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NATIONAL TRANSPORTATION SAFETY BOARD

Washington, D.C.

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT

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NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety
Washington, D.C. 20594

December 21, 1994

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT OF INVESTIGATION

A. ACCIDENT DCA-94-MA-076

Location:	Aliquippa, Pennsylvania
Date:	September 8, 1994
Time:	1904 Eastern Daylight Time
Aircraft:	Boeing 737-300, N513AU

B. SYSTEMS GROUP

Chairman:	Greg Phillips National Transportation Safety Board Aviation Engineering Division Washington, DC
Member:	Roff Sasser National Transportation Safety Board Southeast Regional Office Atlanta, GA
Member:	Richard Babunovic Boeing Commercial Airplane Group Seattle, WA
Member:	Captain Ed Bular USAir Pittsburgh, PA
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Member:	Dale A. Hoth Federal Aviation Administration Flight Standards District Office Coraopolis, PA
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C. SUMMARY

On September 8, 1994, at 1904 Eastern Daylight time, USAir flight 427, a Boeing 737-3B7 (737-300), N513AU, crashed while maneuvering to land at Pittsburgh International Airport, Pittsburgh, Pennsylvania. The airplane was being operated on an instrument flight rules (IFR) flight plan under the provisions of Title 14, Code of Federal Regulation (CFR), Part 121, on a regularly scheduled flight from Chicago, Illinois, to Pittsburgh. The airplane was destroyed by impact forces and fire near Aliquippa, Pennsylvania. All 132 persons on board were fatally injured.

The systems group was formed at an organizational meeting held on September 9, 1994. The on-scene phase of the investigation was conducted at the accident site and in a USAir hangar at the Pittsburgh airport on September 9 through 16, 1994.

As of the date of this report, the investigation has been conducted in five phases. Each phase has consisted of the systems group members convening for aircraft systems testing and examinations of the accident aircraft's components. A detailed report of each phase follows in this report. Three appendices to this report describe the details of the main rudder power control unit (PCU) examination (Appendix 1), the NTSB metallurgist's control column fracture examination (Appendix 2), and a flight controls system description (Appendix 3).

During Phase I of the investigation, the systems group met daily at the accident site and the USAir hangar in Pittsburgh to document the condition of the airplane prior to the removal of components for additional testing. During the investigation at the accident site, the aircraft wreckage was examined and systems components were identified, photographed, and critical measurements were recorded.

At the hangar, aircraft systems components were examined and separated by system function. The main rudder power control unit was removed from the vertical stabilizer by cutting away deformed structure after the position of the input lever arm was secured with shims. The removal was performed in the presence of and at the direction of the systems group and Parker representatives Wally Walz and Steve Weik. The standby rudder actuator was also removed from the vertical stabilizer. The rudder trim actuator was removed from the wreckage for further examinations.

During Phase II, the systems group reconvened on September 19, 1994, at the Boeing EQA facilities in Renton, WA, to examine components removed from the accident airplane during Phase I investigations. The main rudder PCU, rudder trim actuator, and standby rudder PCU were examined (and tested if possible).

During Phase III, the systems group reconvened on September 21-22, 1994, for examination and testing of the main rudder power control unit (PCU) at Parker Hannifin in Irvine, CA.

During Phase IV, the systems group reconvened on October 3-7, 1994, at the Boeing EQA facilities in Renton, WA, to continue the examination of components removed from the accident airplane. The rudder feel and centering unit, pilot's cable drum assembly, copilot's transfer mechanism, spoiler mixer and ratio changer, flight spoiler actuators, ground spoiler actuators, aileron power control units, slat control valve, ground spoiler control valve, autopilot actuators, and cockpit hydraulic system pressure indicator were examined (and tested if possible).

During Phase V, the systems group reconvened for testing and examination at Boeing EQA facilities on November 15-18, 1994. The pilot's and copilot's control columns, standby rudder PCU, main rudder PCU hydraulic fluid filters, autoslat valve, wing leading edge slat actuators, rudder jackshafts, rudder torque tube, rudder pedals, and rudder feel and centering unit were examined (and tested if possible).

Additional examinations and testing by the systems group are planned during the first three months of 1995. The results of those tests will be reported in an addendum to this factual report.

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D. DETAILS OF THE INVESTIGATION

1.0 Phase I, Examinations and Testing at Pittsburgh, September 9-16, 1994.

During Phase I of the investigation, the systems group met daily at the accident site and the USAir hangar to document the condition of the airplane prior to the removal of components for additional testing.

During Phase I, the following components were packaged for shipment via USAir to Boeing facilities for further testing and examination (all components were removed, packaged, transported, and stored under NTSB chain-of-custody control procedures):

Main rudder power control unit (PCU)
Standby rudder power control unit (PCU)
Main rudder PCU vernier control rod
Aileron power control units (2)
Rudder trim actuator switch
Main rudder PCU attach mount

The rudder trim actuator and the forward attachment bearing for the main rudder PCU were hand carried to Boeing by NTSB systems group chairman Greg Phillips for examination.

The following observations were made during the accident site and hangar examinations of the airplane's wreckage on September 9 through 16, 1994. Additional examinations and testing during later phases (away from the accident site) may provide additional details concerning the condition of the components and in some cases these findings may supersede the observations noted during Phase I.

1.1 Left Wing

1.1.1 Spoilers

All of the left wing spoilers were located and examined. At the accident site, all flight and ground spoiler panels appeared to be faired with the top surface of the wing.

The #0 ground spoiler actuator was found fully retracted.
The #1 ground spoiler actuator (s/n 1348) was found fully retracted.
The #2 flight spoiler actuator (s/n 4850) was found extended 0.4 inches from seal plate to gland nut (= 0 degrees from Boeing reference data).
The #3 flight spoiler actuator (s/n 4864) was found extended 0.4 inches from seal plate to gland nut (= 0 degrees from Boeing reference data).
The #4 inboard ground spoiler actuator was found extended 1.88 inches (= 4 degrees, from Boeing reference data).
The #4 outboard ground spoiler actuator was found extended 1.85 inches (= 4 degrees from Boeing reference data).

1.1.2 Left Aileron

The left wing aileron surface was impact damaged and unburned.

1.1.3.1 Outboard Flaps

The #1 flap ball-screw was located near the #1 engine. The nut was attached. The nut to yoke-stop distance was measured and found to be 3 inches (flaps 1 position, from Boeing reference data). A distance of 19.3 inches was measured from the ball nut to the end-stop.

The #2 flap ball-screw was found buried in the ground under the left wing. During examination in the hangar a measurement of 3.6 inches was recorded from the ball nut to the yoke stop. (flaps 1 from Boeing reference data).

1.1.3.2 Inboard Flaps

The inboard flap jackscrew was found under the left main landing gear. It measured 4.6 inches of exposed threads. The ball nut was not examined nor located at the accident site. Examinations at the hangar found a measurement of 3.9 inches from the ball nut to the yoke stop (flaps 1 from Boeing reference data).

The outboard flap jackscrew was found with 27 inches broken off at the aft end of the ball nut. A distance of 5 inches was measured from the ball nut to the yoke-stop (flaps 1 position, from Boeing reference data).

1.1.4 Leading Edge Slats and Flaps

The leading edge slat surfaces were not examined in detail at the accident site. The slat actuators were not attached to their aircraft structural locations. The slat actuator positions were not verified at the accident site. The following documents the general condition of the slat actuators at the accident site.

A slat actuator was found with the outer piston extended approximately 5 inches from the gland nut. The inner piston was extended and broken off. The actuator was fire damaged.

Another slat actuator was found with the outer piston extended from the gland nut. The inner piston was extended and broken off. The actuator was fire damaged.

A third slat actuator was not located during systems group work at the accident site. The actuator was later recovered and a report of its examination follows in this report. See section 5.5.

Two Krueger flap actuators were observed while the group was at the accident site. The location of their installation could not be determined while at the accident site. One of the Krueger flap actuators was found next to the left wing. The piston was extended 7.25 inches (approximately fully extended, from Boeing reference data). After inspection in the hangar, the second Krueger flap actuator was determined to be fully extended.

The remaining two Krueger flap actuators were not observed by the systems group while the group was at the accident site. They were located during the hangar phase and both were determined to be in the fully extended position.

1.2 Right Wing

1.2.1 Spoilers

(All surface position data was obtained from Boeing reference data).

The #5 ground spoiler inboard actuator was found extended 1.8 inches (= 5 degrees).
The #5 ground spoiler outboard actuator was found extended 1.8 inches (= 5 degrees).
The #6 flight spoiler rod length was found extended 0.46 inches (spoiler retracted).
The #7 flight spoiler rod length was found extended 0.46 inches (spoiler retracted).
The #8 ground spoiler was found retracted.
The #9 ground spoiler was found fire-damaged and fully retracted.

1.2.2 Right Aileron

The right aileron was not examined at the accident site. A portion of the aileron was examined at the hangar. The aileron was impact damaged. See structures group chairman's report for details.

1.2.3 Flaps

Four flap ball screws were examined at the accident site. Their locations on the airplane were unknown because they were separated from their normal installed positions. The lengths and equivalent positions from Boeing reference data are:

Total length =27 inches. Extension was 3.4 inches from ball nut to yoke stop (outboard flap). (= Flaps 1).
Total length =27 inches. Extension was 3.6 inches from ball nut yoke stop (outboard flap). (= Flaps 1).
Total length = 35.3 inches. Extension was 4.9 inches from ball nut to yoke stop (inboard flap). (= Flaps 1).
Total length =35 inches. extension was 27 inches from nut to aft stop (inboard). (= Flaps 1).

1.2.4 Leading Edge Slats

The slat actuator installation locations were not verified at the accident site. Two leading edge slat actuators were located and examined at the accident site. For Krueger flap actuator information, see left wing for on-site data.

One slat actuator outer piston was extended 4.38 inches; the inner piston was extended and broken off.

A second slat actuator outer piston was extended 4.75 inches; the inner piston was extended and broken off.

A third slat actuator was located when the wreckage was examined in the hangar, no measurement data was taken.

1.3 Control Wheel and Control Columns and Aileron Power System

Both aileron power control units (PCU's) were found without rod ends. Gland plate marks were found 1.38 inches from the rod end where the rod end had separated from the PCU.

1.3.1 Control Wheel and Control Columns

Several parts from the control column and control wheel mechanisms, forward elevator quadrants, aileron transfer mechanisms (left and right) were located. A measurement of the bladejoint position at the base of both columns indicated a control wheel position of 30 to 40 degrees control wheel right.

1.4 Empennage

1.4.1 Horizontal Stabilizer Trim System

The forward cable drum was examined at the accident site. The control cables were missing from only two cable grooves 2 inches from the top of the drum.

The aft cable drum was examined. The control cables were missing from the cable grooves in the center of the drum starting 4.25 inches from the top to 6 inches from the top of the drum.

1.4.2 Horizontal Stabilizer Trim System Actuators

The main electric trim motor was examined and found broken from the gear box.

The autopilot trim motor was examined and found broken from the gear box.

The horizontal stabilizer trim gear box assembly configuration was identified as "post-airworthiness directive (AD)" (with ratchet and pawl auxiliary brake). The ball screw was found broken off at the base of the gearbox. The measurement from the screw from upper stop to the fracture was 10.5 inches; the measurement of the middle portion of the jackscrew (with nut) was 10 inches and the bottom part of the screw measured 14 inches. Fourteen inches of safety rod remained attached to the gearbox.

1.4.3 Elevators

The LH and RH elevator power control unit (PCU) exposed rod lengths were measured at 0.5 to 0.6 inches respectively which is approximately = 14 degrees aircraft nose up (per Boeing reference data).

The mach trim actuator, bolt to bolt centerline distance was = 7.125 inches.

The RH lower aft quadrant forward edge was measured at 7.5 inches aft of the Sta 1156 bulkhead.

The elevator output rods appeared to be undamaged.

The elevator tabs were found connected.

The aft quadrant was found in an aircraft nose up attitude position.

The LH elevator surface was found broken in an aircraft nose up attitude position. The autopilot rod was found connected to the aft quadrant 6 inches from the lower bolt to retaining flange.

1.4.4 Rudder

The main rudder PCU (p/n 65-44861-9, s/n 1596A) was found with 2.38 inches of actuator rod protruding (fore and aft). The rod was bent forward.

All hydraulic lines were found intact with torque putty in place on all fittings.

No other damage to the PCU was noted other than the bent actuator rod end near the PCU's attachment point to the rudder.

The main rudder PCU input crank position was shimmed and its hydraulic lines were removed and capped prior to removal for shipment and testing.

1.4.5 Standby Rudder Power Control Unit (actuator)

The standby rudder actuator piston extension was 4.8 inches. The input bearing lockwire was found intact. The input arm linkage was found attached and the input arm moved fore and aft without any apparent binding. The input bearing did not move with input shaft movement.

1.4.6 Rudder trim actuator

The rudder trim actuator rod was found extended 2.25 inches (measured from the center of the rod end bolt to the face of the body shaft. The body shaft housing was found broken free. The rudder trim actuator was removed for further testing.

1.5 Cockpit Equipment

The following cockpit equipment indications were recorded:

The radio magnetic indicator (RMI) indicated 212 degrees.
Two airspeed indicator digital displays indicated 264 kts.
One orange airspeed bug was set at 192 kts.
B system hydraulic pressure 3150 psi (A-system needle broken off gage).
Autopilot Mode Control Panel-A/P OFF
GPWS switch-ON
Flight Director Switch-ON

All other cockpit and electronic equipment was too badly damaged to report readings based on accident site examination.

2.0 Phase II, Examinations and Testing at Boeing, September 19, 1994.

On September 19, 1994, the systems group reconvened at the Boeing EQA facilities in Renton, WA, to examine components removed from the accident airplane during Phase I investigations.

All Phase I participants and the following additional persons participated in the testing and examinations: John Calvin, John Ford, Philip Bookout, Ryck Whisler, Paul Hermanson, Scott Hanowski, Steve Slagel, Rick Krantz, and Paul Cline of the Boeing Commercial Airplane Group, Seattle, WA. and Steve Weik of Parker, Irvine, CA.

2.1 Rudder Trim Actuator Tests and Examinations

The rudder trim actuator (manufactured by Machined Parts Corporation (MPC), p/n 10-62025-3, s/n 0487, delivery code 8727, revision P, functional test date 7/01/87) was examined at the direction of and in the presence of the systems group. USAir maintenance records indicated that the unit had been installed on the accident airplane for 23,846 hours and 14,489 cycles at the time of the accident.

The unit was photographed and physical damage noted. An external dimensional check was performed and the unit was X-rayed. Continuity tests were performed on the motor circuits and the position of the rotary variable transducer. The unit was disassembled and dimensional verification, condition of the internal components, and witness marks were documented.

There was an indentation through the housing cavity. Visual examination of the electrical connector indicated a side load. The base of the electrical connector was lifted. The backshell was found broken. (Safety wire had been added at the accident site to secure the actuator gear position. The moving clevis was not safety-wired into position).

The rear portion of the housing (fixed end) was found damaged and pieces were missing. The clevis on the fixed end was broken free. The rear bearing plate had been rotated. The bearing plate and gear were slightly cocked. The rear (outer) bearing was found out of the housing. There was an impact mark on the bearing housing. The bolt on the clevis end had been removed. The manufacturer's security mark was broken.

The safety wire remained attached to the three remaining bolts on the fixed end. The safety wire on the connector was intact. The two housing bolts were missing and the holes through the housing where the bolts pass through were missing.

The unit exhibited axial freedom of movement of the rod relative to the gear. There was approximately 0.25 inch movement of the rod relative to the housing. It was unknown if the added safety wire restricted the motion.

Dimensional checks located the position of the actuator. The 9.1355 inch dimension (relative to the bolt) indicated that the actuator was 0.020 inches from the neutral rig position of 9.11 inches.

Visual inspection under a microscope indicated a sheared gear tooth. The X-ray review indicated that the acme shaft appeared to be bowed. The rotary variable displacement transformer (RVDT) was displaced; its wiring and the screw appeared intact.

After verifying airplane wire identification markings with a wiring diagram, a pin-to-pin continuity test was performed with the following results:

Pin	to Pin	with connector(ohms)	without connector (ohms)
6	7	116.009	115.917
4	5	21.289	21.18
2	13	110.08	109.968
3	13	110.15	110.054
2	3	220.11	220.00
15	case	ground	ground

All circuits were isolated from ground.

Following isolation of all circuits, 26VAC/400Hz was applied to pins 4 and 5 and the voltage was measured with a phase angle voltmeter. The reading across pins 4 and 5 was 70.36 millivolts off null. The nominal voltage rise is 2.797 volts/inch. The reading was consistent with an actuator positioned to 0.025 inches in the extend direction (referenced to RVDT null position).

2.2 Standby Rudder Power Control Unit (actuator)

The standby rudder power control unit (manufactured by Hydraulic Units Inc. (HUI) p/n 1U1150, s/n 1619A) was examined.

Visual examination and photo documentation indicated that all inspection seals and lockwires were present. Slight wear was noted on the anti-rotation lugs; this was characterized by the manufacturer as normal in-service wear.

A hydraulic fluid sample was collected from the actuator and forwarded to Monsanto for analysis. The input lever force with the actuator unpressurized was measured at 0.43 pounds in the extend direction. The input lever force with the actuator unpressurized was measured at 0.49 pounds in the retract direction.

The actuator was tested to verify the unit's capability to function normally. The following tests were performed and results noted:

2.2.1 Bypass valve adjustment

The bypass valve adjustment was tested by measuring the point at which fluid flow stopped. Flow stopped at 850 psi. (Nominal value is 850 +/- 50 psi.). Return port flow increased gradually.

2.2.1.1 Bypass valve bleed flow

The bypass bleed flow was tested by measuring the flow by the method specified in the Dowty test procedure. Flow was measured at 0.155 gpm. (Nominal value is 0.15 to 0.25 gpm).

2.2.1.2 Bypass valve closing pressure

Return port flow increased gradually to cutoff. The cutoff pressure was 850 psi. (Nominal value is 850 +/- 50 psi.).

2.2.2 Servo valve dead spot limits

A test was performed to determine the point at where the servo valve began to operate. The opening point was measured as: Extend -0.1005 inches, Retract -0.1005 inches (0.056 to 0.101 is the nominal value).

2.2.3 Internal leakage

Return port leakage was tested. The test specification requires that return port leakage shall not exceed 20 cc/min. (at 0.020 inches on each side of the approximate neutral position). The unit was measured at 4 cc/min in the extend direction, and 18 cc/min in the retract direction.

The test specification requires that return port leakage shall not exceed 60 cc/min (at 0.397 each side of neutral-approximate full travel). The unit was measured at 68 cc/min in the extend direction, and 32 cc/min in the retract direction.

2.2.4 Actuation and travel

The test specification requires that the standby actuator input arm load shall not exceed 0.5 lbs at 3000 psi. The unit was measured at 0.20 lbs in the extend direction, and 0.32 lbs in the retract direction.

The test for the fully extended length of the actuator met the test specification requirements and was verified as 21.350 inches (minimum).

The test for the fully retracted length of actuator met the test specification requirements and was verified as 16.650 inches (maximum).

2.3 Main Rudder Power Control Unit PCU

The main rudder PCU (p/n 65-44861-9, s/n 1596A) was manufactured by Parker Hannifin, during October of 1987. The PCU was functionally tested by Parker on October 1, 1992 during servicing by Parker.

During the examinations at Boeing, the unit was visually examined, x-rayed, and photo-documented.

All critical areas on the PCU had the original equipment manufacturer's overhaul facility green metal inspection seals on the lockwire.

The tail end piston rod extension was measured and determined to be approximately 1.28 inches.

The input crank to manifold stop gap dimensions were measured and determined to be 0.274 for the retract stop and 0.224 inches for the extend stop.

The piston rod was bent and the connecting input levers were deformed from impact forces.

All critical plugs, caps, and retainers were found bottomed.

A continuity check was performed on the connector portion of wiring leading from the PCU to the airplane and the PCU spare (unused) electrical connector. The following results were recorded (all values were determined to be acceptable):

Pin	Pin	Airplane wire R (ohms)	PCU spare connector R (ohms)
1	2	77	76.9
5	6	----	1006
7	8	----	1006
9	10	103.4	103.3
11	12	82.9	82.9
1	4	shorted	shorted
5	8	2012	

Note: The airplane wire bundle incorporates a wire that interconnects pins 6 and 7 at the airplane wire bundle electrical connector.

3.0 Phase III, Examinations and Testing at Parker, September 21-22, 1994

During Phase III examination and testing at Parker Hannifin, Irvine, CA, on September 21-22, 1994, the main rudder power control unit (PCU) was examined and functionally tested in the presence and at the direction of the systems group. Appendix 1 documents the examination and testing. This section summarizes the significant findings of this phase.

3.1 PCU Examination and Test Preparation

The unit was visually inspected. Electrical resistance tests were performed and hydraulic fluid was removed from the unit after cleaning. The hydraulic fluid filters were removed and visually examined. Hydraulic fluid was drained from the cavities containing the hydraulic filters.

The unit was thoroughly cleaned around the gland area. The unit was then installed in a Parker assembly fixture upside down so that trapped hydraulic fluid had a path of free flow out of the PCU. Trapped hydraulic fluid was removed from the cylinder cavity. The piston and rod were removed and the interior of the unit was examined with a borescope. The surface finish appeared to be intact. There was no evidence of impact marks or abnormal wear.

A new piston assembly, rod end assembly, end gland, retainer nut, summing lever, H-link, snubbing ring, and associated nuts, bolts and washers were installed in order to test the unit. The unit was then installed on the test fixture and connected to the hydraulic bench.

The cover plate assembly was removed. There appeared to be small shiny metallic particles in the linkage cavity fluid. System B hydraulic fluid was taken out of the linkage cavity by using a laboratory sealed syringe. The remaining fluid was poured from the cavity. The fluid sample was collected in a clean container provided by Parker and shipped to the fluid manufacturer (Monsanto) for testing. Results of that testing are documented in a separate report.

The gap between the servo valve external stops and the primary summing lever on both the retract and extend sides was measured with pin gages (the gap position had been secured in the hangar at Pittsburgh). The measurements of the gaps were recorded as, extend side gap (left rudder) = 0.132 inches, retract side gap (right rudder) = 0.090 inches.

3.2 Input Force Tests

A spring force test with no hydraulic pressure applied to the actuator was conducted. The linkage cavity cover plate assembly was removed. With hydraulic supply pressure = 0 psi a pull scale was attached to the input arm. The shims were then removed.

