.

In addition, according to the note in Paragraph F(2), units removed from service may have maximum leakage rates increased 100 percent.

Test 6a: OK/passed

Test 6b: Left - ~4 lb. maximum
Right - ~3.5 lb. maximum

Test 6c: OK/passed

Test 6d: OK/passed

The examination results for the control valve, including the Dual Valve Pressure Gain Curve, were included in Attachment 9.

1.10 Lights

The two nose landing gear lights were fractured. Pieces of the light lens were found in the airplane's path between the runway and the final position.

Delta indicated that the right wing tip light was damaged during the airplane recovery. The left wing tip light was also damaged; Delta indicated that the left wing tip light was damaged prior to the airplane's recovery.

1.11 Navigation

The EGPWS computer was identified and removed from the airplane for further examination.

Delta indicated that, according to their records, the EGPWS computer had been returned to Delta Tech Ops for service once. The removal date was November 10, 2009 and the reason for removal was for the installation of a new terrain database.

For the EGPWS examination at Delta Tech Ops, a visual inspection was performed and the unit appeared in good condition. All connector pins appeared straight.

The unit was tested according to the Honeywell Component Maintenance Manual, Section 34-45-36, revised June 30, 2008. The EGPWS computer was attached to Delta's EGPWS Ground Support Equipment and the EGPWS Automated Test Procedure (ATP) program was conducted on the unit. The EGPWS unit passed all tests with no faults found. Delta provided print and electronic copies of the automated test procedure results.

Examination of the ATP results computer text file (E5_01820) provided the following information:

a. Flight Warning History: Flight Leg 139 had the most-recently recorded flight warning.
b. There were no recorded Current Faults related to the operation of the EGPWS computer.
c. The following Current Faults, External to the EGPWS Computer, were recorded:
    - IRS Bus Inactive
    - Air Data Bus Inactive
    - ILS Bus Inactive
    - Radio Altimeter Bus Inactive
d. The EGPWS (total) Operating Time was 87142:05:48 (HHHHH:MM:SS)
e. Flight Ground History: The following ground history information was recorded:
    - Terrain Not Available, recorded at 87141:59:48
    - TA&D Inoperative, IRS Bus 1 Inactive, recorded at 87141:56:56

- Windshear Inoperative, IRS Bus 1 Inactive, recorded at 87141:56:56

The non-volatile memory was queried and downloaded using the Delta software. No warnings were noted for the accident flight leg, numbered as “flight leg 1” in the EGPWS memory.

After the group departed, the NTSB asked Delta to download the landing history data from the EGPWS. The information recorded the time during each flight leg when the airplane’s radio altitude data decreased below 50 feet.

The information captured for Flight Legs 1 and 2 of the flight landing history are as follows:

FLIGHT LEG 1:	(87142: 5: 6)
Application S/W Version:	212
Configuration S/W Version:	212
Terrain Database Version:	473
Envelope Mod. Database Version:	B07
Aircraft Configuration Number:	0
Audio Menu Index:	0
Callout Menu Index:	0

FLIGHT LEG 2:	(87141:56:40)
Application S/W Version:	212
Configuration S/W Version:	212
Terrain Database Version:	473
Envelope Mod. Database Version:	B07
Aircraft Configuration Number:	118
Audio Menu Index:	4
Callout Menu Index:	91

2.0 AIRPLANE STRUCTURES

Delta Air Lines notified the NTSB that the airplane was damaged during the recovery and movement from the accident site to the maintenance hangar. Therefore, the documentation of the airplane’s structure included efforts to differentiate between accident damage and post-accident damage. Delta Air Lines provided briefings on the damage incurred during the recovery process.

In general, all damage noted forward of body station 332 was due to the accident. In general, all damage aft of body station 332 was assessed to be from the airplane recovery. Damage noted with a red paint transfer was indicative of damage during the recovery, as the airplane was impacted with a red recovery crane.

2.1 <u>Doors</u>

The forward doors, 1L and 1R were open when the group arrived at the hangar. A visual inspection noted no anomalies to either door.

