NATIONAL TRANSPORTATION SAFETY BOARD
OFFICE OF AVIATION SAFETY
WASHINGTON, D.C. 20594

6-30-2009

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB ID No.: DCA09MA021

A. ACCIDENT:

Location:	Denver International Airport
Date:	December 20, 2008
Time:	About 18:18 PM Mountain Standard Time (MST)
Aircraft:	Boeing 737-500

B. SYSTEMS GROUP:

Chairman: Mike Hauf
National Transportation Safety Board
Washington, D.C.

Member: Paul Bowman
Continental Airlines
Houston, Texas

Member: Jeffrey J. Mee
ALPA

Member: Kenneth Fairhurst
Federal Aviation Administration
Seattle, Washington

Member: Chris Dubuque
Boeing Commercial Airplanes
Seattle, Washington

C. SUMMARY:

On December 20, 2008, at 1818 Mountain Standard Time, Continental flight 1404, a Boeing 737-500 (registration N18611), equipped with CFM56-3B1 engines, departed the left side of runway 34R during takeoff from Denver International Airport (DEN). The scheduled, domestic passenger flight, operated under the provisions of Title 14 CFR Part 121, was enroute to George Bush Intercontinental Airport (IAH), Houston, Texas. One of the five crewmembers was seriously injured, and four of the 110 passengers were seriously injured. There were 37 minor injuries, and no fatalities. The airplane was substantially damaged and experienced post-crash fire. The weather observation in effect at the time of the accident was reported to be winds from 290 at 24 knots with gusts to 32 knots, visibility of 10 miles, and a few clouds at 4000 feet and scattered clouds at 10,000 feet. The temperature was reported as -4 degrees Celsius.

The Systems group was formed at an organizational meeting held on December 21, 2008 in Denver, Colorado. During the period of December 22 - 23, 2008, the systems group conducted their investigation at the accident site.

Figure 1 Photograph of N18611 at the accident site

Airplane N18611 came to rest essentially intact in a drainage basin about 40 feet below the runway elevation. The fuselage, wings, and engines sustained extensive impact damage resulting in fuselage and wing fracture, buckling, and major component separation (Figure 1). Additionally, during the crash sequence, both main landing gears and the left engine separated from the wings. Finally, the nose landing gear (NLG) strut was driven aft into the E&E bay resulting in the fuselage collapse and denying wheel well access. Due to this lack of accessibility, the nose gear and its associated components and the components within the main landing gear wheel well were not examined at the accident site. To ensure the NLG would remain in-place during recovery of the airplane, it was secured[1] with a cable between airplane structure and the NLG strut. During the period of January 2 -3, 2009 the airplane was recovered from the accident site and relocated to the Continental Maintenance Hangar located at the Denver International Airport.

During the period of January 3 – 4, 2009 the Systems group re-convened at the Denver Continental Maintenance Hangar to complete the documentation of the aircraft systems. At this time, the airplane was supported, which allowed investigation of equipment that was previously inaccessible while the airplane was resting on its fuselage.

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1 Holes were cut into the cabin floorboards to gain access to the nose gear since it could not be accessed from the lower fuselage.

D. <u>DETAILS OF THE INVESTIGATION:</u>

D.1 Runway Investigative Observations:

Examination of runway 34R revealed tire deposits on the runway surface near the area where N18611 veered off the runway at approximately 2,500 feet from the threshold. Tire deposits consistent with those from the left and right main landing gear were observed on the runway surface starting at approximately 1,910 feet from the threshold and continued approximately 600 feet in a direction towards the left side of the runway (Figure 2). Tire marks consistent with those from the nose gear were observed on the left side of the runway near the point of runway excursion. The main gear tire marks continued through a grassy area at an approximate angle of 10° to the left of the runway centerline, with no nose gear mark visible.

Figure 2 Tire marks on runway 34R

D.2 Nose Landing Gear and Nose Wheel Steering System:

Investigation of the Nose Landing Gear (NLG) and the Nose Wheel Steering (NWS) Systems were conducted in two phases. The first phase consisted of on aircraft inspection of the systems during the period of January 3 – 4 2009 at the Denver Continental Maintenance Hangar. At this time, the conditions of the systems were documented and components identified and removed for further examination. The second phase consisted of conducting laboratory examination of the components identified in phase 1.

D.2.1 Nose Landing Gear (NLG):

D.2.1.1 Description:

The NLG supports the fuselage and provides directional control while the airplane is on the ground. The NLG includes a drag brace, lock links, a shock strut, and torsion links. It is hydraulically actuated to retract forward and up into the wheel well recessed into the lower nose section of the airplane. The shock strut consists of inner and outer cylinders. The upper part of the outer cylinder is "Y" shaped with arms extended to the sidewalls of the wheel well. Trunnion pins connect the gear to airplane structure. The "Y" arms and pins provide lateral stability. The gear rotates about the trunnion pins during extension and retraction. Shocks and bumps during taxi, takeoff, and landing are absorbed by the shock strut, which contains oil and is charged with compressed air or nitrogen. Longitudinal stability is provided by a hinged drag brace which folds upward and forward during gear retraction. The nose landing gear has a slight forward rake, meaning that the shock strut is positioned five degrees from vertical, with the lower part of the strut forward of the upper part of the strut when the gear is down and locked.

D.2.1.2 Investigative Findings:

The NLG was found folded aft and impacted into the lower fuselage (Figure 3). The tires had entered the main electrical equipment ("E/E") compartment through the E/E door and frame, lodging on the edge of the doorframe. The structure between the nose gear wheel well and the E/E door was found crushed upward and around the displaced NLG strut. Both the forward and the left side E/E racks were found displaced upward.

The aft portion of the NLG wheel well, where the nose gear shock strut trunnions are located, was found displaced upward approximately 12 inches, thus allowing the gear to pivot aft without fracturing either the upper or lower drag braces (both of the drag braces were found damaged, but intact). All gear trunnion points (upper drag brace trunnions and shock strut trunnions) were found intact and rotated without difficulty using hand pressure.

The aft lock link was found fractured and bent[2]. Both the NLG actuator and the lock actuator were found intact and remained connected to their respective attachment

[2] With the aircraft relocated and now shored at the Denver Continental Maintenance Hangar, the cable securing the NLG was removed allowing the NLG to extend under the influence of gravity. Due to the extensive damage, the upper and lower drag braces were disconnected to allow full extension of the gear

fittings. The upper and lower nose gear torsion links were found intact and the nose gear air/ground target and sensor (installed on upper torsion link and the outer cylinder) were both intact with no observable damage. The shock strut outer cylinder remained intact and in place. Continental maintenance checked the nose gear shock strut pressure and found it to be deflated (less than 50 psi).

Figure 3 - Photograph of the nose gear folded into the fuselage

D.2.1.3 Wheels and Tires:

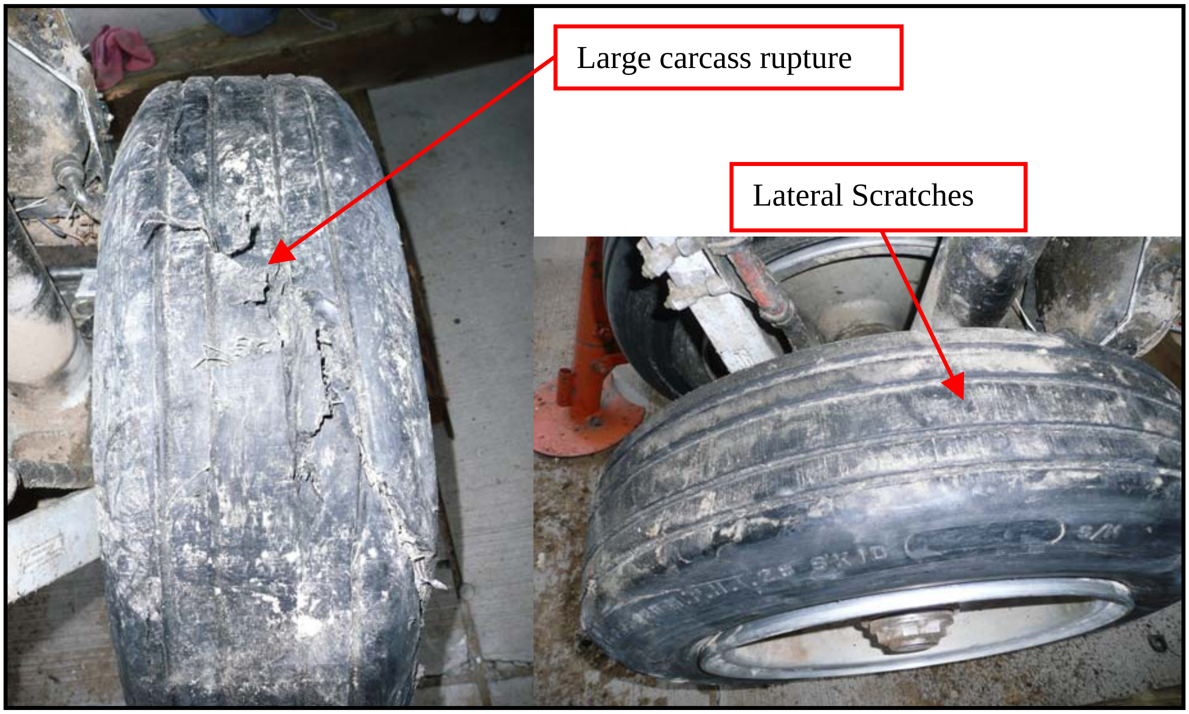
Airplane N18611 was equipped with two Bridgestone part number (P/N) APS01207 nose wheel tires[3] and two Honeywell P/N 2607825-1 nose wheels. Both of the nose wheels and tires were present, remained installed on their respective axles and contained numerous abrasions. The serial number (S/N) on the right tire was Y03FT151 and the S/N on the left tire was illegible. The serial numbers were 5621 and 0901 for the right and the left wheels respectively.

[3] The tire size, ply and speed rating markings were molded into the rubber of the tire and were:27X7.75-15, 12 ply, and 225 mph.

The left nose wheel tire remained inflated with a pressure of approximately 175 psi[4]. Minor lateral marks (scratches) were found on the tread surface around the full circumference. The retread level of this tire was R-5 (fifth retread).

The right nose wheel tire was found deflated and contained a large carcass rupture extending from shoulder-to-shoulder (Figure 4). The center of the rupture was in the crown of the tire with broken cords not on the bias angle of the tire, which is consistent with puncture or foreign object damage. The tire had significant lateral marks (scratches) on the surface of the tread around the full circumference of the tire. The outboard side of the innermost tread groove had notable damage, consistent with runway abrasion. This damage resulted in a 45-degree chamfer worn off of the top edge of this tread groove. The tire had a discolored area of about 2 inches by 1 inch on its outboard sidewall. The appearance of this area is consistent with heat damage but not from runway abrasion. It appears more consistent with external burn damage. The retread level of this tire was R-6 (sixth retread).

