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NATIONAL TRANSPORTATION SAFETY BOARD

Washington, D.C.

Systems / Powerplants Group Chairman's Factual Report of Investigation

(58 Pages)

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

November 22, 2000

SYSTEMS GROUP CHAIRMAN'S FACTUAL REPORT OF INVESTIGATION

A. ACCIDENT: NTSB Case No. DCA00MA023

LOCATION: Near Port Hueneme, California

DATE: January 31, 2000

TIME: 1621 Pacific Standard Time (PST)

AIRCRAFT: McDonnell Douglas MD-83; N963AS

B. GROUP MEMBERS

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C. SUMMARY

On January 31, 2000, about 1621 Pacific Standard Time, N963AS, a McDonnell Douglas MD-83, operating as Alaska Airlines flight 261, crashed into the Pacific Ocean near Port Hueneme, California. All 83 passengers and 5 crewmembers were fatally injured. The flight, from Puerto Vallarta, Mexico, to Seattle, Washington, with an intermediate stop in San Francisco, California, was operating under Title 14 Code of Federal Regulations Part 121.

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[1] Mr. Bartron was present for all documentation and discussions related to the powerplants only. He did not participate in any other systems activities
[2] Mr. Cline participated in all Systems Group examinations and documentation except for the examination of the trim brake switch, one of the trim brake relay examinations, and the Oakland/Trig endplay exercise.
[3] Mr. Simmons participated in the documentation of the elevator system only.
[4] Mr. Mabuni participated in the examinations of the trim system relays only, in lieu of Mr. Lam.
[5] Mr. Sinclair participated in the examination of the trim system gearbox only, in lieu of Mr. Lam.
[6] Mr. Boffo participated in the examination of the primary trim motor only, in lieu of Mr. Lam.

Data from Air Traffic Control (ATC) transcripts, the cockpit voice recorder (CVR), and the digital flight data recorder (DFDR) indicate that the airplane experienced a flight control system malfunction.

The Systems Group was formed on February 2, 2000, to document the systems (including the flight control system) and the powerplants of the accident airplane. The group met initially at the Boeing Commercial Airplane Group factory in Long Beach, California, on February 3 - 5, awaiting the underwater recovery of wreckage. On February 5, 2000, the Group moved to the NTSB Command Post at Port Hueneme, California, to begin the documentation of the wreckage.[7] The initial phase of the on-scene investigation concluded on February 14, 2000.

While wreckage continued to be recovered, the Group reconvened in Tulsa, Oklahoma, from February 29 to March 1, 2000, to perform ground tests on an MD-80 flight control system.

The Group met again at Port Hueneme on March 2, 2000, to continue the documentation of wreckage as it continued to be recovered. The Group concluded its documentation of the recovered wreckage on March 12, 2000. During this second visit to Port Hueneme, some group members left for two-day trips to conduct remote Systems Group activities. On March 7, 2000, Group members participated in a ground test (flap/slat sequencing) of an Alaska Airlines MD-82 in Seattle, Washington, under the supervision of an NTSB investigator from the Northwest Regional Office. On March 10, 2000, Group members traveled to Alaska Airlines heavy maintenance facility in Oakland, California, to gather data regarding maintenance procedures associated with the longitudinal trim system.

The Group subsequently reconvened in the Los Angeles area during the weeks of April 10 and May 1, 2000, to examine of various trim system components (alternate trim motor, gearbox, cockpit trim switch, and trim relays) from the accident airplane. The Group also visited Port Hueneme for a third time on Tuesday, May 2, to perform additional examinations of specific system components, and to interface with the Structures Group as they re-examined the tail section. On May 10, 2000, the Group traveled to Rockford, Illinois, to examine the accident airplane's primary trim motor at the Hamilton-Sundstrand manufacturing facility.

This factual report summarizes the Systems Group findings for all on-site and follow-up activities to date. Supporting documentation includes the following exhibits: Photographs; Drawings and Schematic Diagrams of the MD-80 Longitudinal Trim System; Aircraft Certification Regulations from CAR 4b; Observations of Accident Airplane Flight Control Lubrication; Tulsa Ground Test Data; Seattle Flap/Slat Sequence

[7] On February 6, Group members traveled to Reno, Nevada, to investigate an air turn-back of an Alaska Airlines MD-83 which experienced a trim system malfunction. The team returned to Port Hueneme on the following day. The results of the investigation will be documented separately as NTSB Case no. DCA00SA086.

Test Data; Boeing Messages and Temporary Revisions Regarding End Play Check; 1967 Douglas Design Memorandum Addressing Excessive Acme Screw and Nut Wear Rate; Manufacturer's All Operator Letters Addressing Lubrication of Acme Screw and Nut; Excerpts of Airworthiness Directive and Alert Service Bulletin Regarding Excessive MD-11 Acme Screw and Nut Wear; FAA Airworthiness Directives Issued Post-Accident; and Excerpts of Records Regarding Pertinent End Play Results from All Airlines.

D. DETAILS OF THE INVESTIGATION

1.0 System Description

1.1 Horizontal Stabilizer

The horizontal stabilizer for the Douglas DC-9 series, McDonnell Douglas MD-80/90 series, and Boeing[8] B-717 series airplane is located at the top of the vertical stabilizer (schematic diagrams and drawings attached). It is hinged forward of the rear spar so that the leading edge can be moved up and down to provide longitudinal trim for the airplane. Movement of the horizontal stabilizer is controlled from the cockpit by the longitudinal trim control system through a range of 12.2 degrees leading edge down to 2.1 degrees leading edge up. The system is electrically operated and consists of a primary system, an alternate system, the actuating mechanism, an indicating system, a takeoff warning system, and a motion warning system.

