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

August 6, 2020

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT

DCA19MA143

A. ACCIDENT

Location:	Jacksonville, FL
Date:	May 3, 2019
Time:	2142 EDT
Aircraft:	Boeing 737-81Q (S/N 30618)
Registration:	N732MA

B. SYSTEMS GROUP

Chairman:	Adam Huray National Transportation Safety Board Washington, DC
Member:	Scott Wolf Boeing Seattle, WA

C. ACCIDENT SUMMARY

On May 3, 2019, at 2142 eastern daylight time, Miami Air International flight 293, a Boeing 737-81Q, registration N732MA, was landing on runway 10 at Jacksonville Naval Air Station, Jacksonville, Florida, when it departed the end of the runway, contacted a stone embankment, and came to rest in shallow water in the St. Johns River. The 2 pilots, 4 flight attendants, 1 mechanic, and 136 passengers were not seriously injured. The airplane was substantially damaged. Flight 293 was a non-scheduled passenger flight from Leeward Point Field, Naval Station Guantanamo Bay, Cuba, operating under the provisions of 14 Code of Federal Regulations Part 121 Supplemental. Instrument meteorological conditions prevailed at the time of the accident, and rain was occurring during the landing.

D. DETAILS OF THE ON-SCENE INVESTIGATION (ACCIDENT LOCATION)

D.1 Flight Deck:

The flight deck was accessed by the Systems and Operations Group Chairmen while the aircraft was still in the St. Johns River. The entire flight deck remained above water and was in good condition. The control wheels were in a right turn position. The control wheels could not turn easily to the left of the as found position and force was not used to move past this position. The control wheels could be moved to the right and back to the as found position. The wheels moved together when turned. The columns both appeared to be in a slightly forward position and moved together when pushed forward and aft. Both rudder pedal assemblies were in a position with the left pedal forward of the right pedal. The pedals moved together when the left and right pedals were pressed. The speed brake lever was found in the forward "DOWN" position and was partially in the detent. It could be moved to about the "FLIGHT DETENT" position but it was difficult to move. The thrust reverser handles could be moved to approximately the interlock position but would not move further. The speed brake lever popped out of its detent when the thrust reverser handle was in the interlock position.

D.1.1 Settings and Switch Positions:

The following settings and switch positions were observed:

Emergency Exit Lights guarded and set to "ARMED" position
#1 Engine Start set to "CONT" position
#2 Engine Start set to "CONT" position
L Wiper set to "HIGH"
R Wiper set to "PARK"
Auto Throttles (A/T Arm) set to "OFF"
L Flt Dir set to "ON"
R Flight Director set to "ON"
L Main Panel DUs set to "NORM"
L Lower DU set to "NORM"
R Main Panel DUs set to "NORM"

R Lower DU set to "NORM"
Autobrake set to "2"
Standby altimeter set to "29.97" inHg
Landing gear lever set to "DN"
Both throttle levers were full aft
Both thrust reverser levers were stowed
Speed Brake lever was full forward and partially in "DOWN" detent
Fuel Cutoff levers were set to "IDLE"
All three fire handles pulled and turned
Cargo Fire FWD Arm pushbutton was not pressed
Cargo Fire AFT Arm pushbutton was not pressed
Parking Brake lever was stowed
Flap lever was in "30" detent
Flaps indicator displayed "30"
Weather Radar set to "WX/TURB MAP"
Weather Radar gain set to "AUTO"
Weather Radar Tilt set to approximately 1 unit up
Nose Wheel Steering switch was guarded and set to "NORM"
Stab Trim switch was guarded and set to "NORM" with trash bag drawstring hooked on it under guard
Stab Trim indicators on both sides read approximately 7.5 units nose up
Rudder Trim indicator displayed "OFF"
Aileron Trim indicator displayed approximately 3 units right trim on both columns
Battery "BAT" switch guard open and switch set to "OFF"
Flight Spoiler Shutoff Valve A switch was guarded and set to "ON" (safety wire was broken)
Flight Spoiler Shutoff Valve B switch was guarded and set to "ON" (safety wire was broken)

D.1.2 Flight Deck Circuit Breakers:

The following circuit breakers on the pilot-side circuit panel (P18-2) were pulled by NTSB personnel prior to flight deck examination:

C7 – Voice Recorder Relay
C8 – Flight Recorder Position Sensor
C9 – Flight Recorder AC
C10 – Flight Recorder DC
D7 – Voice Recorder

The following circuit breakers were found open (there were no circuit breakers found to be collared):

<u>Pilot-Side Circuit Breaker Panel (P18-1)</u>

D1 - AFCS Sys A – "Warn Light (Bat)"

<u>First Officer Side Circuit Breaker Panels</u>

<u>P6-1 Panel</u>

E11 –Display – "F/O INBD"
E12 – Display – "CTR LWR"

<u>P6-2 Panel</u>

A21 – Fire Protection – "MA WRN and CONT"
B1 – AFCS Sys B – "Warn Light (BAT)"

<u>P6-3 Panel</u>

E16 – Landing Gear – ANTISKID "INBD"

<u>P6-4 Panel</u>

A14 – Aux Power Unit – "CONT"
E15 – DC Bus Indication – "HOT BAT"
F11 – Generator – "CONT UNIT 2"
F13 – Generator – "BUS PWR CONT UNIT"

D.1.3 Deferred Maintenance Items in Flight Deck:

The left thrust reverser lever was safety wired in the stowed position. There was not a "REVERSER INOP" tag present on the left thrust reverser lever. Sticker #11700, titled "#1 Eng Rev", was located next to the number 1 engine reverser light. Sticker # 11701, titled "Rt Pack", was located next to the right pack switch. The right pack switch was in the "OFF" position. Sticker #11702, titled "ETOPS", was located over the captain's side "10 Min Engine-Out Takeoff Thrust Limit" placard. There was no sticker over the first officer's placard. Sticker #11703, labeled "Duct Mess", was located next to the number 1 bleed switch. The switch was in the "OFF" position. Sticker #11698, labeled "Sat Comm", was located on the aft panel of the center console.

E. DETAILS OF THE INVESTIGATION (AFTER AIRCRAFT RELOCATION TO REYNOLDS PARK MARINA)

The airplane, in its entirety, was transported by barge to Green Cove Springs, Florida, and placed in an open field at Reynolds Park Marina. The aircraft was examined at this location by the Systems group from May 9 – May 12, 2019. The following observations were taken at this location.

E.1 Landing Gear and Brakes:

Brake cable continuity was verified by pressing the brake pedals and observing the movement of the input levers to both metering valves. The hydraulic connections downstream of the metering valves were either inspected during component removal or by visual inspection. The hydraulic lines were breached on both sides where the gear was disconnected from the airframe. Miami Air stated that the #1, #2, and #4 brakes were all replaced on 2/4/2019.

E.1.1 Left Main Gear:

The left main gear was separated from the aircraft (see Structures Group Chairman's Factual Report for details). The strut pressure measured 500 psi. The gauge filled with fluid due to the angle of the strut. The chrome extension of the strut measured 15-3/4 inch. The wheel speed sensor wiring was pinched by structure between the torsion link and axle. The wiring conduit was pulled away from the wires where it passed through structure near the torsion link. At the terminal end of the wires (aircraft connection), the yellow/blue wires still had connector pins, while the red and red/blue wires had no pins.

The #1 tire was a Bridgestone H44.5 x 16.5 – 21, P/N APS06017. There was no retread indicated. The tire surface appeared intact with no obvious signs of flat spotting or reverted rubber. Some surface roughness and areas of chevron cuts were noted on the tread. A few areas of missing rubber chunks were also identified. The tire pressure measured 220 psi approximately 4 hours after recovery from the water and while still on the barge. Per tire inflation task 12-15-51-780-801, the nominal tire inflation pressure should be set to 205 +/- 5 psig. From left to right, the tread grove depths measured 7/32 inch, 8/32 inch, 7/32 inch, 7/32 inch. The brake was P/N 2-1587-1, S/N 0111, DOM 06/98. The brake wear pin extension beyond the guide measured 1-17/32 inch (brakes not compressed). The brake pistons were flush against the pressure plate.