The primary input was moved through the bias spring force to the secondary slide pick-p point by moving the input lever towards the forward end (rudder extend). The primary slide moved into the servo body without friction or binding at a force measured at 0.55 - 1.00 lbs. The primary moved back toward the aft end (right rudder) when the input was released without friction or binding. This was normal (the PCU is designed with a bias spring in the servo to take out backlash between the drive ball & primary slide). The primary bias spring is designed for the primary slide to be driven up against the secondary pick-up point in the PCU retract direction. By design, the pull test performed during this examination can not be performed in the PCU retract direction without moving the secondary slide.

The force required to move the secondary bias spring in the extend and retract direction was tested. The PCU was extended by pulling the input lever until the secondary detent spring was compressed and the secondary slide moved. In the extend direction (left rudder), the input lever was pulled toward the forward end of the PCU. Between 7.0 - 7.5 lbs of force were required to move through the secondary detent spring. In the PCU retract (right rudder) direction (input toward aft end), 5.5-5.25 lbs of force were required to move through the secondary detent spring. The test operator noted that the movement was smooth with no apparent binding in both directions of travel.

3.3 PCU Actuator Direction of Travel Testing

A test cover plate was installed in place of the link cavity cover plate and the unit was set up on the test stand. Hydraulic fluid from system A was removed and sent to Monsanto. System B was pressurized and cycled to remove additional hydraulic fluid from System A. No fluid came out of the return port. System A was pressurized and hydraulic fluid was collected from the return port of System A. The linkage and servo valve's reaction to pressure was observed. When the unit was cycled, the linkage and servo valve components appeared to function normally.

With the hydraulic system pressure to the PCU set at 360 psi, the PCU operated normally in the extend and retract directions. A full-rate demand rudder pedal input to the input link was made in an attempt to cause the PCU to operate in a reversed direction. There was no PCU reversal in either the retract or extend direction. Several applications of hard right and left rudder commands were input by the investigation team with no actuator piston reversal. Both primary and secondary slides were cycled. The motion of input lever was smooth with no apparent binding noted.

The hydraulic supply pressure to the PCU was increased to 3000 psi. A full-rate demand rudder pedal input to the input link was made in an attempt to cause the PCU to operate in a reversed direction. There was no PCU reversal in either the retract or extend direction. Several applications of hard right and left rudder commands were input by the investigation team with no actuator piston reversal. Both primary and secondary slides were cycled. The motion of input lever was smooth with no apparent binding noted.

3.4 Yaw Damper System Testing

The PCU successfully passed each of the following tests (see Appendix 1 for additional detail and test data).

Rig Neutral Cylinder Stroke and Clearance.
Linkage Breakout Friction.
Transducer Null.
Transducer Output.
Yaw Damper Authority.
Yaw Damper Engage.
Phase Check.
Yaw Damper System Phase Lag.
Yaw Damper System Repeatability and Linearity.
Intersystem Leakage.
Internal Leakage Yaw Damper On.
Internal Leakage Yaw Damper Off (Bypass Test).

A test was devised and conducted to determine the yaw damper velocity authority (rate of actuator movement with full step signal input to the yaw damper). The nominal rate is approximately = 50 degrees/sec. This test is not part of normal Parker's test procedures. The unit passed the test.

The unit did not pass the input force vs input travel test. The primary slide picked up the secondary slide on the extend side 0.002 inch sooner than allowed by Parker test specifications.

3.5 Servo Control Valve Testing

The PCU was removed from the test fixture and installed on an assembly fixture. The linkage cavity was examined with a borescope while the systems group members observed the examination on a television monitor. The secondary internal summing lever came into contact with the servo external stop on both the retract and extend side. There were no abnormalities noted. A side force was applied to the secondary lever to simulate a misalignment. With a side force, the secondary slide made contact with its stops; there were no abnormalities noted in either the retract or extend direction during the simulation.

It was noted that the lockwire on the servo valve nut was installed backwards. The torque stripe was properly aligned. A torque test was performed on the nut in a clockwise direction to verify that the nut was tight. Over 600 in-lbs of torque was applied with no motion of the nut. The nut was untorqued and retorqued to the torque stripe. In the direction that tightened the nut, 575 in-lbs. of torque was required to match the torque stripe.

The servo valve nut was removed. Hydraulic fluid was found in the back of the servo cavity (this is normal). The back of the servo cavity was examined. No abnormalities were noted.

The PCU was disassembled to remove the servo valve. Normal segment cam wear was observed during the examination. A wear mark was found on the shaft at the ball end. The following components were examined with no anomalies noted: 69-35605-1 Segment, 69-35613-1 Fork Lever, 69-35608-1 Seat Cam, 69-34502-1 Lever Assy Primary, 69-35603-1 Lever Assy Secondary, 68045 Guide (noted some deformation on 2 of the 5 corners of the scallop cuts), 68021 Nut, 83347 Spring, 68046 Guide, 83311 Housing.

The primary slide and secondary slide assemblies were examined under a microscope. There were no anomalies noted under the magnification used (less than 26X). The return "communication" hole in the assembly was flushed with fluid; nothing flushed out indicating that there was no existing blockage and the absence of foreign material. A light was shined through the passage as systems group representatives from the FAA and Parker witnessed. There were no apparent obstructions.

A video borescope was used to view the metering edges and flow holes of the servo valve components. The components were checked for burrs or erosion. Slight erosion was noted on system A 1st return metering edge (C retract to Return). Slight erosion was noted on system A 1st pressure metering edge (Pressure to C retract). No anomalies were noted under the magnification used. All dimensions checked were within acceptable limits. It was noted that the

68010-15 Spring Slot (for 83348 Spring Tang) appeared to have had burrs generated during assembly/disassembly. The burrs were removed with a stone.

The servo valve was reassembled and installed into the Parker servo test fixture. The following summarizes the significant results of the servo valve testing (see Appendix 1 for test data sheets). The servo passed all Parker 68010-5003 ATP and -5005 ATP test 2A tests except: it failed the Flow Gain test in the servo extend direction because the overlap between the primary and secondary extends out of the acceptable test envelope. The unit also failed the Primary Friction test because the friction was 0.5 ounces too high.

The Yaw Damper Assembly was disassembled to remove and inspect the following yaw damper components: 69-35609-2 Lap Assy, 69-35611 Sleeve, 59188-3 Diaphragm, 10-60810-1 Transducer Assy, and 59174-5 Cap.

The aft piston, end gland, retainer, nut, summing lever, H-link and rod end assembly (installed for testing purposes) were removed and the original piston, end gland, retainer, rod end assy and nut were reinstalled. The aft nut was hand tightened. The summing lever and H-Link were not reinstalled. All open cavities in the PCU were plugged and all disassembled hardware were repackaged with the PCU and submitted to NTSB officials for storage.

4.0 Phase IV, Examinations and Testing at Boeing, October 3-7, 1994

On October 3-7, 1994, the systems group reconvened at the Boeing EQA facilities in Renton, WA, to continue the examination of components removed from the accident airplane. The following components were tested or examined.

4.1 Rudder Feel and Centering Unit

On October 7, 1994, the rudder system feel and centering unit recovered from the impact site was inspected in the presence and at the direction of systems group in the Boeing Equipment Quality Analysis (EQA) labs. Additional participants were: Philip Bookout (a Boeing controls system engineer), and Ryck Whisler (Boeing EQA).

The rudder feel and centering (F and C) unit provides centering (neutral) capabilities for the rudder system and feel forces to the rudder pedals. The F and C unit provides these two functions by a cam profile, roller and springs. The cam has a valley in the center of its profile that creates the neutral position. When inputs from the rudder pedals are provided, the cam rotates either clockwise or counter-clockwise depending upon which pedal provided the input. The rotation of the cam displaces the roller from the neutral position. The roller's resistance to being displaced is provided by two compression springs. This resistance provides the feel forces to the rudder pedals. Looking from the top of the F and C unit down, a clockwise rotation of the cam would result from a left rudder pedal input and a counter-clockwise rotation would result from a right rudder pedal input.

The F and C unit was identified as part number 65C25410-2, serial number SC A 191, dated February 27, 1987. All but one part of the F and C unit assembly was intact. The missing part was the upper bearing housing (the inner race of the bearing was still attached to F and C unit). The structure which the upper bearing housing attaches to was found detached and was in a separate location at the hangar.

Both of the lower mounting brackets were still attached. The top lower bracket was bent and was still attached to the rear spar of the vertical fin at the crash site (the removed bolts were attached). The lower bracket was also bent and the lower third of the bracket was broken off and not examined.

The connecting rod from the F and C unit to the torque tube was still attached. About one inch of the crank arm of the torque tube was still attached to the connecting rod. The F and C unit was bent. The bend of the F and C unit and lower brackets coincide with the direction of impact of the airplane. In order to closer inspect the F and C unit, the lower mounting brackets were removed. Installation torque putty was still intact on all the nuts. There were no marks on the F and C unit that would explain the bending of the housing. There was a mark on the housing next to the main rod. This mark aligned with a bolt head on the main rod. Both of the compression springs were intact.

Because the F and C unit housing was bent at impact, the cam was stuck in a clockwise-commanded direction position. From angular measurements of a photograph, it was determined that the position would correlate to an approximate 2.26 degrees left rudder position at impact.

During inspection of the cam profile, a mark was observed on the side of the cam opposite of the position the roller was resting. A similar mark on the roller was also found. These marks were examined further at a later date (see Section 5.6.5 for results).

4.2 Pilot's Cable Drum Assembly

From October 5 through October 7, 1994, the pilot's aileron cable drum assembly recovered from the impact site was inspected at the direction and in the presence of the systems group in the Boeing EQA labs. assisting the systems group were Paul Hermanson (Boeing Flight Controls Engineer) and Ryck Whisler (Boeing EQA).

The pilot's cable drum assembly provides a connection between the pilots control wheel inputs to the lateral control system. A bus drum connected to the copilot's control wheel transmits the copilot's control wheel inputs to the lateral control system under normal operation. A force transducer measures the pilot's and copilot's wheel forces for control wheel steering autopilot function. A force limiter limits the amount of control wheel force (as a function of flight condition) applied to the lateral control system.

All parts of the assembly showed evidence of fire damage. They were blackened and charred. There were no part numbers remaining. The following part numbers were identified using Boeing engineering drawings. The proper drum assembly for line #1452 (the accident airplane) is 65-55731-55 installed by 65-45101-751. The following parts called out by the installation drawing (65-45101-751) were present; 10-61816-5 limit mechanism, a 6-inch piece of the mechanism forward of the shaft remained, the entire aft housing except the electric motor and internal components, the upper housing (inboard portion), clutch assy, and a portion of the 10-61072-8 Force Transducer was separated from the main shaft and fire-damaged.

The following parts called out by Boeing engineering drawing 65-55731-55 were present: 6-60428-2 Fork Assembly. 65-55476-6 Bearing Housing (partial), 65-55476-7 Bearing Housing (partial), 65-55710-4 Shaft (partial-minus travel stop), 65-55711-2 Bus Drum (partial-hub only), 65-55729-4 Aileron Drum (partial-hub only), 66-24952-1 Spacer (complete), 69-40961-1 nut (complete), 69-41762-2 Cable Guard (partial), 69-42919-1 spacer (complete), 69-46712-1 Spacer (assumed to be still in assy), BACB10A235 Bearings (partial inner and outer race only), BACB10A661 Bearing (complete), BACB10A822 Bearing (seal broken).

The force transducer, lower support, and middle support provided no useful information on control wheel position at impact. The upper support region showed a witness mark on the shaft assembly from the upper support. The shaft assembly was positioned so that a measurement of the position of the shaft position relative to the upper support could be taken. The witness marks indicated a range of 21 degrees to 35 degrees control wheel right. The shaft assembly was bent aft approximately 20 degrees above the force transducer clutch. Additional control wheel position confirmation could not be determined from the bend.

4.3 Copilot's Aileron/Spoiler Transfer Mechanism

From October 5 through October 7 1994, the copilot's transfer mechanism assembly recovered from the impact site was inspected at the direction and in the presence of the systems group members in the Boeing EQA labs. Additional participants were Paul Hermanson (Boeing flight controls engineer), and Ryck Whisler (Boeing EQA).

The copilot's transfer mechanism performs the following functions: it transmits the copilot's control wheel forces to the aileron bus cable, provides an override function that allows the copilot's control wheel input, (in the case of an aileron control cable jam) to be transmitted to the spoiler control cables above a specified force level and after a specified wheel motion.

The transfer mechanism was in two separate parts (identified as Part A and Part B for the purpose of this examination). The part numbers were identified using Boeing engineering drawings where no part number was visible.

4.3.1 Part A

Part A consisted of the upper portion of the assembly as identified by the part listing below. The components were not fire-damaged and some part numbers were present: The following documents the components general condition. 65-60428-2 Fork (complete), 65-54206-5 Shaft Assembly, (partial-broken below the upper spline, the remaining portion found in Part B). 65-55476-7 Bearing Housing (partial-broken off at fwd end), 66-24781-1 spacer (complete), 66-

24952-1 Spacer (complete), 69-40832-1 Washer (complete), 69-40961-1 Bolt (complete), 69-41772-1 Aft Beam (partial-cracked), 69-41859-1 Aft Rib (partial), 69-61339-1 Retainer (complete), BACB10A235 Bearing (complete with seal damaged), BACB10A823 Bearing (complete), BACB10BX Bearing (complete).

The 65-55476-7 support arm had witness marks which matched the lower corners of the 66-60428-2 fork assembly. If the fork assembly lines up with the marks, the resulting copilot's control wheel position is right of neutral. The resulting control wheel position from these marks could not be measured accurately since the marks were made by a vertical/tilting motion that could not be duplicated.

4.3.2 Part B

Part B consisted of the lower portion of the assembly. The components are listed below. The components were fire-damaged. No part numbers were visible. 65-54207-2 Spoiler Drum (complete-some damage), 69-61352-1 Spring Cartridge (lower cartridge housing and spring), 65-54206-5 Shaft (lower portion below top spline without bottom spline), 66-24942-1 Nut (complete), BACB10A821 Bearing (seals missing, locked in position).

The spoiler drum, p/n 65-54207-2, had two notches that matched the dimension of the flanges on rib 69-41858-1. The rib is normally fixed directly forward of the spoiler drum. The notches located 54 degrees from the location where the rib would be in if the spoiler drum was located at neutral. This matches a copilot's wheel position at impact of 54 degrees wheel right. Similar damage (that is not as definite on the rear of the spoiler drum 65-54207-2) matched a wheel position of 36 degrees right at impact.

The remains of the 65-54206-5 shaft assembly was locked in place within the spoiler drum. The missing tooth on the spline indicated that the shaft had been rotated 39 degrees (copilot's wheel right) from the nominal position. This measurement was considered questionable. If the spoiler lost motion device is attached, the spoiler drum should only rotate 12 degrees relative to the shaft.

A fracture was identified on the 65-54206-5 shaft assembly below the upper spline. This fracture was analyzed by NTSB metallurgists. See Appendix 2 for a report of the findings.

4.4 Spoiler Mixer and Ratio Changer

On October 7, 1994, the lateral control system spoiler mixer and ratio changer which was recovered from the impact site was inspected under the direction and in the presence of the systems group in the Boeing EQA labs. Additional participants were Scott Hanowski (a Boeing flight controls engineer) and Ryck Whisler (Boeing EQA).

The spoiler mixer/ratio changer is provided aileron and speed brake handle position inputs. The spoiler mixer/ratio changer outputs are flight spoiler control and ground spoiler control valve position. The flight spoiler panels are positioned to any position between 0-40 degrees by individual hydraulic actuators controlled by cables from the ratio changer. Operation of the ailerons drive the ratio changer input crank to operate the spoiler mixer linkage. The linkage rotates spoiler cable quadrants mounted on the ratio changer to signal the flight spoilers to move up on the wing.

Counterclockwise rotation of the speedbrake input quadrant (speedbrake lever to aft) operates the linkage in the spoiler mixer to cause an UP signal at both spoiler cable quadrants. Clockwise rotation causes a DOWN signal at both quadrants. In either case, the linkage also positions the ground spoiler control valve through the rotation of the spoiler control valve crank on the spoiler mixer. Ground spoiler panels are two position devices. They are either full down or full up.

Boeing engineering drawings indicated that the proper components for line #1452 the accident airplane) were: 65-49173-5014 Spoiler Mixer Installation, 65-46360-7 Spoiler Mixer Assembly, and 65-46370-16 Ratio Changer Assembly.

The following parts from the Spoiler Ratio Changer Assembly and the Spoiler Mixer Assembly were examined at the EQA lab: Spoiler Ratio Changer Assembly, 65-46363 Left Upper Spoiler Cable Quadrant (partial), 65-46363 Right Upper Spoiler Cable Quadrant (complete), 65-46362 Right Lower Spoiler Cable Quadrant (complete), 65-53856 Aft Frame (partial, a 4 inch by 6 inch portion of the frame above the left spoiler quadrant), 65-53854-7 Forward Frame (partial, a 3 inch by 6 inch portion of the frame outboard of the "no back"). The following linkages: 65-46365, 65-46366-4,69-40326, 65-46364, 65-75324, 65-46367-2, 65-75325, 65-51686,65-50857, 65-52299 were all in generally good condition.

Examination of the following spoiler mixer assembly components was performed in the EQA labs: 65-46354 or 65-53852 Aft Housing (partial-two fragments surrounding the speed brake shafts approximately 3 inch in diameter. All linkage components of the mixer mechanism were examined in the EQA labs with the exception of the 65-66519-1 Cam.

The parts listed below each displayed witness marks which were thought to indicate their position at the moment of impact.

The cable attachment side of the left upper spoiler cable quadrant was deformed (compressed). A matching deformation was observed on the housing of the spoiler ratio changer assembly which enclosed the quadrant. A second matched pair of deformations between the quadrant and the housing also occurred near the rotation point of the quadrant. Matching the two damaged areas of the quadrant and housing gave a left quadrant position relative to the housing at the point of aircraft impact. However, drawing layouts showed that the upper left spoiler quadrant on the spoiler mixer assembly could not normally move to the position required to cause the observed damage.

The ground spoiler control valve crank showed paint removed from one of its underside edges. The piece of the spoiler mixer assembly housing attached around the rotation axis of the crank had the top coat enamel removed. The crank edge and removed enamel line matched.

Such a match gave an apparent position of the ground spoiler control valve crank. However, drawing layouts showed that the ground spoiler control valve crank on the spoiler ratio changer assembly could not normally move to the position required to cause the observed damage.

Metal was scarred and removed from the lower right spoiler quadrant of the spoiler ratio changer assembly near the cable grooves on the quadrant. The spoiler cable guide from the housing of the spoiler ratio changer assembly had corresponding damage.

A puncture mark indentation on the aft face of the lower right spoiler quadrant was observed. One of the spoiler mixer linkage joints was located aft of the position of the indentation mark. Drawing layouts showed that the lower right spoiler quadrant on the spoiler mixer assembly could physically move to the position required to cause the observed damage. The position of the right spoiler quadrant appeared to be between 20 degrees and 23 degrees counterclockwise from the neutral position (i.e. ailerons at neutral and speed brake handle at the DOWN position).

4.5 Spoiler Actuators

The spoiler actuators were examined on October 4 and 5, 1994, in the Boeing EQA labs at Renton, WA, at the direction and in the presence of systems group members. The following documents the conditions noted during the examinations.

4.5.1 #2 Flight Spoiler Actuator

The #2 flight spoiler actuator was identified as p/n 65-44561-15, s/n 4850. All lockwire and inspection seals were intact except at the piston rod end gland bolts. The end gland was lockwired but there was no inspection seal. There was no severe impact damage; however, there was slight indentation on the rod end gland and at the lockwire end on one end gland bolt (possible area where inspection seal might have been located).

X-rays were taken to examine the hold-down check valves and servo valve areas. No unusual conditions were observed. The anti-cavitation check valve retainer caps were removed. The static o-rings exhibited slight extrusion nibbling. The check valve seats and poppets were properly seated.

The hold-down check valve retainer cap was removed and fluid collected from the cavity including fluid from the retract side by manually pulling the piston rod towards the extend direction. An anomaly (what appeared to be slight "coining" of the valve seat) was noted on the hold-down check valve seat. This condition was further evaluated by conducting a pull test on the actuator to determine the integrity of the seat.

The unit was connected to hydraulic pressure and operated in a normal condition. No unusual or abnormal conditions were observed. A hold-down check valve leakage test per paragraph 3.4.6, sheet 3, page 11 of drawing 65-44561 was performed using the specified 4400 pound tension load. The unit performed acceptably. The "coining" condition noted above did not affect the performance of the unit.

4.5.2 #3 Flight Spoiler Actuator

The #3 flight spoiler actuator was identified as p/n 65-44561-15, s/n 4864. All lockwire and inspection seals were intact. Minor impact damage was observed at the anti-cavitation check valve retainer caps. X-rays were taken to examine the hold down check valve, anti-cavitation check valves, and servo valve areas. No unusual conditions were observed. The anti-cavitation check valve retainer and hold-down check valve caps were removed to collect fluid samples. The piston rod end was extended manually to collect fluid from the retract side.

The unit was connected to hydraulic pressure and operated. No unusual or abnormal conditions were observed. A hold-down check valve leakage test per paragraph 3.4.6, sheet 3, page 11 of drawing 65-44561 was performed using the specified 4400 pound tension load. The unit performed acceptably.

4.5.3 #4 Inboard Ground Spoiler Actuator

The #4 ground spoiler actuator (inboard actuator) was identified as p/n 65-44851-7, s/n 4905. The unit was x-rayed and the lock segments were found engaged. The actuator end cap, piston, and locks were removed. The piston seal and bore appeared normal; there were no witness marks.

4.5.4 #4 Outboard Ground Spoiler Actuator

The #4 ground spoiler actuator (outboard actuator) was identified as p/n 65-44851-7, s/n 5064. The unit was x-rayed and the lock segments were found engaged. The actuator end cap, piston, and locks were removed. The piston seal and bore appeared normal, there were no witness marks.

4.5.5 #5 Inboard Ground Spoiler Actuator

The #5 ground spoiler actuator (inboard actuator) was identified as p/n 65-44851-7, s/n 5056. The actuator end cap was removed. The exterior of the housing was fire-damaged. The piston retainer gland seal was extruded which prevented the piston assembly from being removed.