2.2 Fuselage

The following damage was noted on the forward fuselage:

- The radome was shattered due to impact with the berm.
- Forward bulkhead was penetrated.
- Nose landing gear wheel well was substantially damaged.
- Severe scrapes on underside fuselage from station 160 to 332.
- All antennas forward of station 332 were gone, separated from airplane.

Damage aft of station 332, due to the recovery of the aircraft post-accident

Left hand side of fuselage, aft of station 332:

- Buckled skin from crane strap between stations 484 and 560
- The antenna under station 1155 broken off
- Scrapes on lower fuselage between stations 1271 and 1287, due to strap damage
- Skin penetration at around station 1287 due to impact with truck

Right hand side of fuselage, aft of station 332:

- Skin penetration at station 522 due to crane beam impact
- Skin penetration forward of cargo door between station 617 and 636
- Skin penetration at cargo door between stations 655 and 674

2.3 Stabilizers

No visible damage was noted on the horizontal or vertical stabilizer.

2.4 Windows

No visible damage was noted to the airplane windows.

2.5.0 Wings

2.5.1.1 Damage to Left Wing

Damage to left wing due to accident

Slats

- #3 center slat link sheared off at slat attachment point

- #3 slat sheared off at location 35 inches outboard
- #4 slat pushed aft due to impact
- #5 slat at 2nd link sheared off to aft lug
- #5 slat completely sheared off and departed aircraft
- #5 two most outboard slat links pushed aft
- Scuff marks starting about 12 inches of inboard #2 slat, continues to wing tip on all remaining slats

Wing

The scrape marks on the lower skin of left wing were documented. The angles prescribed by the scrapes were determined between forward spars to lower wing skin attachment fasteners as follows:

- From XRS 545.380 to wing tip: 26 degrees
- Between outboard Fuel tank boundary to ~XRS 509.500: 55 degrees
- Between XA 402.882 and outboard Fuel tank boundary: 23 degrees
- Between XRS 267.000 and XA 402.882: 46 degrees

Light scuff marks on upper skin between inboard fuel tank boundary and XRS 111.500 on the leading edge, as well as between XRS 137.760 and XRS 160.000. The source of this damage could not be attributed definitely to the accident.

- Small dent around XRS 353.000
- Leading edge damage with parts sheared off, damage down to forward spar. Damage in area between XRS 353.000 to about half way between XRS 414.000 and XRS 383.000
- Cracked and depressed leading edge and lower skin damage down to forward spar in area around the outboard fuel tank boundary
- Sheared off section between XRS 509.500 and XRS 531.700
- Damage around landing light, with the landing light missing
- Strobe light cover missing
- Buckled Skin & Tear to aft spar in area between XW 486.250 and XW 536.861

Flaps, Spoilers and Ailerons

- Complete inboard flap twisted
- Outboard flight Spoiler split around flap due to leading edge impact with flap
- Aileron leading edge pushed aft about 8 inches from normal position at 40 degrees
- Aileron control tab leading edge scrapes
- Aileron control tab shows two impact points due to contact with berm
- Cable pulleys severed in section between XRS 509.500 and XRS 531.700
- Aileron cables severed at section between XRS 509.500 and XRS 531.700
- Trailing edge of aileron control tab is delaminated due to scraping

- Trailing edge of outboard flap bent upwards about 10 inches from neutral position
- Tear on upper and lower skin in most outboard section of outboard flap
- 2-foot section of trailing edge of outboard flap between XTE 251.996 and XFS 304.031 damaged with upper and lower skin sheared off and several ripped out rivets

Damage to Left Wing Due to Recovery of Aircraft

Slats

- #1 and #2 slats disconnected, alignment pins disconnected, air duct disconnected
- #2 slat bent upwards, left side sheared off, right side of #2 sheared off rivet heads XIS 136.421 and XIS 186.374
- #2 center link lower cap sheared off due to crane impact XIS 136.421 and XIS 186.374
- Red scuff marks on #2 slat between XIS 136.421 and XIS 186.374
- #2 outboard link cutout damaged by link pushed aft
- Edge plate between #2 and #3 is bent underneath #3 slat.