The #2 tire was a Bridgestone H44.5 x 16.5 – 21, P/N APS06017, retread 2 (date 01-19). The tire surface appeared intact with no obvious signs of flat spotting or rubber reversion. Some surface roughness and areas of chevron cuts were noted on the tread. A few areas of missing rubber chunks were also identified. There was a cut on the inboard sidewall (sidewall opposite of brake pistons) in the area of the speed rating/skid depth label. A chunk of rubber was missing on the outboard sidewall (sidewall on same side as brake pistons) in the area of the "Bridgestone" label. An area of discoloration was noted on the tread surface adjacent to the speed/load rating label. The discolored area was the width of the tread and approximately 20 inches in length. The outboard sidewall had signatures of water runoff (rust and dirt trails) from the rim to the center of this area, and the discolored tread area had adhered rust residue. There was no difference observed in the surface texture in this area compared to the rest of the tire. The manufacturing mold bead located on the center tread was still protruding slightly from the tire surface in this area similar to the rest of the tire. The tire pressure measured 215 psi approximately 4 hours after recovery from the water. From left to right, the tread grove depths measured 9/32 inch, 12/32 inch, 11/32 inch, 9/32 inch. The brake was P/N 2-1587-1, S/N 2427, DOM 11/03. The brake wear pin extension beyond the guide measured 1-17/32 inch (brakes not compressed). The brake pistons were flush against the pressure plate. The hub cap was missing from the #2 wheel.

E.1.2 Right Main Gear:

The right main gear was separated from the aircraft (see Structures Group Chairman's Factual Report for details). The strut pressure measured 500 psi. The gauge filled with fluid due to the angle of the strut. The chrome extension of the strut measured 15-3/4 inch. The wheel speed sensor wiring was damaged where it passed through structure near the torsion link. At the terminal end of the wires (aircraft connection), the yellow wire still had a connector pins, while the red and blue wires had no pins. The red/blue wires had no pins, and the blue wire was severed about 12 inches from this location.

The #3 tire was a Bridgestone H44.5 x 16.5 – 21, P/N APS06017, retread 3 (date 07-18). The tire surface appeared intact with no obvious signs of flat spotting or rubber reversion. Some surface roughness and areas of chevron cuts were noted on the tread. A few areas of missing rubber chunks were also identified. A cut with chunking was noted on the tread surface in line with the speed/load rating. There appeared to be an area of faint discoloration on the tread surface in line with the Boeing P/N label. The area was the width of the tread and at least 12 inches in length (continued under tire). Another area of slight discoloration was noted on the tread surface in line with the retread date label. The area was the width of the tread and approximately 11 inches in length. The outboard sidewall (sidewall on same side as brake pistons) had signatures of water runoff (rust and grime trails) from the rim to one edge of this area, and the edge of the discolored tread area had adhered rust residue. There was no difference observed in the surface texture in these areas compared to the rest of the tire. The tire pressure measured 230 psi approximately 4 hours after recovery from the water. From left to right, the tread grove depths measured 7/32 inch, 8/32 inch, 7/32 inch, 6/32 inch. The brake was P/N 2-1587-1, S/N 1642, DOM 09/01. The brake wear pin extension beyond the guide measured 19/32 inch (brakes not compressed). The brake pistons were flush against the pressure plate.

The #4 tire was a Bridgestone H44.5 x 16.5 – 21, P/N APS06017. There was no retread indicated but there was a date of 09-18. The tire surface appeared intact with no obvious signs of flat spotting or rubber reversion. Some surface roughness and areas of chevron cuts were noted around the tread. A missing chunk of tire and cut was noted on the tread surface in line with the retread label. A large chunk of rubber was blown outward on the inboard sidewall (sidewall on same side as brake pistons) adjacent to the rim in the area of the "Bridgestone" label. A bubbly texture was noted on the outboard sidewall (sidewall opposite of brake pistons) along and next to the tire bead (about ¼ the circumference of the tire) and on the inboard sidewall along and next to the tire bead (about 6 inches in length). The inboard and the outboard bubbly texture areas were on opposite sides of the rim circumference. There appeared to be an area of faint discoloration on the tread surface in line with the retread label. The area was the width of the tread and at least 14 inches in length (continued under tire). There was no difference observed in the surface texture in this area compared to the rest of the tire. The tire pressure measured 0 psi. From left to right, the tread grove depths measured 5/32 inch, 2/32 inch, 4/32 inch, 5/32 inch. The brake was P/N 2-1587-1, S/N 0264, DOM 11/98. The brake wear pin extension beyond the guide measured 1-21/32 inch (brakes not compressed). The brake pistons were flush against the pressure plate.

E.1.3 Nose Gear:

The nose gear was folded aft and lodged in the forward electronics bay. The strut pressure measured 400 psi. The gauge filled with fluid because of the angle of the gear. The gear pressure was released and the gear was dropped by the recovery team. Once down, the steering angle on the guide measured 7 degrees left turn; however, the entire steering collar could move by hand. The strut chrome extension measured approximately 8 inches after the gear was moved to the dropped position. The ID label was destroyed and could not be read. The nose wheel steering cable "NWSA" was severed at the steering collar and "NWSB" was severed at the trunnion. The left nose wheel steering actuator remained connected at the rod end but the rod end was bent. The right nose wheel steering actuator remained connected at the rod end but the unit was separated from the steering collar. Severed wires were found at the junction box.

The left tire was made by Bridgestone. It had multiple damaged areas and had no pressure. There were no indications of abnormal wear identified on the tread. From left to right, the tread grove depths measured 5/32 inch, 5/32 inch, 5/32 inch, 5/32 inch. It was labeled as retread 9 dated 04-18. The wheel rim was broken around the bead area.

The right tire was made by Bridgestone and labeled as 27 x 7.75 - 15. About 1/3 of the tire was damaged and the tire had no pressure. There were no indications of abnormal wear identified on the tread. From left to right, the tread grove depths measured 5/32 inch, 4/32 inch, 5/32 inch, 5/32 inch. There was no retread identified on the label. The tire was dated 04-18.

E.2 Flight Controls:

E.2.1 Left Wing:

The inboard and outboard trailing edge flaps appeared to be aligned. The inboard flap was damaged and the aft segment was broken. The inboard flap, outboard jackscrew measured 10-5/8 inches from the forward stop to the base of the ballnut. Per Boeing, this correlates to an inboard flap position of approximately 33° and is consistent with a flaps lever setting of 30. The outboard flap, inboard jackscrew measured 9-5/8 inches from the forward stop to the base of the ballnut. Per Boeing, this correlates to an outboard flap position of approximately 35° and is consistent with a flaps lever setting of 30. The aileron in the as found position was deflected 1.5 inches trailing edge down from the trailing edge of the wing. The aileron trim tab was deflected 1/2 inch trailing edge up from the aileron surface. The aileron and trim tab moved correctly when the control wheels were turned. The leading edge slats were in an extended position and were in line with each other. The outboard slat (#1 slat) had a large puncture on the inboard half of the slat. The inboard slat (just outboard of the engine pylon) actuator chrome extension measured 14-3/8 inches from the base of the actuator housing to the forward side of the jam nut at the eye end. Per Boeing, this correlates to a slat position of fully deployed. The leading edge flaps were removed prior to examination. The inboard leading edge flap actuator chrome extension measured 10-11/16 inches from the base of the actuator housing to the forward side of the jam nut at the eye end. Per Boeing, this correlates to an inboard leading edge flap position of fully extended and is consistent with a flaps lever setting of 30. The outboard leading edge flap actuator chrome extension measured 11.5 inches from the base of the actuator housing to the forward side of the jam nut at the eye end. Per Boeing, this correlates to an outboard leading edge flap position of fully extended and is consistent with a flaps lever setting of 30.