4.5.6 #5 Outboard Flight Spoiler Actuator

The #5 outboard flight spoiler actuator was identified as p/n 65-44851-7, s/n 5057. In an attempt to remove the actuator end cap the spanner lugs were broken off. This prevented further disassembly of the actuator. Based on the similarities of the findings involving the other spoiler actuators examined, the systems group decided not to perform additional disassembly of the actuator.

4.5.7 #6 Flight Spoiler Actuator

The #6 (inboard right wing) flight spoiler actuator was identified as p/n 65-44561-14, s/n 432915. The anti-cavitation check valve retainer caps were removed to examine the valves and manifold bores. Seal extrusion and deterioration as a result of fire and heat caused the check valve body to cock in bore (noted in x-ray examination). USAir maintenance records indicated that the unit was overhauled by Tramco and installed on the accident airplane on November 14, 1990.

The hold-down check valve cap o-ring was fire-damaged. The fire-damaged poppet was easily removed. Seat marks were observed. The actuator rod end measurement from top of the retainer to the piston was within 0.005 inch of a new unit in the retracted position, indicating that the piston was retracted at the time of the fire damage. The O-ring and backups relative to the piston were completely burned. The servo valve was unable to be removed. The piston was cleaned with freon and the lower area of the piston was examined for witness marks. No witness marks were observed. The upper bearing was removed from the piston. The bearing surface was cleaned and the bearing was examined for impact damage. No witness marks were observed.

4.5.8 #7 Flight Spoiler Actuator

The #7 flight spoiler actuator was identified as p/n 65-44561-15, s/n 4777. The check valve caps were removed. Seal decomposition due to fire damage was noted. The check valves could not be removed because of fire related damage. Hold-down check valve seal decomposition due to fire damage was noted. Neither the poppet nor the seat could be removed because of fire damage. A measurement taken from the top of the retainer to the end of the piston was within 0.001 inch of a new unit in the retracted position.

The control valve and centering pistons were frozen in place due to fire damage. In general, #7 was fire damaged more than the #6 spoiler previously documented in section 4.5.7. A pull-fixture was used to remove the piston from the barrel. The end gland bearing was removed and all surfaces were examined for impact damage. There were no witness marks observed.

4.6 Ground Spoiler Actuators

The ground spoiler actuators were examined at the direction and in the presence of the systems group members on October 6, 1994 in the Boeing EQA labs at Renton, WA. The following documents the conditions noted during those examinations.

4.6.1 Actuator s/n 1359

The actuator was identified as p/n 65C26864, s/n 1359. The unit was fire-damaged and in the fully retracted position.. The piston was bent. X-ray examination revealed that the locking keys were in their locked positions.

4.6.2 Actuator s/n 1348

The actuator was identified as p/n 65C26864-2, s/n 1348. The unit was not fire-damaged; it was in the fully retracted position. The piston was fractured. X-ray examination revealed that the locking keys were in their locked positions. The actuator was pressurized and fully extended. Visual examination revealed normal wear on the piston outer diameter.

4.6.3 Actuator s/n 1352

The actuator was identified as p/n 65C26864, s/n 1352. The unit was not fire-damaged; it was in the fully retracted position. The piston rod was bent. X-ray examinations revealed that the locking keys were in their locked positions. The actuator was pressurized and fully extended. Visual examination revealed normal wear on the piston rod outer diameter.

4.7 Aileron Power Control Units (PCU's)

Both aileron PCU's manufactured by Parker were examined at the direction and in the presence of the systems group members on October 6, 1994, in the Boeing EQA labs at Renton, WA. The following documents the conditions noted during the examinations.

4.7.1 A-System Aileron PCU

The unit was identified as p/n 65-44761-21, s/n 6958 (missing data plate, number derived from Parker records), servo valve s/n 5151. The PCU exhibited severe impact damage. The rod end was found broken off from the actuator. A portion of the control input rod remained attached to the input lever. The head-end housing was pulled loose from the manifold. Inspection seals were found attached to lockwire at all critical points.

The unit had servo valve p/n 65-44828-4 E4, s/n 5151, installed to manifold s/n 5037. A impact mark was found 1.372 inches from the back of the rod end. The rod was approximately 0.528 inches retracted from center (neutral) position at impact. A dimension from the head-end of the rod to the manifold was measured as 2.653 inches. Impact witness marks were found on the head-end of the piston outer diameter at 180 degrees on the opposite side. The width of the end gland bearing was measured as 0.462 +/- .015 inches.

Engineering calculations based on the impact marks indicates the following control positions at impact: 25% aileron PCU retract (= 20 degrees control wheel right), 5.25 degrees left aileron down, 4.75 degrees right aileron up, 5 degrees right spoilers up. Boeing engineers noted that full control wheel motion available is 108 degrees to the right or left (spoilers begin engagement at 11 degrees control wheel motion). Boeing engineers stated that when the control wheel is at 80 degrees rotation, the ailerons and spoilers are completely up.

The hydraulic fluid filter bowl was removed and examined. The input lever and input shafts were severely deformed by impact. The primary and secondary input shafts were found bent external to manifold with the bearing broken and damaged. An arbor press was required to press the primary input shaft from the inside of the secondary shaft. Some damage occurred to the primary and secondary spools as a result of the unit's disassembly and examinations. Based on the hydraulic fluid color (similarities noted in other identifiable component examinations), the PCU was believed to be from the A system.

The servo valve was disassembled. Other than the damage and marks that occurred as a result of disassembly, no other significant damage to the metering edges were noted. Velocity marks on the lands were noted.

4.7.2 B-System Aileron PCU

The unit was identified as p/n 65-44761-21, s/n 4998. The unit was severely impact damaged. The rod end was found broken off from the actuator. The clevis end rod was found bent. The internal manifold was found open to the atmosphere. The primary and secondary servo slides were bent and broken. The bypass valve was free to move but travel was limited as a result of impact related corrosion and contamination.

The PCU servo valve was missing except for the slides noted above. Examinations indicated that the actuator is 40% off neutral in the retracted position. A distance from the manifold to the broken end of the piston (rod end) was measured at 0.495 inches. A distance from the rod end manifold to the piston head was measured at 3.531 inches. Hydraulic fluid removed from the rod end cylinder was amber in color similar to the # 2 flight spoiler (believed to be B system).

4.8 Slat Control Valve

The slat control valve was examined at the direction and in the presence of the systems group members on October 7, 1994, in the Boeing EQA labs at Renton, WA. The following documents the conditions noted during the examinations.

The only identification of the part was the supplier p/n 4163, all other identifying data was destroyed by impact. The unit was in the fully retracted position. The clevis was broken at the lock pin. All ports were open to the atmosphere. The lockwire was cut and the end cap was removed. The slide and sleeve (s/n 775) were removed. No binding was noted.

A witness mark was located 0.766 inches from the lead-in chamfer. The witness mark was 0.490 inches long and parallel to the centerline of the actuator's axis. The valve housing exhibited exterior impact damage. The control slide was broken off at the notch for the control lever bolt attachment. The end cap (1697B5) was frozen in the housing by impact damage and was not removed. The hydraulic fittings were removed what appeared to be burnt oil residue was found inside the ports. Based on this examination, it was determined that the valve was in the normal air-mode position at impact.

4.9 Autopilot Actuators

The autopilot actuators were examined on October 3, 1994, at the direction and in the presence of the systems group members in the Boeing EQA labs at Renton, WA. The following documents the conditions noted during the examinations. Paul Masters and Dennis Tesch of the autopilot actuator manufacturer Abex provided technical support during the examination.

The Boeing 737 airplane incorporates two actuators for autopilot commanded lateral control. Both actuators were x-rayed. X-ray examinations revealed that the detent pistons were retracted and the mod pistons were near center. In this position, the servos would not provide lateral input to the airplane.

4.9.1 Outboard Autopilot Actuator

The outboard autopilot actuator unit was identified as p/n 75130, s/n 3388. The output arm was wrapped around the unit approximately 80 degrees. Approximately 3.25 inches of connecting rod was attached and broken. About 6 inches of the pressure tube was attached but it was twisted and kinked. The return tube fitting was broken off at the o-ring. The o-ring was wedged in the groove. Most of the mounting bracket remained attached. The output was jammed.

The electrohydrostatic valve (EHSV) was examined. The bolts were found attached. The torque motor and cap were missing. Both solenoids were broken off the actuator, the attachment screws were intact. The linear variable displacement transformer (LVDT) coil was broken off above the flange; two of the bolts were sheared. The probe and extension were separated and the armature was missing. The pressure regulator was broken off above the manifold. The filter cap appeared normal. The electrical connector was crushed and pulled away from the housing. The housing was dented and cracked adjacent to the electrical connector.

4.9.2 Inboard Autopilot Actuator

The unit was identified as p/n 75130, s/n 71986. The output arm was broken off. The pressure tube was broken off at the B-nut. The return tube was at the union with threads exposed. The mounting feet were missing. The output was found jammed. The EHSV torque motor was missing. Both solenoids were missing. The solenoid normally closest to the electrical connector had the flange pulled away from the manifold.

The LVDT coil was broken off above the flange. The armature was broken off of the extension. The extension was bent towards the base of the EHSV. The pressure regulator and filter caps appeared to be normal. The electrical connector was broken off at the flange. The mounting feet were broken off the housing.

4.10 Hydraulic Pressure Indicator (Cockpit)

The A and B system hydraulic pressure indicator was examined at the direction and in the presence of the systems group members on October 7, 1994, in the Boeing EQA labs at Renton, WA. The following documents the conditions noted during the examinations.

The Boeing p/n was identified as 10-3223-43, SRDL-0C7E, s/n 810. The indicator was manufactured by US Gauge. The indicator was severely damaged by impact forces. The glass lens was missing along with the system A pointer. The indicator housing was partially crushed. The end piece that contains the electrical connector had been dislodged from the indicator housing. The two servo motors were outside of the indicator housing along with part of their attachment frame. The frame contains the gear train used to drive the indicator pointers. The gear train and servo motors were damaged to the extent that they could not be operated.

The examination of the dial face revealed two witness marks. The witness marks were at the 3,100 psi mark on the dial face. The dial face had also been slightly crushed from the edge at the 3,000 psi mark. The witness marks consisted of white paint on the black background paint. The witness marks in the shape of the end of the pointer were on the flat surface of the dial and a witness mark that had the shape of the tip of the pointer was on the horizontal surface recess in the dial face. The tip of the 'B' system pointer had been impacted.

A third witness mark was observed on the B system pointer. This witness mark was just below the B system pointer. The mark consisted of an impression with particles of white paint. The examination of a new A system pointer revealed that the edge of the white paint used on the back of the pointer coincided with the impression and particles of white paint found on the B system pointer.

4.11 Leading Edge Slat and Flap Actuators

On October 7, 1994, the leading edge slat and flap actuators were examined at the direction and in the presence of the systems group at the Boeing EQA facilities in Renton, WA, in an attempt to verify their position and integrity at impact.

Six leading edge slat actuators were examined. The examination consisted of measurements of the extension of the actuator rod and identification (name plate review). All actuators were found in the extended position.

The #1 slat actuator was identified as p/n 65-44760-10, s/n 0763. Its intermediate piston extended length was recorded as 4.54 inches.

The #2 slat actuator was identified as p/n 65-44760-12, s/n 0532. Its intermediate piston extended length was recorded as 4.27 inches.

The #3 slat actuator was identified as p/n 65-44760-11. The s/n was unknown because of a missing name plate. Its intermediate piston extended length was recorded as 5.00 inches.

The #4 slat actuator was identified as p/n 65-44760-11, s/n 0740. Its intermediate piston extended length was recorded as 4.99 inches.

The #5 slat actuator was identified as p/n 65-44760-12. The s/n was unknown because the name plate was missing. Its intermediate piston extended length was recorded as 4.91 inches.

The #6 slat actuator was identified as p/n 65-44760-10. The s/n was unknown because the name plate was missing. Its intermediate piston extended length was recorded as 4.51 inches.

The leading edge flap (Krueger flap) actuators were also examined. Four actuators were examined. They were all found in the extended position. The following observations were noted:

Identification	Measurement	Bend angle(headend, towards press ports)
YW 163 E470	3.66 inches	90 degrees
YS 634 A99	3.67 inches	110 degrees
YW 724 B06	3.68 inches	unknown
YW 163 F01	4.361 inches	315 degrees

4.12 Ground Spoiler Control Valve

On October 7, 1994, the ground spoiler control valve (manufactured by Sargent) was examined in the Boeing EQA labs at Renton, WA, at the direction and in the presence of the systems group.

The valve housing was impact damaged. The control slide was found broken off at the notch for the control lever bolt attachment. The valve serial number plate was missing.

The end cap could not be removed because it was frozen in the housing. A burnt oil residue was found inside the hydraulic fluid fittings. The valve position was verified as in the air-mode position based on a measurement of 0.484 inches from the valve slide lead-in chamfer to the retaining nut.

5.0 Phase V, Examinations and Testing at Boeing, November 15-18, 1994

During Phase V testing and examination at Boeing EQA facilities on November 15-18, 1994, the following components were tested and observations noted.

5.1 Pilot's and Copilot's Control Columns

On November 16, 1994, the pilot's and copilot's control columns were examined at the direction and in the presence of the systems group in an attempt to verify control position and integrity at impact. Paul Hermanson of Boeing performed the handling and technical support of the identification of the components.

Examination of the pilot's paddle fitting, the copilot's paddle fitting, and the pilot's control wheel remnant all indicate a control wheel position of approximately 40 degrees right at impact.

The drawings that controlled the installation of the columns in the accident airplane were identified as: 65-45126-31 control column installation, and 65-45121-47 and -48 pilot's and copilot's assembly.

The pilot's column was found burnt, the part number plate was unrecognizable. The copilot's part number plate was legible, the last digit was scratched away however a round bottom of the digit was recognizable. The pilot's column had a 9-67622 outer tube attached to the 15-14521 ELL fitting and a 9-48032 dust shield. The lower portion of the 6-84648 torque tube was found inside with the 6-60429 paddle fittings attached. The lower portion of the 65-24190 remained in the outer tube.

The outer tube and torque tube were smashed flat in a fore and aft direction. The 6-60429 paddle was locked in a position that indicated a control wheel position of approximately 40 degrees right aileron at impact.

The upper portion of the 6-84648 torque tube was still attached to the 9-48067 column gear. Both pieces were loose from the lower portion of the column and from the 65-24190 gear housing.

The 65-24190 gear housing was broken away from the outer tube. It contained the 9-49084 pinion gear with a portion of the hub of the pilot's wheel attached. The hub of the control wheel indicated a wheel position of approximately 40 degrees to the right (based on the remnant of the slot in the wheel casting for the wire bundle).

The pilot's control wheel was found broken into four main pieces, including the piece in the gear housing.

Neither the pilot's nor copilot's stick shakers or mounting straps were examined.

The copilot's column was found broken into three pieces. The 15-14521 ELL fitting contained the bottom terminal (6-60488) of the torque tube, the 6-60429 paddle and the 9-48032 dust shield. The paddle fitting was locked in a position that indicated approximately 40 degrees right wheel angle.

The main portion of the 9-67622 outer tube contained the 6-84648 torque tube, without its bottom terminal. The outer tube was folded lengthwise and smashed flat.

The 65-24190 gear housing contained the 9-48067 column gear and was attached to the upper remnant of the 9-67622 outer tube. The pinion gear and column wheel were not examined.

5.2 Standby Rudder Power Control Unit (PCU)

Additional examinations of the standby PCU p/n 1U1150, s/n 1619A, manufactured by Dowty were conducted under the direction of and in the presence of the systems group on November 16, 1994, at the Boeing EQA labs. Tom Redick of Dowty observed the testing and examinations and provided technical support. The following documents the observations of the examinations.

5.2.1 Standby Rudder PCU Input Crank (shaft) Disassembly:

Prior to disassembly, it was noted that all safety wire and manufacturer's security seals were found intact. The input shaft was removed for examination. The input bearing (p/n 1087-22) breakaway torque was measured at 600 in-lb (overhaul manual requirement is 500-600 in-lbs) on disassembly. Visual and low power (less than 40X magnification) optical examination of the input shaft outer diameter surface confirmed the presence of two areas of surface material disturbance. The areas were located at nearly diametrically opposed points approximately 0.2 inches from the shoulder. Each measured approximately 0.25 inches in length with a circumferential orientation. The disturbance was characterized as shallow abrasions combined with regions of smeared material which appear to be raised slightly relative to the adjacent shaft surface. The configuration of the bearing hindered direct examination of the area on the inner diameter surface which mated with the condition noted on the shaft. The condition noted was located in an area that during normal operation is wetted by hydraulic fluid. The input shaft and bearing were submitted to the NTSB materials laboratory for further examination. A separate report is being prepared.

The servo spool/sleeve was removed from the unit and visually examined. There was no evidence of abnormal wear or corrosion. The by-pass spool/sleeve was removed from the unit and visually examined. There was no evidence of corrosion or abnormal wear. The actuator piston was removed and visually examined. There was no evidence of abnormal wear or damage. The housing bore was visually examined. There was no evidence of abnormal wear or damage.

5.3 Main Rudder PCU Hydraulic Fluid Filters

On November 18, 1994, the hydraulic fluid filters installed in the 65-44861-9 main rudder power control unit (s/n 1596A) were removed at the direction and in the presence of the systems group by John Ford (Boeing EQA) for additional visual examination and performance testing to be accomplished at a later date.

The following identification markings were noted on the filters after their removal:
Yaw damper filter: s/n 30498, BAC 10-60808-3, TFS 05228-7500271, FT 5-28-94
B system pressure filter: s/n 14139, BAC 10-60808-4, PTI p/n 7500272, FT 7-18-90
A system pressure filter: s/n 14443, BAC 10-60808-4, PTI p/n 7500272, FT 7-18-90

All filters were identified by vendor code VO 5228 which identifies their manufacturer as Purolator, Aerospace Division, Newbury Park, CA. This information was obtained from Boeing document OHM 27-20-01.

5.4 Autoslat Valve

On November 18, 1994, the 65C26869-2 autoslat valve from the accident airplane was examined at the direction of and in the presence of the systems group. The nameplate identified the part as s/n FAH 0357, assembly date 2-Q-87.

The solenoid near the nameplate was damaged. The housing was torn open. The solenoid was removed. No additional damage was noted. The second solenoid on the opposite side of the housing was torn open. The solenoid's wiring was exposed. The solenoid was removed. There was no other additional damage noted other than normal wear at the backup outer diameter on the pressure to cylinder o-ring.

Impact damage to the valve precluded functional testing of the component. Hydraulic fluid was poured by hand into the C2 port. The fluid flowed out of the C2 ACT port. Hydraulic fluid poured into the C1 port did not flow out of the C2 ACT port. This test indicates that the valve was found in the normal OFF position.

The short slide and sleeve were removed. There was no damage except some dirt from the open port on the sleeve. The spring was intact and appeared to be normal. The long slide and sleeve were removed. There was no damage noted other than dirt from the open port was found on the sleeve. The spring appeared to be normal.

5.5 Wing Leading Edge Slat Actuators

On November 16 and 17, 1994, the wing leading edge slat actuators were examined at the direction and in the presence of the systems group. The actuators were removed from the accident site for additional visual examination. Measurements of the extended lengths of the actuators were taken as an earlier systems group activity. Rex Rhodes from Parker, the slat manufacturer, provided technical assistance in the examinations.

The parts were identified as Boeing p/n's 65-44760-10,11 & 12 depending on their installed positions.

The following identifies the slat actuators relative to their positions on the airplane's wings.

Slat #1, left wing outboard
Slat #2, left wing center
Slat #3, left wing inboard
Slat #4, right wing inboard
Slat #5, right wing center
Slat #6, right wing outboard

The term "intermediate piston" refers to the smaller diameter piston that extends first when commanded by the C1 pressure. The term "primary piston" refers to the larger diameter piston; it does not extend due to a command to C1 pressure. This piston extends at a C2 command.

Marks identified as impact markings are visual observations of physical upset of material on the barrel wall from the back part of the primary piston. Physical evidence of bent or broken pistons indicates extension to the fully extended position.

All actuators indicated damage consistent with impact forces. Four out of six units (#1, #2, #3, and #4 slats) had the blocking valve/manifolds sheared away from the piston barrel assembly at the attach points. On the remaining two units (#5 & #6) the blocking valve/manifold remained attached to the piston/barrel assembly.

Fluid samples were taken from the #5 and #6 slat actuators C2 chamber (from behind the primary piston in the barrel assembly). These hydraulic fluid samples were not tested during the investigation.

5.5.1 #1 Slat Actuator

The #1 slat actuator was identified as p/n 65-44760-10, s/n 0763. The blocking valve and manifold were found separated from the barrel assembly. The intermediate piston was broken off at full extension. The primary piston was fully extended. The full extension position was verified by measurement of an impact mark in the barrel. Impact marks were noted opposite the blocking valve/manifold attach point. A measurement of 2.65 inches was consistent with piston full extension (up against the gland).

The barrel (I.D. YS 438) had a threaded sleeve rework where the lock stud nut retains the lock stud. The rework appeared to be satisfactory.

5.5.2 #2 Slat Actuator

The #2 slat actuator was identified as p/n 65-44760-12, s/n 0532. The blocking valve and manifold were found separated from the barrel assembly. The intermediate piston was found at a full extension position broken and bent over approximately 45° in a direction away from barrel blocking valve/manifold mounting pad. The gland nut was difficult to remove from the barrel possibly because of impact related barrel distortion.

The primary piston was found fully extended in the barrel. The full extension position was verified by measurement of an impact mark in the barrel. The impact mark indicates full extension and was on the side of the blocking valve/manifold attach point. A 2.36 inch measurement was consistent with full piston extension (up to gland within 0.100).

5.5.3 #3 Slat Actuator

The #3 slat actuator was identified as p/n 65-44760-11. The s/n was unknown because of a missing name plate. The blocking valve and manifold were found separated from the barrel assembly mounting point. The intermediate piston was found broken off at full extension. The primary piston was found fully extended in the barrel. The position was verified by measurement of impact marks. The impact marks indicate full extension and are on the side opposite the blocking valve/manifold attach point. A measurement of 2.25 inches was consistent with full piston extension (up against the gland).

The barrel (I.D. YS 438) had a threaded sleeve rework where the lock stud nut retains the lock stud. The rework appeared to be satisfactory.

5.5.4 #4 Slat Actuator

The #4 slat actuator was identified as p/n 65-44760-11, s/n 0740. The blocking valve and manifold were found separated from the barrel assembly mounting point. The intermediate piston was completely broken off at full extension. The primary piston was found fully extended in the barrel. The position was verified by measurement of impact marks. The impact marks indicate full extension and are on the same side as the barrel side mounting pad. A measurement of 2.26 inches was consistent with full piston extension (up against the gland).