Wing

- Red scrapes in area around aft spar XRS 214.000
- Small section of buckled skin around XRS 137.750
- Damage to leading edge in area around XFS 129.718

Flaps

- #2 inboard flap actuator fairing damaged, red paint marks on fiberglass visible
- Scrapes half way between XFLS 187.406 and XW 154.583

Finally, the measured height from the left wing tip, next to landing light, to the hangar floor was measured as 105 inches. The measured height at XRS 383.000 aft spar to the hangar floor was measure as 92 inches.

2.5.1.2 Damage to Right Wing

The damage to the right wing occurred during recovery of the airplane. The following damage was noted:

- Damage to slats between XIS 83.534 and XIS 136.421. Slat leading edge crushed and sections sheared off, lower skin cracked to trailing edge
- Edge plate between #2 and #3 slats bent
- Dent half way between XOS 332.588 and XOS 380.229
- Scrape marks on wing tip

- Dents and scrapes to the leading edge of the wingtip, from XW 560.861 and further “outboard” on the wing tip
- RH landing light was noted as missing

Finally, the measured height from the right wing tip, next to landing light, to the hangar floor was measured as 103 inches. The measured height at XRS 383.000 aft spar to the hangar floor was measure as 92 inches.

3.0 AIRPLANE POWERPLANTS

The airplane was fitted with 2 powerplants, mounted on each side of the aft fuselage. Engine number 1 is the engine on the left side of the airplane (looking forward) and engine number 2 is the unit on the right side.

3.1 Powerplants

The powerplants were identified as follows:

Engine Number 1	
Engine Manufacturer:	Pratt & Whitney
Model:	JT8D-219
Serial Number:	725552

Engine Number 2	
Engine Manufacturer:	Pratt & Whitney
Model:	JT8D-219
Serial Number:	726934

Each engine was visually examined with no visible accident damage noted. The left engine cowl was noted with some damage which was considered to be from the airplane recovery.

No damage was noted to the engine inlets and exhaust areas of the powerplants.

3.2 Engine Controls

The engines and engine thrust are controlled by use of the throttles on the center pedestal in the flight deck. The throttles are connected the engines on the aft fuselage by a series of cables, levers, cams, and pulley quadrants. The throttles connect to the engine fuel control assembly on each respective engine.

The throttles are moved forward for increasing thrust and pulled back for decreased thrust.

While on-scene the group confirmed the continuity of the engine controls for both powerplants, with no faults and visible damage noted.

3.3 Engine Indicating System

The engine indicating instruments are located on engine display panel, located on the center instrument panel. The instruments on the engine display panel, from top to bottom of the panel, are the engine pressure ratio (EPR) instruments, N1 (in % of revolutions per minute (RPM)), exhaust gas temperature (EGT) in degrees Celsius, N2 in % RPM, and FUEL FLOW/USED. Each engine has an instrument for each indication. See Figure 11.

Figure 11 - Engine Display Panel (Sister Ship)

The EPR gauges provide a digital readout of the value in the center of the gauge. Around the circumference of the gauge is a dial gauge with incremental marking for EPR values between 1.0 and 2.2. A yellow “needle” moves around the outside of the circumferential to provide a visual indication. See Figure 12.

Figure 12 - Close Up Photo of EPR Gauges (Sister Ship)

The EPR gauges are used for both forward and reverse thrust indications.

3.4.0 Engine Thrust Reversers

The airplane's thrust reversers (one per engine) provides the engines with the ability to alter the direction of fan air and exhaust gas flow. The reversers are connected to the aft section of each engine, and are mechanically initiated and hydraulically operated. The reversers are controlled by lever movement at the throttle quadrant, manually activated by the flightcrew.