The #3 spoiler panel was broken and the #4 spoiler panel was missing. The actuator for the #4 panel was severed (rotated up and missing the piston) and the actuator cable quadrant was out of position. The remaining flight spoilers and both ground spoilers were flush with the wing surface. The #1 and #6 panels are ground spoilers. These actuators remained connected to the related panel. The hydraulic connections to these actuators appeared to be intact. The #2 panel is a flight spoiler. The actuator was connected to the panel and all cables were connected to the cable quadrant. The cable (WSA1) running from the #2 quadrant to the #4 quadrant was severed in the area of the broken # 4 actuator. The #5 panel is a flight spoiler. The actuator remained connected to the #5 panel and all cables were connected to the quadrant. Cable continuity from the speedbrake lever to the mixer was verified by moving the speedbrake handle and observing movement at the mixer. Cable continuity from the mixer to the flight spoiler actuators was verified by visual inspection. With the exception of the broken cable between the #2 and #4 spoilers, all cables had continuity. A cable (WSB1) running from the mixer to the #2 flight spoiler quadrant had slack but was not disconnected. This cable ran through the damaged area of the #4 spoiler. In addition, cable (WSA1) that connects the mixer to the out of position #4 spoiler quadrant was also loose.

E.2.2 Right Wing:

The right wing flight controls remained mostly intact with the exception of the inboard trailing edge flap and inboard ground spoiler which were damaged. The inboard and outboard trailing edge flaps appeared to be aligned, but the inboard flap was bent upwards in the middle. The inboard flap, outboard jackscrew measured 10.5 inches from the forward stop to the base of the ballnut. Per Boeing, this correlates to an inboard flap position of approximately 33° and is consistent with a flaps lever setting of 30. The outboard flap, inboard jackscrew measure 9-5/8 inches from the forward stop to the base of the ballnut. Per Boeing, this correlates to an outboard flap position of approximately 35° and is consistent with a flaps lever setting of 30. The aileron in the as found position was deflected 1-1/8 inches trailing edge up from the trailing edge of the wing. The aileron trim tab was deflected ¼ inch trailing edge down from the aileron surface. The aileron and trim tab moved correctly when the control wheels were turned. The leading edge slats were in an extended position and were in line with each other. The inboard slat (just outboard of the engine pylon) actuator chrome extension measured 14.5 inches from the base of the actuator housing to the forward side of the jam nut at the eye end. Per Boeing, this correlates to an inboard slat position of fully deployed. The leading edge flaps appeared to be in line with each other. The inboard leading edge flap actuator chrome extension measured 10-5/8 inches from the base of the actuator housing to the forward side of the jam nut at the eye end. Per Boeing, this correlates to an inboard leading edge flap position of fully extended and is consistent with a flaps lever setting of 30. The outboard leading edge flap actuator chrome extension measured 11.5 inches from the base of the actuator housing to the forward side of the jam nut at the eye end. Per Boeing, this correlates to an outboard leading edge flap position of fully extended and is consistent with a flaps lever setting of 30. The bullnose on the forward side of the outboard leading edge flap was broken.

The inboard ground spoiler was bent and in a slight up position. The inboard ground spoiler actuator chrome extension measured 1.5 inches from the base of the actuator housing to the jam nut. All four flight spoilers and the outboard ground spoiler were flush with the wing surface. Cable continuity from the speedbrake lever to the mixer was verified by moving the speedbrake handle and observing movement at the mixer. Cable continuity from the mixer to the flight spoiler actuators was verified by visual inspection. The #7 and #12 spoilers are ground spoilers. These actuators remained connected to the related panel. The hydraulic connections to these actuators appeared to be intact.

E.2.3 Empennage:

The empennage remained intact and was in good condition. The rudder surface trailing edge was positioned slightly to the right of center. The rudder surface moved when the rudder pedals were pressed. The stabilizer jackscrew measured approximately 18.5 inches from the ballnut gimble to the lower stop, and about 34-3/4 inches from the ballnut gimble to the lower grease fitting. Per Boeing, these measurements correlate to a horizontal stabilizer position of approximately 3.5 degrees aircraft nose up and are consistent with a stabilizer trim setting of about 7.5 units. The horizontal stabilizer rig line was roughly 5 inches below the neutral rig line on the fuselage, which was also consistent with a stabilizer trim setting of approximately 7.5 units. The horizontal stabilizer moved correctly when the trim wheels were moved. The trailing edge of both elevators were slightly below the stabilizer surface and the trailing edges of both elevator trim tabs were slightly above the elevator surface. The elevators and trim tabs moved correctly when the control columns were moved.

E.3 Forward Electronics Bay Circuit Breakers:

The P91 and P92 circuit panels in the forward electronics bay were inspected for open circuit breakers. No open or collared breakers were observed.

F. BRAKE SYSTEM AND BRAKE COMPONENT TESTING

F.1 Brake System Description:

Flight Data Recorder (FDR) data revealed that the active brake system during the accident was the normal braking system. In addition, the FDR data revealed that the autobrake engaged approximately three seconds after touchdown for approximately two seconds before disengaging. At that time, the pressure from the left normal (manual) brake metering valve increased to more than 750psi, and by design the autobrakes disengaged. The brakes were manually controlled by the crew for the remainder of the landing. Therefore, the following systems descriptions will focus on manual braking with the normal braking system. The brakes and wheel assemblies are identified from left to right as 1, 2, 3, and 4, with 1 referring to the left outboard and 4 referring to the right outboard.

F.1.1 Normal Braking System:

The normal braking system uses hydraulic system B as its hydraulic source. The brakes are controlled by the flight crew using the brake pedals in the flight deck. Brake pedal movement is transmitted by cables to the left and right brake system metering valves located in the main landing gear wheel well. The left brake metering valve supplies metered hydraulic pressure to the left main gear wheel brake assemblies in response to the input from the control cables. The right brake metering valve supplies metered hydraulic pressure to the right main gear wheel brake assemblies in response to the input from the control cables. The metered hydraulic pressure passes through a shuttle valve and then to the respective inboard and outboard antiskid valves. Between the shuttle valve and the antiskid valves is a brake pressure transducer (one for the left brake system and one for the right brake system). This is the location where the brake pressures recorded on the FDR originate. Between each antiskid valve and brake assembly there is a hydraulic fuse to prevent hydraulic fluid loss if there is an external leak downstream of the fuse, and there is an alternate brake shuttle valve to allow brake pressure to come from the alternate brake system if required. Each wheel has one brake assembly. The brake assemblies are rotor-stator units that use hydraulic pressure to push the rotors and stators together, causing the wheel to slow. See Figure 1.