Movement of the primary or alternate longitudinal trim controls energizes the electrical circuit to the respective actuating mechanism to drive the horizontal stabilizer to the desired position. The indicating system shows the position of the horizontal stabilizer, and acts as a follow-up system to actuate the travel limit switches to de-energize the electrical circuit when the stabilizer reaches the travel limits. The indicating system also drives a sensing device that provides an audible signal in the cockpit when the stabilizer is in motion.[9]

1.2 Primary Longitudinal Trim Control System

The primary longitudinal trim control system (schematic diagrams and drawings attached) consists of two control wheel switches in the outboard horn of each aileron control wheel; dual control handles located on the left side of the control pedestal; primary trim brake switch located on the aft left side of the control pedestal; two contactors, two brake control relays; a manual override and shutoff control, and a brake switch, all located behind the lining in the forward lower cargo compartment;

[8] The Boeing Commercial Airplane Company acquired the McDonnell Douglas Corporation in 1997. The McDonnell Douglas MD-95 designation was subsequently changed to the Boeing 717.

[9] According to Boeing, the audible signal is produced when the horizontal stabilizer continuously moves through about 1.1 degrees, and is produced again about every 0.55 degrees of continuous movement thereafter.

and the primary longitudinal trim actuator motor mounted on the actuating mechanism in the vertical stabilizer. Rate-of-trim using the primary system is approximately 1/3-degree per second.

Electrical control is provided by the control wheel switches. One switch in each control wheel is a motor control switch; the other is a brake control switch. Both switches must be moved simultaneously, in the same direction, to move the horizontal stabilizer. Moving the motor control switch on either control wheel energizes the airplane nose-up (ANU) or airplane nose-down (AND) contactor. Energizing either of the contactors completes the circuit to the motor section of the primary actuator motor. Moving the brake control switch in the same control wheel energizes the corresponding brake control relay and completes a circuit to release the brake in the actuator motor. Energizing this relay disengages the autopilot. The control wheel switches are arranged in the circuit so that when the switches on one control wheel are being operated, operation of the switches on the other control wheel in the opposite direction opens the circuit and stops the movement of the horizontal stabilizer.

Mechanical control (via mechanical linkage to an electrical switch) is provided by the dual handles, commonly referred to as "suitcase handles," on the control pedestal. The two-way cable system attached to the inboard handle is routed to a bell crank and a shaft behind the lining in the forward lower cargo compartment. A link rod connects the shaft to the manual override control located between the contactors. When the inboard handle is moved, the manual override control actuates the contactor and completes the circuit to the motor section of the primary actuator motor. The two-way cable attached to the outboard handle is routed to another bell crank and shaft in the forward lower cargo compartment. A lever connects the shaft to the brake switch. When the outboard handle is moved, the brake switch is actuated, and a circuit that bypasses the brake control relays is completed to release the brake in the primary actuator motor. To move the horizontal stabilizer, both control handles must be operated simultaneously in the nose up or nose down position. Operation of the control handles in a trim direction opposite that of the control wheel switch trim direction will override the control wheel switches and command trim in the direction of the control handles.

The trim indicating system follow-up cable is connected mechanically to the manual override and shutoff control to provide shutoff control as the horizontal stabilizer reaches the limit of travel. The follow-up cable is routed around a drum in the forward lower cargo compartment. A link rod connects the shaft of the drum to the shutoff control. As the stabilizer reaches the limit of travel, the link rod operates the shutoff control on the contactors, breaking the circuit to the primary actuator motor.

The primary trim brake switch, which is a red guarded switch located on the aft left portion of the center pedestal, provides a means to stop horizontal stabilizer movement if a malfunction occurs in the primary longitudinal trim control system. The primary trim brake switch is a two-position normally closed switch installed in the circuit to the primary longitudinal trim contactors and brake coil in the actuator motor. Movement of the switch to the STOP position opens the circuit and interrupts electrical power to the contactors and brake coil, causing the brake to engage and stop movement of the stabilizer.

1.3 Alternate Longitudinal Trim Operating System

The alternate longitudinal trim control system consists of a motor control switch and a brake switch, both connected to levers located on the control pedestal, an alternate longitudinal trim actuator motor mounted on the actuating mechanism in the vertical stabilizer, and two limit switches located at the lower end of the trim indicating system control drum shaft. Rate-of-trim using the alternate longitudinal trim system is about 1/10 degree per second.

The control switches, when in the center normal positions, are included in the autopilot trim circuit. No action occurs unless the autopilot trim and brake relays are energized. The motor control switch is a three-position rotary switch with the center position maintained in the autopilot circuit. Moving this switch either way bypasses the autopilot trim relays and completes a circuit through the motor limit switches to the motor section of the alternate longitudinal trim actuator motor. The brake control switch is a two-position switch with the normal position maintained in the autopilot circuit. Moving the brake control switch bypasses the autopilot brake relays and completes a circuit to release the brake in the actuator motor. To move the horizontal stabilizer, the crew must operate both control handles simultaneously to the full nose up or nose down position.

The two limit switches provide the shutoff control for the alternate system to prevent over travel when the horizontal stabilizer reaches the limits of the travel range. As the horizontal stabilizer reaches the limits of travel, the actuator roller on the limit switch is moved by an arm attached to the drum shaft and the circuit to the actuator motor is opened.