The barrel (I.D. YS 438) had a threaded sleeve rework where the lock stud nut retains the lock stud. A thin sliver was raised from but not detached from the reworked sleeve. Otherwise, the rework appeared to be satisfactory.

5.5.5 #5 Slat Actuator

The #5 slat actuator was identified as p/n 65-44760-12. The s/n was unknown because the name plate was missing. The blocking valve/manifold was found attached to the barrel. The blocking valve/manifold was removed from the barrel after the area around the blocking valve was cleaned. Hydraulic fluid was removed in the same manner as noted above.

Approximately 5 cc's of hydraulic fluid was drained from the C2 port. The other two ports were covered. The barrel was then pushed to force the C2 fluid out of the port. The fluid was collected in two clean fluid sample bottles. One bottle was filled with approximately 220 cc's of fluid. The other bottle was filled with approximately 130 cc's of fluid.

The intermediate piston was broken off at a fully extended position. The primary piston was found in a fully extended position. The extension of the primary piston was verified by the measurement of impact marks inside the actuator barrel. The impact marks were found on the same side as the barrel mounting pad. A measurement of 2.36 inches was consistent with full piston extension. The barrel (I.D. YS 438) had a threaded sleeve rework where the lock stud nut retains the lock stud. The rework appeared to be satisfactory.

5.5.6 #6 Slat Actuator

The #6 slat actuator was identified as p/n 65-44760-10. The s/n was unknown because the nameplate was missing. The blocking valve/manifold was found attached to the barrel. Impact forces caused one attach point to break. The mounting screws were difficult to remove. The blocking valve/manifold was removed from the barrel after the area around the blocking valve was cleaned. The unit was clamped across the flats to a bench with the primary piston and the actuator barrel turned up. This caused the blocking valve/manifold attach point to face down.

Approximately 5 cc's of hydraulic fluid was drained from the C2 port. The other two ports were covered. The barrel was then pushed to force the C2 fluid out of the port. The fluid was collected in two clean fluid sample bottles. One bottle was filled with approximately 220 cc's of fluid. The other bottle was filled with approximately 160 cc's of fluid.

The intermediate piston was broken off at a fully extended position. The primary piston was found in a fully extended position. The extension of the primary piston was verified by the measurement of impact marks inside the actuator barrel. The impact marks were found on the same side as the barrel mounting pad. The measurement of 2.65 inches was consistent with full piston extension.

The barrel (I.D. YS 438) had a threaded sleeve rework where the lock stud nut retains the lock stud. A thin sliver was raised from but not detached from the reworked sleeve. Otherwise, the rework appeared to be satisfactory.

5.6 Rudder System Mechanical Components

5.6.1 Rudder Aft Control Quadrant

On November 17, 1994, the rudder aft control quadrant was examined at the direction and in the presence of the systems group in an attempt to verify control position and integrity at impact. Philip Bookout (Boeing flight controls engineer) performed the handling and technical support of the identification of the components.

The rudder aft control quadrant transmits the motion of the rudder control cables to the dual path rudder control torque tube. The assembly consists of a quadrant bolted to a shaft. The shaft is mounted vertically in the horizontal stabilizer. Rotation of the quadrant pushes or pulls the quadrant input rod attached to a crank on the rudder control torque tube.

The rudder aft control quadrant installation consists of a tube, quadrant, and two bearing/houses. The tube was found intact. Small impact marks were noted on the tube. Both ends of the quadrant where the rudder cables attach were found broken off. The two ends were not present with the quadrant. The crank area which connects to the torque tube through a control rod was found intact. About five inches of the control rod was still connected to the crank. The quadrant remained rigidly attached to the tube.

The left side bearing was not attached to the tube and was not present. The left bearing slid freely onto the tube and was not retained in the installation. This apparently prevented damage to the end of the tube.

The right bearing was present but not intact. The inner race was still attached to the tube. The outer race and bearing housing were separated from the inner race. Markings were observed on the nut that retains the inner race and on the bearing housing that indicate the position the aft quadrant was in when the separation occurred. These markings indicated that this angle was 32 degrees of quadrant rotation in the right pedal/surface direction. This translates (according to Boeing engineering documentation) to a rudder position of 23.25 degrees right (full travel is 26 degrees) at separation.

5.6.2 Rudder Jackshaft

On November 18, 1994, the rudder jackshaft was examined in an attempt to verify control position and integrity at impact. The examination was performed at the direction and in the presence of the systems group. Paul Hermanson of Boeing flight controls performed the handling and technical support of the identification of the components.

The Boeing drawing effectivity of the installation was identified as 5-97614-3003 (pilot's) and -3004 (copilot's) installations. The pilot's jackshaft assembly was identified as 5-97613-3019. The copilot's jackshaft assembly was identified as 5-97613-3020.

5.6.2.1 Pilot's Components

The jackshaft assembly had the inner races of the bearings attached at each end. Foreign-impact related material found packed in the bearing races indicated that both bearings came apart at impact. There was no pieces of structure or outer races attached. The cross tube (9-47362) was missing. Its clevis on the jackshaft was found broken off. One remnant of a clevis ear indicated that the cross tube was pulled away to the right. The rear ends of both 6-58993 push rods were still attached. The remnants of both tube ends remained attached by rivets to the rod ends.

Most of the jackshaft assembly was coated in a dark substance. There was a paint scrape and impact mark on the edge of the 5-63067 shaft assembly, just above the lower bearing, oriented right of center (clockwise as seen from above) approximately 45 degrees from the front. The presence contamination indicated that the damage was done during the impact. A scratch was observed on the opposite end of the shaft assembly just below the upper bearing. The scratch appeared clean and was not covered with the dark substance.

The left arm of the 65-7208 yoke was found bent towards the shaft centerline. The arm nearly contacted the shaft. The right arm did not appear to be bent. The yoke was found clocked approximately 5 degrees left of horizontal as viewed from the rear.

The 65C10041 quadrant was found in two main pieces. The cable groove end was found separated from the assembly. The hub end was still attached to the 5-63067 shaft although the rivets had been sheared and the quadrant hub had rotated about the shaft approximately 45 degrees to the right moving the quadrant arm aft. The quadrant arm was twisted.

5.6.2.2 Copilot's Components

The 65C10041 Quadrant arm was found in one piece. The arm and cable groove were bent. The hub was smashed. The hub still contained the upper portion of the 5-63037 shaft. The rivets holding the arm to the shaft were found sheared. The rivet holes indicate that the arm moved up about 1.5 inches relative to the shaft and was rotated aft (counterclockwise) approximately 5 degrees. The upper bearing was missing.

The 5-63067 shaft was found with the lower bearing and 3-74664 bearing retainer attached. The clevis for the cross tube was found broken off. The arms of the 65-7208 yoke were not noticeably bent and contained the 9-47382 push rod ends.

5.6.3 Rudder Control Torque Tube

On November 17, 1994, the rudder control torque tube was examined at the direction and in the presence of the systems group in an attempt to verify rudder control position and integrity at impact. Philip Bookout of Boeing flight controls performed the handling and technical support of the identification of the components.

The rudder control torque tube provides a dual load path for rudder control linkage inputs. The torque tube consists primarily of two aluminum tubes bonded and swaged together. A dual-load-path plug forms each end of the torque tube and is double-riveted in place. The torque tube is mounted on two bearings retained by two nuts installed on each dual plug. The tube is mounted in a vertical position in the vertical fin. Three cranks are bolted to the tube. The lower crank is connected to the input rod from the aft control quadrant and to the rudder feel and centering mechanism. The center crank is connected to the main rudder power control unit linkage. The upper crank is connected to the standby power control unit linkage. The cranks are of dual construction with the two halves bonded and riveted together. Rudder pedal input causes the torque tube and cranks to rotate. This provides input to the rudder PCU, the feel and centering unit, and the standby actuator.

Only the lower crank of the torque tube was present. The crank remained attached to the torque tube which was broken off approximately 1.5 inches above the crank. All four bolts which hold the crank to the tube were tight and the inspection putty was in place.

The crank has two arms. One receives input from the cockpit and the other attaches to the rudder feel and centering unit. The end of the arm (clevis) that attaches to the feel and centering unit was broken off. The clevis was found attached to the control rod from the feel and centering unit. The other arm was partially broken. There was an impact mark which corresponded to the fracture of the arm. A large impact marking was found opposite the two arms which appeared to be the result of contact with a bolt (thread indentations were present). The position of this marking at impact is consistent with the crank being twisted and breaking the tube. Another impact mark on the opposite side of the first (between the arms) and at the top of the crank, indicated contact with the torque tube bearing housing. This provided additional confirmation of the direction at which the crank was broken off from the torque tube.

5.6.4 Rudder Pedal Assemblies

On November 17, 1994, the rudder pedal assemblies were examined at the direction and in the presence of the systems group in an attempt to verify rudder control position and integrity at impact. Paul Hermanson of Boeing flight controls performed the handling and technical support of the identification of the components.

The pilot and copilot are each provided with a pair of rudder pedals used for controlling the airplane about the directional axis. Each pair of pedals consist of right and left pedals mounted on a shaft. The pedal shaft is attached to the upper end of the pedal arm assembly. The lower end of the pedal arm assembly is mounted on a support shaft attached to structure below the floor. Fore and aft movement of the pedals is transmitted by the two pushrods to the jackshaft yoke. The rotary motion of the jackshaft yoke is passed to the forward quadrant by means of the jackshaft. The two sets of rudder pedals are bussed together by means of a bus pushrod connecting the two jackshaft assemblies.

A majority of the major pieces of the four rudder pedal arms were recovered and examined. The parts were identified and grouped. All of the 65-33517 arm assemblies were broken in half approximately at cockpit floor level. All 6-60402 brake rod assemblies were missing. All clevises on the 65-38243 bellcranks and on the 65-24573 pedals were broken off. The 65-38243-8 (copilot's right foot) bellcrank was missing. All of the 65-24573 rudder pedals had most or all of the pan broken off; only the hub remained.

All of the 65-33517 arms were broken in a fore and aft direction, except for the pilot's right foot arm which was broken in a side-to-side direction. The portion of the pilot's right rudder arm that was broken below the floor, connected to the bellcrank was broken in a fore-and-aft direction. The portion above the floor was broken in a side-to-side direction.

The rudder pedal shafts, 69-26660, were broken off at the upper arm terminal (69-26599) on both the pilot's and copilot's left foot pedals. Both right foot pedals remained attached. Both of the right foot pedal shafts were bent forward approximately 20 degrees.

The pilot's left foot pedal was bent forward approximately 20 degrees before separation. The copilot's left foot pedal shaft was bent forward less than 5 degrees before separation. The entire remaining upper portion of the copilot's left foot rudder arm was fire damaged. The Boeing Materials Specification (BMS) 5-16 plug in the pedal shaft was missing. The upper portion of the pilot's left foot rudder arm was fire-damaged over its lower half. The upper portion of the pilot's right foot rudder arm was covered with a dark substance. It did not appear to be burned. All remaining parts were not burned.

The pilot's main arm shaft, 69-6801, was found broken in half. Each half was locked in its corresponding arm bottom fitting. The copilot's arm shaft was also broken in half. The right foot side rotated freely in the bottom fitting. The left foot side was missing.

All of the forward clevises of the four 6-58993 pushrods remained attached to the arm bottom fittings. The pushrod tube remnant of the pilot's left foot arm was found jammed forward covering the shank of the rod end fitting. All of the other three pushrod tube remnants remained riveted to their respective pushrod fittings. Splintered wood was found jammed into the copilot's right foot pedal (65-24573). Sheet metal from the arm support structure was found crumpled and still attached to the pilot's right foot arm pivot.

5.6.5 Rudder Feel and Centering Unit

On November 18, 1994, the rudder feel and centering unit was examined at the direction and in the presence of the systems group in an attempt to verify rudder control position and integrity at impact. This examination was a continuation of the examination described in section 4.1 of this report. Philip Bookout of Boeing flight controls performed the handling and technical support of the identification of the components.

It was noted that the manufacturer's inspection putty was intact on all nuts. The dual load path bolt which connects the spring assembly to the roller arm was removed. The spring assembly was then removed. The lockwire on the roller arm pivot bolt was cut in two places. A single swipe mark was noted on the rudder panel right side of the cam. There were multiple cam to roller marks on the rudder panel left side. These multiple marks were located where the feel and centering unit came to rest.

The shaft assembly was pulled and the cam removed. The position of the swipe mark was determined to equate to a rudder position of approximately 10 degrees right rudder by use of Boeing engineering drawing documentation.

6.0 Systems Group Hydraulic Fluid Sampling and Testing

This section describes the collection and testing of hydraulic fluid samples taken from the accident airplane's components. All samples were collected by or in the presence of the systems group.

Hydraulic fluid samples were taken at Boeing from the following components:

Standby rudder power control unit
Main rudder power control unit
Left wing flight spoilers (# 2 and 3)
Left wing ground spoilers
Both aileron PCU's

6.1 Standby rudder power control unit

Two fluid samples were drained from the unit into glass containers on September 20, 1994, at the Boeing EQA labs. The collection was performed by Boeing technicians under the direction of and in the presence of the systems group. One sample which consisted of a total of about 10 cc's was removed from the unit's pressure and return hoses.

The second sample was taken by cycling the main actuator piston to remove the fluid sample. Approximately 45 cc's were drained into a glass container. The samples were sealed and shipped to the attention of Joe Gardina of Monsanto in St. Louis, MO, via Federal Express. Monsanto reported that they received the samples on September 29, 1994. The samples were tested by Monsanto on September 30, 1994; the systems group did not participate in the testing at Monsanto.

6.2 Flight Spoilers, Ground Spoilers, and Aileron power control units.

Flight Spoiler #2 and #3 fluid samples were obtained on October 4 and 5, 1994, by removing the anti-cavitation check valve plug and cycling the actuator piston. Fluid was drained into glass bottles, labeled and sealed by Boeing EQA technicians.

There were no fluid samples taken from ground spoilers #0, #1, inbd #4 or outbd #4.

Two samples were taken on October 6, 1994, from each aileron PCU. Both PCU's were sampled using the same procedures. One sample was taken from the filter bowl. The second sample was taken by removing the piston end cap and end bearing and pouring the fluid from the housing/manifold. All four samples were labeled and sealed by Boeing EQA technicians.

The flight spoiler and aileron samples were not tested because of accident related (open hydraulic lines, fittings, and body housings) contamination of the fluid contained within the units.

6.3 Main Rudder PCU

The following hydraulic fluid samples were taken at Parker. The samples were identified for testing as:

#1 A Pressure and Return Line
#2 B Pressure and Return Line
#3 A System Filter
#4 B System Filter
#5 Yaw Damper Filter
#6 Link Cavity (collected with syringe)
#7 Link Cavity (poured)
#8 A system Manifold from Bench
#9 B system Manifold from Bench
#10 Parker Test Bench

All fluid samples from the main rudder PCU, return hoses, and the B system pressure hoses. (samples #1 and #2) were collected into glass bottles by Roff Sasser of the NTSB on September 21,1994. The A system pressure filter cap was cleaned with solvent and blown dry. The A system filter cap was removed, the filter visually checked for signs of debris (none was noted), the fluid in the filter cavity was drained into a glass container by pouring the fluid out (sample #3). The A system cap was reinstalled. The B system filter cap was cleaned using solvent and blown dry. The cap was removed and the filter was visually checked for signs of debris (none noted), and the fluid in the cavity was drained into a glass container by pouring the fluid out (sample #4). The B system yaw damper filter cap was cleaned using solvent and blown dry. The cap was removed, the filter was visually checked for debris (none noted), the fluid in the filter cavity was drained into a glass container by pouring the fluid out (sample #5).

The link cavity cover was cleaned with solvent and blown dry with air. The cover was removed and the fluid in the cavity was removed with a laboratory sealed syringe and placed into a glass container (sample #6). The remaining fluid in the link cavity was poured into another sample glass bottle (sample #7). A plug was removed from the link cavity for a drain hose to be installed. The link cavity cover was reinstalled. The bent piston was removed and replaced with a new piston in order to run the unit on the bench. The unit was then mounted on the test bench. A system was pressurized and the initial return fluid was drained into a glass sample bottle (sample #8). B system was pressurized and the fluid that filled the link cavity was drained into a glass sample bottle (sample #9). A sample of fluid from Parker's test bench was also taken (sample #10).

Samples #3, #4, #5, and #7 were provided to Parker for analysis by their Quality Assurance. These samples were tested on 9/23/94. During the analysis the samples were drawn through filter patches for particle typing.

All remaining samples collected were tested by Monsanto and the test results are reported in a separate report to this investigation prepared by the NTSB materials laboratory.

[redacted]

Gregory Phillips
Aerospace Engineer
National Transportation Safety Board

12/21/94

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NATIONAL TRANSPORTATION SAFETY BOARD

Washington, D.C.

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT
APPENDIX 1

Main Rudder Power Control Unit
Test Data

ENGINEERING INVESTIGATION

MAINTENANCE AGREEMENT ☐ YES ☒ NO ____ DAY ADVANCE NOTIFICATION ☐ YES ☒ NO

REPAIR ORDER #	CUSTOMER NAME	CUSTOMER CODE	DATE	AIRCRAFT
7047188-100	USAir	008010	09/21/94	737

QTY.	P/N REC'D / PARKER P/N	CHG. LTR.	SERIAL #	PART NAME	CARD INITIATED BY
1	P/N REC'D 65-44861-9 PARKER P/N 65-44861-9	T	1596A	RUDDER	RD (STAMP)
	ASSY QTR. 1 Q87	☐ TSO ☐ TSR ☐ TSN ☒ TT 21077.33 HOURS		WARRANTY CLAIM ☐ YES ☒ NO	REP/OHL ESTIMATE REQ'D ☐ YES ☒ NO
QTY.	P/N REC'D ____ PARKER P/N ____	CHG. LTR.	SERIAL #	FINAL INSPECTION REQ'D ☐ GSI ☒ FAA ☐ CUST	DUE DATE 10/14/94

REASON FOR RETURN (REJECTION CODE 470)

☐ REPAIR ☐ OVERHAUL ☐ TEST/CERT. ☒ OTHER

NTSB INVESTIGATION.

INITIAL ACTION REQUIRED

☐ FLUID SAMPLE REQUIRED ☒ SEE M.D.R.

☐ FUNCTIONAL TEST/INSPECT TO CONFIRM REASON FOR RETURN

☐ INVESTIGATION

☐ OTHER

TO BE WITNESSED BY ☐ QC ☐ ENG ☐ DCAS ☐ CUSTOMER

CONDITION AS RECEIVED (INCLUDING LRU'S)

☐ DAMAGED OR BROKEN HARDWARE. REPORTED TO Q.E./ENG.

DIRTY, SHIMMED ON EXTERNAL SUMMING LEVER. LOCKWIRE SEALS ARE CSO. TORQUE STRIPE ON SERVO DOES NOT ALIGN. THE ELECT. CONN'S & PORTS WERE COVERED. L/W ON SERVO IS REVERSE.

25130 S/N 33193

GM1032 S/N 15928, P/N 59600-5003 S/N CY5532

INSPECTED BY ____ STAMP ____ DATE 9-21-9

CFE AS RECEIVED

4 TUBES

3 BRACKETS

7047188-100

RECEIVING TEST AND/OR INSPECTION RESULTS

☐ SEE CONTINUATION SHEET FOR ADDITIONAL RESULTS

TEST / INSPECTION BY ____ STAMP ____ DATE ____

WITNESSED BY

QC ____ DATE ____ DCAS ____ DATE ____

ENG ____ DATE ____ CUST ____ DATE ____

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Form 1162-3 (6-89) ☐ AIR ☐ FUEL ☐ HYD ☐ PRM CC-89-2377

Observation of P/N 65-44861-9, S/N 1596A, US Air 737 Rudder PCU Examination on September 21, 1994.

	Visual Inspection
10:50 - 11:05	Test 11B pin to pin resistance test performed - see test data sheets.
11:05 - 11:15	Clean Filter Cap with Isopropyl Alcohol
11:20 - 11:35	Untorque Filter Cap (Air cleaned Filter Cap) Cleaned sealant around the filter cap with knife & air cleaned.
11:40 - 11:45	Drained fluid out of lines - John Calvin & Roff Sasser A System hoses (2) in one beaker B System hoses (2) in another beaker
11:50	Removed system A filter cap P/N 59120. A System - Filter P/N 10-60808-4 inspected, no large particles- drained fluid around filter. O-ring/backups visually inspected, no deficiencies. Reinstalled cap on A System Filter.
12:00	Removed system B filter cap P/N 59120. B System - Filter P/N 10-60808-4 inspected - no large particles - drained fluid around filter. O-ring/backups visually inspected, no deficiencies. Reinstalled cap on B System Filter.
12:04	Removed system B electrical filter cap P/N 59119. System B - Electrical System - Filter P/N 10-60808-3 inspected - no large particles. Drained fluid around filter. O-ring/backups visually inspected, no deficiencies. Reinstalled cap on B Electrical System Filter.
12:06	Lunch. Decision was made to replace bent piston rod & summing lever to perform functional ATP. The following is disassembly and assembly of piston.
12:45	Removed brace - Tim Hurley Removed Screw NAS334CPA40, 68093 Washer & HL73-8 Collar Removed P/N 66-22727-1 Bolt, NAS679A6 Nut & AN960PD616 Washer (between Summing Lever P/N 69-35567-1 & H-link 69-35563-1). Had to use C-Clamps, punch, hammer & press to remove the 66-22727-1 Bolt.
1:16	Removed Screw - P/N NAS333CPA25-5, Nut HL73-6 & Washer 68094. Removed Bolt 66-22728-1 & NAS679A5 Nut, (between piston rod & summing lever). Had to use hammer, punch & clamp to remove 66-22728-1 Bolt.
1:20	Removed Screw NAS334CPA32, Collar HL73-8 & Washer 68093. Removed Bolt 66-22727-2, Nut NAS679A6, Washer AN960PD616. (between H-link and 69-35566 Lever), had to use hammer & punch to remove 66-22727-2 Bolt.
1:20	Removed H-link & Summing Lever
1:25	Cleaned end gland area with alcohol (going to try to trap fluid as it come out when we remove the Aft Nut P/N 69-35540-1 & Gland P/N 69-35533-1.) Q-tips were used in the cleaning processes. Cleaned out debris from outside of gland area.
1:34	Installed in assy fixture.