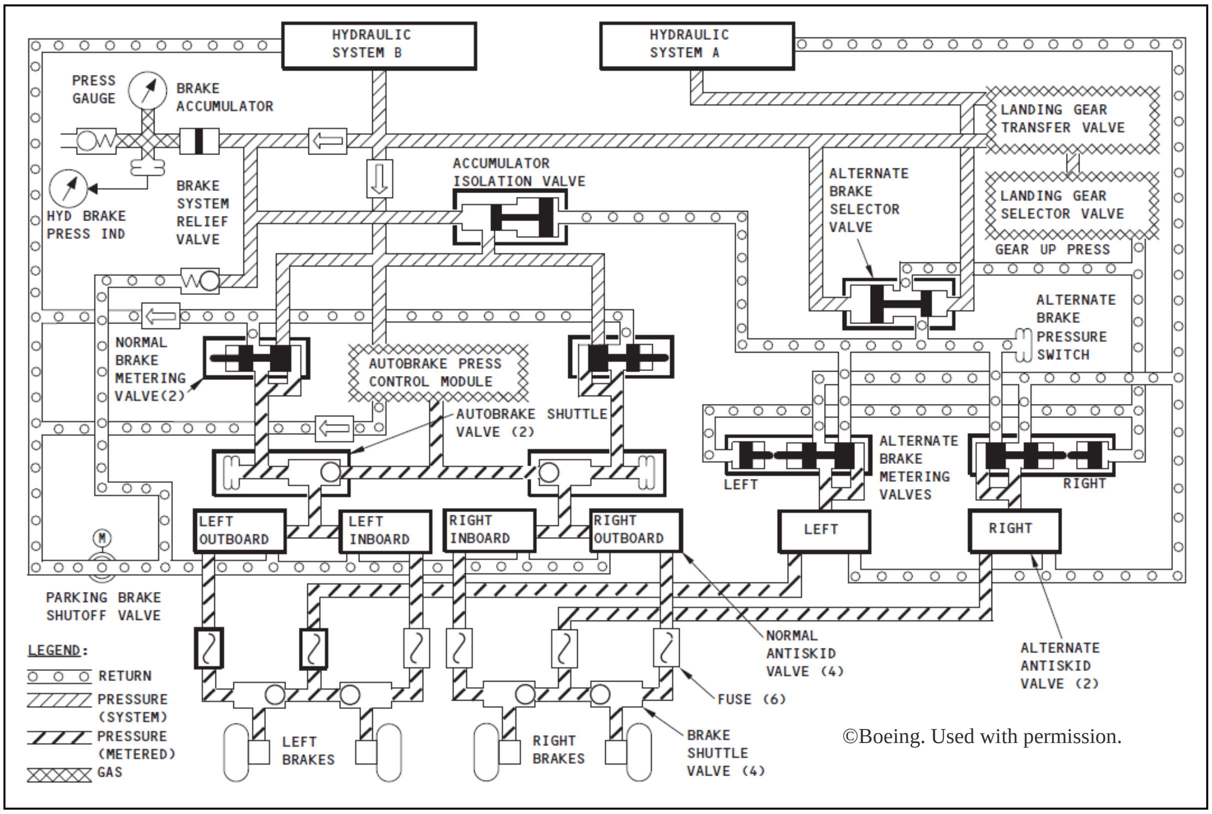

Figure 1: Hydraulic Brake System

F.1.2 Antiskid System:

The antiskid system monitors wheel deceleration and controls the metered brake pressure to prevent wheel skids during brake application. Boeing stated that the tire pressures observed on-scene would not affect the antiskid operation. The antiskid system is operational whenever the associated electrical buses are powered and requires no flight crew action. When brake pressure is released to a wheel that is skidding, the wheel speed is permitted to increase which stops the skid condition. When the normal braking system is active there is an antiskid valve for each wheel brake. The antiskid valve releases pressure to its associated wheel brake when commanded by the Antiskid Autobrake Control Unit (AACU). The unwanted pressure is released through the parking brake valve. A transducer in each main landing gear wheel axle supplies wheel speed data to the AACU. The ANTISKID INOP amber light comes on if the built-in test card in the AACU detects a fault in the antiskid system. The AACU monitors for faults related to system power, Antiskid Inop light operation, sense relays, wheel speed transducers, parking brake lever and parking brake shutoff valve disagree, the pressure switch on the alternate brake selector valve, antiskid valves, and the AACU itself. If the antiskid inboard or outboard circuit breaker is open the AACU will detect it as a fault. When certain faults (including an open antiskid inboard or outboard circuit breaker) are detected in the antiskid system the autobrake system becomes inoperative. These are the antiskid functions (see Figure 2):

- Skid control operates at a ground speed of more than eight knots to control each wheel deceleration independently during normal braking system antiskid operation. Skid control compares the calculated wheel speed velocity with a velocity model to control wheel deceleration. If a wheel slows down too quickly, the skid control releases brake pressure until the wheel speed increases.

- Locked wheel protection compares the wheel speed of the two outboard or the two inboard pair of wheels. If the slower wheel speed decreases to less than 30 percent of the faster wheel speed, the locked wheel protection releases brake pressure from the slower wheel. Locked wheel protection does not operate at a ground speed less than 25 knots.

- Touchdown protection releases brake pressure from wheels 2 and 4 while the airplane is in the air and remains active until 0.7 seconds after the corresponding wheel spins up to 70 knots, or when the ground mode has been sensed continuously for three seconds. The Proximity Sensor Electronics Unit supplies the air or ground signal to the AACU.

- Touchdown/hydroplane protection compares wheel speed data to Air Data Inertial Reference Unit (ADIRU) ground speed data. Wheel 1 is compared to the left ADIRU data and wheel 3 is compared to the right ADIRU data.[1] When the wheel speed decreases to 50 kts less than ground speed, the touchdown/hydroplane protection releases pressure to the corresponding brake. The hydroplane function supplies protection to wheels 1 and 3 only.

Indirect hydroplane protection to wheels 2 and 4 are provided through the locked wheel protection function. If all wheels spin down due to hydroplaning, the direct hydroplane protection will release pressure to the brakes for wheels 1 and 3. This allows these wheels to speed up which will then trigger the locked wheel protection to release the brakes for wheels 2 and 4.

Touchdown protection (releasing brakes until the airplane is on the ground) is also inherently provided for the wheels 1 and 3 by hydroplane protection since the speed difference between the wheel speeds and the ADIRU ground speed is greater than 50 kts until wheel spin-up is achieved at touchdown. Touchdown protection persists for half a second on all wheels after the airplane is on the ground to allow the tires to spin up to airplane synchronous speed before braking is allowed.

[1] The right ADIRU groundspeed value is not recorded by the FDR. However, if the right ADIRU groundspeed is flagged invalid it will cause an "IRS FAULT" light which is recorded on the FDR. Per the accident FDR data, the right ADIRU groundspeed was not flagged invalid. In addition, Boeing stated that a difference in ADIRU groundspeed would also create a difference in ADIRU position. The position difference would be identified by the Flight Management Computer and would generate a scratchpad message "VERIFY POS: IRS-IRS" along with the FMS MESSAGE light. This would also trigger a Master Caution.

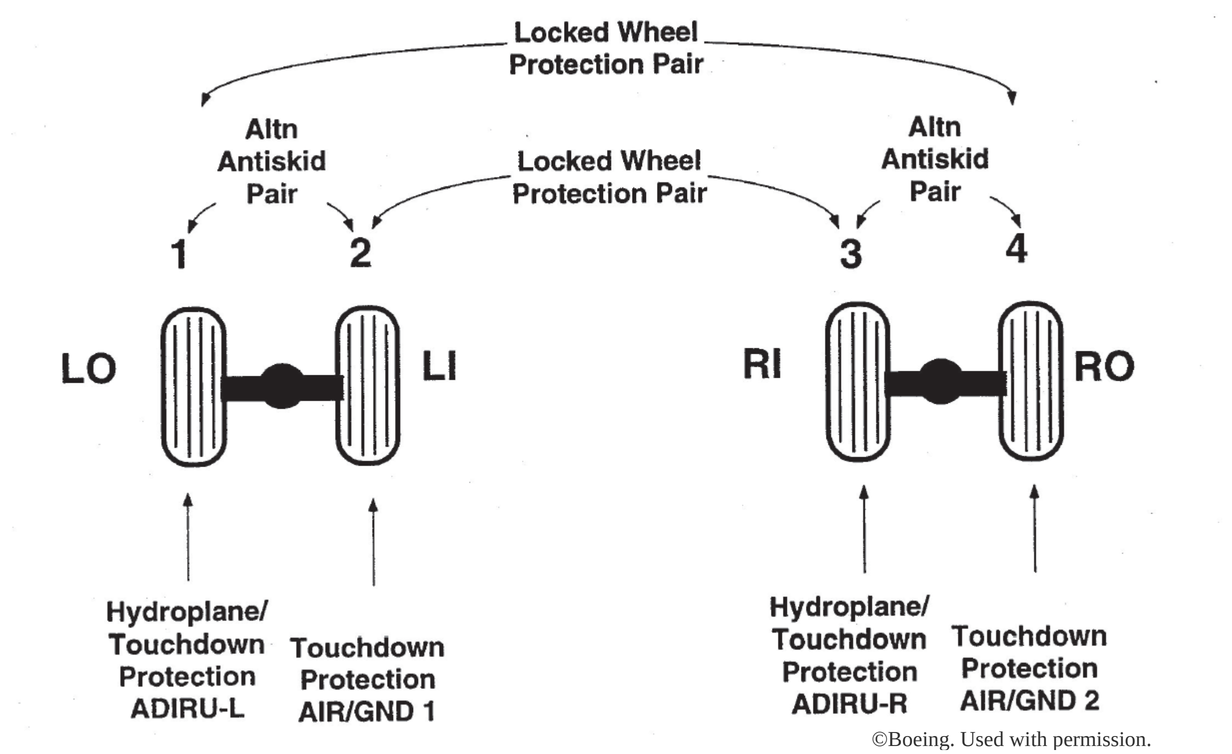

Figure 2: Antiskid Functions

F.1.3 Autobrake:

The autobrake system supplies metered brake pressure to stop the airplane after the airplane lands or if a rejected takeoff (RTO) occurs. The autobrake system monitors wheel deceleration and controls metered pressure to maintain the target deceleration rate selected by the pilot on the AUTO BRAKE select switch until the airplane comes to a full stop. The pilot can select a setting of RTO, OFF, 1, 2, 3, or MAX depending on the desired deceleration rate. The autobrake system arms for landing when there are no associated faults in the autobrake system or the normal antiskid system, and all the following conditions occur:

- The AUTO BRAKE select switch is moved to a landing deceleration position (1, 2, 3, or MAX)
- Both air/ground systems in air mode, or both thrust levers at idle, or one or both air/ground systems in the ground mode for less than or equal to three seconds
- Valid input from the left ADIRU
- Normal brake metered pressure is less than 750 psi

The autobrake function applies the brakes when these conditions occur:

- Landing autobrake is armed
- Both thrust levers at idle

- Either air/ground system continuously indicates ground for 0.2 seconds (if wheel spin-up occurs more than one second before ground is sensed) or 0.7 seconds (if wheel spin-up occurs less than one second before ground is sensed).
- Wheel spin-up detection occurs or the spin-up latch sets. Wheel spin-up detection occurs when one wheel on each main landing gear increases to 60 kts or greater and the wheel speed stays above 30 kts. The spin-up latch sets 3 seconds after the air/ground system is in the ground mode and the wheel spin-up detection occurs.

F.2 Brake Component Testing:

The Antiskid Autobrake Control Unit (AACU), four antiskid valves, and four wheel speed transducers were removed from the accident aircraft for component level testing. The AACU was removed from the forward electronics bay and required a pry bar to release it from its deformed rack. The nose gear folded aft causing damage to the forward electronics bay. The antiskid valves were removed from the main landing gear wheel well. The location of each valve was as follows:

#1 (lower left location): P/N 39-353; S/N 18315
#2 (upper left location): P/N 39-353; S/N 23467 (accidently dropped in dirt with fittings removed)
#3 (upper right location): P/N 39-353; S/N 23597
#4 (lower right location): P/N 39-353; S/N 23582

The wheel speed transducers were removed from each main gear wheel hub. It was noted that water came out of the wheel hubs when the transducers were removed. The location of each transducer was as follows:

#1 Transducer: P/N 140-025-2; S/N 0296346
#2 Transducer: P/N 140-025-2; S/N 0291415
#3 Transducer: P/N 140-025-2; S/N 021569
#4 Transducer: P/N 140-025-2; S/N 0291443 (Backshell of connector on wire end was loose)

The above listed components were manufactured by Crane Aerospace. Investigators met at the Crane Aerospace facility, Burbank, CA, on May 28-30, 2019 for the examination of these components.

F.2.1 Antiskid Autobrake Control Unit (AACU):

Part Number: 42-935-2
Serial Number: 690
Date of Manufacturer: 11-99
Date of Install on S/N 30618: 04/06/2010

The Built-In Test Equipment (BITE) instructional card was present in the card slot located on the front face. The upper and lower covers were removed, and an initial visual inspection was performed. The circuit boards were in place and did not reveal any noticeable physical damage. No moisture was observed inside the unit. The unit was connected to the test bench and the non-volatile memory (NVM) was downloaded using software version FD-299-043951-00.01. Memory blocks are created in NVM only when the reset button on the front panel is pressed; therefore, it is possible for many flight legs and power ups to be associated with one block. Crane confirmed that any faults identified by the AACU BITE during the landing should have been recorded to NVM even if there was a sudden power loss as long as there were no short circuits during or prior to the power loss. The current block (the block associated with the accident flight) was labeled Block 82 and contained no faults. The previous block (Block 81) also contained no faults. Blocks 80-78 contained a fault labeled "Fault 51-invalid index". The NVM decoder did not have this fault defined and therefore labeled it as "invalid index". Crane determined that this fault was related to the left inboard speed switch control, and it occurs when the voltage measured at the left inboard speed switch control disagrees with the speed switch output. Crane stated that the speed switches are used to enable deployment of the speed brakes. Boeing confirmed that a speed switch failure would result in the activation of the Antiskid Inop light.

An acceptance test (TP42-935-2, Rev F) was performed on the unit. A total of 19 faults were identified during this test and 15 faults were recorded to the NVM during this test. Crane confirmed that some of these faults would have caused the autobrake to become inoperative.

The circuit boards were removed. Various discolorations were observed on the circuit boards. Considerable sand and debris were observed inside the unit and adhered to the circuit cards. There were no bent pins, burn marks, or other signs of mechanical damage. The circuit cards were brushed and vacuumed to remove debris and the pins were cleaned with alcohol. The circuit cards were placed back in the unit and a second ATP was performed. A total of 45 faults were identified. Due to a possible error that was made during the manual portion of this test, the AACU was connected to a different test box and an ATP was again performed. This test resulted in 50 faults.

Crane had no records of repair for this unit. Miami Air had one record of repair from Lufthansa Technic with a repair tag dated April 1, 2010. The reason for repair was that sometimes the speed brake would not auto-extend.

F.2.2 Antiskid Valves:

Part Number: 39-353
Serial Number: 18315
Date of Manufacturer: 3/9/1998
Date of Install on Aircraft S/N 30618: 11/22/2008

The antiskid valve was visually inspected with no physical anomalies identified. The unit was connected to the test bench and the internal hydraulic fluid was flushed. The flushed fluid was tested for particle count and was found to be Class 4 per SAE AS4059. Test procedure TP-39-353, Rev K, was performed. The insulation resistance, dielectric strength, and proof pressure tests were performed last. The unit passed the coil resistance test and the leakage at 1500psi test. The leakage at 1500psi test required a 3000psi input at the pressure port, 1500psi output at the brake port, and required 90cc/min to 600cc/min leakage at the return port in order to pass. The unit then failed multiple test steps. It was observed that with a 3000psi input pressure, the maximum brake output pressure was initially about 1500psi, and then stabilized at about 250psi.

A teardown was performed. The electrohydraulic servo valve (EHSV) was removed. The brake port, pressure/filter port, and return port o-rings were removed and the ports were flushed with alcohol. No particulates were visually identified. The o-rings were returned to their respective ports and the EHSV was placed on a test fixture. Pressure was applied to the EHSV and it allowed a maximum brake port output of 13 psi (expected output was 3000 psi).

The EHSV from antiskid valve S/N 23467 (after testing of S/N 23467 was completed) was removed and placed on the control valve body for S/N 18315. Test procedure TP-39-353, Rev K, was performed. The unit passed all test steps except for step 11.3-2 [Pressure vs. Current-Brake Port (Pressure Gain)]. This test plots the output pressure as it corresponds to the current supplied to the EHSV. According to Crane these results, although outside of ATP requirements, would not significantly affect the antiskid valve operation.