1.4 Longitudinal Trim Actuating Mechanism

The longitudinal trim actuating mechanism (schematic diagrams and drawings attached) is located within the vertical stabilizer forward of the horizontal stabilizer front spar. The actuating mechanism consists of an acme screw and nut[10], drive torque tube (also commonly referred to as the "quill shaft") located inside the acme screw, main gearbox, sandwich gearbox, primary longitudinal trim actuator motor, alternate longitudinal trim actuator motor, and support. The main gearbox and the acme screw and nut assembly are common to both the primary and alternate systems. The nut is attached to the empennage structure by a gimbal ring and retaining pins, and the acme screw is installed in the nut and attached to the support, which is installed at the

[10] The acme screw and nut have two independent thread spirals along their lengths.

stabilizer front spar center section. The main gearbox is a dual planetary gear assembly and is installed on the support directly above the acme screw.

The actuator motors are mounted on the top of the gearbox and connected to the first stage of the gear system. Reduction is obtained through the planetary gear arrangement. The second stage of the gear system is connected to the acme screw by a spheroidal spline drive adapter. When either actuator motor is actuated, the gear system is driven and the acme screw is rotated (via the drive torque tube which transmits the torque from the motors to the screw via an internal spline connection just above the retaining nut) within the acme nut, moving the stabilizer leading edge up or down. Each motor is protected from excessive heating by a thermal cutout unit that interrupts 3-phase electrical power during an overheat condition. The alternate actuator motor thermal unit will open after approximately 60 seconds of overheating. The primary motor thermal unit will open after 15 to 30 seconds of overheating.

1.5 Autopilot Interface with Pitch Control and Autopilot Trim Light

According to the manufacturer, automatic pitch control is provided by autopilot engagement. Pitch control signal information is induced into the Digital Flight Guidance Computer (DFGC), where computed signals are processed and transmitted to a duplex elevator servo drive. The elevator servo drive is coupled to the elevator control system, and when commanded signals from the DFGC cause the servo drive to move, the elevator control tabs are moved the proper amount to execute the commanded pitch maneuver. Whenever a signal to the elevator control tabs is sustained for more than 5 seconds, the autopitch trim moves the horizontal stabilizer in a direction to relieve the elevator deflection.

The airplane is also equipped with an autopilot trim annunciator in the cockpit. This amber light illuminates to alert the flightcrew that a horizontal stabilizer out-of-trim condition exists with the autopilot engaged.[11] According to Boeing, both the Captain's and the First Officer's Flight Mode Annunciators contain two lamps and an "A" and "B" channel to annunciate "A/P TRIM." The trim warning is based on the magnitude of the elevator autopilot servo current and the trim warning counter. When the magnitude of the servo reaches a value of 380 milliamps, an electronic trim warning counter begins to run. The counter will continue to run (decrement) for 9.9 seconds. When and if the counter reaches zero, the trim warning output discretes are set and the lights are activated. If the servo counter is reset, the output discretes are cleared, and the trim warning lights will extinguish. If a cockpit light test is in progress, the trim warning output discretes are not changed.

[11] Interviews of airline personnel by the Maintenance Records Group indicate that the flightcrew reported the illumination of this light during the accident flight.

1.6 Trim Motor Capability

Both primary and alternate trim motors have prescribed speed versus torque characteristic curves. When reflected at the acme screw (via the gearboxes), these motors have significant capability in terms of torque generation. In the case of the alternate motor, it will rotate the acme screw at speeds of about 10 rpm and has a maximum torque capability (stall torque) of 4,400 in-lbs. The primary motor can rotate the acme screw at speeds of around 35 rpm and has a maximum torque capability of 18,850 in-lbs. (For comparison, the torque required to move the stabilizer on the ground is only 300 to 500 in-lbs.).

The differential gearbox, onto which both primary and alternate motors are attached, is a speed sum device, not torque sum. In normal operation, only one trim motor will be operated at a time. The opposite motor will remain fixed with its brake engaged thus providing a ground for the functioning motor to drive the acme screw. In the case of running both motors together in the same trim direction, the two speeds sum together resulting in a higher trim rate than the primary motor alone. This sum, however, applies only up to the point where the resisting air loads (or a mechanical jam) provide a reaction torque at the acme screw equal to or greater than the capability of the alternate motor (about 4,400 in-lb dependent upon motor condition, etc.) At this point, the stronger primary motor will begin to back drive the alternate motor; the acme screw, held by the resisting air load, will stop rotating and act as a grounding point. Upon subsequent release of the alternate motor command (and reapplication of its brake) or the reduction of the air load lower than the equivalent 4,400 in-lb, the alternate motor becomes the ground point once again, and the acme screw would begin to rotate as powered by the primary motor.

According to Boeing, in the case of commanding both motors at the same time but in the opposite trim direction (for example: commanding the primary trim toward the airplane nose down direction while simultaneously commanding the alternate trim switch toward the airplane nose up direction), the acme screw would turn at a slower rate than as commanded by the primary motor alone as a function of the speed sum characteristic of the gearbox. The "ground point" restriction as noted above would also apply in this case.

2.0 Certification of the DC-9 and its Longitudinal Trim System

2.1 DC-9 Certification Basis

According to the FAA Type Certificate Data Sheet (TCDS No. A6WE, excerpts attached) for the original Douglas DC-9 (series –11), the DC-9 was certified under Civil Aeronautics Rule (CAR) 4b, as revised through Amendment 12, dated March 30, 1962. Subsequent models of the DC-9 (–12, -13, -14, -15, -15F, -21, 31, -32, -32F, -33F, -34, -34F, -41, -51) were also certified under requirements cited under CAR 4b. More recent models of the DC-9, including the McDonnell Douglas MD-83 (designated by the TCDS as DC-9-83, excerpts attached) and the Boeing B-717 (initially designated as the MD-95) were certified under Federal Air Regulations (FAR) Part 25 and applicable amendments.