1:37	Break torque on Aft Nut P/N 69-35540-1 with a special tool. Approximately 200 ft-lbs untorquing.
1:45	Broke Redundant Rod loose, removed Redundant Rod Nut NAS679A6 & Washer MS20002-6, removed outer Redundant Rod Nut BACN10BY510 & Washer MS20002-10.
1:48	Removed PCU from assy fixture, and reinstalled upside down so trapped fluid had a path of free flow, out of the way of solenoid valve.
1:50	Removed (rod end side) End Gland P/N 69-35533-1 & Retainer P/N 69-35534-1, trapped fluid in cylinder cavity.
1:51	Removed Aft Piston Assy 65-44867-1 and Rod End Assy 69-35535-1. Trapped fluid in cylinder cavity, (System B & A). Looked inside with a borescope - John Calvin / Roff Sasser. Hard anodize intact. No evidence of impact marks or abnormal wear marks.
2:00	Reinstalled new aft piston assembly, rod end assembly, end gland, retainer, nut, summing lever, H-link, snubbing ring, & associated nuts, bolts and washers.
2:20	Unit back together. Installed pressure & return fittings on to match test setup.
2:30	Discussion
3:00	Cleaned cover plate area on PCU.
3:30	Removed Cover Plate Assy P/N 69-54772-24 and associated nuts, bolts and washer. There appeared to be shiny minute metallic particles in the linkage cavity fluid.

Looked into cavity with TV monitor

Magnified and video recorded microscopic metal particles in fluid in linkage cavity.

System B fluid taken out of leakage cavity. Shipped to Monsanto for evaluation.

Measured the of gap between the servo valve external stops (on P/N 83311 Servo Housing), and the Primary Summing Lever P/N 69-35602-1, on both the Retract and Extend sides. This was done with pin gages. (Lever position was secured in Pittsburgh, in hanger).

a) Extend, left rudder
.132
b) Retract, right rudder
.090

(Inspection on 9/22/94 Servo stop to stop gap measured .681 & Primary Rudder stop width measured .461).

Observation of P/N 65-44861-9, S/N 1596A, US Air 737 Rudder PCU Examination on September 22, 1994.

7:10 - 7:35	Discussion
7:35 - 8:05	Set up 0 psi spring force test with video. Removed cover Plate Assy P/N 69-54772-24 and associated nuts, bolts and washers.
8:08-8:19	Supply pressure = 0 psi - Pull scale on the input test, pulled input toward forward end, extending PCU; left rudder command. Removed shims, (put in place in hanger). Moved primary thru bias spring to secondary pick up point by moving input lever toward forward end (rudder extend) - primary moved into the servo body without friction or binding, force measured at .55 - 1.00 lbs. Moves back toward aft end (right rudder) when input was released without friction or binding. (bias spring). (PCU is designed with a bias spring in the servo to take out backlash between drive ball & primary slide). (Primary bias spring has primary slide up against the secondary pick-up point in the PCU retract direction, primary slide out of servo body). By design can not perform pull scale test in the PCU retract direction without moving secondary.
8:30	Video recorded force required to break secondary bias spring extend and retract. Extended PCU by pulling input lever until secondary, detent spring was compressed and secondary slide moved. Extend PCU (left rudder) Input lever pulled toward forward end, 7.0 & 7.5 lbs to break secondary detent spring - smooth, no binding. Retract PCU (right rudder) Input toward aft end, 5.5 & 5.25 to break secondary detent spring - smooth, no binding. Special cover plate was installed.
8:45	Discussion
9:50	Unit set up on test stand, video recorded. (Special cover has been installed.)
10:05	Removed fluid from System A. (Sent to Monsanto). Pressurized System B, cycled to remove fluid from System A. No fluid came out return port. Pressurized System A and collected fluid out of return port of System A. Special cover was put on to see the linkage & servo valve's reaction to pressure. Everything worked smoothly. When the unit was cycled the linkage and servo valve acted normally. Pressure at 360 psi (Note: Had to remove Plug P/N 69-35625-1 when pressure was applied with special cover so return fluid could drain out).
10:15	Removed special cover plate, replaced with U.S. Air Cover Plate P/N 69-54772-24.

10:38 Attempted rudder reversal with hardover command simulating full pilot input rate.
Put supply pressure at 3000 psi to unit.
Applied full rate command to input link, no reversal in either retract or extract direction. This was done with mechanical input only.
Hardover right rudder - no piston reversal.
Hardover left rudder - no piston reversal.
(both primary & secondary slides were cycled). Motion of input lever is smooth, no binding.

10:45 12E. Rig Neutral Cylinder Stroke and Clearance Test
(Refer to test data sheets). Passed.

10:50 12F. Linkage Breakout Friction
(Refer to test data sheets). Passed.

10:54 12M. Transducer Null
(Refer to test data sheets). Passed.

10:56 12L. Transducer Output
(Refer to test data sheets). Passed.

10:58 12N. Yaw Damper Authority
(Refer to test data sheets). Passed.

11:05 12T. Yaw Damper Engage
(Refer to test data Sheets). Passed.

11:10 12S. Phase Check
(Refer to test data sheets). Passed.

11:15 12Q. Yaw Damper System Phase Lag
(Refer to test data sheets). Passed.

11:19 12R. Yaw Damper System Repeatability and Linearity
(Refer to test data sheets). Passed.

11:26 Special Yaw Damper Velocity Authority Test
(Rate of ram with full step signal to yaw damper).

11: 55 Lunch

12:30 Special Yaw Damper Velocity Authority Test
(See special data sheet) 50 degrees/sec.

12:55 12G. Input Force vs Input Travel
(Refer to test data sheets). No Pass, Secondary Slide picked up on extend side sooner than allowed by .002.

12:58 Discussion Gary Schaul to finish testing.
12K. Intersystem Leakage - Passed, 12J. Internal Leakage Yaw Damper On - Passed.
12J. Internal Leakage Yaw Damper Off, & 12U. Bypass Test - Passed.
Removed PCU from test fixture and install on assembly fixture.

2:40 Removed Cover P/N 69-54772-24. Examined linkage cavity with a borescope/monitor.
Secondary internal summing lever contacting the servo external stop (83311 Servo Housing). The secondary internal summing lever ~~does~~ GP comes into contact with the servo external stop, both retract and extend side, no abnormalities found. Pulled up on secondary lever to simulate worst case - secondary slide still made contact with stops, no abnormalities, both retract and extend direction.

3:20	Video taped lockwire on Servo Valve Nut 83344. Lockwire installed backwards. Torque stripe 83344 Nut - current position as received.
3:25	Torqued 83344 Nut clockwise to see if there is any motion. Over 600 in-lbs. No motion.
3:27	Untorqued 83344 Nut. Retorqued to torque stripe, 575 in-lbs. (new torque stripe).
3:30	Removed 83344 Nut - Trapped fluid in back of servo cavity. Removed 83347 Spring, 68173 Bias Spring, 68171 Guide & 68170 Pin. Inspected back of servo cavity with lights/monitors. Looked at 68045 Guide and 68021 Nut to see if nut is lined up in scallops on guide: corner of nut is flush with scallop point. Had contact. Picture taken of back of servo valve.
3:45	Disassembly of 737 US Air Rudder PCU (Enough to remove Servo Valve 68010-5003)
3:55	Checking torque of 68021 Nut. Torque: Around 200 in-lbs. Removed 68021 Nut and 68045 Guide Inspected 69-35607-1 Segment Cam with monitor. Wear mark found on shaft at ball end. This is not an abnormal wear mark. Checked the following parts with the monitor:

69-35605-1	Segment - no anomalies
69-35613-1	Fork Lever - no anomalies
69-35608-1	Seat Cam - no anomalies
69-34502-1	Lever Assy Primary - no anomalies
69-35603-1	Lever Assy Secondary - no anomalies - Chamfer is small stop engagement is good.
68045	Guide - Some deformation on corner of scallop cuts. (2 out of 5 corners).
68021	Nut - no anomalies
83347	Spring - no anomalies
68046	Guide - normal wear pattern
83311	Housing - no anomalies - normal/good contact points/ engagements.

Ref. dimensions measured of primary slide and servo housing.

(69-35602	Primary	Stop	.461	ref. only B/P .463 +/- .001)
(83311	Housing	Stops	.681	ref. only B/P .680 +/- .001)

4:35	Moved 68010 Servo Valve to servo test area.
4:40	Looked at 83349 Primary Slide & 68010-15 Secondary Slide Assembly under microscope. 83349 Primary Slide - no anomalies
4:45	Flushed return 'communication' hole in 83311, nothing flushed out, therefore, no blockage and no differential pressure across the secondary slide. FAA & Parker witnessed shinning of light through hole satisfactory apparent, no restriction were noted.

5:15 Set up 68010-13 on a video borescope. Looking at the metering edges and flow holes.
Checking for burrs/erosion.
System A 1st return metering edge (C retract to Return)
Slight erosion
System A 1st pressure metering edge (P to C retract)
Slight erosion
Stop inspection to test 68010-5003
Used microscope and borescope to inspect the 68010-15 metering edges & flow holes. No anomalies found.
Took reference measurements on I.D. of 68010-15 .2496 to .24965
Took reference measurements on I.D. of 68010-13 .7498 to .000025.
Both referenced dimensions within acceptable limits.
68010-15 Spring Slot (for 83348 Spring Tang)
Had burrs kicked up during assembly/disassembly.
Had to remove burrs with stone.

5:55 Assembly of the 68010-5003 S/N 2956.
Installed 68010 Servo into servo test fixture.
Test 2 Secondary Detent, Stroke and Internal Bypass
(from 68010-5005T)
Test 2B
See test data sheets.

Observation of P/N 65-44861-9, S/N 1596A, US Air 737 Rudder PCU Examination on September 23, 1994.

8:00 Cut lockwire to remove Yaw Damper Assy. Disassembled enough to remove Yaw Damper Assy P/N 69-35609-6. Yaw Damper components removed and inspected with the monitor.
69-35609-2 Lap Assy (without the 69-35611 Sleeve)
No anomalies
69-35611 Sleeve
No anomalies
59188-3 Diaphragm
No anomalies
10-60810-1 Transducer Assy
No anomalies
59174-5 Cap
No anomalies
Removed Parker's aft piston, end gland, retainer, nut, summing lever, H-link and rod end assy. Installed US Air piston, end gland, retainer, rod end assy and nut. Hand tightened Aft Nut P/N 69-35540.
Summing Lever and H-Link not reinstalled. All open cavities in US Air PCU were plugged and all disassembled hardware sent with PCU.
Tested 68010-5003 US Air Servo Valve.
See test data sheets.
Test on servo P/N 68010-5003 passed all 68010-5003 ATP and -5005 ATP Test 2A except:
Flow Gain - Failed in the servo extend direction. Overlap between primary and secondary extends out of envelop.
Primary Friction - Failed - .5 ozs. too high.

Servo Test Fixture: 68010TF5
S/N L5036
Calib: 3/24/94

Rudder Test Fixture: TF83300
SN: L5029
Calib: 2/24/94

Summary Conclusions

- Testing and examinations conducted on the Rudder Power Control Unit validated that the unit is capable of performing its intended functions as specified by BCAG.
- Testing validated that the unit was incapable of uncommanded rudder reversal, or movement.
- The Yaw Damper System components of the unit functioned normally and the yaw deflection limit of +/- 3 degrees was verified.
- Subcomponent performance variations noted during testing did not affect overall PCU function.

[redacted]

9/23/94

Parker **TEST DATA RECORD**

Customer Support Operations

Rev. G

Drawn [redacted] Date 4-20-93

Test Approval [redacted] Date 4-20-93

Eng. Approval [redacted] Date 4-29-93

Overhaul Manual 27-20-01 ☐ Overhaul ☑ In-Service Work Order 704-7188-100

PART NO. 65-44861-9/65C37052- **SERIAL NO.** 1596A **DATE OF TEST** 9-21-94

PART NAME RUDDER ACTUATOR PCU **INSPECTOR** [redacted]

☒ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
11.A. Continuity Check	Check Continuity per wiring diagram (Fig. 403, Fig. 403A).	OK		
11.B. Pin to Pin Resistance FRC JMC Jaw TL	Check pin to pin resistance at 70°F and record. Must comply with noted valve.		PAS 74 EFT	

Pins	Resistance	Sys "A"	Sys "B"
1-2	71-87 OHMS (Solenoid PN 59600)		78
1-2	49-62 OHMS (Solenoid PN 45080 @ 77°F)		
1-2	79-115 OHMS (Solenoid PN 45080-1 @ 77°F)		
5-6	900-1100 OHMS		1007
7-8	900-1100 OHMS		1008
1-4	0 OHMS (Short)		—
9-10	80-165 OHMS		104
11-12	60-135 OHMS		84
3-Other	Infinite (No Connection)		∞

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
11.C. Dielectric Strength	Slowly apply noted voltages and hold for a period of 5 seconds (Fig. 403), 1 min (Fig. 403A). There must be no arcing or insulation failure.			
	1500 VAC Body to Pin 2 (Fig. 403)	OK		
	1500 VAC Body to Pin 1 (Fig. 403A)	OK		
	1000 VAC Body and Pins 2,5,7 connected to common lead to Pin 9 (Fig. 403)	OK		

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PART NO. 65-44861-9/65C37052-	SERIAL NO. 1596A	DATE OF TEST 9-21-94
PART NAME RUDDER ACTUATOR PCU	INSPECTOR [redacted]	

☒ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
11.C. Dielectric Strength (Continued)	1000 VAC Body and Pins 2,5,7 connected to common lead to Pin 11 (Fig. 403)	______ OK		
	1000 VAC Body and Pins 1,5,7 connected to common lead to Pin 11.	______ OK		
	1000 VAC Body and Pins 1,5,7 connected to common lead to Pin 9.	______ OK		
	800 VAC between Pins 5 and 7	______ OK		
	800 VAC between Pins 9 and 11 (Fig. 403A)	______ OK		
11.D. Insulation Resistance	500 VDC between noted pins 100 Megohms min resistance 10 megohms min in-service. (Fig. 403)			
	Pins	Sys "A" Sys "B"		
	1,5,7,9 & 11 to Body	______ ______		
	1 to 5,7,9 & 11	______ ______		
	5 to 7,9, & 11	______ ______		
	7 to 9 & 11	______ ______		
	9 to 11	______ ______		

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PART NO. 65-44861-9/~~65C37052-~~	SERIAL NO. 1596A	DATE OF TEST 9-22-94	
PART NAME RUDDER ACTUATOR PCU		INSPECTOR [redacted]	
☒ PHOSPHATE ESTER	☐ MIL-H-5606	☐ MIL-H-83282	☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.B. Return Pressure	No external leakage or permanent set, each test. No intersystem leakage.	3000 3 ± 2 Sys A _____ _____ Sys B _____ _____		
12.C.3 & .4 Secondary Stroke/Flow Test	Measure and record leakage at open ports Ra and Rb. The unit shall not move after 25 lbs. is applied to lever TF83300-53, and leakage at each return port must not exceed the following: (a) 300-700 cc/min for overhauled unit. (b) 300-1085 cc/min for unit in service.	Extend (12.C.3) Ra_____cc/min Rb_____cc/min Retract (12.C.4) Ra_____cc/min Rb_____cc/min Movement_____in.		
12.D. Proof Pressure	No external leakage or permanent set, each test. No intersystem leakage.	5400 3 ± 2 Sys A _____ _____ Sys B _____ _____		
12.E. Rig Neutral Cylinder Stroke and Clearance Test	Output Rig Pin must fit. (27.46 - 27.54 Ref.) No binding or interference at 3000 PSI. Stoke 26° ± ½° each direction. Visual snubbing. No binding or interference at 0 PSI.	Rig Pin OK Surface Indicator at 0° 0° Binding Yes No Ext 25.39 Ret 25.30 Snubbing ✓ OK ✓ OK	530 27 ETO	

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Parker **TEST DATA RECORD** (CONTINUATION) 56

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Overhaul Manual 27-20-01 ☐ Overhaul ☑ In-Service Work Order 704-7188-10C

PART NO. 65-44861- /65C37052-	SERIAL NO. 1596A	DATE OF TEST 9-22-94	
PART NAME RUDDER ACTUATOR PCU		INSPECTOR [redacted]	
☒ PHOSPHATE ESTER	☐ MIL-H-5606	☐ MIL-H-83282	☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.F. Linkage Breakout Friction	19 oz. maximum to start extend. 1 oz. minimum in extend direction to start retract.	Ext. 11 Oz. Ret. 2 Oz.	CSO 27 FTO	
12.G. Input Force vs. Input Travel	Force plot must fall within limits shown on Fig. 723.	✓ OK	CSO 27 FTO	
12.H. Cylinder Rod Leakage	1 drop/25 cycles max at each End Gland (In-Service 1 drop/5 cycles) 2 drops/25 cycles max. at Center Gland (In-Service 2 drops/5 cylces) 1 drop/100 cycles max at Input Shaft (In-Service 1 drop/25 cycles)	Rod Gland ____ drops Aft Gland ____ drops Center Gland ____ drops Input Shaft ____ drops		
12.J. Internal Leakage	(1) Rig neutral leakage at RA & RB. 300 cc/min - overhaul. 3000 cc/min - In-Service. (2) & (3) Input lever at extend & retract stops 300-700 cc/min - overhaul. 300-1085 cc/min - In-Service.	Neutral RA 185 RB 190 RA / RB Ext. 400 430 Ret. 420 350	CSO 27 FTO	

Parker TEST DATA RECORD (CONTINUATION) 57

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Overhaul Manual 27-20-01 ☐ Overhaul ☒ In-Service Work Order 704-7188-100

PART NO. 65-44861-9/65~~C37052~~-	SERIAL NO. 1596A	DATE OF TEST 9-22-94
PART NAME RUDDER ACTUATOR PCU	INSPECTOR [redacted]	

☒ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.J. Internal Leakage (Continued)	(4) 65-44861 and 65C37052 System A energized 1370 cc/min - overhaul 2000 cc/min above leakage measured in step (1) (b) In-Service	Neutral RA ____ RB ____		
	(5) 65-44861 and 65C37052 System A energized 1370 cc/min - overhaul 2000 cc/min above leakage measured in step (1) (b) In-Service	RA / RB Ext. ____ ____ Ret. ____ ____		
	(6) 65-44861-5 thru -9,-11 and 65C37052-5 thru -9 System B energized 1370 cc/min - overhaul 2000 cc/min above leakage measured in step (1) (b) In-Service	RB Neutral 680		
	(7) 65-44861-5 thru -9,-11 and 65C37052-5 thru -9 System B energized 1370 cc/min @ RB 2000 cc/min above leakage measured in step (1) (b) @ RB for In-Service	RB Ext. 630 Ret. 690		

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Overhaul Manual 27-20-01 ☐ Overhaul ☒ In-Service Work Order 704-7188-10C

PART NO. 65-44861-9/65C37052-	SERIAL NO. 1596A	DATE OF TEST 9-22-94
PART NAME RUDDER ACTUATOR PCU	INSPECTOR [redacted]	

☒ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.K. Intersystem Leakage	Combined leakage from PB and RB 10 cc/min maximum.	Ø cc/min	27	
12.L. Transducer Output	4.2-4.8 VAC 65-44861-2 and 65C37052-2 (Sys A & B) 1.95-2.55 VAC 65-44861-3,-4 and 65C37052-2,-3,-4 (Sys A & B) 1.95-2.55 VAC 65-44861-5,-8 and 65C37052-5,-8 (Sys B) 3.07-3.67 VAC 65-44861-6,-7,-9,-11 and 65C37052-6,-7,-9 (Sys B)	"A" Extend ____ VAC Retract ____ VAC "B" Extend 3.58 VAC Retract 3.48 VAC	27	
12.M. Transducer Null	50 MV maximum at null for each system - Overhaul. 150 MV maximum at null for each system In-Service.	"A" Null ____ VAC "B" Null .010 VAC	27	

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Parker **TEST DATA RECORD** (CONTINUATION)

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Overhaul Manual 27-20-01 ☐ Overhaul ☑ In-Service Work Order 704-7188-100

PART NO. 65-44861-9/65C37052- **SERIAL NO.** 1596A **DATE OF TEST** 9-22-94

PART NAME RUDDER ACTUATOR PCU **INSPECTOR** [redacted]

☒ **PHOSPHATE ESTER** ☐ **MIL-H-5606** ☐ **MIL-H-83282** ☐ **AIR** ☐ **PD680**

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.N. Yaw Damper Authority	Per Figure 1, Ref. Test Data Sheet 10. Actuator must be stable and within .050" of neutral.	Ext. Ret. Sys "A ____ Sys "B" .239/.232 Sys "A & B" ____ Stable Yes No Position from "ZERO" ____ Ext. Ret. Stable Yes No Position from "ZERO" ____ Ext. Ret.		
12.P. Manual Hysteresis 4.3.14	Hysteresis shall not exceed .004 inch each direction - Overhaul. .006 inch each direction In-Service	Extend .0 Inch Retract .001 Inch		
12.Q. Yaw Damper System Phase Lag	Phase shift to be: 25 degrees (Sin Ø, 0.423) 30 degrees (Sin Ø, 0.500) for in-service units. No crossover in plot.	"A" sin Ø ____ "B" sin Ø .270 Crossovers: Yes No		

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Parker **TEST DATA RECORD** (CONTINUATION)

Customer Support Operations

Rev. G

Overhaul Manual 27-20-01 ☐ Overhaul ☑ In-Service Work Order 704-7188-100

PART NO. 65-44861-9/65G37052- SERIAL NO. 1596A DATE OF TEST 9-22-94

PART NAME RUDDER ACTUATOR PCU INSPECTOR [redacted]

☑ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.R. Yaw Damper System Repeatability and Linearity	Pattern must repeat within 0.8" (.008" at actuator) - Overhaul. 1.2" (.012" at actuator) In-Service	"A" ____ in max "B" .230 in max		
	Overshoot after reversal must be 0.2" maximum (.002" at actuator)	"A" ____ in max "B" .100 in max		
	Average input/output slope of any 10% segment must fall within the slope limits shown.	"A" Yes No "B" Yes No		
12.S. Phase Check	Manual control "extend", actuator must extend.	Extend "A" ok "B" ok		
	Manual control "retract", actuator must retract.	Retract "A" ok "B" ok		
	Operation smooth and stable for A and B.	Stable: Yes No		
12.T. Yaw Damper Engage 4.3.17	.004" maximum movement-overhaul .010" maximum movement-In-Service.	"A" ____ in "B" .004 in		
12.U. By-Pass Valve Operation	(4) No piston movement at 250 PSI differential for either system.	"A" Yes No "B" Yes No		
	(6a) Noticeable decrease in flow at less than 460 PSI differential.	"A" 390 PSI "B" 370 PSI		
	(8) Piston rod movement of 1.00 inch or more in either direction.	Ext. ✓ OK Ret. ✓ OK		