Each thrust reverser has two doors, one attached to the upper engine fairing and one attached to the lower engine fairing. The doors are hydraulically actuated by two thrust reverser door actuators.

A visual examination of both thrust reversers was conducted, with no anomalies noted.

See Section 4.0 - <u>FLIGHT CONTROL, MANUAL BRAKE, AND THRUST REVERSER ACTUATION EXERCISE</u> regarding the actuation of both thrust reversers during the on-scene hydraulic exercise.

3.4.1 Engine Thrust Reverse Control

Thrust reverser control system levers are mounted on the engine throttles on the center pedestal in the flight deck. The levers, one for each engine, are hinged to each throttle lever. The levers are connected to the same cable systems used by the throttles, and each lever has a total travel of 120 degrees. The thrust reverser levers can only be operated when the throttles are in the idle position, either on ground or in flight.

The thrust reverser control system consists of thrust reverser levers, throttle cable system, control valves, hydraulic accumulator, interlock mechanism and push-pull cable. See Figure 13.

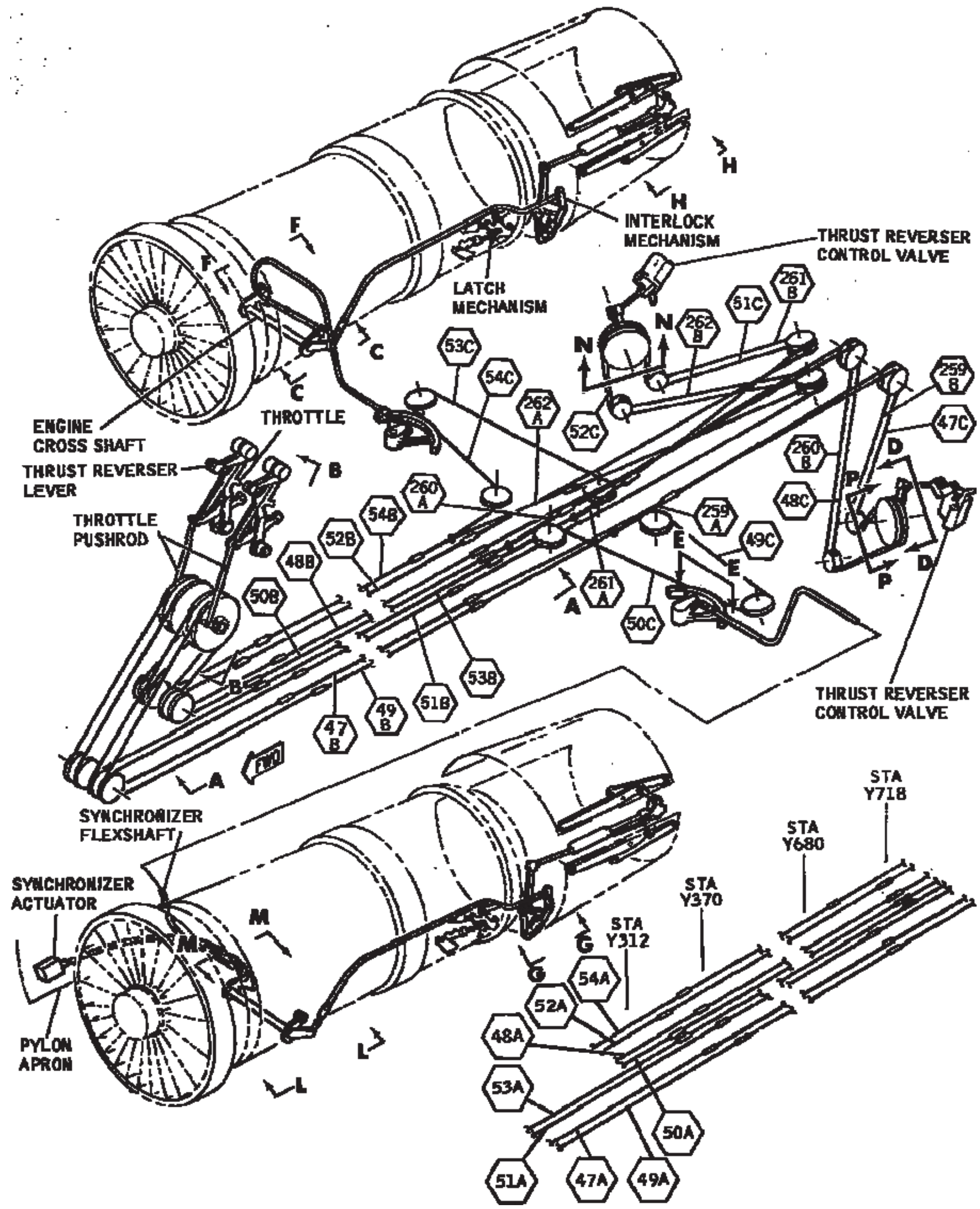

Figure 13 - MD-80 Thrust Reverser Control (Reprinted Courtesy of Boeing)

3.4.2 Boeing Service Bulletin to Provide 1.3 EPR Detent on Thrust Reverser Levers

On June 14, 1996 Boeing Published Service Bulletin MD80-78-068R01, which provided a method for operators to install an intermediate detent on the Thrust Reverser Control Levers

Cam that would correspond to approximately 1.3 EPR reverse thrust value. A copy of the Service Bulletin is included in Attachment 10.

After receiving negative feedback from operators regarding the effectiveness of the 1.3 EPR detent, On April 7, 1997, Boeing issued MD80-COM-0006, Subject: DC-9 Series 90 Thrust Reverser, Replacement of Thrust Reverser Intermediate Detent Cam Support Assembly. The message recommended that operators removed the cam with the intermediate detent and replace with an original cam. Boeing indicated that two operators had difficulty obtaining the 1.3 EPR value because of the slope of the cam, the inherent hysteresis within the system, and was difficult to rig. A copy of the Boeing communication is included in Attachment 11.