2.2 Control Systems Requirements

According to CAR 4b, section 4b.322, entitled “Trim Control and Systems”:

> (a) Trim controls shall be designed to safe guard against inadvertent or abrupt operation… (e) Trim devices shall be capable of continued normal operation in the event of failure of any one connecting or transmitting element of the primary flight control system.

Section 4b.325, entitled “Control System Stops”, states

> All control systems shall be provided with stops which positively limit the range of motion of the control surfaces. (b) Control system stops shall be so located in the systems that wear, slackness, or take-up adjustments will not affect adversely the control characteristics of the airplane because of a change in the range of surface travel. (c) Control system stops shall be capable of withstanding the loads corresponding with the design conditions for the control systems

2.3 Structural Strength Requirements

According to CAR 4b, section 4b.270 (excerpts attached), entitled “Fatigue evaluation of flight structure”: “The strength, detail design, and fabrication of those portions of the airplane’s flight structure in which fatigue may be critical shall be evaluated in accordance with the provisions of paragraph (a) or (b) of this section.” Paragraph (a) is entitled “Fatigue Strength,” and paragraph (b) is entitled “Fail safe strength.” Paragraph (b) states:

> It shall be shown by analysis and/or tests that catastrophic failure or excessive structural deformation, which could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single principal structural element. After such failure, the remaining structure shall be capable of withstanding static loads corresponding with the flight loading condition specified [in CAR 4b]. These loads shall be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise taken into consideration.

2.4 Certification Data Regarding the DC-9 Longitudinal Trim System

The Group Chairman's review of DC-9, MD-80/90, and 717 series airplane certification documents[12] indicated that the manufacturer provided material strength and loads information, stress calculations, and analyses to support assertions that the acme screw and torque tube are designed such that any one of these two elements, for load path redundancy purposes, will withstand tensile and compressive loads well beyond that which could be generated by aerodynamic forces acting on the horizontal stabilizer. The documents also indicated that these assertions are based on the assumption that the acme screw and acme nut threads remain engaged as a load path.

The certification documents indicate that the manufacturer supported (with calculations and analyses) the following conclusions:

Assuming an unused, intact acme screw and acme nut that meet drawing specifications, the actuator is capable of carrying limit loads, as generated by the horizontal stabilizer, with a margin of safety, in either pure tension or compression, without failing, for each of the following scenarios:

- A fractured acme screw -- by designing the inner torque tube to have the capability of supporting limit operational tension and compression loads (in addition to the normal operational torsion loads) as generated by the horizontal stabilizer.

- A fractured inner torque tube -- by the immediate cessation of trim operation. Under this failure effect, the loads generated by the horizontal continue to be carried by the acme screw and acme nut and the airplane can be aerodynamically controlled in the pitch direction by the elevator.

- The loss of 90 percent of acme screw and nut threads (i.e. 3.2 threads remaining) – by providing more threads than needed.

- The failure of one set of threads in the acme screw or acme nut – by the incorporation of two independent thread spirals along its length.

The Group Chairman's review of the April 15, 1965, DC-9 Flight Controls System Fault Analysis (revised July 14, 1997), Douglas Report No. LB-32160, indicated that the failure mode of excessively worn acme nut threads were not formally considered, and no stripped acme nut thread failure mode was specifically addressed in the Fault Analysis. The review also revealed that the acme nut was designed with a

[12] The Safety Board requested all available documentation addressing the certification of these airplane designs. The documentation included the April 15, 1965, DC-9 Flight Controls System Fault Analysis (revised July 14, 1997), the 1964 DC-9 Control System Design Criteria, and the April 1998 MD-95-30 Control Systems Loads Criteria reports. Additionally, internal "design memorandums" that addressed the trim system were reviewed.

softer material than the acme screw and was meant to wear. The wear of the acme nut was designed to be measured periodically by the operator so that appropriate repair or replacement of the assembly could be performed based on this maintenance and specified wear limits.

2.5 Comparisons of the DC-8 and DC-9 Longitudinal Trim System Design

According to Boeing, the type of longitudinal trim actuation system used on the DC-9 was also used on earlier Douglas DC-8 airplanes, except that the DC-8 incorporated two separate acme screw and nut assemblies. According to Boeing, for the DC-9, the structural redundancy achieved by the DC-8 dual acme screw design was accomplished with a single acme screw and nut assembly by the addition of a torque tube within the acme screw. According to Boeing, the screw/torque tube combination provides a redundant load path if any one of these elements fail.

3.0 Description of System Wreckage

3.1 Flight Controls

3.1.1 Flight Control Cables and Cockpit Flight Controls

Generally, the flight control cables were separated along the length of the fuselage in three specific locations: 1) just aft of the cockpit, 2) the overwing area, and 3) the aft pressure bulkhead area.

Sections of both cockpit control columns above the column hinge points were recovered. Both columns were heavily damaged and corroded. Both column hubs were broken off but remained loosely attached as the aileron drum and cables were present within the columns. The column crank arms below the hinge points were not recovered. Both columns were recovered with the forward fuselage cables. The aileron bus torque tube, which connects both cockpit control wheels, was also recovered. This torque tube was fractured into approximately three pieces and was recovered with the forward fuselage cables.

A portion of the rudder pedal stature adjustment crank arm was recovered with a portion of the cable still attached. A single rudder pedal was recovered with no mechanisms attached.