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Parker **TEST DATA RECORD** (CONTINUATION)

Customer Support Operations

Rev. G

Overhaul Manual 27-20-01 ☐ Overhaul ☒ In-Service Work Order 704-7188-100

PART NO. 65-44861-9/65C37052- SERIAL NO. 1596 A DATE OF TEST 9-22-94

PART NAME RUDDER ACTUATOR PCU INSPECTOR [redacted]

☒ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

TEST & REF.	REQUIREMENT	RESULTS	ACC	REJ.
12.V. Duty Cycle (optional) Not Required for In-Service Unless Actuator Seals Were Replaced	3.0 cc/8 hours maximum at each piston rod seal. 6.0 cc/8 hours at center gland - Overhaul. 1.8 cc/hr maximum at each piston rod seal. 3.6 cc/hr at center gland - In-Service.	Output End _____cc_____hrs Fixed End _____cc_____hrs Center _____cc_____hrs		
12.W. Low Pressure Leakage (not same as 12.V.)	No external leakage in 8 hours - overhaul. No external leaks in one hour In-Service.	_____cc_____hrs		

Parker

TEST DATA RECORD (CONTINUATION)

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Customer Support Operations

Rev. G

Overhaul Manual 27-20-01 ☐ Overhaul ☒ In-Service Work Order 704-1596-100

PART NO. 65-44861-9/65C37052- SERIAL NO. 1596A DATE OF TEST 9-22-94

PART NAME RUDDER ACTUATOR PCU INSPECTOR [redacted]

☒ PHOSPHATE ESTER ☐ MIL-H-5606 ☐ MIL-H-83282 ☐ AIR ☐ PD680

OVERHAULED UNIT			
PCU Assembly	Actuator Output Stroke (inches)		
	System A	System B	System AB
65-44861-2 65C37052-2	0.294 to 0.334	0.294 to 0.334	0.335 to 0.375
65-44861-3,-4 65C37052-3,-4	0.137 to 0.177	0.137 to 0.177	0.294 to 0.334
65-44861-5,-8 65C37052-5,-8	--------	0.137 to 0.177	--------
65-44861-6,-7,-9,-11 65C37052-6,-7,-9	--------	0.215 to 0.255	--------

IN-SERVICE UNIT			
PCU Assembly	Actuator Output Stroke (inches)		
	System A	System B	System AB
65-44861-2 65C37052-2	0.274 to 0.354	0.274 to 0.354	0.315 to 0.394
65-44861-3,-4 65C37052-3,-4	0.117 to 0.197	0.117 to 0.197	0.274 to 0.354
65-44861-5,-8 65C37052-5,-8	--------	0.117 to 0.197	--------
65-44861-6,-7,-9,-11 65C37052-6,-7,-9	--------	0.196 to 0.276	--------

Actuator Output Stroke Limits
Figure 1

DRAWN: 10-2-90 APPR: 10-2-90 TEST APPR: 10-2-90

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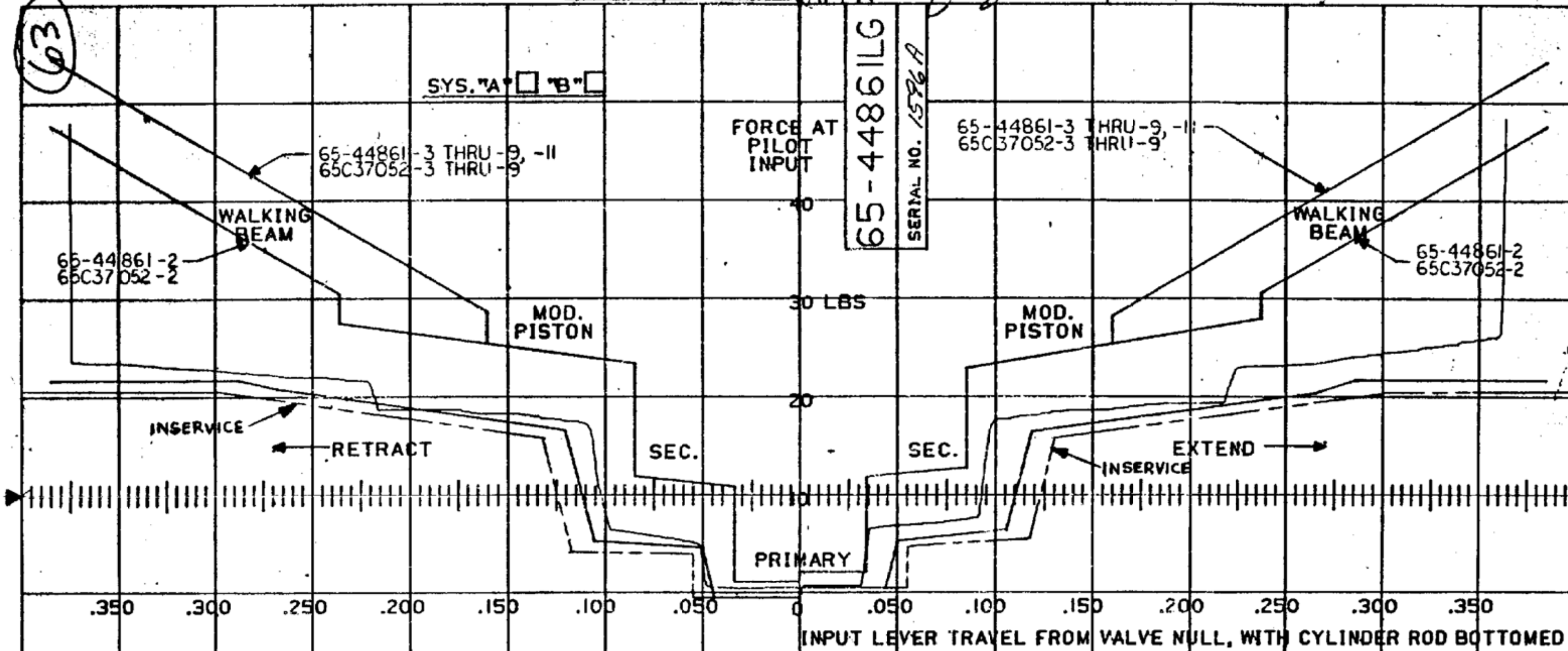


INPUT FORCE VS INPUT TRAVEL
TEST 12G

PAGE 8 OF 10 ACCEPTANCE TEST DATA SHEETS				
DATE TESTED: 9-22-94	SIZE B	BERTEA CORPORATION IRVINE, CALIFORNIA		
INSP: [redacted] 704-7188-100	FSCM NO. 82106	65-44861LG (1)		REV G
PART NO. & DASH NO. 65-44861-9		PROJECT 737	PAGE 743	

OHM 27-20-01

DRAWN: *[signature]* DATE: 10-2-80 ENG. APPR: *[signature]* DATE: *10-2-80* TEST APPR: *E. Cooper* DATE: *10-2-80*

(64)

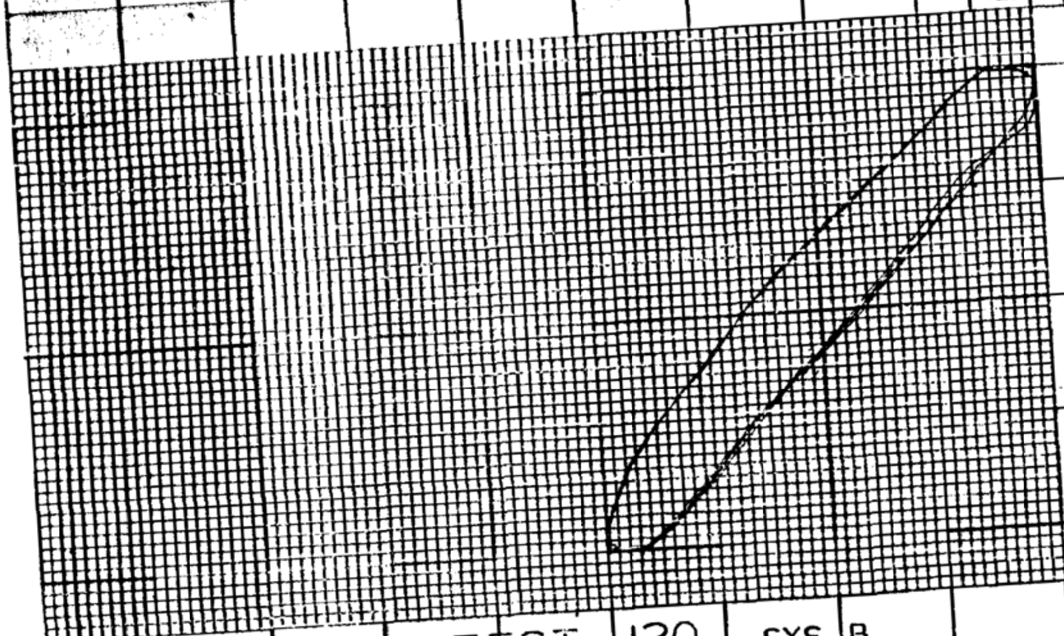
SINE Ø MAXIMUM PER TEST REQ'MT .423 OVERHAUL .500 INSERVICE

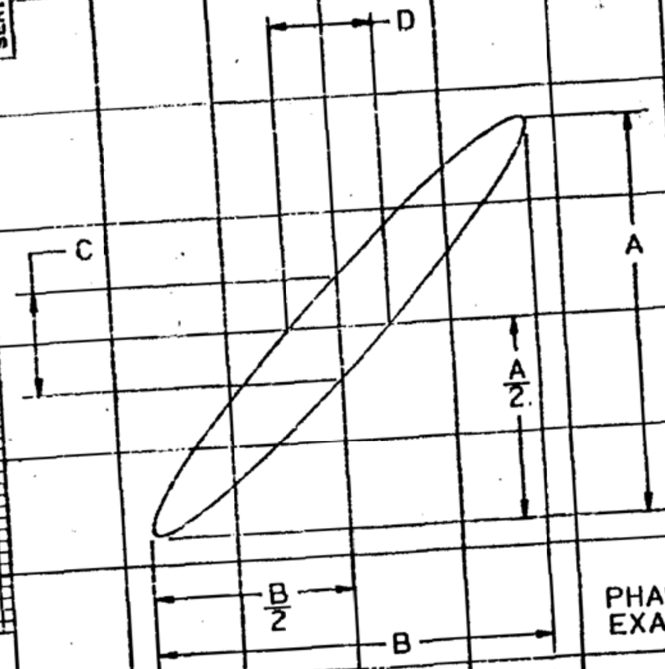
65-44861LG

SERIAL NO. *1596A*

$2\sqrt{\frac{C}{A} \cdot \frac{D}{B}}$ = SINE Ø

CALCULATION





PHASE LAG EXAMPLE

TEST 12Q

PHASE LAG

SYS A	SYS B
SINE Ø .	SINE Ø *.270*

(-5 ASSY, 1 SYSTEM ONLY)

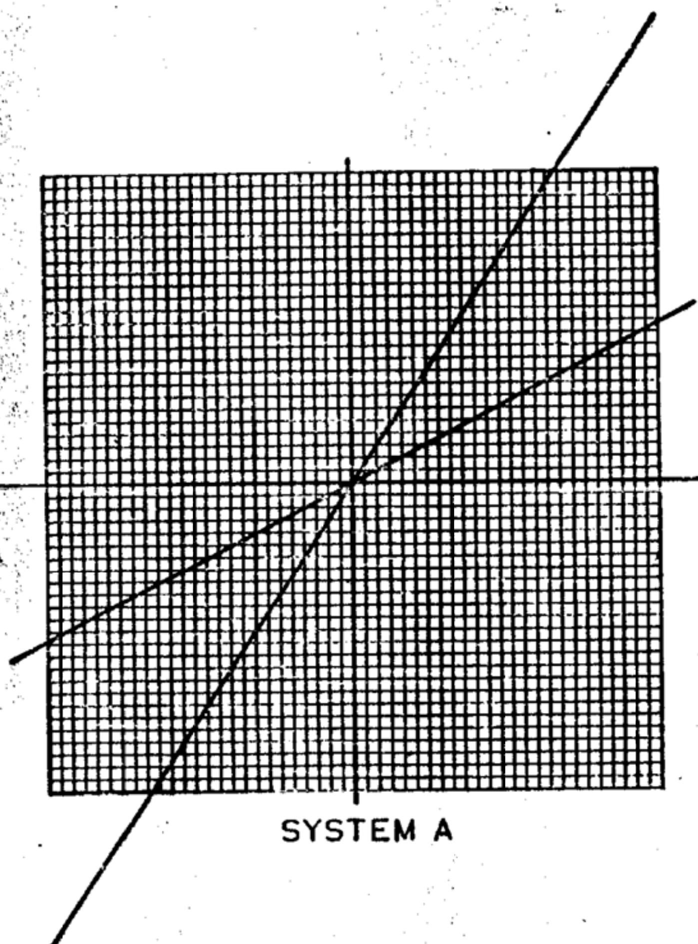
PAGE 9 OF 10 ACCEPTANCE TEST DATA SHEETS

DATE TESTED: *9-22-94*	SIZE B	BERTEA/CORPORATION IRVINE, CALIFORNIA	
INSP: [redacted] *704-7188-100*	FSCM NO. 82106	65-44861LG (2)	REV G
PART NO. & DASH NO. *65-44861-9*		PROJECT 737	PAGE 750

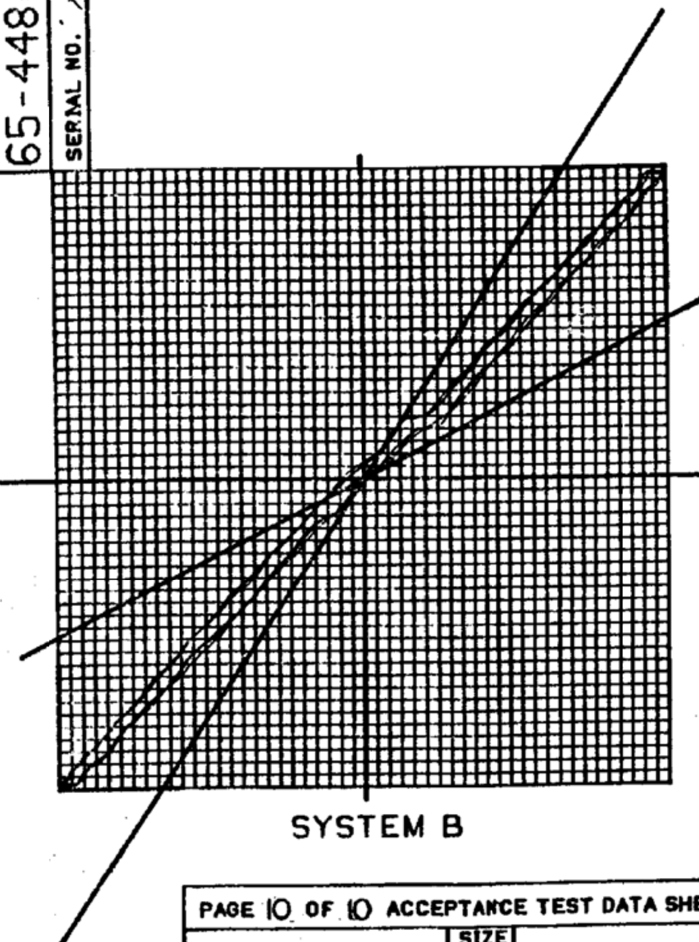
OHM 27-20-01

DRAWN: DATE: 10-2-80 ENG. APPR: DATE: 10-2-80 | TEST APPR: DATE: 10-2-90

65-44861LG SERIAL NO. 1596A



SYSTEM A



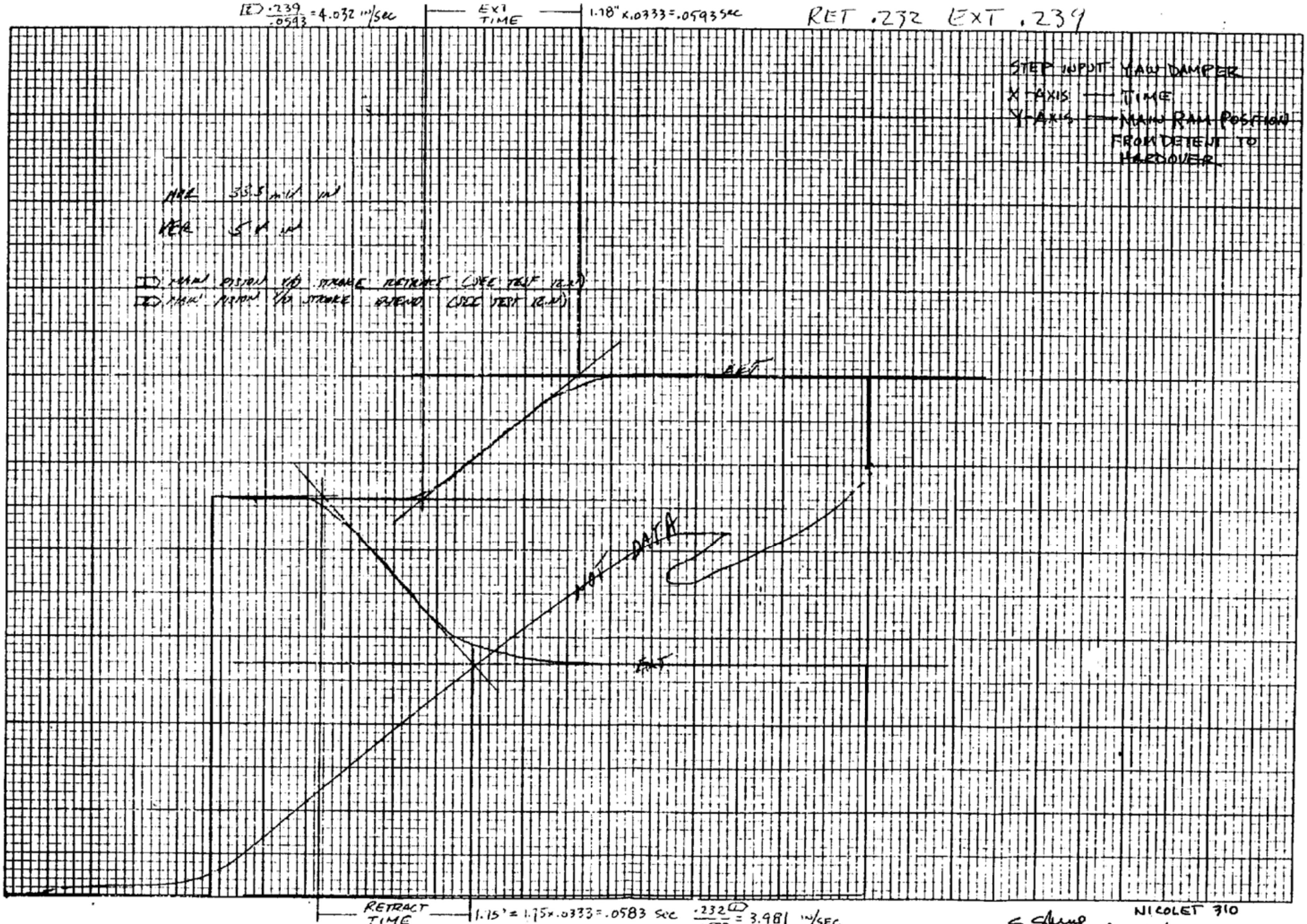
SYSTEM B

TEST 12R
REPEATABILITY AND LINEARITY
(-5 ASSY, 1 SYSTEM ONLY)

PAGE 10 OF 10 ACCEPTANCE TEST DATA SHEETS			
DATE TESTED: 9-22-94	SIZE B	BERTEA/CORPORATION IRVINE, CALIFORNIA	
INSP: [redacted] 27 704-7188700	FSCM NO. 82106	65-44861LG (3)	REV G
PART NO. & DASH NO. 65-44861-9		PROJECT 737	PAGE 751

OHM 27-20-01

RET .232 EXT .239



S. Sheng 9.22.1994

NICOLET 310

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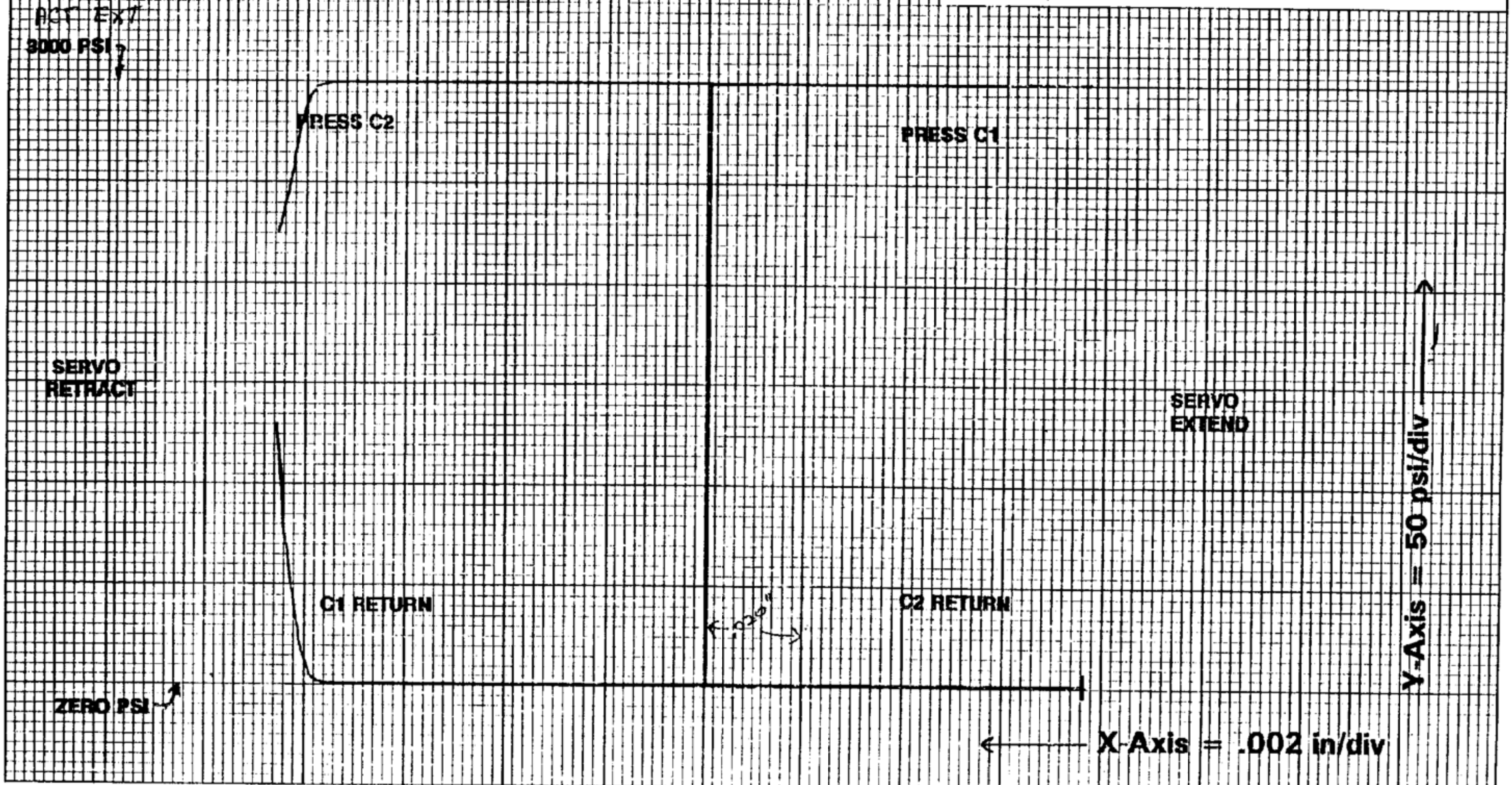
SYS B