4.0 FLIGHT CONTROL, MANUAL BRAKE, AND THRUST REVERSER ACTUATION EXCERCISE

After a review of information collected on-scene, the group identified systems to be considered for hydraulic testing. On March 10, 2015, the Airworthiness Group conducted an exercise in which hydraulic power and limited electrical power were applied to the accident airplane's hydro-mechanical systems. The purpose of the exercise was to actuate the airplane's flight controls, brakes, and thrust reversers to determine range of motion and operation.

The following tests were conducted, given the limitations of the damaged airplane (specifically, the inability to apply electrical power to the airplane):

1. Manual Brake application
2. Actuation of rudder control surface
3. Actuation of the elevator control surface
4. Actuation of Thrust Reversers
5. Actuation of Spoilers using manual spoiler deployment

For the exercise, alternate hydraulic power was applied to airplane via a hydraulic ground cart (mule). It was noted that the pressure indication on the hydraulic mule fluctuated from 2,900 pounds per square inch (psi) at no hydraulic system load to varying pressures as low as 1,000 psi, depending on hydraulic load applied. The rapidly fluctuating mule hydraulic supply pressures did not accurately represent the airplane system pressures and flow rates that would be expected in normal airplane hydraulic system operation.

For the test, the right hydraulic system was first pressurized. Then the left was powered via the right hand system through the Power Transfer Unit (PTU). Finally, the hydraulic cart was moved and connected to the left hydraulic system. The same actuations were repeated using the left hydraulic system.

Manual brake application was accomplished with no anomalies noted.

First the rudder was actuated rudder using flight deck controls. The rudder was put into manual mode and the control tab was moved. The rudder was actuated full throws left and right. The surface moved as commanded from the rudder pedal movement during the test. The control tab moved as commanded from the rudder pedal movement during the test. The rudder trim tab was actuated and the trim tab surface moved with the corresponding rudder trim knob input. In all cases using the rudder, no issues were noted.