The EHSV from S/N 18315 was examined. The cover was removed and corrosion was observed on the magnet. The o-ring was present on the base of the servo body. The o-ring on the base of the canon plug was present but deformed. The ground wire separated from the back of the canon plug connector during cover removal. The wire tips at the break were shiny and necked. A bulb of black and metallic material that contained some wire strands was adhered to the ground wire connector spade. The same material appeared to be on the wire side of the break as well. The magnet base was not flush with the body and resulted in a gap approximately 0.048 in. The magnet alignment pins were not aligned with the alignment holes, but the magnet was secured in place by the installation screws. Torque was applied to the magnet installation screws to see if they could be tightened and the screws remained fixed. Torque was applied to loosen the screws. A breakout torque was observed followed by the screws loosening. The magnet set screws were removed and the interior coil was removed. A yellow sludgy liquid consistent with water was observed under the coil.

The potting that seals access to the nozzles was removed. The return nozzle was removed. A relatively large piece of foreign object debris (FOD) was observed to be adhered to the nozzle cone near, but not covering, the return orifice. During the attempt to remove the FOD, the FOD popped off the nozzle and was lost. The return nozzle tip (orifice circumference) had two areas of deformation. The pressure nozzle was also removed and inspected with no FOD identified or other anomalies noted.

The filter RTV sealant, filter plug, retainer ring, & o-ring were removed. The filter was removed and a slight kink was observed at the open end. The filter was flushed in the forward flow direction and no particles were observed. The filter was flushed in the reverse flow direction and three particles were observed measuring in a range from 0.0023in to .0156in. The particles broke apart when touched with a probe.

The two magnet alignment pins and the fluid barrier end cap nut were removed. A dark sludgy fluid was identified under the end cap nut. The fluid barrier was removed and fluid resembling hydraulic fluid was observed within the cavity. The armature was removed and rinsed with alcohol. Three fiber like pieces of FOD (the largest measuring approximately 0.033 inch in length), some very fine black dust particles, and a very small amount of an orange colored residue were observed. The armature had witness marks on both faces corresponding with the nozzle tips. The upper portion of the armature shaft on the return side demonstrated pitting. The thermal plate and fluid barrier o-ring were removed. The o-ring was observed to be slightly flattened and a bit stiff.

The magnet from the EHSV from antiskid valve S/N 23467 was replaced with the magnet from S/N 18315 and the EHSV was placed on the test bench. The magnet cover was not reinstalled. The Pressure Gain plot followed the general slope of the min/max envelope but was outside of the envelope bounds. The magnet was then readjusted and approximately orientated as originally found in S/N 18315. The Pressure Gain plot again followed the general slope of the min/max envelope but was even farther outside of the envelope bounds. The ground wire was then disconnected. This resulted in no data for the pressure gain (pressure vs current plot) test. The Pressure Gain plot showed only a single data point of approximately 2840psi at 0 mA.

Crane had no records of repair for this unit. Miami Air had one record of repair from Delta Air Lines with a repair tag dated November 13, 2007. The reason for repair was not noted on the presented card.

Part Number: 39-353
Serial Number: 23467
Date of Manufacturer: 12/13/00
Date of Install on Aircraft S/N 30618: Original Install

The antiskid valve was visually inspected. There were some scratches on the solenoid cover, but no major physical anomalies were identified. The unit was connected to the test bench and the internal hydraulic fluid was flushed. The flushed fluid was tested for particle count and was found to be Class 6 per SAE AS4059. Test procedure TP-39-353, Rev K, was performed. The unit passed all test steps except for step 11.3-2 [Pressure vs. Current-Brake Port (Pressure Gain)]. This test plots the output pressure as it corresponds to the current supplied to the EHSV. According to Crane these results, although outside of ATP requirements, would not significantly affect the antiskid valve operation.

Crane had no records of repair for this unit.

Part Number: 39-353
Serial Number: 23597
Date of Manufacturer: 01/17/2001
Date of Install on Aircraft S/N 30618: Original Install

The antiskid valve was visually inspected with no physical anomalies identified. The unit was connected to the test bench and the internal hydraulic fluid was flushed. The flushed fluid was tested for particle count and was found to be Class 3 per SAE AS4059. Test procedure TP-39-353, Rev K, was performed. The unit passed all test steps except for step 11.3-2 [Pressure vs. Current-Brake Port (Pressure Gain)]. This test plots the output pressure as it corresponds to the current supplied to the EHSV. According to Crane these results, although outside of ATP requirements, would not significantly affect the antiskid valve operation.

Crane had no records of repair for this unit.

Part Number: 39-353
Serial Number: 23582
Date of Manufacturer: 01/17/2001
Date of Install on Aircraft S/N 30618: Original Install

The antiskid valve was visually inspected with no physical anomalies identified. The unit was connected to the test bench and the internal hydraulic fluid was flushed. The flushed fluid was tested for particle count and was found to be Class 5 per SAE AS4059. Test procedure TP-39-353, Rev K, was performed. The unit passed all test steps except for step 11.3-2 [Pressure vs. Current-Brake Port (Pressure Gain)]. This test plots the output pressure as it corresponds to the current supplied to the EHSV. According to Crane these results, although outside of ATP requirements, would not significantly affect the antiskid valve operation.

Crane had no records of repair for this unit.

F.2.3 Wheel Speed Transducers:

Part Number: 140-025-2
Serial Number: 0296346
Date of Manufacturer: 2/1/1996

The transducer assembly was visually inspected with no anomalies noted. The unit was connected to the test bench and functional test procedure TP-140-025, Rev J, was performed. The unit passed all test steps that were related to the RPM output signal. The unit failed the dielectric leakage test and the insulation resistance test. The unit was opened, and water/moisture was observed in the cannon plug housing, coil cavity, housing cavity, and rotor. White corrosion was observed on the rotor.

Crane had no records of repair for this unit.

Part Number: 140-025-2
Serial Number: 0291415
Date of Manufacturer: 2/1/1991

The transducer assembly was visually inspected. The only observed external anomaly was that the connector flange screws were corroded. The unit was connected to the test bench and functional test procedure TP-140-025, Rev J, was performed. The unit passed all test steps that were related to the RPM output signal. The unit failed the dielectric leakage test but passed the insulation resistance test. The unit was opened, and water/moisture was observed in the cannon plug housing, coil cavity, housing cavity, and rotor. White corrosion was observed on the rotor.

Crane had no records of repair for this unit.

Part Number: 140-025-2
Serial Number: 021569
Date of Manufacturer: 2/1/2015

The transducer assembly was visually inspected with no anomalies noted. The unit was connected to the test bench and functional test procedure TP-140-025, Rev J, was performed. The unit passed all test steps that were related to the RPM output signal. The unit failed the dielectric leakage test and the insulation resistance test. The unit was opened, and water/moisture was observed in the cannon plug housing, coil cavity, and the housing cavity. Reddish corrosion was observed on the rotor assembly bearing and white corrosion was observed on the rotor.

Crane had no records of repair for this unit.

Part Number: 140-025-2
Serial Number: 0291443
Date of Manufacturer: 2/1/1991

The transducer assembly was visually inspected with no anomalies noted. The unit was connected to the test bench and functional test procedure TP-140-025, Rev J, was performed. The unit passed all test steps that were related to the RPM output signal. The unit failed the dielectric leakage test and the insulation resistance test. The unit was opened, and water/moisture was observed in the cannon plug housing, coil cavity, housing cavity, and rotor.

Crane had no records of repair for this unit.