Figure 4 - Photograph of the rupture in the right nose gear tire



The left wheel could be rotated, but its bearings were found damaged and the inboard hub was found broken in the wheel bearing area. The wheel bearings on the right wheel appeared normal; the wheel rotated freely.

After the removal of the tires, examination revealed that the left axle of the nose gear was found bent upward about two inches as measured at the axle tip. The right axle remained intact and was found unremarkable.

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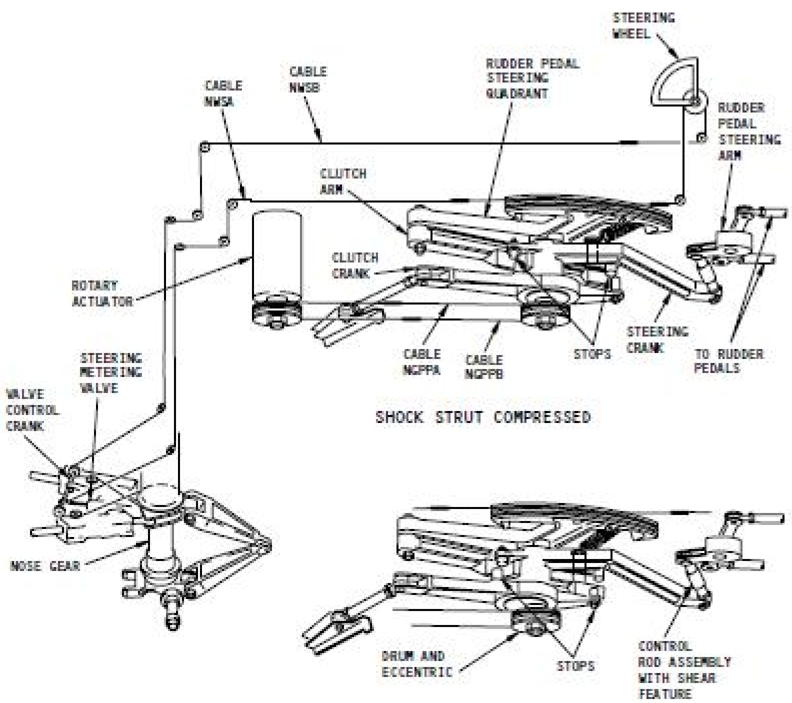
[4] The tire pressure gauge is in cert until 3/09

D.2.2 Nose Wheel Steering System (NWS):

D.2.2.1 Description:

The NWS provides airplane directional control during ground maneuvers and taxiing (Figure 5). "A" system hydraulic power is used to turn the nose wheels to either side from zero to 78 degrees. Steering is controlled by a control wheel (tiller) on the left side of the flight deck and by an interconnect mechanism from both sets of the rudder pedals. The steering system is spring-loaded to the center in the interconnect mechanism. Internal cams in the shock strut center the nose gear when the shock strut is fully extended.

Figure 5 Figure of the nose wheel steering system



Movement of the steering control wheel in either direction is transmitted by cables to a steering metering valve, which directs 3,000 psi hydraulic fluid to the nose wheel steering cylinders for turning the steerable portion of the nose gear. A steering control wheel movement of 95 degrees will result in 78 degrees of nose wheel deflection. The steering wheel always overrides the rudder pedal input to the NWS system.

Rudder pedal steering is available during takeoff, landing, and taxiing where limited directional changes are required. Full deflection of the rudder pedals commands about seven degrees of NWS. The rudder pedal nose wheel steering interconnect mechanism is engaged to allow nose wheel steering when in ground mode, as sensed by the squat switch located on the nose gear shock strut. When the nose gear is compressed, a squat switch senses the movement of the upper torsion link. The ground mode causes the rotary actuator to reposition the stops mounted on the clutch crank in the interconnect mechanism. In this position, any movement of the rudder pedals will be transmitted into steering system inputs and result in steering of the nose gear. When in the air mode, the interconnect mechanism does not pass rudder pedal inputs through to the steering system.

The maximum load that can be applied to the cable during operation of the NWS system is a function of the static rig load plus the force applied to the system (via the tiller). The rig load for the NWS cable is set to a nominal 40 pounds at ambient temperature (AMM 32-51-00). Boeing states that the maximum force a pilot could apply to the tiller is 150 pounds, which equates to a cable load of 240 pounds.

D.2.2.2 Steering Cylinders Investigative Findings:

Airplane N18611 was equipped with left and right nose wheel steering cylinders. Both of the two steering cylinders are connected to the steering collar, which is held clamped around the outer cylinder in an annular recess immediately below the trunnions. When force is applied to the steering collar, by either steering cylinder, the collar transfers the force through the torsion links to turn the inner cylinder to the right or left respectively to which cylinder force is applied to give steering action to the nose wheels. Both steering cylinders were found in the debris path of the airplane. The steering cylinders were photographed and then retained for examination. A visual examination revealed the lower steering collar had fractured and separated from the nose gear assembly; the section had separated was found in the debris field near the airplane. The upper steering collar was found bent downward about 30 degrees. Fractured rod ends from both steering cylinders remained attached to the steering collar.

Examination of the steering cylinders was conducted on February 25, 2009 at the Boeing Equipment Quality Analysis (EQA) facilities located in Seattle, Washington. In attendance were representatives from the NTSB, the Federal Aviation Administration, Boeing, ALPA, and Continental Airlines.

D.2.2.2.1 **Steering Cylinder "Right":**

The steering cylinder identified as the "right cylinder" was missing its identification tag. However, the part was recognized as being from the 65-44710 series. Visual examination revealed that a swivel assembly S/N 3192 had separated from the steering metering valve and remained attached to the cylinder housing. The swivel assembly was removed from the cylinder to allow testing of the cylinder. Examination of the cylinder rod revealed that the rod end had separated from the cylinder rod. Measurements of the cylinder rod indicated that the forward rod extended 4.755 inches from end plate and the aft rod was found extended 2.460 inches from its end plate.

The cylinder assembly passed all hydraulic functional testing per Boeing's CMM 30-50-11, with the exception of proof pressure being performed at 3,000 PSIG as opposed to 4,500 PSIG. Reducing the proof pressure to 3,000 PSIG was an onsite engineering decision made by the Systems group as the cylinder had been subjected to unknown loads during the crash sequence. The friction test was not performed since the rod moved under 50 PSIG. The low-pressure test was performed at 50 PSIG instead of 2 PSIG due to test bench limitations. The system's group elected not to disassemble the cylinder assembly because it passed functional testing.

D.2.2.2.2 **Steering Cylinder "Left":**

The steering cylinder identified as the "left cylinder" was missing its identification tag. However, the part was recognized as being from the 65-44710 series. Examination found that the cylinder housing trunnion mount had fractured. Examination of the cylinder rod revealed that its rod end had separated from the cylinder rod. A ~5.525 inch section of the cylinder rod remained extended out of the forward end of the cylinder housing[5]; the rod was found bent approximately 3½ degrees. A ~1.6 inch section of the cylinder rod remained extended out of the aft end of the cylinder housing.

Because the cylinder rod was found bent, the steering cylinder was not functionally tested. The System's group determined to disassemble the cylinder and document the findings. Following the removal of cylinder rod from the cylinder housing a measurement was taken to determine the location of the "bend" from the end of the cylinder rod. The "bend" was noted to be ~6.50 inches from the end of the cylinder rod.

Visual examination revealed that all seals were intact and in good condition, with dirt and debris found on the inside of the end caps outside of the seal. Using a microscope, the trunnion mount was examined and the fracture was determined to be the result of both a ductile overload with post-fracture smearing.

D.2.2.3 Steering Metering Valve Investigative Findings:

Airplane N18611 was equipped with a Sargent Controls & Aerospace nose wheel steering metering valve (SMV) having part number 4129RA9 (BAC part number 10-60590-4) and serial number 4164. According to maintenance records provided by Continental Airlines, valve S/N 4164 was installed on airplane N18611 on 10-2-2008. At this time, the valve had previously accumulated a total of 63,666.55 flight hours, and 33,708 flight cycles. The valve had been overhauled by Fortner Engineering & manufacturing Inc. on 09-08-2008.

A visual inspection of the SMV was conducted while it remained attached to the airplane. Investigation revealed that the fiberglass cover[6] that normally is installed over the summing mechanism[7], located on the front of the SMV, was found broken and sections were missing. The entire "summing mechanism" linkage was not located in the wreckage. The SMV remained attached to the nose gear upper support bracket by one of its three mounting bolts; the remaining mounting bolts were found sheared. Both the SMV and the upper support bracket remained attached to the nose landing gear. Visual inspection revealed abrasions and structural damage on the exterior of the valve body. Normally, two swivels (left and right) are attached to the metering valve housing and to their respective

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[5] Measurements were taken from the cylinder housing end plate to the end of the cylinder rod.
[6] The purpose of the cover is to preclude jamming from foreign object debris (FOD)
[7] , The summing mechanism transmits inputs from the cable system to the steering metering valve.

steering cylinder. The hydraulic swivels on each side of the SMV were found detached from the SMV with the swivel mounting hardware fractured. The right swivel assembly (housing and spindle) did not remain attached to the metering valve housing, its attachment bolts were found sheared. However, the complete swivel assembly remained attached to the right steering cylinder assembly. The left swivel housing[8] did not remain attached to the metering valve housing, its attachment bolts were found sheared. However, the left swivel spindle remained attached to the left steering cylinder housing. The unit's input rod[9] was found restricted and bent about 30 degrees downward. The SMV was removed from the airplane, photographed and then sent to Boeing for further examination.

A detailed examination of the SMV was conducted at the Boeing Equipment Quality Analysis (EQA) facilities located in Seattle, Washington on February 25, 2009. Participants in the examination included representatives from the NTSB, the Federal Aviation Administration, Boeing, ALPA, and Continental Airlines.

An inspection confirmed that the unit's input rod remained restricted in the same position as found during the on-scene activities. The metering valve assembly was X-rayed to determine the "as received" position of the slide within the valve. A review of the x-ray revealed that the slide was intact, and positioned approximately at its "full extend" position. In this position, the valve would port pressure commanding the nose wheel to a left turn. No mechanical anomalies were identified on the sleeve.

Functional testing of the steering metering valve was deemed not feasible due to the bent input slide and broken swivels. Removal of the slide and sleeve assembly was not performed to avoid destructive disassembly.