The elevator was observed to move full travel to the down position with full forward column movement. The elevator boost cylinders were actuated and the surface moved trailing edge down, as commanded, from the control column forward movement during the test. No issues were noted with the elevator actuation portion of the exercise.

Also deployed both thrust reversers, in tandem and separately. Both thrust reversers deployed and retracted normally when actuated with the thrust reverse levers on the left and right throttles. No issues were noted with the thrust reverser actuations.

To actuate the inboard ground spoilers, an alternate electrical power source (30 volts) was connected to S3-13, stud #5 and S3-12, stud #5 for left and right inboard spoiler deploy valve assemblies.

The inboard and outboard flight spoilers were observed to deploy normally in response to spoiler handle movement. When an attempt to deploy ground spoilers was made using a single hydraulic system, ground spoilers did not deploy. It was found that ground spoilers would deploy when the hydraulic Power Transfer Unit (PTU) valve was moved to the on position. Selecting the PTU valve to ON enabled pressurization of the left and right hydraulic systems. In this condition Ground Spoilers were observed to travel to the full up position. Further attempts to deploy ground spoilers using a single hydraulic system noted slight Ground Spoiler panel movement could be observed, but full deployment could not be achieved. The Ground Spoilers are normally actuated by both hydraulic systems through a dual piston actuator. With both systems pressurized simultaneously, actuator force exerted to deploy the ground spoiler panel is higher than is available with a single hydraulic system. In this test using a hydraulic mule with limited flow rates resulting in fluctuating pressures due in part to hydraulic leak at the connection to the aircraft, the ground spoiler panels required both hydraulic systems to be pressurized simultaneously to achieve spoiler deployment.

For the test, an MD-88 rated pilot was used to manipulate the flight deck controls. The pilot's observations regarding the exercise were as follows:

Flight Control Hydraulic Actuation, Cockpit actuations and actions:

"As part of the hydraulic activation of controls and surfaces, actuation and movement of various controls in the cockpit was necessary. Electrical power was not supplied to the aircraft busses due to concerns over damage to the E &

E compartment. A pilot member of the investigative group was asked to manipulate certain flight controls and brake controls while hydraulic pressure was provided to parts of the hydraulic system utilizing a “mule”. The sequence of actuation followed a schedule agreed upon by members of the Systems group.

The Spoiler Handle was found to be in a retracted, forward position. However it displayed some red paint range marking that indicated that it may not have been fully retracted. Panels on the CA side of the center control panel were removed to allow access for visual inspection of the actuation mechanism of the spoiler control. The electrically driven ground spoiler actuator was found to be in the extended or deployed position. The actuator motor was manually driven by rotation with a wrench to the fully retracted position. The spoiler handle was then observed to be in the normal retracted position.

During the first observation, hydraulic power was applied to the right hydraulic system using attachment fittings on the airplane. Due to lack of electric power, no actuator lights or gauges inside the cockpit were functioning. The ground spoiler actuators received power from attached batteries to open the shutoff valves.

The right brake pedal, left brake pedal and both brake pedals were depressed three times each in sequence. The pedals were depressed fully and held for about one second. Observers at the main wheels observed movement of the brake calipers was consistent with the operation of the pedals. The pedals moved easily to full actuation. The resistance felt less than normal.”

Tom Jacky
Aerospace Engineer

Attachment 1 - Spoiler Control Box Test Results Sheet
Attachment 2 - Auto Spoiler Actuator Test Results
Attachment 3 - Flight Deck Documentation
Attachment 4 - PSEU ATP Test Sheets
Attachment 5 - Brake Pressure Transducer Examination Results
Attachment 6 - Anti-Skid Control Box Examination Results
Attachment 7 - Anti-Skid Dual Control Valve Examination Results
Attachment 8 - Main Gear Wheel Speed Transducer Examination Results
Attachment 9 - Nose Wheel Steering Control Valve Examination Results
Attachment 10 - Boeing Service Bulletin – 1.3 Thrust Reverser Detent
Attachment 11 - Boeing Communication to Rescind 1.3 Reverser Detent