G. SPEED BRAKE DO NOT ARM LIGHT

There are four flight spoilers and two ground spoilers on each wing. The spoilers decrease lift and increase drag. If the speed brake lever is in the "Down" position during landing, the auto speed brake system moves the speed brake lever to the "Up" position and all the flight and ground spoilers deploy together when these conditions occur:

- main landing gear wheels spin up (more than 60 kts)
- both thrust levers are retarded to IDLE
- either reverse thrust lever is positioned for reverse thrust (interlock position or farther)

The "Speed Brake Do Not Arm" (SBDNA) light indicates an abnormal condition or test input to the automatic speed brake system. This light is deactivated when the speed brake lever is in the down position. SBDNA logic is based on parameters from 12 different sources, including air/ground relays, antiskid operation, throttle position (both throttles must be at idle stop), reverser position, and individual wheel speeds on the landing gear.

FDR data showed that the SBDNA light activated briefly on three different occasions during the landing rollout phase of the accident. The middle occurrence, which triggered at a ground speed of approximately 120 kts, occurred when many of the logic parameters were in a steady state and the aircraft was on the runway. At this time, the throttles were at idle, and all three landing gear weight-on-wheel switches indicate on ground.

According to Boeing, there have been past landing events in service where the SBDNA light illuminated once or more during the landing rollout while on a runway with low friction. The SBDNA light logic monitors the four main wheel speeds for spin rates above 60 kts. Signals from the left outboard (wheel 1) and right inboard (wheel 3) are paired, and the left inboard (wheel 2) and right outboard (wheel 4) are paired. These pairings correspond to the wheels that receive direct hydroplane protection (wheels 1 and 3), and those that receive hydroplane protection indirectly through locked wheel protection (wheels 2 and 4). If all four main wheels slow while ground speed remains high, hydroplane protection will allow wheels 1 and 3 to speed up, followed by locked wheel protection allowing wheels 2 and 4 to spin up. Boeing stated that the time needed for each wheel pair to spin up above 60 kts can vary depending on runway conditions and that this time difference can create a temporary difference of wheel speeds that can activate the SBDNA light.

H. COCKPIT VOICE RECORDER

H.1 Cockpit Voice Recorder System Description:

The Cockpit Voice Recorder (CVR) unit collects the captain's microphone and headphone audio, the first officer microphone and headphone audio, the observer microphone and headphone audio, and the cockpit area microphone audio. It stores the last 120 minutes of communication data in solid state memory. It erases the communication data automatically so that the memory stores only recent audio.

The inputs from the captain, first officer, and observer microphones go to the Remote Electronics Unit (REU). The REU mixes each station's microphone audio with that station's headphone audio. The REU then increases the audio signal and sends it to the voice recorder unit. The cockpit area microphone on the Cockpit Voice Recorder Panel collects flight compartment sounds, such as voices and aural warnings. The cockpit voice recorder panel increases the audio signal from the area microphone and sends it to the voice recorder unit. See Figure 3.

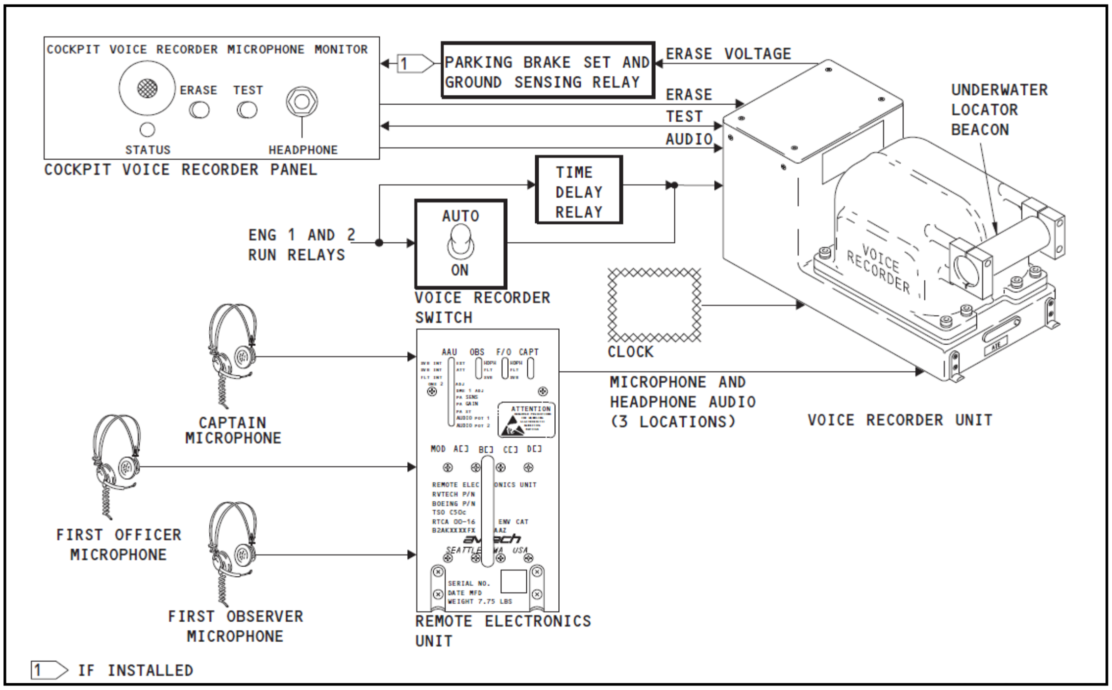

Figure 3: Cockpit Voice Recorder System Components

H.2 CVR Datalink Recording Modification:

Miami Air was in the process of modifying the CVR system to a solid state CVR with Aircraft Communications Addressing and Reporting System (ACARS) message recording capability. Miami Air had completed Work Order 24279 on 4/27/2019, which included the wiring installation and activation for the modification. Miami Air stated that this modification to the accident aircraft was being used to certify a proposed Supplemental Type Certificate (STC).[2] For this purpose, a CVR unit (P/N HFR5-V, S/N 08092) with datalink communications recording capabilities was installed, used to test the modification installation, and sent away for readout. After this unit was sent away for readout, Miami Air reinstalled an old type CVR (P/N 980-6022-01, S/N 01481) unit, which was not capable of datalink communications recording. This is the unit that was on the aircraft at the time of the accident. According to the FAA, since the STC was not yet approved at the time of the accident, the aircraft should not have been operating in this configuration without additional approval. The FAA further stated that no additional approval was given. The accident CVR recording was extracted by the NTSB and the HOT microphones and cockpit area microphone recordings were found to be of insufficient quality (see the CVR Group Chairman's Factual Report for more details).

The NTSB received a copy of the CVR recording that was intended to certify the installation of

[2] STC number STO4461AT was issued on September 24, 2019.

the STC. The recording did not contain the use of the HOT microphones and the recording occurred while the aircraft was on the ground with no operational noises present. Therefore, this recording was not able to be used by investigators to confirm if it exhibited the same defects as the accident CVR.

In addition, Miami Air sent a CVR (P/N 980-6022-01, S/N 12609) from a sister aircraft (registration N733MA) to the NTSB for download and comparison to the accident CVR recording. The aircraft had the same CVR configuration as the accident aircraft (the STC wiring upgrade was installed but the aircraft was using the old type CVR without ACARS messaging recording capability). The recording was again of insufficient quality (see the CVR Group Chairman's Factual Report for more details).

Finally, Miami Air sent another CVR (P/N HFR5-V, S/N 08135) from a second sister aircraft to the NTSB for download and comparison to the accident CVR recording. This aircraft had the same STC wiring upgrade installed with the new type CVR that included ACARS messaging recording capability. This recording, while not optimal, was likely of sufficient quality to be used in an accident investigation. Attempts to further investigate the CVR issues on Miami Air aircraft were discontinued when Miami Air suspended operations.

H.3 Cockpit Voice Recorder System Component Testing:

H.3.1 Cockpit Voice Recorder Panel:

Manufacturer: Honeywell
Part Number: 980-6116-001
Serial Number: 2091
Modification Status: 1, 2, 3

The Cockpit Voice Recorder Panel from the event airplane was examined at Otto Instruments in Ontario, CA on 8/15/2019. Honeywell owns the design for this component but contracts Otto Instruments to perform component repair and overhaul. The examination was witnessed by a representative from Miami Airlines.