D.2.2.4 NWS Cable:

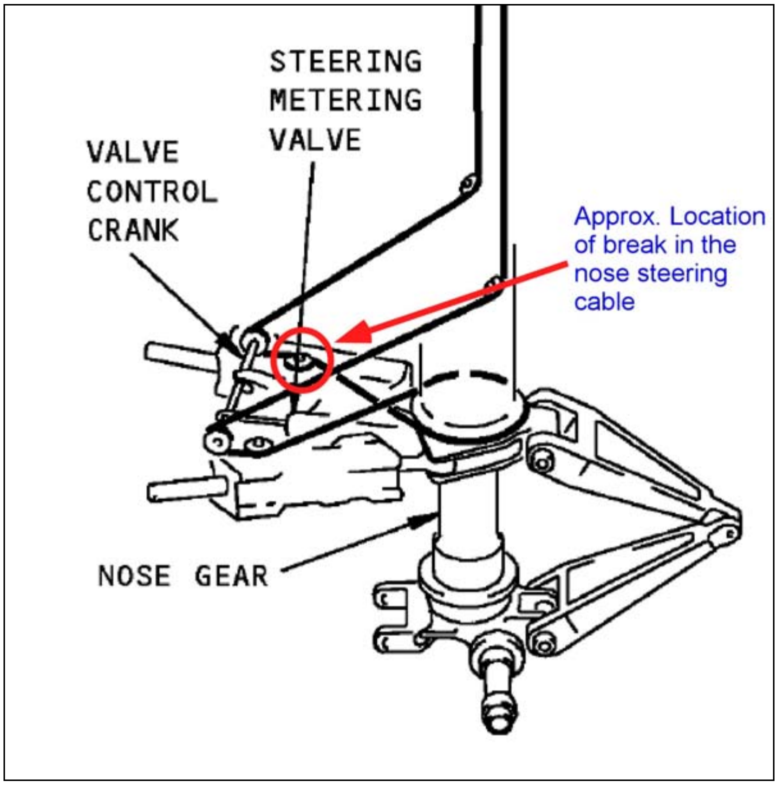
The NWS system is controlled by one continuous cable, identified as cable (NWSA/B), which is made of Corrosion Resistant Steel (CRES) and is of a 7x7 wire cable construction. The cable is connected to the control wheel (tiller) and then routed down and through the pressurized area of the airplane on the left side of the nose landing gear wheel well. The cable is routed through the left side of the nose landing gear trunnion to the strut and downwards towards the steering mechanism by two pulleys mounted at the top of the shock strut. The cable is then routed back following the same path to the control wheel (tiller).

Inspection of airplane N18611's NWS cable revealed that cable NWSB (right section of cable NWSA/B) was broken in the area of the horizontal pulley mounted on the right side of the lower steering plate (Figure 6) with the remaining cable found intact. Boeing states a cable break at this location may result in the nose gear steering (rotating) left approximately seven degrees. Cable (NWSA/B) was inspected in the area of the pressure seal located at the left strut trunnion. No notable wear or damage was found on the

8 The left swivel housing was never located.
9 Referred to as 'slide' in the Boeing EQA Analysis and subsequent paragraphs of this report.

cables in this location. Two sections of cable NWSB were cut (near the break) and sent for NTSB laboratory analysis (Reference NTSB Materials Laboratory Factual Report number 09-006 in the Safety Board's public docket for accident, DCA09MA021).

Figure 6 Figure showing the location of the cable break



Continuity of the NWS cable system was verified between the steering control wheel and the exposed cables in the nose wheel well. The continuity was verified by witnessing the movement of the nose wheel steering cable (in the nose wheel well) when the steering control wheel (tiller) was rotated in both the left and right directions.

A review of Continental's maintenance records revealed that the NWS cables (NWSA/B) were installed on airplane N18611 on May 24, 2000, and were last inspected on October 27, 2008 without any defects reported.

A review of Continental's FAA approved maintenance program revealed that their NWS cable inspection program requires more frequent inspections of the NWS cables than required by the current Boeing Maintenance Planning Document (MPD) B32-00-00A, dated March 2009. Continental's maintenance program requires that the NWS cables are to have a detailed visual inspection[10] (DVI) at each "C" check[11] (1C, 2C, 4C, 8C) and separately, every 18 months (not embedded within these "C" checks) as compared to the

[10] Detailed Visual Inspection, sometime abbreviated (DVI) is defined as a critical visual examination of a specific area, installation, or assembly to detect damage, failure, or irregularity.
[11] A "C" check is required every 4,000 flight hours.

Boeing MPD requirement which only requires a detailed visual inspection at each "C" check (1C, 2C, 4C, 8C).

Continental's DVIs are to be performed in accordance with Continental Engineering Specification 2000S00226, rev D, titled "aircraft control cable system procedures". This document states that the control cables must be first cleaned and then displaced through their full range of travel for inspection. Control cable inspection criteria include: fray or wear, kinks, broken wires, corrosion, slackness, proper routing and freedom of movement. The specification document also states that CRES cables shall not be lubricated.

The NWS cables are replaced when the NLG is replaced (life limited - 21,000 cycles, or 10 years), or on an "On-Condition[12]" basis. The On-condition inspection criterion is specified in the Boeing 737 Aircraft Maintenance Manual, Section 20-20-31. Continental Airlines utilizes two categories of parts, "Rotables" and "Expendables". "Rotables" are tracked by serial/part numbers when received by the company, when installed in an aircraft, removed, sent to OEM, etc. "Expendables" are not tracked as their value does not warrant it. Control cables are considered "Expendable", therefore, not specifically tracked. The expendable Cable part is not tracked, but all associated work processes performed on the aircraft pertaining to the Cable (installation, removal, maintenance, and/or inspection) is tracked.

D.2.2.5 Interconnect Mechanism:

The functionality and mechanical integrity of the NWS interconnect mechanism was verified by witnessing its operation when rudder pedals inputs were made. The rudder pedal NWS interconnect mechanism was accessed by cutting a 20 inch by 20 inch hole in the left side of the airplane between Body Station (BS) 227 and BS 247 and below Water Line (WL) 208. The interconnect mechanism was found intact, with no apparent damage. Rudder pedal inputs resulted in movement of the steering cable quadrant in the mechanism in both the left and right directions. The rotary actuator, which engages and disengages rudder pedal inputs from the steering cable system based on air/ground state, was inspected and found to be intact with no apparent damage. The rotary actuator index mark on the pulley was found rotated approximately 100 degrees from the index mark on the housing, indicating GROUND mode (in the AIR mode, the index marks would be aligned).

[12] On-Condition maintenance is applicable to components which can be periodically inspected, checked, or tested on wing to appropriate standards (wear, torque, rate of flow, etc.) which predict continued operation until the next scheduled check.

D.3 Brake System:

Investigation of the brakes was conducted at the accident site and at the Continental Maintenance hangar.

D.3.1 Description:

Each main wheel has a brake unit bolted to a flange on the axle. The brakes are multiple disc type, with stationary carrier and divided lining discs, and segmented rotating brake discs. Each brake includes six pistons, which actuate the brakes when hydraulic pressure is applied. The brakes also include combination return springs and automatic adjusters. The automatic adjusters compensate for brake wear. Each brake metering valve directs system hydraulic pressure up to 3000 psi to the brakes of the main gear it serves.

D.3.2 Investigative Findings:

The wheel and brake positions are identified as 1, 2, 3, and 4 defined as left-outboard position across the airplane to the right-outboard position. Details of each brake follow:

1. **Brake Number 1:**

   a. Wear Indicator Pin:
   The wear indicator pin measurement was found to be 1.0 inch. Wear pin measurement is normally performed with hydraulic pressure applied to the brake which pushes all of the brake rotors and stators together to obtain an accurate wear pin measurement. On 04 January 2009, the wheel/tire assembly was removed thus allowing the rotors and stators to be manually pushed together in a way that more accurately represents a pressurized brake.

   b. Brake Manifold/Pistons:
   The brake manifold/pistons had no obvious defects or any sign of hydraulic leakage. The brake lines at the manifold had no signs of leakage, however the brake hoses and hydraulic tubing on the shock struts were found damaged in several areas.

   c. Wheel Speed Transducer:
   The aero fairing (hubcap) was removed exposing the wheel speed transducer hubcap. No obvious damage to the wheel speed transducer was observed. However, while subsequently manipulating the inner cylinder to remove the wheel/tire assemblies, the wheel speed transducer drive coupling contacted the ground and was bent. The antiskid electrical wiring was in the conduit and present.

d. Brake's Heat Sink Elements:
   On 04 January 2009 when the wheel/tire assembly was removed, the brake's heat sink elements were found to have no signs of damage and the rotors spun freely using hand pressure.

2. Brake Number 2:

a. Wear Indicator Pin:
   The wear pin was measured and found to be 0.875 inches.

b. Brake Manifold/Pistons:
   Brake manifold/pistons had no obvious defects or any sign of hydraulic leakage. The brake lines at the manifold had no signs of leakage, however the brake hoses and hydraulic tubing on the shock struts were found damaged in several areas.

c. Wheel Speed Transducer:
   Wheel speed transducer hubcap was present and found lock wired in-place. The antiskid electrical wiring was in the conduit and present.

d. Brake's Heat Sink Elements:
   On 04 January 2009 when the wheel/tire assembly was removed, the brake's heat sink elements were found to have no signs of damage and the rotors spun freely using hand pressure.

3. Brake Number 3:

a. Wear Indicator Pin:
   The wear indicator pin was measured and found to be 1.0 inch.

b. Brake Manifold/Pistons:
   Brake lines at the manifold had no signs of leakage, however the brake hoses and hydraulic tubing on the shock struts were found damaged in several areas.

c. Wheel Speed Transducer:
   Wheel speed transducer hubcap was present and lock wired. The hubcap was removed and no obvious damage was found on the wheel speed transducer. The antiskid electrical wiring was in the conduit and present.

d. Brake's Heat Sink Elements:
   On 04 January 2009 when the wheel/tire assembly was removed, the brake's heat sink elements were found to have no signs of damage and the rotors spun freely using hand pressure.

4. **Brake Number 4:**

   a. Wear Indicator Pin:
      The wear indicator pin was measured and found to be 0.5 inch.

   b. Brake manifold/pistons:
      Brake lines at the manifold had no signs of leakage, however the brake hoses and hydraulic tubing on the shock struts were found damaged in several areas.

   c. Wheel Speed Transducer::
      The aero fairing (hubcap) was removed exposing the wheel speed transducer hubcap. No obvious damage to the wheel speed transducer was observed. The antiskid electrical wiring was in the conduit and present.

   d. Brake's heat sink elements:
      On 04 January 2009 when the wheel/tire assembly was removed, the brake's heat sink elements were found to have no signs of damage and the rotors spun freely using hand pressure.