3.1.2 Lateral Flight Controls

3.1.2.1 Ailerons and Aileron Trim

3.1.2.1.1 Right Aileron and Aileron Trim

The right-hand-side (RHS) aileron control sectors were recovered intact within a portion of the right wing. A portion of the aileron and aileron control tab were also present. Both upper (aileron bus follow up) and lower (tab drive) sectors were intact and able to articulate. The tab drive crank and pushrod were present. The sector clapper spring was not found in the mechanism.

The RHS aileron trim tab jackscrew was recovered intact within a portion of the right wing. The cable jackscrew was present with the drive cables still attached. The jackscrew adjustable rod end remained attached to the tab pushrod and was still connected to the tab surface input crank arm. The amount of jackscrew extension was recorded as 2-1/2 inches from the center of the rod end bearing to the aft face of the jackscrew body. According to Boeing, this corresponds to a right wing down aileron trim command.

3.1.2.1.2 Left Aileron and Aileron Trim

The left-hand-side (LHS) aileron trim tab actuator (jackscrew) was found intact in the left wing except for the drive rod end, which was fractured. The drive cables, though severed inboard along the rear spar, were still wrapped about the cable drum. The broken rod end was attached to the trim tab pushrod. The trim tab pushrod was intact as well, terminating at the tab drive crank. Only a portion of the aileron and aileron trim tab were present. The trim actuator position was measured: from the body of the actuator to the rod end jamb nut aft face was 1-7/32 inches. According to Boeing, this corresponds to a left wing down aileron trim command.[13]

The LHS aileron control tab drive mechanism was largely intact. One of the two drive cables was lockwired in place to the sector; the second cable was present in the mechanism but hanging loosely. The control tab pushrod was present and was attached to the mechanism and to the tab drive crank. The sector override spring was present and attached at only one lug; the opposite lug was broken away from the clapper mechanism.

3.1.2.2 Spoilers

3.1.2.2.1 Left Spoiler

[13] The aileron trim command is either left wing or right wing down; it cannot be both during normal operation.

The LHS ground spoiler actuator and mechanism were not recovered. The outboard flight spoiler actuator was present in the rear spar of the wing. The inboard flight spoiler actuator mechanism was not present except for the panel stow torque tube.

3.1.2.2.2 Right Spoiler

The RHS ground spoiler actuator and mechanism were not recovered. Both the inboard and outboard flight spoiler actuators and mechanisms were present in the rear spar.

3.1.2.2.3 Miscellaneous Spoiler Components

One spoiler hydraulic actuator was recovered separately; its location was not determined. Both spoiler lockout mechanisms, one found with the forward cables and one found separately, were recovered. A large portion of the lockout mechanism pull cable was also recovered separately.

3.1.3 High-Lift Devices

3.1.3.1 Flaps

3.1.3.1.1 Left Flap

All hinges (three on outboard flap and one on inboard flap) and inboard panel flap track were present. All hydraulic actuators (two on outboard flap, one on inboard flap) were intact at the hinges. A 24-inch section of the retract cable was lockwired into the flap track. The extend cable was not present. The flap position sensor remained attached at the inboard hinge of the outboard flap section. The black-booted wire bundle, input pushrod, and sensor crank arm remained intact.

The LHS inboard flap drive mechanism, including the hydraulic actuator and linkage, was present on the fuselage structure.

3.1.3.1.2 Right Flap

All hinges (three on outboard flap and one on inboard flap) were present. All hinge-mounted hydraulic actuators (two on the outboard flap, one on the inboard flap) were intact at the hinges. The flap position sensor was present at the inboard hinge of the outboard flap section. The black booted wire bundle was disconnected from the sensor and was found hanging with an adjacent wire bundle. The input pushrod was present but broken approximately at its midpoint. The sensor crank arm was still attached.

The RHS inboard flap’s drive mechanism, including the hydraulic actuator and linkages, was not present on the fuselage structure and was not recovered. The RHS inboard flap track and roller bearing was recovered. A 4-inch piece of the retract cable was still attached to the track by a secured cable ball. The extend cable was not present.

3.1.3.1.3 Flap Bus

The majority of the flap bus cable was intact. The flap bus fuse link was still intact as was the flap position follow-up bridal cable fitting. The LHS extend cable was found intact with the cable ball end still attached. The LHS retract cable was found severed approximately 10 feet outboard of the fuse link. (This matches the 2-foot section found in the LHS flap track.) The RHS extend cable was found intact with the cable ball still attached. The RHS retract cable was found severed approximately 8 feet outboard of the bridle fitting.

3.1.3.1.4 Flap Actuators

All hydraulic flap actuators from the left and right hand side exhibited a line of demarcation that was defined by the border between the shiny chrome piston surface and the dirty chrome piston surface. The dimension in all cases was 1-3/4 inches from the line to the forward face of the clamp-up washer. According to Boeing, this corresponds to a fully retracted flap position. No score marks or bent piston rods were noted, and several of the actuators could be manually extended and retracted. All actuators were recovered except for the right inboard actuator.

3.1.3.2 Slats

3.1.3.2.1 General Description of Slat Surfaces

About 90 percent of the leading edge slats from the left wing were recovered, and all exhibited evidence of water impact damage. About 20 percent of the slats from the right wing were recovered. Numerous pulleys, pulley brackets (portions and complete assemblies), and wing-mounted slat track mechanisms associated with the leading edge slat installation were also recovered.

3.1.3.2.2 Slat Drive Hydraulic Actuation Components

3.1.3.2.2.1 Hydraulic Actuators

Both LHS and RHS actuators were attached to their respective structural supports and to the main bull wheel input crank arm. No damage to these components was noted. Both actuators were found in the retract position.