The unit was functionally tested per Honeywell CMM 012-0666-001 (23-70-43), Rev 7. The unit passed this functional test in its entirety. The examination identified some melting of the microphone alignment bushing and also determined that the ERASE and TEST switches were noisy when pushed (the noisy condition was not present when the switches were not pushed). Melting of the microphone bushing is a known issue and Honeywell released Service Information Letter SIL D201709000023 in October of 2017 to increase awareness. According to the SIL, the problem identified with the melted bushing is that a black sticky substance emanates from the area of the bushing. There is no mention of degraded microphone performance due to the melted condition.

H.3.2 Remote Electronics Unit:

Manufacturer: AvtechTyee
Part Number: 5140-1-10
Serial Number: 2773
Modification Status: No mods marked
Date of Manufacturer: Oct 1999

The Remote Electronics Unit (REU) from the event airplane was examined at the AvetechTyee facility in Everett, WA. The purpose of the examination was to check the gain settings and functionality of the REU with regards to the captain's and first officer's hot microphones.

The unit arrived at AvtechTyee on 6/5/2019 and AvtechTyee performed an incoming inspection on 6/6/2019. They reported that power was applied to the unit and that they completed preliminary manual diagnostic tests and checked for basic functionality. No failures were noted during the manual bench tests and all front panel calibration controls were within specified limits. The unit powered up in normal mode with flight interphone selected on all three stations (captain, first officer, and observer). Acceptance Test Procedure 5140-4, Rev AY, was performed. The steps that directly correlate to the gain settings, CVR frequency response and distortion, and hot mic functionality passed their respective test steps. Several test steps related to the captain's, first officer's, and observer's auxiliary speaker outputs and auxiliary headphones were slightly outside of required limits. AvtechTyee stated that it was typical for these parameters to be slightly out of calibration on older units. In addition, they stated that the test results were not far enough out of specification limits to impede function. The unit passed all other test steps. During this initial phase of testing, the cover was removed, but the printed circuit boards were not removed, and no adjustments of the calibration controls were made. Corrosion was identified inside the unit during the visual inspection.

Representatives from Miami Airlines and Boeing arrived at AvtechTyee on 6/12/2019 to discuss the Acceptance Test Procedure results and perform the tear down activities. The printed circuit boards were removed, and additional corrosion was identified within the unit. The AvtechTyee technician noted that 3 of the 4 printed circuit boards were corroded/damaged beyond repair. The 4th printed circuit board had some corrosion that could potentially be repaired.

AvtechTyee had one record of repair for this unit. It was returned on 11/7/2000 for a complaint related to the audio and video system. The REU does not handle video signals. The unit was tested with no fault found. AvtechTyee also noted that the unit was initially manufactured as P/N 5140-1-6, and that they had no records of performing the work that modified it to a 5140-1-10.

I. ADDITIONAL AIRCRAFT COMPONENT EXAMINATIONS

I.1 Enhanced Ground Proximity Warning System (EGPWS):

Manufacturer: Honeywell
Part Number: 965-0976-003-212-212
Serial Number: 10280
Modification Status: 1-9
Date of Manufacturer: 0103
Terrain Database Version: 431

The Mark V Enhanced Ground Proximity Warning System (EGPWS) provides aural and visual alerts and warnings to prevent controlled flight into terrain (CFIT) and for low altitude wind shear conditions. The EGPWS uses aircraft inputs such as position, attitude, airspeed, and glideslope along with an internal terrain and obstacle database to predict potential conflicts in the aircraft's projected flight path. Audible alert messages and visual clues alert the crew if a potential collision is detected. The EGPWS contains non-volatile memory that records information related to system faults, warnings, and airplane status.

The location of the EGPWS as installed on the airplane was in the forward electronics bay which was submerged under water after the aircraft came to rest in the river. The unit's non-volatile memory was downloaded on May 22, 2019, at the Honeywell Aerospace facility in Redmond, WA. Representatives from the NTSB and Boeing witnessed the download. Visual inspection revealed that the external case was intact. Small amounts of debris and corrosion were observed both externally and internally.

The circuit cards were cleaned and dried. The U210 chip containing non-volatile memory was removed and placed in a universal device programmer. The device programmer was used to read the chip and create a copy of the chip image. EGPWS flight history data was extracted from the copy of the chip image. The U210 chip was then re-installed on the original board and a standard flight history download was performed per Honeywell procedure 060-4199-115, Rev D.

The download revealed that a takeoff record for the accident flight was recorded at EGPWS total operating time of 87071:01:45. A takeoff record is created when the EGPWS air/ground logic switches to "in air" mode. For this aircraft type, "in air" mode will be selected if weight on wheels is false and radio altitude is greater than 5 feet for one second, or if airspeed is greater than 90 knots and radio altitude is greater than 5 feet for 11 seconds. A landing record for the accident flight was recorded at EGPWS total operating time of 87073:27:08. A landing record is created and stored to NVM when the aircraft transitions through approximately 50 ft radio altitude during descent.

A "warn" log is recorded in NVM when an alert condition occurs. There were two "warn" records recorded for the accident flight leg. At EGPWS total operating time of 87071:01:57 a M6BA (Bank Angle) alert was recorded. This EGPWS was configured to provide bank angle alerts at 35, 40, and 45 degrees. The recorded data in the log showed that at the time of the alert the bank angle was approximately 35 degrees. At EGPWS total operating time of 87073:27:03 an M1SK (Sink Rate) alert was recorded. This alert occurs when the sink rate performance envelope based on radio altitude and descent rate is breached. The recorded data in the log showed that at EGPWS total operating time of 87073:27:02 (one second prior to the alert time stamp), the radio altitude was approximately 155 feet and the descent rate was -1455 ft/min. Honeywell confirmed that these alerts occurred at the expected times based on the recorded data and alert laws. There were no wind shear alerts recorded for the event flight.

The memory file obtained using the device programmer method revealed one recorded fault for the accident flight leg. A "fault" log is recorded in NVM when the EGPWS determines there is a system failure with the EGPWS or with an input provided from an external system. The fault was titled "RADIO ALTIMETER BUS 2" and occurred while the aircraft status was "in air". This fault was also recorded on multiple previous flight legs throughout the fault log history. This fault occurs when the EGPWS determines the signal from the radio altimeter is invalid. There are two redundant radio altimeters that provide input to the EGPWS, and the EGPWS can operate normally with only one of the radio altimeters being valid. Radio altitude is a parameter that is recorded in the EGPWS NVM when an alert is recorded. The recorded radio attitude at the time of the Bank Angle advisory alert and at the time of the Sink Rate caution alert appeared to be valid. The memory file obtained using the standard download procedure contained one additional fault labeled "SUPPORT TASK FAILED". This fault occurs when the EGPWS detects a software error in a support task that is not needed for the EGPWS to perform its primary functions. This fault was observed during the download procedure at Honeywell; and, since it was also not present in the memory file obtained using the device programmer, the group determined that this fault was introduced during the download procedure at Honeywell.

An "event" log is recorded in NVM when an EGPWS function is declared inoperative by the EGPWS due to required data being not present/not valid. The memory file obtained using the device programmer method did not contain any events for the accident flight leg. The memory file obtained using the standard download procedure contained the following events for the accident flight leg at EGPWS total operating time 87073:28:02 (54 seconds after the landing record was written): GPWS INOP, WINDSHEAR INOP, MODE 6 INOP, BANKANGLE INOP, TAD INOP, and ENV MODE INOP. Based on the time stamp of these events, and the lack of these events in the file obtained using the device programmer, the group determined that these events were introduced during the download procedure at Honeywell.

I.2 Antiskid Inop Light:

The "Antiskid Inop" light illuminates when the AACU determines there is a problem with the antiskid system. The annunciator light contained two bulbs. The bulbs were examined using x-ray. There was no stretch identified on either bulb filament. One filament had a single break.

Adam Huray
Mechanical Engineer