D.3.2.1 Summary:

A summary of the brake part numbers, serial numbers, and wear pin measurements (taken on 04 January 2009) is provided in the following table:

Brake Position	Brake Part Number	Brake Serial Number	Wear Pin Measurement
1	2606672-4 (Honeywell)	6763	1.0"
2	2606672-4 (Honeywell)	5196	0.875"
3	2606672-4 (Honeywell)	6829	1.0"
4	2606672-4 (Honeywell)	4923	0.5"

D.3.3 Brake Control System:

D.3.3.1 Brake System Cables:

On January 4, 2009, the System's group checked for continuity of the brake control system cables. Five of the eight brake control cables were found broken and they showed characteristics consistent with tensile overload. The broken cables are identified in the following table along with the approximate location of the break:

Brake Pedals	Cable Identification	Approximate Break Location
Captain's side	LGBA1	Seat Row #2
Captain's side	LGBA2	Seat Row #2
Captain's side	LGBB1	Seat Row #2
Captain's side	LGBB2	Seat Row #2
First Officer's side	LGBA2	Six feet aft of brake pedal quadrant (which is above nose gear wheel well)

D.3.3.2 Brake Metering Valves:

The brake metering valves and linkages, located on the aft wall of the main gear wheel well, were examined and no damage or anomalies were noted. The “NORMAL” and “ALTERNATE” brake-metering valve input slides were moved by hand and were found to move smoothly and easily. The “ALTERNATE” brake system metering valves were verified to have positive clearances at the rigging gap but the gaps were not measured. The “NORMAL” system brake metering valve rigging gaps were checked and were found to be within the allowable range of 0.002-0.023 inches:
Left metering valve: 0.020 inches.
Right metering valve: 0.006-0.007 inches.

As directed by the System’s group, a Continental maintenance technician removed the left and right “NORMAL” brake metering valves from the airplane on January 4, 2009. The valves were placed in a shipping container and brought (via NTSB) to the Nabtesco Aerospace, Inc. for examination.

Investigation of the “NORMAL” brake metering valves was conducted on February 24, 2009 at the Nabtesco Aerospace, Inc. (formerly Teijin Seiki America) facilities in Redmond, Washington. The investigation of the brake metering valves was conducted under the supervision of the NTSB and witnessed by representatives from Federal Aviation Administration, Boeing, ALPA, Continental Airlines, and Nabtesco Aerospace.

D.3.3.2.1 **Right Metering Valve Examination:**

The valve was manufactured by Teijin Seiki and was identified as part number (P/N) 1316200-3, and serial number (S/N) 6101. According to the data plate found attached to the unit, the valve was manufactured on 1-19-1994.

A visual examination revealed that the original safety wire remained attached to the unit; with its metal seal intact having a “TS” stamp indicating Teijin Seiki. Nabtesco Aerospace has no previous repair records for valve S/N 6101. The exterior surface of the “as received” valve was found coated with oil and dirt residue.

The valve was functionally tested per the “testing and troubleshooting section” of the Teijin Seiki component maintenance manual “32-41-01”, dated December 25, 1989. With the exception of internal leakage tests, the valve assembly passed all of the functional tests. The following describes the discrepancies noted during the leakage tests:

a. Test 3.C.(1) - Internal leakage test:
   1. Test (a). With the slide fully extended, the fluid leakage rate was 62 drops within 1 minute. The test procedure states that the leakage rate shall be 40 drops maximum during the first minute. According to Nabtesco Aerospace representatives, this discrepancy would result in increased internal leakage, but would not affect normal brake operation.

2. Test (b). With the slide fully extended, the fluid leakage rate was 57 drops per minute within 5 minutes. The test procedure states that the maximum leakage rate should be 20 drops per minute or less within 15 minutes. According to Nabtesco Aerospace representatives, this discrepancy would result in increased internal leakage, but would not affect normal brake operation.

b. Test 3.C.(2) - Internal leakage test:
   1. Test (b). With the slide fully depressed, the fluid leakage rate was 13 drops per minute within 5 minutes. The test procedure states that the maximum leakage rate should be 10 drops per minute or less within 15 minutes. According to Nabtesco Aerospace representatives, this discrepancy would result in increased internal leakage, but would not affect normal brake operation.

D.3.3.2.2 **Left Metering Valve Examination:**

The valve was manufactured by Teijin Seiki and was identified as P/N 1316200-3, and S/N 6122. According to the data plate found attached to the unit, the valve was manufactured on 1-19-1994.

A visual examination revealed that the original safety wire remained attached to the unit; with its metal seal intact having a “TS” stamp indicating Teijin Seiki. Nabtesco Aerospace has no previous repair records for valve S/N 6122. The exterior surface of the “as received” valve was found coated with oil and dirt residue.

The valve was functionally tested per the “testing and troubleshooting section” of the Teijin Seiki component maintenance manual “32-41-01”, dated December 25, 1989. With the exception of an operation and internal leakage tests, the valve assembly passed all of the functional tests. The following describes the discrepancies noted during the leakage tests:

a. Test 3.B.(6)(c) - Operating test:
   With the slide against the stop, the slide has moved 0.30 inches from the fully extended position. The limit was 0.25 +/- .03. The result would be a slightly higher brake pedal angle to bottom the brake-metering valve. This discrepancy would not affect normal brake operation.

b. Test 3.C.(1) - Internal leakage test:
   1. Test (a). With the slide fully extended, the fluid leakage rate was 45 drops within 1 minute. The test procedure states that the leakage rate shall be 40 drops maximum during the first minute. This discrepancy would result in increased internal leakage, but would not affect normal brake operation.
   2. Test (b). With the slide fully extended, the fluid leakage rate was 42 drops per minute within 15 minutes. The test procedure states that the maximum leakage rate should be 20 drops per minute or less within 15 minutes. This discrepancy would result in increased internal leakage, but would not affect normal brake operation.

D.3.3.3 Brake Accumulator:

The brake accumulator is mounted on the lower right side of the aft wall of the main gear wheel well. The brake accumulator was intact; however the gas-side tubing and charging station for the accumulator were found torn from the aft wall. The gas-side tubing was broken and kinked in several areas thus the brake accumulator was found de-pressurized.

D.3.3.4 Other Brake System Components:

The remainder of the brake system components in the main gear wheel well had no observed damage.

D.4 Main Gear Tire and Wheel Assemblies:

D.4.1 Description:

Each main gear has two tire and wheel assemblies designed to withstand high rolling speeds. A pressure relief valve, which will rupture at 375 to 450 psi, is provided to protect against over inflation. Heat shields consisting of thin steel plates mounted between the rotor drive keys protect the tire against excessive heat build up during braking. Four thermal fuses located on the inside wheel half will protect the tire against explosion due to excessive heat build up during abnormal braking conditions.

Airplane N18611 was equipped with four Bridgestone P/N APS01337 main gear tires, which are tubeless and are designated H40 x 14.5 -19 rated 24 or 26 ply 225 mph and four wheels, manufactured by Honeywell, identified with part number 2606671-2.

The Continental 737-500 Aircraft maintenance manual (AMM) section 12-15-57, page 301A, (Main landing Gear Tires) shows the following tire inflation pressure requirements:

1. Nominal 190-210 PSI
2. Re-inflation 181 – 189 PSI
3. Deferred Inspection 171 – 180 PSI
4. Remove tire from service 140 – 170 PSI
5. Remove adjacent tire 130 – 139 PSI

D.4.2 On-Scene Findings:

On December 23, 2008, the systems group performed a visual examination of the four landing gear tires. This examination did not reveal any significant pre-impact anomalies. Each of the tires exhibited normal wear such as chevrons, minor nicks, and scrapes.

1. Left Gear Outboard Wheel and Tire (#1):
   The tire, identified with serial number 308NH027, had a measured tire pressure of 184 PSI[13], and had a R-0 retread level. A measurement of the "as found" tire tread depth revealed that all four tire grooves were 7/16 inch to ½ inch in depth. The outboard side of the tire had three cuts, one six inches long, the other two cuts were two inches long along the radial length. These cuts did not penetrate the full carcass depth. There was no evidence of dry or wet braking flat spots on the tire tread. As requested by the NTSB, a Continental mechanic removed the valve stem and deflated the tire.

   The wheel identified with serial number B10400, was unremarkable.

2. Left Gear Inboard Wheel and Tire (#2):

   The tire, identified with serial number 907NH254, had a measured tire pressure of 189 PSI[14], and had a R-2 retread level. A measurement of the "as found" tire tread depth revealed that all four tire grooves were 3/16 inch in depth. There was no evidence of dry or wet braking flat spots on the tire tread. As requested by the NTSB, a Continental mechanic removed the valve stem and deflated the tire.

   The wheel identified with serial number B10287, was unremarkable.

3. Right Gear Inboard Wheel and Tire (#3):

   The tire, identified with serial number xx3NH022 (first two characters illegible), had a measured tire pressure of 0 PSI[15], and had a R-5 retread level. A measurement of the "as found" tire tread depth revealed that all six tire grooves were 1/8 inch in depth. The tire had several large tears in the carcass, extending from one sidewall across the crown to the other sidewall. Most of the tears were on the diagonals that make up the tire "bias" ply structure. This is consistent with a pressure burst where the fabric tends to tear on a bias angle. At the apex of the two intersecting tears in the crown, there are indications of a deep cut that went through all layers of the carcass and inner liner. The ends of the fabric in this area show sharp smooth edges consistent with a cut. Also, the cut area is progressing nearly transverse across the tread surface, further indicating a cut as it is not following the bias angles. There were no indications of heat or over deflection thermal damage on the tire. There was no evidence of dry or wet braking flat spots on the tire tread.

   The wheel identified with serial number B2940, was unremarkable.

[13] The tire pressure gauge is in cert until 3/09.
[14] The tire pressure gauge is in cert until 3/09.
[15] The tire pressure gauge is in cert until 3/09.

4. Right Gear Outboard Wheel and Tire (#4):
The tires serial number was illegible, had a measured tire pressure of 192 PSI[16], and had a R-1 retread level. A measurement of the "as found" tire tread depth revealed that all six tire grooves were 3/16 inch in depth. There was no evidence of dry or wet braking flat spots on the tire tread. As requested by the NTSB, a Continental mechanic removed the valve stem and deflated the tire.

The wheel identified with serial number H0135, was unremarkable.

D.5 Flight Control Systems:

D.5.1 Lateral Control System:

The aileron and spoiler control system provides airplane lateral control. There are two flight spoilers and one aileron and balance tab on each wing. Rotation of either control wheel results in movement of the ailerons and movement of the flight spoilers. Two independent hydraulic power control units power the ailerons.