3.1.3.2.2.2 Slat Drive Bull wheel

The slat drive bull wheel was severed from its drive mechanism. The fracture at the mounting flange corresponds to the fracture seen at the drive mounting plate. The upper bull wheel pivot support bracket was still attached although the bracket mounting face was broken. Both proximity sensors were present on this bracket with the connecting wire still attached. The proximity targets were still present on the upper/inner surface of the bull wheel and appeared undamaged.

The forward guard pin support bracket was not present with the upper bull wheel pivot support. Both “A” and “B” proximity sensors that are normally attached to the support bracket were missing.

Only the left half of the wing front spar was recovered. Remnants of the LHS slat drive mechanism support brackets including the upper bull wheel pivot bracket were found still attached to the spar.

All of the mechanical drive components were intact within the bull wheel mechanism including the main input crank, “dog bone” link, and C-shaped drive crank. The upper bull wheel (the component that connects and drives the leading edge slat cables) was not attached to the upper mounting flange of the C-drive crankshaft and recovered separately.

The following observations of the bull wheel grooves were noted: Top grooves (left hand side slat system) -- All swaged lugs of cables found secured to bull wheel with pins. All remaining sections of these cables were measured both to the left and to the right of the swaged lug. Bottom Grooves (right hand side slat system) -- Swaged lugs of slat nos. 2, 3 and 4 cables were found secured to bull wheel with pins. The cable for slat no. 5 was found separately. The bull wheel lug socket for slat no. 5 was found with the lug retention pin missing. The lug socket for slat no. 0 cable was found with the lug retention pin still in place. The lug socket for no.1 slat cable was found with the retention pin missing.

A gouge was identified on the no. 1 right cable groove (right hand side of the bull wheel). This gouge, angling upwards and forwards, spanned the cable groove and cut into the groove walls. This mark was approximately 90 degrees clockwise (CW) from the forward position. A second mark was identified on the lower wall of the no. 1 right cable groove at the zero-degree position. A generally elliptical section of the wall was broken away and not recovered.

3.1.3.2.2.3 Fuselage Interface with Slat Drive Cables

An examination of a portion of the RHS fuselage skin which held several of the RHS slat cable pressure seals indicated that three of the six slat seals were completely intact and surrounded nearly completely by skin structure. This included the no. 0 slat extend and retract cable. One seal had approximately 50 percent of its material present and was identified to be the no. 1 slat cable seal. In this instance, approximately 50 percent of the surrounding structure was present. Two seals were completely missing, as the surrounding structure was not present. An additional fuel crossfeed cable pressure seal was also found to be intact.

The LHS portion of the fuselage where the LHS slat cables pass through the pressure vessel was examined. All six of the dual cable pressure seals were present. No cables were present in the seals. All of the seals exhibited tearing of their elastomer material.

3.1.4 Longitudinal Trim System

3.1.4.1 Horizontal Stabilizer Actuation Components

Approximately 95 percent of the horizontal stabilizer (HS) follow-up cables were recovered and identified. Some segments of the cable runs in the vicinity of the aft pressure bulkhead were missing.

3.1.4.2 Drive Unit Support

The main supporting elements of the drive unit were found intact and still attached to the horizontal stabilizer's front spar "A" frames. The differential gearbox was attached, although the upper housing of this gearbox was completely torn away except for the mating interface with the alternate motor. The acme screw was still attached to the drive support elements. The entire mechanism was deformed toward aircraft left.

3.1.4.3 Primary Trim Motor

The primary trim motor (Sundstrand p/p 9590-6, s/n 2006) was found detached from the differential gearbox. The alternate gearbox (commonly referred to as the "sandwich gearbox") was found attached to the primary motor along with a portion of the upper housing of the differential gearbox. Both electrical connectors to the primary motor were present, although their connection wires were not. The trim motor unit was mostly intact. Damage to the upper edge of the brake cover/cap was noted. When viewed from above, with the aircraft forward direction at the 12 o'clock position, a dent was found at the 1:30 o'clock position.

The primary trim motor was disassembled and examined in detail by the Group at the manufacturer's facility. A measurement of the air gaps between the brake plates revealed gaps that measured between 0.012 and 0.014 inches. According to Sundstrand, these measurements are all within an acceptable range for overhaul; however, Sundstrand representatives indicated that corrosion damage and swelling could affect the air gap.

Brake torque was unremarkable. Electrical continuity check (using 28 VDC) between the brake coil connector and the brake coil revealed an open circuit. Another check of just the coil itself gave a reading of 6.4 ohms.

The rotation monitor switch on the motor was examined. No signs of arcing or thermal damage were noted. The contact pads did not exhibit any evidence of excessive wear or thermal damage. The leaf spring was intact and functional.

The thermal switch[14] was intact and corroded. No evidence of any thermal damage was noted. The 3-phases of the switch were checked for continuity. One phase was found open and the other two were closed.

The main rotor and stator of the motor were intact and did not exhibit any evidence of arcing, burning, or melting. Both exhibited evidence of corrosion.

The brake assembly of the motor was disassembled and inspected. The assembly consisted of five rotating disks, ten reaction plates, and four shims. Many of these components exhibited evidence of discoloration. The color was blue-ish in some locations and dark brown-ish in others. Some of the reaction plates appeared to have friction material adhering to them, and non-uniform wear of the friction material of the rotating discs was noted. The parts were labeled and sent to the NTSB Materials Laboratory for further examination. A micrometer was used to measure the thickness at four locations on each of the five rotating disks. The following results were recorded (in inches):

No. 1 (top)	.056	.058	.058	.059
No. 2	.059	.059	.059	.059
No. 3	.061	.061	.060	.059
No. 4	.056	.057	.059	.059
No. 5	.055	.056	.054	.054

According to representatives of the primary motor manufacturer, these measured thicknesses were not remarkable.