USAIR FLT 427 SYSTEMS GROUP TEST

J/N		Oil Temp	
P/N	68010-5003	Absolute Visc. (CP)	
S/N	2956	Reason For Rejection	
Scale		Verified By	
Test		1st Rejection	
Assy		2nd Rejection	
Trim		3rd Rejection	
Lap			
Hone			
Date	9/22/91		

RECEIVING TEST

SYSTEM 1



47 0703

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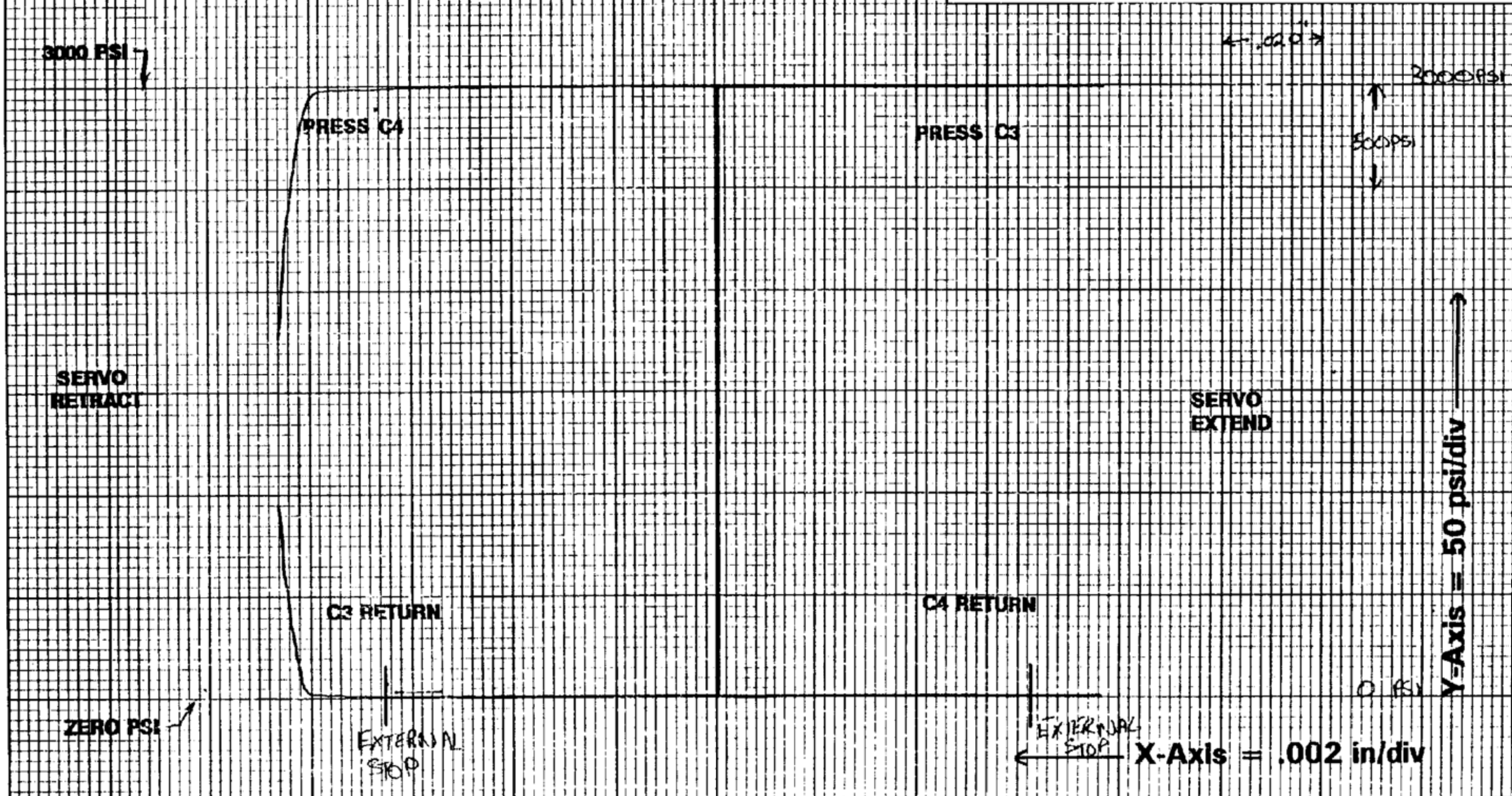
SYS A

USAIR FLT 427 SYSTEMS GROUP TEST

RECEIVING TEST

SYSTEM 2

J/N		Oil Temp	
P/N	68010-5003	Absolute Visc. (CP)	
S/N	2956	Reason For Rejection	
Scale		Verified By	
Test		1st Rejection	
Assy		2nd Rejection	
Trim		3rd Rejection	
Lap			
Hone			
Date	9/22/94		



47-0703

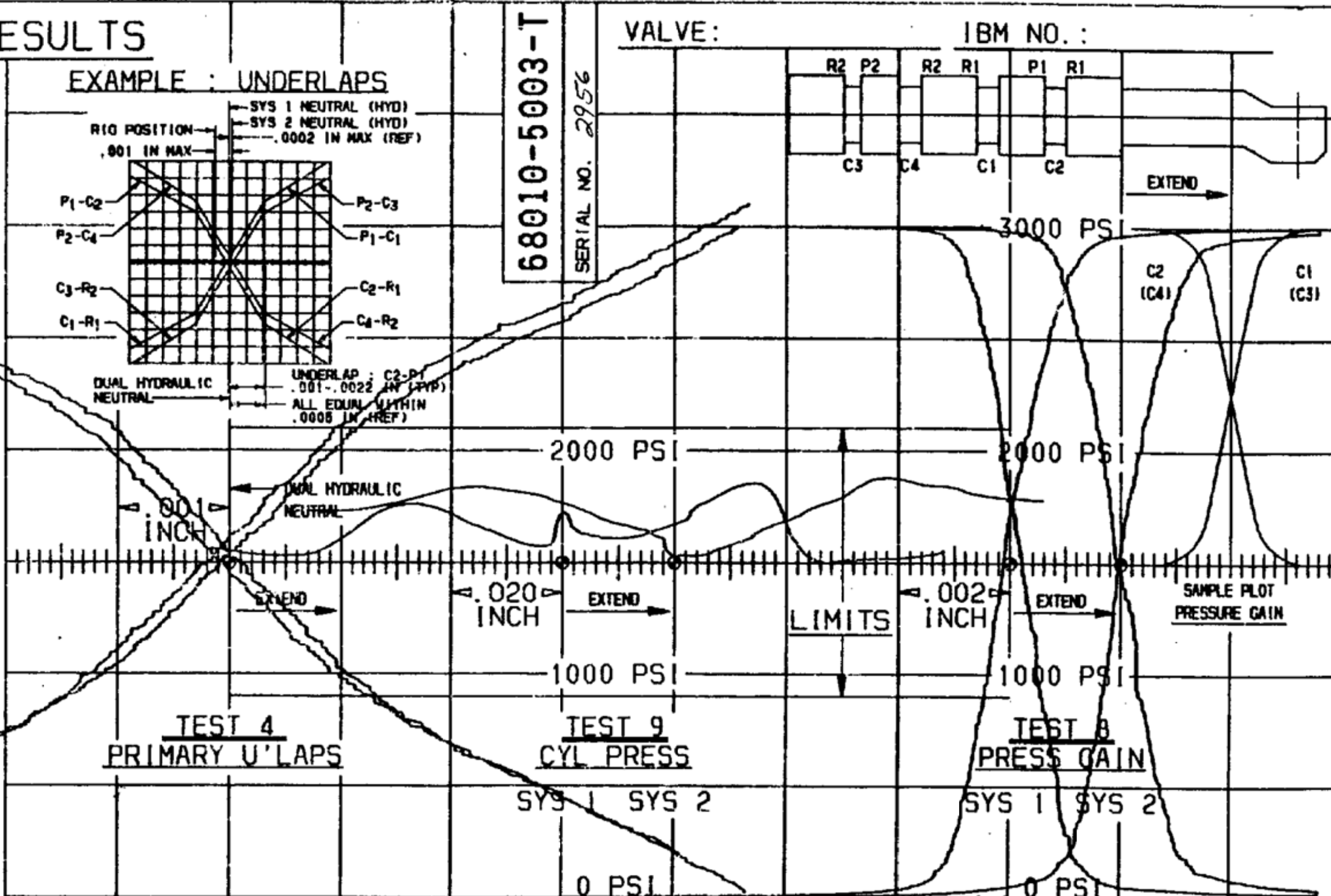
TEST FLUID: TYPE IV ☐ MIL-H-5606 ☒

INSPECTION TEST RESULTS

1. PROOF ______ OK
2. A. DETENT POSITION .001 IN.
   B. SLIDE TRAVEL EXTEND .063 IN.
   RETRACT .064 IN.
3. SECONDARY SLIDE FRICTION AND DETENT SPRING FORCE:
   EXTEND 5 LB RETRACT 6 LB
   RETURN 4 LB RETURN 4.5 LB
   DIFF. 1 LB DIFF. 1.5 LB
4. PRIMARY SLIDE UNDERLAPS ✓ OK
   NEUTRAL POSITION SYS 1 .0005 IN.
   (FROM RIG) SYS 2 .0006 IN.
5. NEUTRAL CYLINDER PRESSURE (PSI):
   C1 1775 C2 1775 C3 1500 C4 1500
6. NEUTRAL FLOW (NOT CORRECTABLE):
   SYS 1 186 CC/MIN SYS 2 190 CC/MIN
7. FULL STROKE LEAKAGE:
   SYS 1 EXT 28 CC/MIN; RET 25 CC/MIN
   (HYJET 132CC/MIN)
   ⚠5 CORRECTED 41 CC/MIN; 36 CC/MIN
   (MIL-H-5606 90CC/MIN)
   SYS 2 EXT 36 CC/MIN; RET 38 CC/MIN
   (HYJET 132CC/MIN)
   ⚠5 CORRECTED 52 CC/MIN; 56 CC/MIN
   (MIL-H-5606 90CC/MIN)
8. PRESSURE GAIN ✓ OK
9. CYLINDER PRESSURE ✓ OK
10. FLOW GAIN: SYS 1 ______ OK SYS 2 ✓ OK
    EXTEND "FLAT" ______ OK ✓ OK
    RETRACT "FLAT" ✓ OK ✓ OK
    SATURATION
    FLOW DIFF. ______ OK ✓ OK
11. PRIMARY FLOW FORCE:
    EXTEND 4 OZ RETRACT 9 OZ
12. PRIMARY SLIDE FRICTION:
    RETRACT 12.5 OZ EXTEND 3 OZ
13. CYLINDER PHASING:

		C1	C2	C3	C4
EXTEND	PSI	3075	0	3050	0
	IN	.001		.001	
RETRACT	PSI	0	3200	0	3000
	IN		.001		.002

14. TEST ACCEPTANCE
    ACPT ______ REJ ______



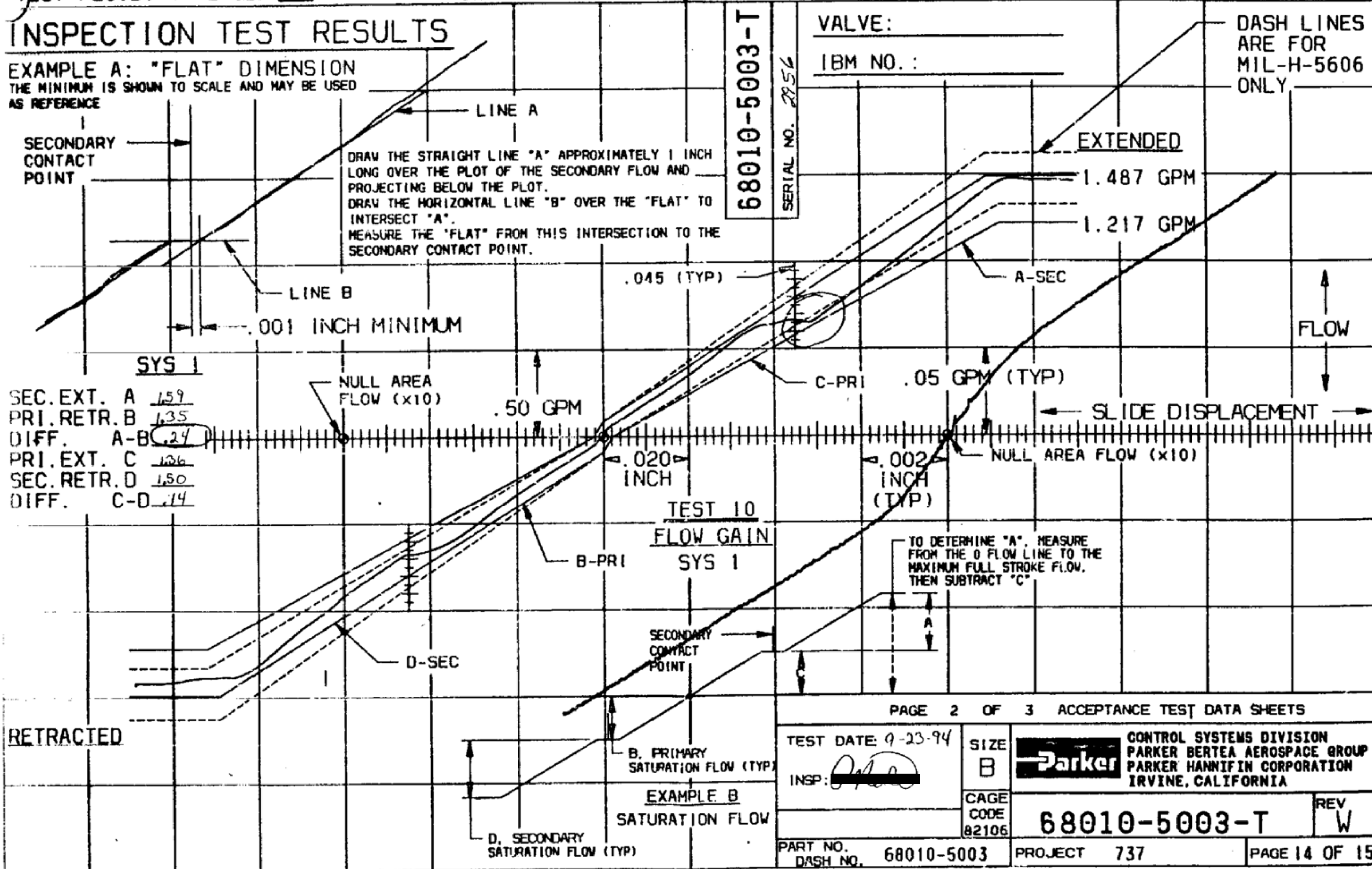
PAGE 1 OF 3 ACCEPTANCE TEST DATA SHEETS

TEST DATE: 9-23-94 INSP: [redacted]	SIZE B	Parker	CONTROL SYSTEMS DIVISION PARKER BERTEA AEROSPACE GROUP PARKER HANNIFIN CORPORATION IRVINE, CALIFORNIA
	CAGE CODE 82106	68010-5003-T	REV W
PART NO. DASH NO. 68010-5003		PROJECT 737	PAGE 13 OF 15

CATIA - CAD CAM SYSTEM CODE W

TEST FLUID: TYPE IV [ ] MIL-H-5606 [X]

INSPECTION TEST RESULTS



SYS 1

SEC. EXT. A 1.59
PRI. RETR. B 1.35
DIFF. A-B .24
PRI. EXT. C 1.36
SEC. RETR. D 1.50
DIFF. C-D .14

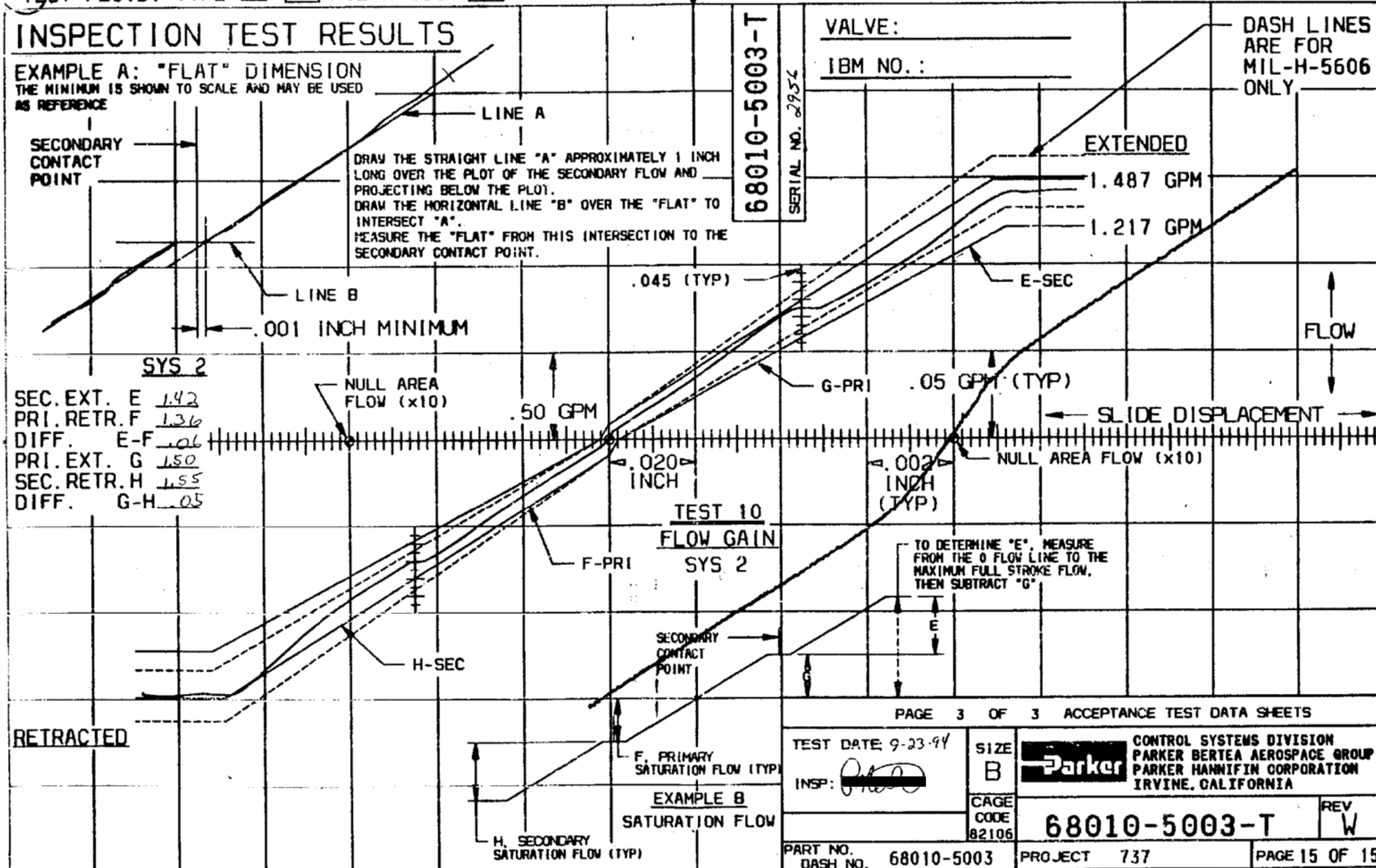
PAGE 2 OF 3 ACCEPTANCE TEST DATA SHEETS

TEST DATE: 9-23-94 INSP: [redacted]	SIZE B	Parker	CONTROL SYSTEMS DIVISION PARKER BERTEA AEROSPACE GROUP PARKER HANNIFIN CORPORATION IRVINE, CALIFORNIA
	CAGE CODE 82106	68010-5003-T	REV W
PART NO. DASH NO. 68010-5003		PROJECT 737	PAGE 14 OF 15

CATIA - CAD CAM SYSTEM CODE W

TEST FLUID: TYPE IV [ ] MIL-H-5606 [X]

INSPECTION TEST RESULTS



SYS 2

SEC. EXT.	E	1.42
PRI. RETR.	F	1.36
DIFF.	E-F	.06
PRI. EXT.	G	1.50
SEC. RETR.	H	1.55
DIFF.	G-H	.05

PAGE 3 OF 3 ACCEPTANCE TEST DATA SHEETS

TEST DATE: 9-23-94

INSP: [redacted]