D.5.1.1 Lateral Control System Continuity:

Continuity of the lateral control system could not be demonstrated due to broken cables in the fuselage where the nose gear impacted the airplane.

D.5.1.1.1 **Forward Section (including flight deck):**

As initially found, both control wheels remained positioned to the right of center and aligned with 5 units of trim[17]. An attempt to rotate the captain's control wheel revealed that it would not turn; it remained in place. An attempt to rotate the first officer's wheel revealed that it could be rotated to its full left (CCW) stop and to its full right (CW) stop, however, it felt like the transfer mechanism was being overridden. After both aileron power control units (PCUs) and both autopilot PCUs were removed from the airplane, the System's group re-inspected the movement of both control wheels. Both the captain's and the first officer's control wheels could be rotated over 90 degrees in the CW and CCW directions.

Inspection of the lateral control system components located below the flight deck revealed that the aileron cables (ACBA and ACBB) remained intact and connected between the captain's aileron control drum and to the transfer mechanism located at the base of the First Officer's control column.

The captain's aileron control cables (AA and AB) remained connected to the captain's aileron control drum. The first officer's spoiler control cables (AA and AB) remained connected to the transfer mechanism. The captain's cable AA was found broken

[16] The tire pressure gauge is in cert until 3/09.
[17] The aileron trim indicator is located at the top of each control column.

approximately 14.5 feet aft of the forward turnbuckle. The break appeared to be consistent with tensile overload. The first officer's cable AB was found broken approximately 30-inches aft of the forward turnbuckle. The break appeared to be consistent with tensile overload.

D.5.1.1.2 **Aileron Power Control Units (PCUs):**

Airplane N18611 was equipped with two Parker Aerospace part number (P/N) 65-44761-21 aileron PCUs. The serial number (S/N) on the lower aileron PCU was 6148A and the S/N on the upper aileron PCU was 13109A. Both aileron PCUs and their associated components (input linkages, external summing levers, input rods) were inspected and found intact and mechanically connected to their respective attachment points via attachment hardware. Visual inspection revealed that the lower (A system) aileron PCU had a deformed (bent) input rod. No structural damage was found on the upper (B system) aileron PCU or linkage. On January 4, 2009, the System's group removed both aileron PCUs from the airplane for subsequent examination.

Investigation of the aileron PCUs was conducted at the Parker Aerospace Alton facility located in Irvine, California during the period of February 11-12, 2009. This investigation was conducted under the supervision of the NTSB and witnessed by representatives from Parker Aerospace, Federal Aviation Administration, Continental Airlines, Boeing and ALPA.

The upper PCU was functionally tested per the testing section of part number 65-44761 overhaul manual 27-09-21 dated 11/1/2008. With the exception of an input friction test and a manual threshold test, the PCU passed all of the functional tests. The following describes the discrepancies noted:

1. <u>Input Friction (Exceeded allowable input force):</u>
   Test procedure 12.F states: "*measure the force (at input crank bearing centerline) required to displace input arm from neutral position to a distance where the spring loaded sleeve of the servo valve is engaged. The input force in either direction shall not exceed 8 ounces*." The recorded force in the extension direction was 11 ounces.

2. <u>Manual Threshold and Hysteresis:</u>
   Test procedure 12.S Step 6 states: "*gradually move the input arm in either direction. When movement of the actuator occurs, hold input lever position. Make sure input arm movement does not continue past the point where initial actuator movement begins. Measure and note input arm travel in inches. This is the manual threshold, and should not exceed 0.006 inch*". The recorded manual threshold was 0.007 inches.

A hydraulic fluid sample was taken from the upper aileron PCU (B System). The sample was analyzed and found to be outside allowable limits for particulate contamination per Parker Repair Station Instruction 097, Rev E, which references SAE Aerospace Standard 4059, Rev E.

The lower aileron PCU was received at Parker with a bent input rod (as found at the accident site) attached to the input arm. This input rod and the boot protecting the input lever were removed to facilitate testing. A hydraulic fluid sample was taken from the unit, analyzed, and found to be outside allowable limits for particulate contamination per Parker Repair Station Instruction 097, Rev E, which references SAE Aerospace Standard 4059, Rev E.

The lower PCU was functionally tested per the testing section of part number 65-44761 overhaul manual 27-09-21 dated 11/1/2008. With the exception of an input arm friction test, cylinder friction test and a actuator velocity test, the PCU passed all of the functional tests. The following describes the discrepancies noted:

1. <u>Input Arm position: (Failed - travel was less than required in both directions)</u>
   Test procedure 12.J Step 6 states: "*measure and record input arm travel (each way) from the valve null position to the manifold stop. Check that measured travel is 0.145-0 - 0.160 inch in each direction*". The measured values were 0.141 inches in the extend direction and 0.138 inches in the retract direction

2. <u>Cylinder Friction (Extension failed, mid-range and retract passed)</u>
   While performing the cylinder friction test with the piston in a fully retracted position, the piston rod could not be moved when applying approximately 50 lbs using a push-pull scale. In an effort to free the piston rod, the input lever was positioned to full extend and the hydraulic pressure was slowly increased until the piston rod began to extend at approximately 1000 psi. The piston could be extended and retracted manually from full extension to a point at which approximately 1.2 inch of chrome was visible between the rod end and the gland flange.

3. <u>Actuator velocity and snubbing (Failed - did not trace curve):</u>
   Test procedure 12.Q requires manually cycling the actuator piston and then recording the piston velocity versus piston position on an X-Y plotter and then ensuring that the tracing complies within specified limits. During hydraulic cycling of the PCU, it was apparent that the piston rod was bent.

The test for external leakage was not performed because of the possibility of the bent piston rod inducing further damage to internal seals. The unit was disassembled to determine if the seals were damaged and to remove the piston rod for inspection. The piston rod was measured with both ends placed in v-blocks and using a dial gage on the piston. The rod was found to be bowed 0.046 inch. The piston rod had three visible parallel partial circumferential witness marks. The first witness mark measured 1.65 inch from the rod end and the last witness mark measured approximately 2 inch from the rod end. Wear was noted to the o-ring underneath the cap seal inside the tail stock end gland. All other seals and the glands in the piston housing did not show signs of damage. The rod end bearing was noted as being tight. The PCU inlet filter was removed and visually examined. No visible contamination was present.

D.5.1.1.3 **Feel and Centering Unit and Trim Actuator:**

The aileron feel and centering unit receives the aileron inputs, controls the PCUs and provides a feel force to the pilot. It also moves the control wheel to a neutral position when there is no force input. The aileron trim actuator is connected to the feel and centering unit

Visual inspection of the aileron feel and centering unit and the aileron trim actuator found them in-place and connected to structure via attachment hardware. The aileron feel and centering mechanism contains two centering springs, which were both found in-place, unbroken and remained attached to the unit. The centering springs were removed to facilitate the removal of the trim actuator. Using a tape measure, the distance from the bolt to bolt on the trim actuator was measured and found to be 7.5 inches. The aileron trim actuator was removed and is identified as follows: Boeing P/N 10-62026-1, S/N 940302

At the request of the NTSB, Continental Airlines performed a lateral trim system test on one of their 737-500 airplanes to determine the amount of lateral trim that was commanded on the accident aircraft. The test was conducted on January 15, 2009 on airplane N14605. Continental set, via the aileron trim switches, the lateral trim actuator to 7.5-inches, bolt-to-bolt centers. The positioning of the lateral trim actuator resulted in the captain's control wheel rotating to a position of about 0.2 units of right lateral trim and the first officer's control wheel rotating to a position of about 0.3 units of right lateral trim.

D.5.1.1.4 **Aileron (Left) and Control Mechanisms:**

Continuity from the left aileron to the aileron quadrant (wheel well) was verified by moving the aileron surface by hand from stop to stop. The following aileron system control components were inspected and found to be unremarkable (no jams or disconnects):

- Cable tension device.
- Aileron bellcrank.
- Connecting rod between aileron bellcrank and aileron surface.
- Aileron cables were under tension and attached to the aileron bellcrank.
- All pulleys between bellcrank and the aileron control quadrant.
- Cables were in the center of all fairleads.

Inspection revealed that the left aileron remained attached to the wing by its four hinges, which were found intact and unremarkable. The aileron had a 1x6 inch crescent shape puncture on its lower surface between the number 2 and 3 aileron hinges. The aileron tab remained attached to the aileron by all four hinges and the tab control rods remained connected to their respective attach points. Using hand pressure, the left aileron could be moved up and down plus/minus 2-inches. When moved, the tab on the aileron moved appropriately in a balanced function.

D.5.1.1.5 **Aileron (Right) and Control Mechanism:**

Continuity from the right aileron to the aileron quadrant (wheel well) was verified by moving the aileron surface up and down approximately two inches (full travel was not possible because of supporting equipment under the wing). The following components were inspected and, with the exception of thermal damage on the inboard portion of the right wing, were found to be unremarkable (no jams or disconnects):

- Cable tension device
- Aileron bellcrank
- Connecting rod between aileron bellcrank and aileron surface
- Aileron cables were intact
- Aileron cables were attached to the aileron bellcrank
- All pulleys between bellcrank and the aileron control quadrant were inspected
- Thermal damage to several pulleys, which affected cable routing and resulted in no cable tension

Inspection revealed that the right aileron remained attached to the wing by its four hinges, which were found intact and unremarkable. The aileron tab remained attached to the aileron by all four hinges and the tab control rods remained connected to their respective attach points. When moved, the tab on the aileron moved appropriately in a balanced function.

D.5.1.1.6 **Spoiler Control System:**

The control inputs to the spoiler system are from the combined lateral control system based on captains and the first officer's control inputs. The input components (aileron spring cartridge, ratio changer input rod, spoiler control quadrant), were inspected and found to be intact and connected to their respective attach points. System continuity was verified.