The auxiliary gearbox was separated from the motor, disassembled, and examined. No evidence of any pre-impact mechanical damage was noted. The gearbox was able to rotate slightly when manually rotated; however, the gears became jammed due to solidified grease remnants. Samples of this grease were taken from various locations inside the gearbox and submitted to the NTSB Materials Laboratory.

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[14] The thermal switch on the motor is specified to open at 150 +/- 7 degrees C (302 degrees F), and close at 110 +/- 15 degrees C (230 degrees F). According to the ATP data plot sheets (attached) for the motor, the thermal switch should activate in 10 to 30 seconds under a stalled motor condition, and then reset in 7 to 27 seconds.

The oil plug was removed and a mixture of oil and water was drained out of the motor and submitted to the NTSB Materials Laboratory. The motor housing was opened and the planetary gear assembly was then removed. A small amount of corrosion was found on the gears. The gears were able to move freely with no apparent binding. Gear teeth and the gear spline shaft were not damaged. Movement of the bearings did not reveal excessive free play.

According to representatives at Sundstrand, 12 service bulletins and one FAA airworthiness directive (AD) have been issued against the motor. The AD addressed output shaft heat-treating. The predominate failure mode of the motor is brake wear. According to Sundstrand records, the accident motor had been overhauled by Sundstrand on February 22, 1994, and again on October 4, 1996.

3.1.3.4 Alternate Trim Motor

The alternate trim motor (Vicker-Eemco p/n: D1775-1, s/n: 905, per DAC Spec 7923799) was found detached from the differential gearbox and was recovered in several pieces. The largest pieces collected included the cylindrical end bell, stator winding, and mounting flange. The rotor brake solenoid, brake disk, and output shaft were recovered as an assembly.

All recovered pieces of the alternate trim motor from the accident airplane were examined. This motor could not undergo the manufacturer's acceptance test plan due to impact damage and corrosion. The impact damage and gouging were predominantly located toward the left (port side), rear portion of the motor.

Only a portion of the upper brake pad was recovered with the motor; the thickness of this portion was measured with a micrometer. The measurement was 0.0527 inch, and the minimum acceptable thickness specification is 0.048 inch. The thickness of the lower brake pad (which was recovered intact and in one piece) was measured to be 0.127 inch; the minimum acceptable thickness specification is 0.125 $\pm$ 0.002 inch.

The upper bearing of the stator was present but corroded. The inner race of the lower bearing was present, but the ball bearings and outer race were missing. No apparent damage was noted at the output gear spline. Electrical continuity of the stator was verified for all three phases with a measured resistance of 6.4 ohms on each.

A check of the stator dielectric was accomplished and the result was that it failed at less than 200 volts alternating current (VAC) at 0.1 miliamp (mA). (The specification threshold is 250 VAC). A dielectric test of the brake coil revealed breakdown at 750 VAC at 0.15 mA. A resistance check of the coil revealed a value of 11.7 ohms (specification calls for a range of 11 to 12.38 ohms.)

Manufacturer records for this motor indicate that it was manufactured and factory tested on May 30, 1985. The records also indicate that the motor had never been returned to Eaton-Vickers for overhaul since 1997 (which is when their overhaul records terminate).

3.1.3.5 Differential Gearbox

External examination of the accident gearbox revealed that the Original Equipment Manufacturer (OEM) lockwired lead seal was intact, indicating that the box had never been opened up after delivery from the manufacturer. The tint of paint color from the top cover of the gearbox was slightly lighter (white) than the lower half of the gearbox assembly (flesh). The differential gearbox was attached to the support member. Due to impact damage, loss of the upper cover, and submersion in seawater, the magnesium housing experienced structural degradation.

No pre-impact mechanical malfunctions or defects were found with the gearbox. The aft locator pin was found bent aft and toward the left (port side - inboard). The primary drive spline socket was cracked along its major axis. The output spline shaft was able to rotate about 45 - 60 degrees, and appeared to bind due to chunks of coagulated grease within the gear set. Once the grease was removed, the shaft could rotate 360 degrees. The drain plug lockwire appeared to be original issue; the service plug lockwire did not appear to be original issue. All of the gear teeth from all internal gears were intact and undamaged. After the gears were cleaned in a solvent, they were reassembled and able to rotate with no binding or detents noted. Grease samples were taken from five locations and submitted to the NTSB Materials Laboratory.

According to representatives of the gearbox manufacturer (Schrillo), the gearbox design was conceived in the early 1960s by the VARD Company for the DC-8. No service bulletins or ADs had ever been issued against the gearbox, and no failures have ever been reported to the manufacturer.

3.1.3.6 Acme Screw

The acme screw was found attached into the drive support assembly. The screw was bent along its major axis toward the left side of the aircraft, and was also bent aft. Upon closer inspection, a circumferential fracture was identified on the acme screw above the upper stop location. The entire length of the screw was discolored with dirt deposits, white material deposits, and red/orange rust deposits.[15] There was no visible[16] evidence of grease on the acme screw along the length where it normally travels[17]

[15] The acme screw was examined by the Systems Group after it had been recovered from the ocean floor, rinsed, and brought to shore. Additional details of the wreckage recovery process are documented in the Structures Group Chairman's Factual Report of Investigation.

[16] Visible to the naked eye, without magnification.