SIZE B

CAGE CODE 82106

Parker

CONTROL SYSTEMS DIVISION
PARKER BERTEA AEROSPACE GROUP
PARKER HANNIFIN CORPORATION
IRVINE, CALIFORNIA

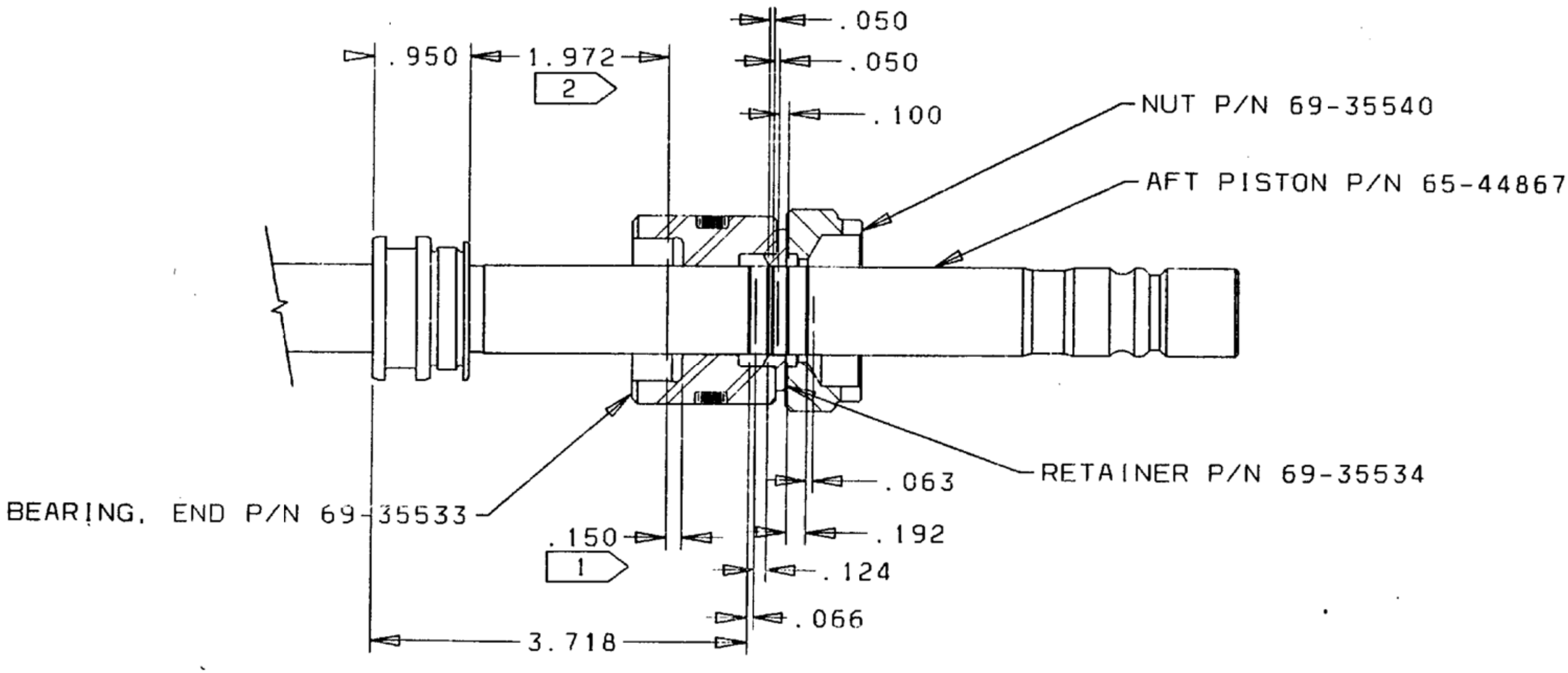
68010-5003-T REV W

PART NO. DASH NO. 68010-5003 | PROJECT 737 | PAGE 15 OF 15

CATIA - CAD CAM SYSTEM CODE W

$\boxed{1}$ $\sin^{-1}\left(\frac{.150}{4.5}\right) = 1.9$ DEGREES RETRACT (RIGHT RUDDER)

$\boxed{2}$ NEUTRAL POSITION OF PCU (0 DEGREES SURFACE)



BOEING CORPORATE OFFICES SEATTLE, WA 98124	SIZE B	FSCM NO 81205	DWG NO USAIR FLT 427 RUDDER PISTON ANALYSIS		
			P CLINE	10/13/94	SCALE 1

BOEING PROPRIETARY SEE PAGE 1 FOR DETAILS

NATIONAL TRANSPORTATION SAFETY BOARD

Washington, D.C.

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT
APPENDIX 2

Control Column Fracture Examination Report

NATIONAL TRANSPORTATION SAFETY BOARD

Office of Research and Engineering
Materials Laboratory Division
Washington, D.C. 20594

October 19, 1994

METALLURGIST'S FACTUAL REPORT Report No. 95-4

A. ACCIDENT

Place	: Aliquippa, Pennsylvania
Date	: September 8, 1994
Vehicle	: Boeing 737-3B7, N513AU
NTSB No.	: DCA 94-M-A076
Investigator	: Greg Phillips, AS-40

B. COMPONENTS EXAMINED

1. First officer's control column lower end shaft, P/N 65-54206-6, with a separation adjacent to the splines.
2. Five pieces of control cables.

C. DETAILS OF THE EXAMINATION

1. CONTROL COLUMN SHAFT

An overall view of the control column shaft is shown in figure 1. The shaft was separated near its lower end at the position indicated by arrows "1" in figure 1. A view of the fracture area on the major portion of the shaft is shown in figure 2. Visual examinations of the fracture faces revealed that slightly less than one half of the fracture surface contained smearing damage and that the remainder of the fracture was matte with an irregular texture, indicative of overstress separations in aluminum alloys. Examination of the fracture faces with a scanning electron microscope (SEM) revealed that the lower face of the fracture was obscured by deposits that were minimally affected by ultrasonic cleaning in acetone and then in a chromic phosphoric acid cleaning solution. The SEM examination of the upper fracture face revealed the presence of scattered deposits. However, the presence of ductile dimples were noted in the portions of the fracture that were matte in appearance, and smearing damage was noted in the remainder of the break. No evidence of preexisting fracture, such as fatigue cracking, was found during the visual and SEM examinations of the fracture.

The splined end of the shaft contained an impact mark at the location indicated by arrow "2" in figure 1. In addition, the lower ends of some of the splines on each side of the impact mark were deformed away from the mark, and at the very end of the splines, the spline crowns and flanks had been smeared in an offset circular pattern. This offset circular pattern is illustrated in figure 3, with the arrows indicating the smearing damage to the lower ends of the splines.

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2. CABLES

Five lengths of cable were submitted and were examined for internal and external wear and for fracture type. Three of the lengths were 1/8 inch diameter and the other two were 3/32 inch diameter. Some of the wires were stiff and heavily oxidized, as if exposed to fire, and some fractures in these overheated wires were brittle. The remaining cable fractures were typical of ductile overstress separations or separating by a cutting mechanism. Minimal wear was found on the cable pieces.

[redacted]
James F. Wildey II
National Resource Specialist - Metallurgy

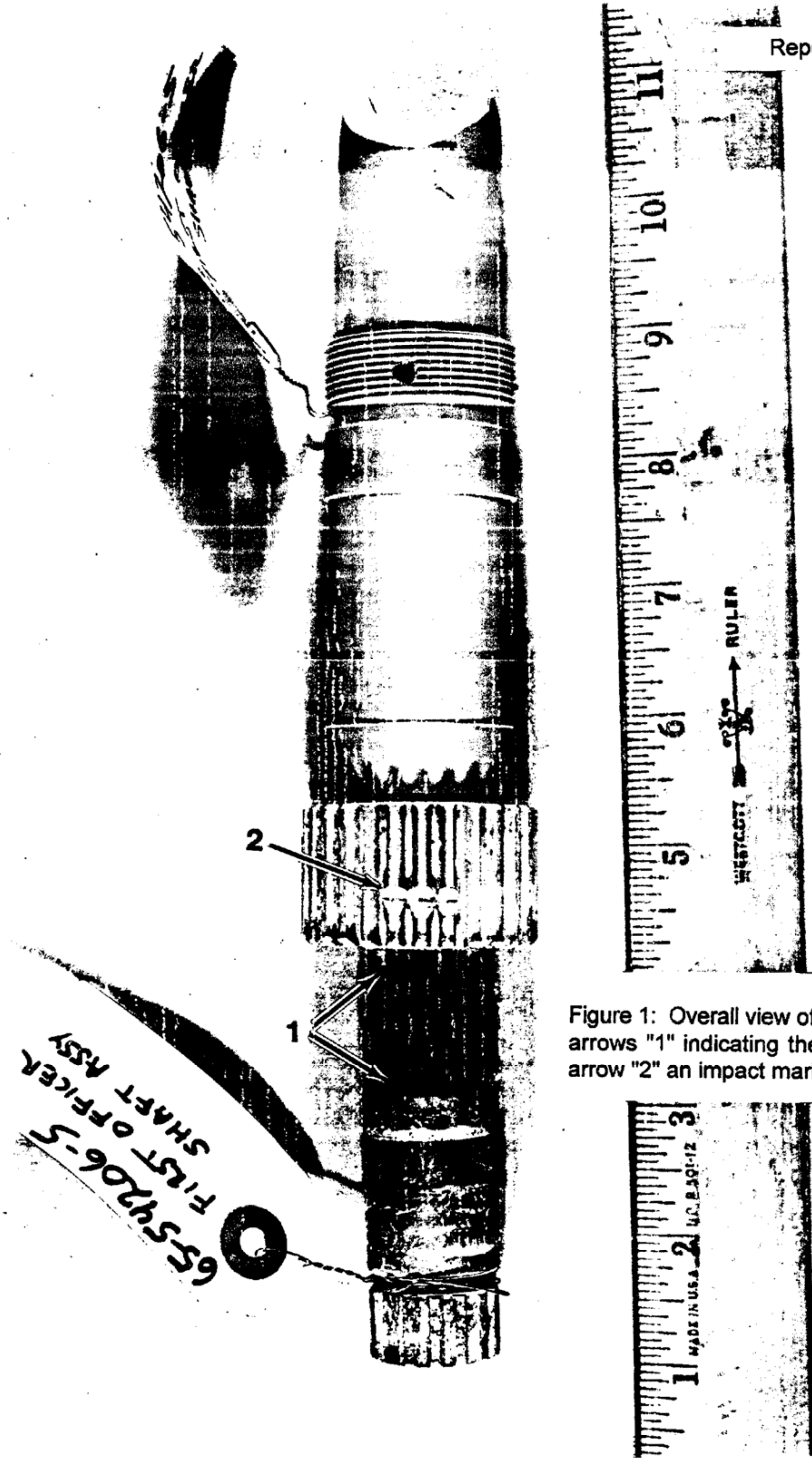

Figure 1: Overall view of the shaft with arrows "1" indicating the fracture and arrow "2" an impact mark.

Figure 2: Fracture surface on the larger piece of the shaft. Smeared portion of the fracture is between the arrows. Magnification, 2X

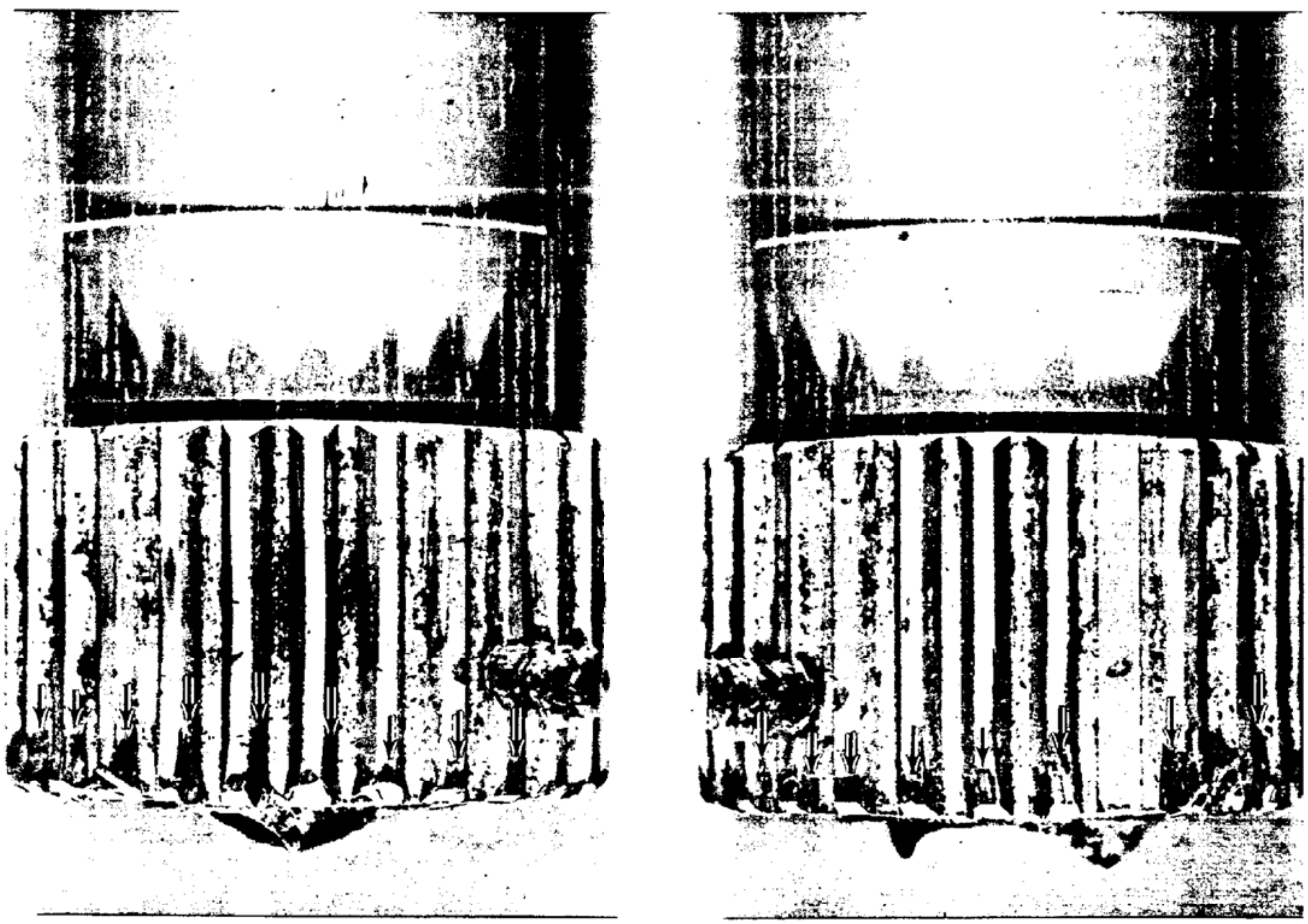

Figure 3: Two views of the spline area. The arrows indicate smearing damage on the lower ends of the splines. Magnification, 2X

NATIONAL TRANSPORTATION SAFETY BOARD

Washington, D.C.

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT
APPENDIX 3

Boeing 737 Flight Controls System Description

Flight Control Systems Description

The following flight control systems descriptions are excerpts from Boeing maintenance manual and maintenance training documentation.

General

The Boeing 737 airplane incorporates a hydraulic powered flight control system which features ailerons and spoilers for roll control, elevators and moveable horizontal stabilizer for pitch control, and a rudder and yaw damper for yaw control, speed brakes for flight and ground aerodynamic braking, and high-lift devices to provide additional lift for takeoff and landing phases.

Primary flight controls (ailerons, elevators, and rudder) are powered by the A and B hydraulic systems. Either hydraulic system alone will power all primary flight control surfaces. In the event of the failure of both hydraulic systems, the aileron and elevator controls revert to a mechanical system. Alternate rudder power is provided by a standby hydraulic system.

Lateral (Roll) Axis

Lateral control is provided by an aileron and two flight spoilers on each wing. These controls are operated by either control wheel in the cockpit. The pilot's and copilot's control wheels are connected by cables to an aileron control quadrant which operates the aileron power control unit through a mechanical linkage. The two cockpit control columns are connected by a bus cable.

The base of the copilot's control column is equipped with a system which allows normal control wheel motion to be transmitted through the left aileron cables only. If a malfunction occurs which jams the aileron control system, lateral (roll) control is accomplished by operating the flight spoilers with the right aileron cables controlled from the copilot's control column. Control wheel movement of more than 12 degrees left or right is required to operate the spoilers through the transfer mechanism.

A spoiler mixer combines lateral input from the aileron system with speed brake lever position to allow the flight spoilers to augment lateral control when simultaneously being used as speedbrakes. The spoiler mixer also functions as a ratio changer which varies the output to the spoiler mixer for a given magnitude of input from the aileron system, depending on speedbrake lever setting. The output decreases as speed brakes are raised.

An aileron spring cartridge provides the mechanical input connection between the copilot's aileron input and the input to the aileron power control units. In normal operation the aileron spring cartridge is not extended or compressed. It would be extended or compressed as a result of control system jamming in the roll axis. The spoiler system is isolated from the aileron system by four shear rivets at the attach point between the spring cartridge and the control quadrant input crank

The ailerons are powered by two independent hydraulic power control units (PCU), one connected to system A and the other connected to system B. Either unit is capable of providing full-range lateral control. Control forces are minimized by aileron balance tabs. Aileron trim is provided by an electromechanical actuator which repositions an aileron centering mechanism.

Two flight spoilers (#2, 3 for left wing) (#6, 7 for right wing) on each wing operate in conjunction with the ailerons to supplement the ailerons for lateral control. The spoiler panels provide increased drag and reduced lift. When the speedbrake handle is in the DOWN detent, the flight spoilers become operational on the up aileron wing at 9 degrees (plus or minus 1 degree) equivalent control wheel rotation. In the FLIGHT detent, the spoilers become operational immediately at any control wheel rotation.

The outboard flight spoilers are operated by hydraulic system B while the inboard flight spoilers are operated by system A. The spoilers also may be operated together to serve as aerodynamic speed brakes. Aerodynamic forces limit panel extension within appropriate limits for the airplanes structural design.

Three ground spoilers (#0, 1, 4 for left wing) (#5, 8, 9 for right wing) are also located on each wing to provide aerodynamic drag for ground operation only. The ground spoilers are protected from airborne operation by a ground spoiler bypass valve connected to the right main landing gear. The ground spoilers are powered by hydraulic system A.

Longitudinal (Pitch) Axis

The Boeing 737's elevators are powered by two independent hydraulic power control units. One actuator within the PCU is connected to hydraulic system A and the other is connected to hydraulic system B. Either unit independently can provide full pitch control.

Pilot input to the power control unit is from the control column through a dual-cable system and torque tube which is connected to both elevators. With both hydraulic systems OFF, the elevator control system automatically reverts to manual function.

Longitudinal trim is provided by a movable horizontal stabilizer, operated by a single ballscrew jack. Power for the jack comes from three sources; the main electric trim motor, the autopilot trim motor, or the manual trim system. Manual stabilizer trim control wheels are located in the cockpit and connect through a cable system to the stabilizer.

A hydraulic "feel" system provides control column forces proportioned to airspeed and center of gravity. Airspeed pressure and stabilizer position (c.g.) are sensed by the elevator feel computer to provide the appropriate control column forces

The elevator installation also incorporates balance tabs which are normally locked to the elevator when hydraulic pressure is applied to the elevator tab lock actuators. The right tab lock actuator is powered by the B hydraulic system. The left tab lock actuator is powered by the A hydraulic system. When hydraulic pressure is removed from the actuators the tabs then become moveable and are mechanically linked to the elevator movement. The tabs are installed to reduce control surface operational forces during manual reversion operation

Directional (Yaw) Axis

Directional control of the airplane is provided by rudder pedals through a hydraulically powered rudder without a tab. A dual-tandem main power control unit is connected directly to the rudder and is powered by hydraulic systems A and B and operates through a dual load-path linkage. Rudder backup power is provided by a standby actuator which is powered by the standby hydraulic system. Any single hydraulic system power source will provide rudder control. The rudder is operated by hydraulic power only, there is no manual reversion capability.

The rudder is also controlled by the yaw damper system which operates through B system hydraulic control in the main power control unit. The yaw damper operates independently of the pilot's control system and does not result in feedback at the rudder pedals. Rudder trim is electrically operated via wires from a control knob on the aislestand to an electro-mechanical actuator attached to the feel and centering mechanism at the rudder.

The rudder power control unit (PCU) provides hydraulic power to position the airplane's rudder. The rudder PCU includes dual tandem hydraulic actuators within the unit. Hydraulic system A provides power to the forward actuator through the hydraulic system A flight control module. Hydraulic system B provides power through the hydraulic system B flight control module to the rear actuator.

Standby Rudder System

The Boeing 737 provides a standby rudder system in the event of failure of A and B hydraulic systems. There is no manual reversion in the Boeing 737 rudder system. The standby rudder actuator which powers the rudder is located above the main rudder power control unit in the vertical stabilizer. The actuator consists of a bypass valve, control valve, and the actuating cylinder.

The standby rudder actuator is not normally powered. When operation is selected by the A or B flight control switches (either switch positioned to STBY RUD) or automatic operation, the actuator is powered through the standby rudder shutoff valve. The standby rudder shutoff valve is automatically opened and the standby pump is started, to pressurize the standby actuator, whenever either primary flight control low pressure switch is low, the trailing edge flaps are not up, and the airplane is either in the air, or on the ground with the wheelspeed above 60 knots. At least one side of the main power control unit is not powered when the standby actuator is powered. No more than two hydraulic systems can be used to operate the rudder.

Inputs from the rudder pedals or rudder trim actuator are simultaneous to the main PCU and the standby actuator. When standby pressure is not available, the bypass valve is in the bypass position. This connects both chambers to the same port of the control valve to prevent a hydraulic lock.

When standby rudder operation is activated, standby pressure opens the bypass valve and connects the actuator chambers to separate control valve ports. Control inputs, operating the external crank, position the control valve to apply pressure in one chamber and open the other to return. The actuator housing strokes on the piston to position the rudder and null the control valve.

Yaw Damper System

The yaw damper operates through the B system side of the main rudder PCU. The components of the system consist of the yaw damper shutoff valve (engage solenoid), transfer valve, yaw damper actuator, and the yaw damper rate sensor. On the Boeing 737-300 series airplanes, the yaw damper is mechanically limited within the main rudder power control unit to a maximum of 3 degrees of rudder deflection in either direction.

The yaw damper is engaged by activating a solenoid which then allows B system hydraulic flow through the transfer valve. Electrical current flow through one of two opposing coil windings within the transfer valve causes the hydraulic fluid flow to be displaced which causes a spool valve to be operated which then causes the primary rudder valve to be driven in one direction or the other. This results in rudder deflection. The airplane may be dispatched with an inoperative yaw damper system for flights below 30,000 feet.