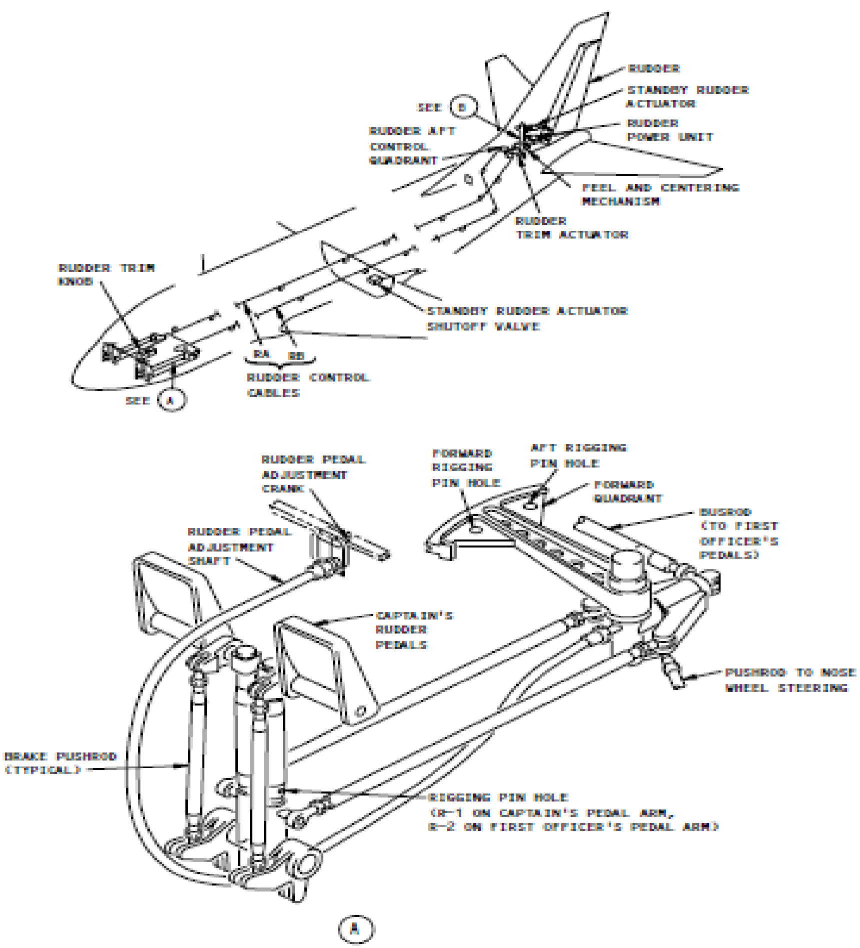
Inspection found all flight and ground spoilers in the down position faired with the wing. All flight spoiler control cables, between spoiler mixer/ratio changer and spoiler input quadrants at each flight spoiler PCU, were inspected and found intact and attached. Flight spoiler actuators remained connected to their respective spoiler panels and all hydraulic lines and input mechanisms were found connected and intact.

Ground spoiler actuators remained connected to their respective spoiler panels and all hydraulic lines were found connected and intact.

D.5.2 Rudder Control System:

Airplane N18611 was equipped with a single conventional rudder surface (without tab) (Figure 7) that provides aerodynamic directional control of the aircraft. The rudder is pedal operated by the captain or the first officer. Pedal movement rotates the forward quadrants, which is transmitted in a closed-loop system through a single cable system to the aft quadrant in the aft portion of the vertical stabilizer. Rotation of the aft quadrant moves a control rod connected to a torque tube. Rotation of the torque tube moves input rods attached to the rudder main power control unit (PCU) linkages, the input rod to the rudder standby PCU, and the feel and centering unit. There are two separate hydraulic systems powering the main PCU. Rudder control backup is provided by a standby PCU, which is powered by a third (standby) hydraulic system. Any one of the three hydraulic systems will provide effective rudder control. The rudder pedals at each pilot position are located on either side of the control column and are protected within a housing located on the floor. Rudder trim is accomplished by operating a trim control knob, which repositions the rudder-centering unit.

Figure 7 Schematic of the Rudder Control System

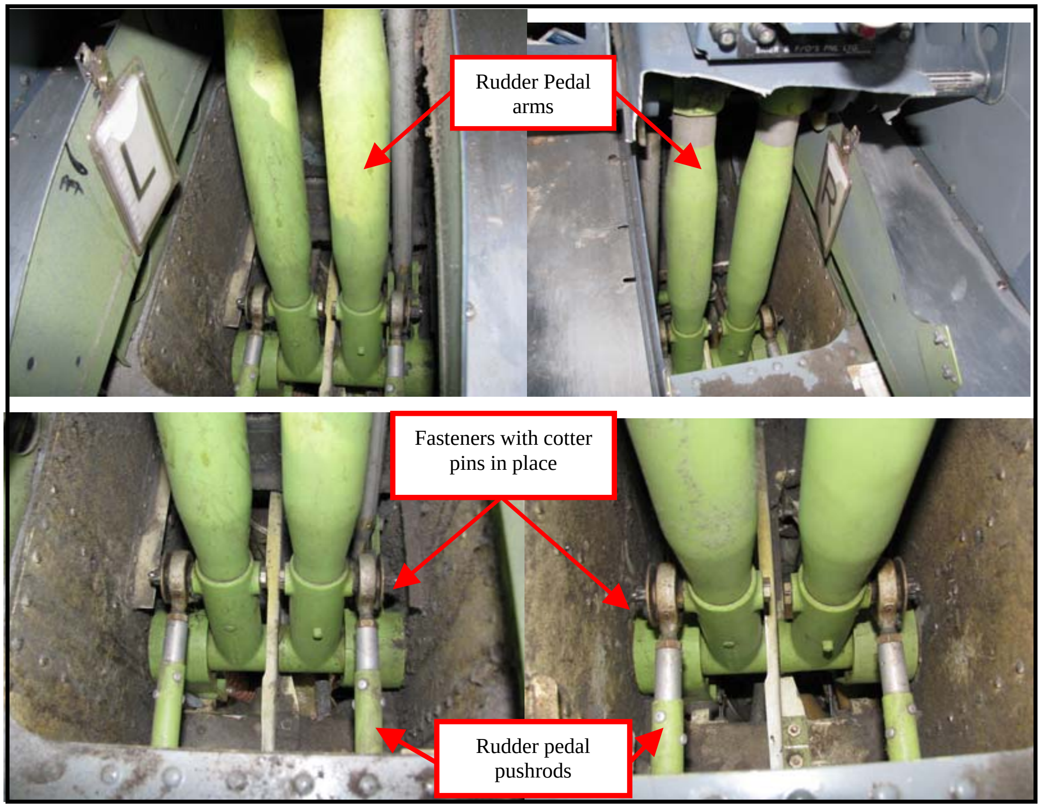


D.5.2.1 Rudder Pedals:

The rudder pedal linkages are affected by Airworthiness Directive (AD) 2001-22-13 and Boeing Service Bulletin 737-27A1214, Revision 1, dated July 1, 1999. This AD was applicable to airplane N18611 and requires replacing the rudder pedal pushrod fasteners for both the captain's and first officer's pedal assemblies with new, improved fasteners. A review of maintenance data provided by Continental Airlines revealed that AD 2001-22-13 was complied with on June 8, 2000.

Visual inspection of the captain's and the first officer's rudder pedal assemblies revealed all attachment hardware was in place and intact as required by AD 2001-22-13. There was no evidence of binding or loose hardware (
Figure **8**).

Figure 8 Photograph's of rudder pedal control



D.5.2.2 Rudder Control System Continuity:

Rudder control system continuity was verified from both the captain's and the first officer's rudder pedals to the aft rudder control quadrant. When the captain's and the first officer's left and right rudder pedals were displaced approximately two to three inches, the opposite set of pedals deflected and the aft rudder quadrant rotated in the respective direction. When the pedals were released from either a left or right input, the pedals did not return to center.

D.5.2.3 Airworthiness Directives:

All applicable airworthiness directives involving the rudder control system were found to be in compliance.

D.5.2.4 Rudder Position:

The "as initially found" position of the rudder was measured prior to moving the rudder pedals or the rudder surface. The rudder was found deflected about 6 5/16 inches left of the rudder index plate[18]. The rudder was then manually cycled, from its "as initially found" position, left and right of the index plate to determine the amount of available surface deflection. The results of the rudder deflection test indicate that the rudder could be deflected 22 inches left and 21 15/16-inches right. The 737-500 aircraft maintenance manual states, the rudder deflection should be in the range of 21.82 $\pm$ 0.80 inches.

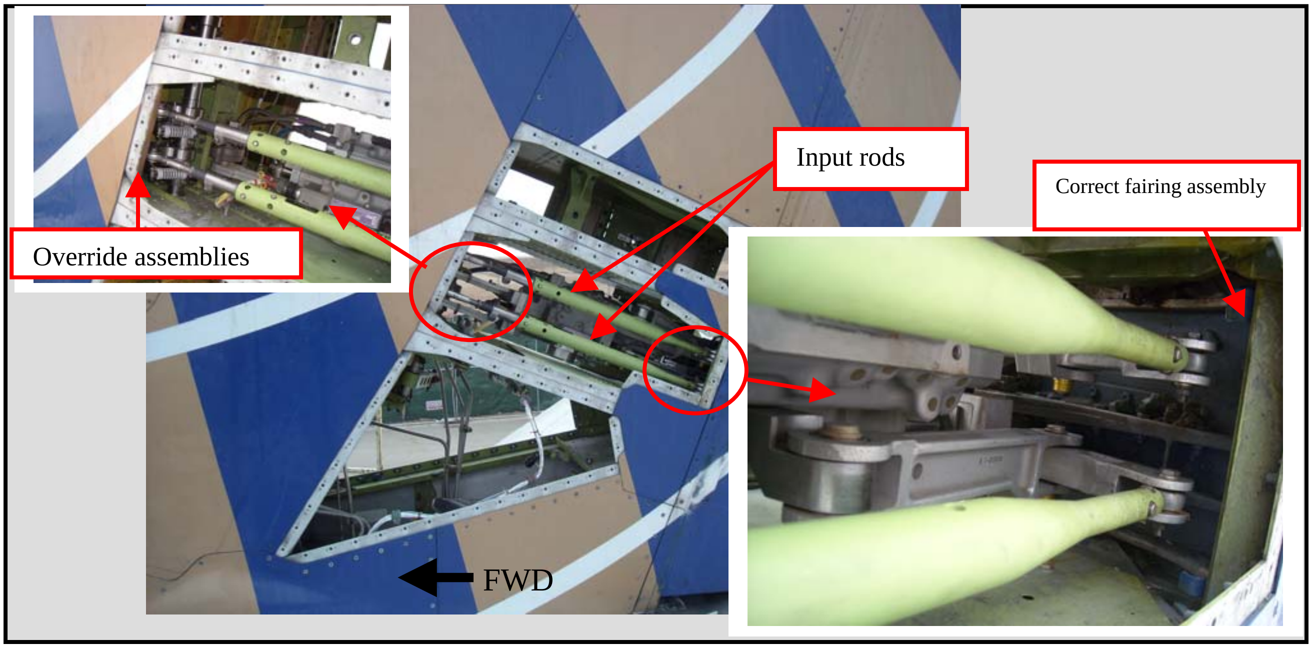
D.5.2.5 Rudder Main Power Control Unit (PCU):

Airplane N18611 was equipped with a rudder main PCU that was manufactured by Parker Aerospace. The main PCU is a dual tandem hydro-mechanical servo, powered by hydraulic systems A and B. The PCU is internally separated into an A and B-side, providing an output force equivalent to two PCUs. Each half of the PCU consists of a single slide main control valve, balanced piston, bypass valve, inlet filter, inlet check valve, solenoid and load limiter relief valve, all contained in a separate manifold for each system. The A and B manifolds are bolted together to make up a single PCU package driving a single output piston. The PCU is controlled by two independent input linkages that drive the two independent main control valves. Each input rod is jam protected by externally mounted breakout devices. The PCU positions the rudder surface in response to mechanical rudder pedal commands through the two separate and independent control paths. The PCU also positions the rudder surface in response to electrical yaw damper commands. Electrical operation is possible only when the "B" hydraulic system is active. The PCU also contains a delta-delta pressure switch, which compares the "A" system and "B" system actuators. If the delta-delta pressure exceeds a predetermined threshold a force fight condition is recognized and the Rudder Standby hydraulic system is activated The rudder main PCU can position the rudder surface to $\pm$ 26° deflection from the neutral trim point under a no load aerodynamic condition.

An on-scene visual inspection of the rudder main PCU (Figure 9) and its associated components (input linkages, external summing linkage, input rods, and override assemblies) revealed that all components were intact, undamaged and remained connected to their respective parts via attachment hardware.

[18] The measurement was taken from the center of the index plate on the tail cone to the lower trailing edge center of the rudder surface.

Figure 9- View of the left side of the vertical stabilizer showing the location of the rudder PCUs



As directed by the Systems group, a Continental maintenance technician removed the rudder main PCU from the airplane on January 4, 2009. The rudder main PCU was identified with P/N 419200-1003 and S/N 419200-00656, Revision E. The PCU was shipped to the Parker Aerospace Alton facility for examination. Upon arrival at Parker Aerospace, the sealed shipping container was placed into a locked storage room.

Examination of the rudder main PCU was conducted at the Parker Aerospace Alton facility located in Irvine, California during the period of February 11-12, 2009. The investigation of the PCU was conducted under the supervision of the NTSB and witnessed by representatives from Parker Aerospace, Federal Aviation Administration, Continental Airlines, Boeing and ALPA. Testing and visual examination found the rudder PCU intact and fully functional. The rudder PCU passed all functional tests per the testing section of the part number 419200 rudder PCU component maintenance manual 27-21-10 dated 1/19/2007.

D.5.2.6 Rudder Fairing:

Boeing Alert Service Bulletin 737-27A1286, dated November 26, 2008 gives instructions to inspect the rudder assembly of each airplane with the Rudder System Enhancement Program (RSEP) installed, to make sure that the correct rudder fairings are installed.

A review of maintenance data provided by Continental Airlines revealed that Boeing Alert Service Bulletin 737-27A1286 was complied with on March 15, 2008. Additionally, Figure 25 of this report shows that this aircraft has the correct rudder fairing installed.

D.5.2.7 Rudder Standby Power Control Unit (PCU):

The FAA released Airworthiness Directive (AD) 97-26-01, which mandated the incorporation of a Dowty Aerospace Service Bulletin (1150-27-04, dated December 5, 1996). This Service Bulletin involves the replacement of an existing input bearing within the rudder standby PCU with a new, improved bearing.

Airplane N18611 had a rudder standby PCU installed at the time of the accident that was affected by this AD. Continental Airlines provided maintenance records showing that this PCU was made compliant with this AD on March 28, 2000.

A visual inspection of the rudder standby PCU and its associated components (input rods, and override assemblies) was conducted on-scene. All components were intact, undamaged and remained connected to their respective parts via attachment hardware. Inspection also confirmed that the rudder standby PCU was in compliance with AD 97-26-01.

As directed by the Systems group, a Continental maintenance technician removed the rudder standby PCU from the airplane on January 4, 2009. Visual inspection revealed the standby PCU was manufactured by Dowty Aerospace and was identified with part number 1U1150-4E and serial number 2760. The units input rod was manually cycled using hand pressure to determine if it moves freely. Testing found the input rod moved freely from stop-to-stop with no binding. The PCU was placed in a shipping container and left with the main wreckage.

D.5.2.8 Rudder Feel and Centering Unit:

The rudder feel and centering unit is attached to the aft rudder input torque tube located in the vertical fin, forward of the rudder main PCU. This unit holds the rudder at the neutral (or trimmed) position when no rudder pedal force is applied. It also provides a feedback force to the rudder pedals that increases as the rudder pedals are depressed. The pilot's rudder pedal force required for full rudder deflection is about 70 pounds; however, the rudder trim system allows the pilots to maintain a rudder deflection without having to maintain a rudder pedal force.

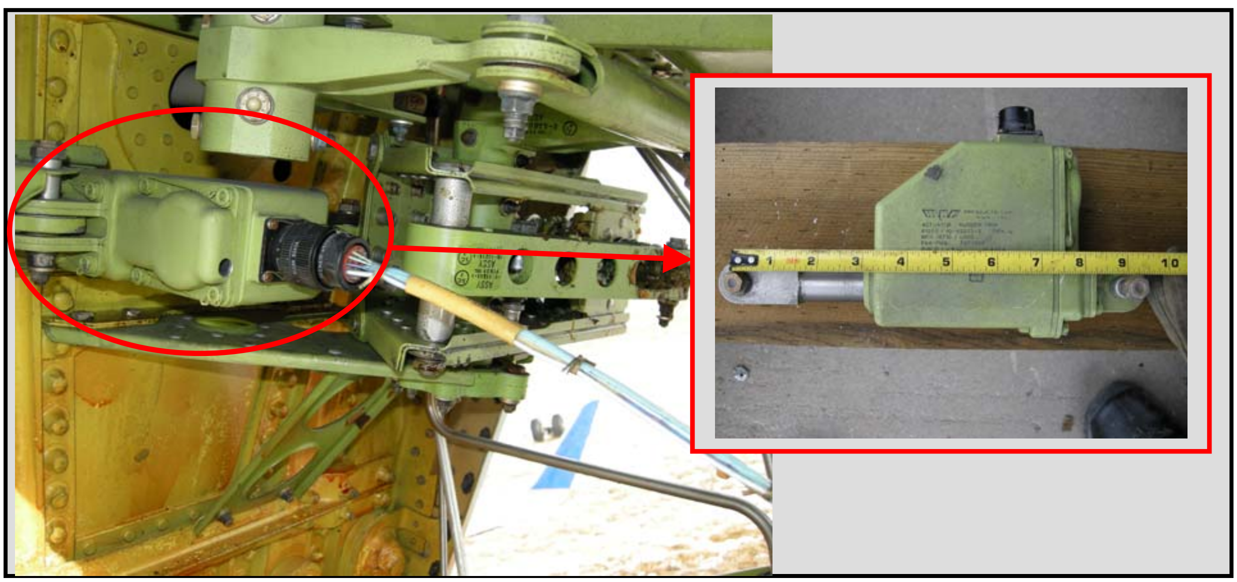
Visual inspection of the rudder feel and centering unit found it intact and connected to structure via attachment hardware. Both the unit's inner and outer springs remained intact and remained attached to the unit. Testing indicated that when either the captain's or the first officer's rudder pedals were displaced by approximately 2 to 3 inches of travel, the rudder aft quadrant, including the cam in the rudder feel and centering unit, would rotate as a result of the pedal displacement. With the pedals displaced, the rudder feel and centering unit cam rotated as commanded with the roller being displaced out of the detent. When the pedals were released, the roller returned to the centering cam detent.

D.5.2.9 Rudder Trim Actuator:

The rudder trim system allows the pilots to command a steady rudder input without maintaining foot pressure on the rudder pedals. The primary purpose for rudder trim is to compensate for the sustained large yawing moments generated by asymmetric thrust in an engine-out situation. Rudder trim is adjusted by rotating the rudder trim knob located on the pedestal in the flight deck. The rudder trim knob activates arming and control switches, which direct electrical power to the linear rudder trim actuator. The rudder trim actuator is connected to the rudder feel and centering unit and to fixed airplane structure. Extension of the rudder trim actuator results in left rudder displacement and retraction results in right displacement. Adjustment of rudder trim repositions the neutral trim point of the rudder feel and centering unit. Adjustment of rudder trim changes the relaxed position of the rudder pedals. The rudder trim indicator displays the trim actuator position. Without electrical power, the trim indicator goes off-scale and "OFF" is displayed. The electric rudder trim moves the rudder at a rate of about 0.5° per second to the desired deflection; maximum rudder trim authority is ± 16°.

Visual inspection revealed the rudder trim knob remained centered and the trim indicator displaying the "OFF" flag. Visual inspection of the rudder trim actuator revealed it was intact and remained connected to the rudder feel and centering unit and to fixed airplane structure via attachment hardware (Figure 10). Measurements of the "as found" actuator extension were taken to determine the last commanded position of rudder trim. Using a tape measure, the distance from bolt to bolt on the trim actuator was measured and found to be about 9.25 inches.

Figure 10 Photograph of the rudder trim actuator



At the request of the NTSB, Continental Airlines performed a rudder trim control system test on one of their 737-500 airplanes. The objective of the test was to determine the amount of rudder trim commanded on the accident aircraft. The test was conducted on

March 15, 2009 on airplane N16650. Continental set, via the rudder trim knob, the rudder trim actuator to exactly 9.25-inches, bolt-to-bolt centers. The positioning of the rudder trim actuator resulted in the rudder surface trailing edge being positioned 0.450-inches left of the rudder index mark. The positioning of the rudder trim actuator resulted in the rudder surface position indicator showing approximately 0.75-degrees of left rudder trim.

D.5.2.10 Yaw Damper:

The yaw damping system provides improved ride comfort, and Dutch roll damping. The enabling control, status indication, and surface monitoring functions are located within the cockpit. The yaw damper system comprises the yaw damper control switch and a yaw damper coupler, which includes a rate gyro that senses airplane motion about the yaw axis and converts the motion to an electrical signal that is sent to an electro-hydraulic servo valve (or transfer valve) connected to the rudder main PCU. The transfer valve converts the electrical signal from the yaw damper coupler to PCU motion by directing hydraulic fluid from hydraulic system B to displace the rudder left or right. The yaw damper system also includes a cockpit indicator of yaw damper activity. In the 737-500 series airplane, the yaw damper can command up to 3° of rudder surface deflection in either direction at a rate of 50° per second. Rudder movements that result from yaw damper system inputs do not backdrive the rudder pedals.

Visual inspection revealed the yaw damper switch, located on the overhead panel, was positioned to “OFF”. The yaw damper is engaged by a powered solenoid. Therefore, when the aircraft loses power, the solenoid will relax and the yaw damper switch automatically transitions to the “OFF” position.

Visual inspection revealed the yaw damper coupler had sustained impact damaged resulting from the nose wheel tires impacting the E&E bay. The bottom of the unit had been punctured and the Master Interconnect Board was found severed. On January 4, 2009, the Systems group removed the yaw damper coupler from the airplane for examination.

Examination of the yaw damper coupler was conducted at the Honeywell Aerospace facility located in Coon Rapids, Minnesota during the period of January 28-29, 2009. This investigation was conducted under the supervision of the NTSB and witnessed by representatives from Honeywell Aerospace, and Continental Airlines.

The coupler was manufactured by Honeywell Aerospace and identified as a Boeing part number (P/N) 10-62253-2, MFR P/N 4084042-911 and serial number 00073467. The MOD status of the unit was marked as Modification 1, 2, 3. Honeywell verified this was the most recent approved configuration. Visual inspection revealed the casing was deformed and the bottom of the unit had been punctured severing the Master Interconnect Board. The manufacturer’s seal from the unit’s last repair was present and indicated the last repair/overhaul was at Honeywell was in September of 2002.

The damage to the Master Interconnect Board prevented the download of NVM using normal procedures. Both non-volatile memory (NVM) chips (processor board locations U13 and U113) were removed and the data was copied to new chips as well as to floppy disks. The copied chips were installed on a known good processor board and this board was installed into the manufacturers yaw damper coupler shop box. The NVM data was then downloaded per the component maintenance manual procedures. A review of the data revealed the most recent fault codes were recorded 80 power cycles previous to the most recent power cycle. The data also indicated that 203 faults had been recorded on the NVM from as far back as 6,335 power cycles.

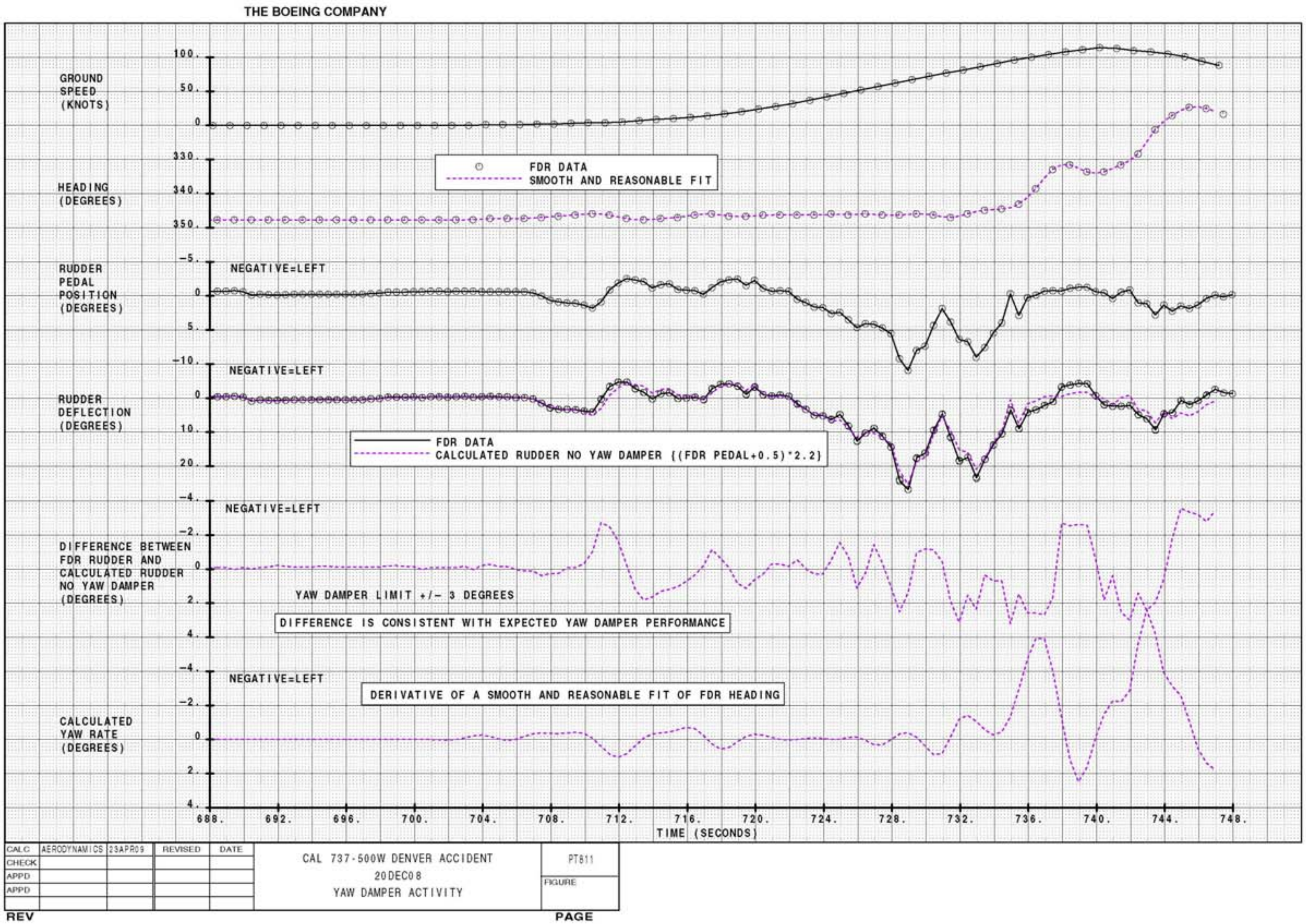
The power supply and the quartz rate sensors were removed and functionally tested on a known good yaw damper coupler shop box. These components passed the functional test indicating a properly functioning power supply and rate sensors. The four DC and two AC EMI filters were tested and passed the insulation resistance test per Honeywell's component maintenance manual.

D.5.2.11 Rudder Position Study:

The NTSB requested Boeing to perform parametric studies of the FDR data for the rudder control system to determine if the actual rudder deflections match the expected rudder deflections for the accident takeoff. Boeing used the FDR heading, rudder pedal, and rudder deflection to perform their studies and to understand the expected yaw damper performance versus the actual yaw damper performance. The yaw damper on the 737-500 is a yaw-rate yaw damper with a rudder authority limit of +/- 3.0 degrees. An optimal control curve fit was used to calculate a smooth and reasonable curve through the recorded heading data. This curve fit was then differentiated to calculate yaw rate (Figure 11).

The FDR rudder pedal data was used to calculate a rudder deflection with no-yaw-damper by using a simplified gearing relationship. First, 0.5 was added to the FDR pedal data to center it about zero. Then the pedal position was multiplied by a factor of 2.2 (pedal-to-rudder gearing) to calculate the no-yaw-damper rudder. An estimation of the yaw damper input was calculated by taking the difference between the FDR rudder deflection data and the calculated rudder. A yaw-rate yaw damper input is expected to oppose yaw rate. The characteristics of the calculated rudder difference (estimated yaw damper input) are consistent with the characteristics of the expected behavior of the yaw damper. The FDR data of the rudder positions and the rudder pedal positions were consistent with, and matched, parametric studies of the rudder system.

Figure 11 Boeing Rudder System Analysis



D.5.3 Pitch Control System

D.5.3.1 Elevator:

Testing[19], on January 3, 2009, revealed that both the left and right elevators could be moved 14 inches Trailing Edge Up (TEU) and 11 inches Trailing Edge Down (TED). The amount of available elevator travel (displacement) was determined by displacing each elevator up and down by hand. Once in its full TEU or TED position, a measurement, from the elevator trailing edge position relative to the elevator index plate was taken. When the left elevator was moved, the right elevator moved in conjunction. Conversely, when the right elevator was moved, the left moved as well.

When the elevator was moved by hand, both elevator tabs moved in the opposite direction, as correctly functioning balanced tabs. With the elevator moved TED (11 inches), the distance from the elevator trailing edge to the tab trailing edge (measured at the inboard end of the tabs) was 1 ¾ inches on both sides.

Control system continuity from the control column to the elevators could not be confirmed due to the damage to the aircraft (fuselage and wings). An attempt (by hand pressure) to move the control column forward and aft was unsuccessful; the column could not be moved.

D.5.3.2 Stabilizer:

The stabilizer was found positioned at about four units of trim, which is consistent with the indication of four units of trim on the captain's and first officer's stabilizer trim indicators. The position of the stabilizer was obtained by measuring the stabilizer jackscrew dimension ("B" dimension) in the stabilizer compartment. The "B" dimension was 40.25 inches, which is equivalent to about four units of trim.

D.5.4 Flap/Slat Control System:

On December 23, 2008 the Systems group examined and documented the trailing edge flap control system. The flap control lever, which is located on the upper right side of the control stand, was found in the 5-unit detent. The left and right wing trailing edge flaps (inboard and outboard), slats, and Krueger flaps were found in a position consistent with the position of the flap control lever. Visual inspection found the flap drive system intact.

D.6 Flight Deck – Controls and Indicators

On December 23, 2008 the Systems group visually inspected and documented the instruments, controls and displays in the flight deck. The following are the results of that inspection:

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[19] The testing was conducted after the airplane had been cut in half by the salvage company.

Speed Brake lever	Full forward (Spoiler Down Position) within the safety blocking detent.
Flap lever	5 degrees and locked in detent.
Flaps Position Indicator	Indicated 5 degrees.
Stabilizer Trim Position Indicator	Pilot and co-pilot's indicated 4 units.
Stabilizer Trim cutoff switches	(Main electric & autopilot cutout): positioned to normal and guarded
Stabilizer Trim Override Switch	Normal position and guarded
Rudder Trim Knob	Centered
Rudder Trim indicator	"OFF" flag shown
Aileron Trim	Toggle switches centered
Captain's Speed Indicator	• Indicated 84 Knots • Internal bug at 128 Knots • External bugs at 138, 140, 165, and 220 Knots) • V_{MO} (barber pole) at 337 Kts
First Officer's Speed Indicator	• Indicated 66 Knots • Internal bug at 215 Knots • External bugs at 138, 140, 160, and 220 Knots) V_{MO} (barber pole) at 337 Kts
Auto Brake Setting	RTO
Anti-Skid	On and guarded
Landing Gear Selector Lever	Down
Hydraulic System Control Switches (P5 Panel)	A System - On B System - On
Hydraulic Brake Pressure (Accumulator)	2,150 PSI
Hydraulic System Pressure	A System pressure: 450 PSI B System pressure: 2,900 PSI
Nose Wheel Steering:	Normal but guard is broken

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