[17] For the purposes of this report, the normally traveled area is the area of the acme screw that contacts the threads of the acme nut along the distance defined by the upper and lower limits of the electrical stop.

within the acme nut. Several flat ribbons of metal, resembling an acme nut thread remnant, were found coiled in the valleys of the acme thread. Two or three of these ribbons were found pulled away from the acme screw and allowed for closer inspection. The thread remnant material appeared bronze in color and measured about 0.025 inches thick and about 0.1 to 0.125 inches in width. The estimated diameter of these helical coils was about 2 inches. The approximate axial location of this thread remnant (as measured from the bottom of the acme screw) was between 9-7/8 inches and 15-7/8 inches. A serial number, P-2663, was identified on the acme screw above the upper stop. The acme screw and acme nut were removed from the airplane and sent to the NTSB Materials Laboratory for detailed examination.

3.1.3.7 Acme Nut and Gimbal Assembly

The acme nut and gimbal assembly was found in a portion of the vertical stabilizer structure. No external damage was noted. The gimbal could be moved on both rotational axes except as prevented by distortion of the surrounding structure. Globules of shiny red grease were found on the exterior of the assembly. Visual inspection of the inner diameter of the nut revealed no threads. There was no visible evidence of grease on the inner diameter of the acme nut. Helical marks along the entire length of the inner diameter were evident. A serial number, P-2663, was obtained from the side of the acme nut element within the gimbal assembly. This number agrees with the serial number of the acme screw and confirms that both the screw and nut remained as a matched set in this aircraft.[18] This component was submitted to the NTSB Materials Laboratory for detailed examination.

3.1.3.8 Upper Acme Screw Stop

The upper stop was found attached to the acme screw. The clamp-up bolt securing the stop to the screw splines was still in place. No apparent damage was noted on the stop. The stop lug stop pad was missing. This component was submitted to the NTSB Materials Laboratory for detailed examination.

3.1.3.9 Lower Acme Screw Stop

The lower stop was found detached and was recovered separately from the acme screw. The stop clamp-up bolt was securely in place and showed distinct scoring on the side nearest the acme screw splines. The vertical face of the stop lug showed no signs of damage and the stop pad mounted in the lug appeared undamaged as well. No evidence of contact with the mating lug stop pad of the acme nut was noted. The upper face of the stop showed several impact/scrape marks. The lower face of the stop, opposite the lower marks, exhibited a round impact mark. Evidence of damage to the splines on the inside diameter of the stop was noted. Additionally, evidence of

[18] It was also later confirmed that the acme screw and nut was not replaced since the time the airplane was delivered from the manufacturer.

lockwire pigtails were seen hanging from the stop clamp-up bolt. This component was submitted to the NTSB Materials Laboratory for detailed examination.

3.1.3.10 Acme Screw Torque Tube

The acme screw torque tube inner shaft (also known as the "quill shaft") was found inside the acme screw with the shaft's lower threaded portion, large flange washer and clamp-up retaining nut broken off.[19] The torque tube was submitted to the NTSB Materials Laboratory for detailed examination by the Materials Group.

3.1.3.11 Primary and Alternate Motor Electrical Wiring

A portion of the vertical tip cap fairing just aft of the horizontal trim actuation system access fairing was recovered. The electrical wire bundle "octopus" that connects to all trim motor electrical connectors was found in this fairing. The primary trim motor heater cap was also attached to this electrical wire bundle.

3.1.3.12 Horizontal Stabilizer Position Indication

3.1.3.12.1 Drive Crank

The curved crank arm that normally attaches to the horizontal stabilizer actuator support assembly was missing. (The other end of the crank arm is the attach point for the indicator cable feedback loop.) A portion of the crank mounting flange remained attached to the acme screw support assembly.

3.1.3.12.2 Indicator Cables

Only one of the two horizontal stabilizer position sensor cables running from the drive crank down the vertical rear spar was found. Segment 129F (horizontal stabilizer shutoff, aircraft nose up) from its fork cable terminal to its first turnbuckle forward was identified in the rear spar. A portion of the horizontal stabilizer position indication crank arm cable bearing was found in the fork terminal end of Segment 129F.

3.1.3.12.3 Stabilizer-In-Motion Sensing Device

The stabilizer-in-motion sensor assembly box was recovered separately. The input shaft to this box was bent and the input cable drum was not present. Examination of the stabilizer-in-motion input cable drum, which was recovered separately, revealed that cables (129C and 129B) were still cotter pinned to the drum. The drum shows a fracture at its mounting flange corresponding to a fracture face found on the input shaft of the motion sensor box.

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[19] The shaft's lower threaded portion, large flange washer and clamp-up retaining nut were never recovered.

The number “5756000-15 R” was found on the device (s/n 1369). The box was intact. The chain drive was still engaged on both gears. The input shaft was bent and jammed into the minor splines shaft. The device could be manually rotated. Both switches “clicked” when manually activated; however, only one of the two switches “clicked” with rotation of the cam. Of the one switch that did "click” upon cam rotation, only one of four cam lobes caused the switch to change state (i.e. “click).

3.1.3.13 Trim Relays

3.1.3.13.1 General Description and History

All of the trim relays were identified in the wreckage. One relay was attached to structure while the other was torn away from its back-to-back mounting orientation. The relay drive cranks were present, but the drive pushrods were not except for the attach fasteners.

According to representatives at the relay manufacturer (Leach), there have been no “unscheduled” returns over the past nine years of up and down trim relays from Alaska Airlines, but there were several returned units from another airline for mechanical shaft deformation. The representatives also stated that these relays were the subject of an AD in 1991 due to several DC-9 incidents involving excessive wear, thermal damage, and the potential for cargo compartment fires. As a result of the AD, the relays were hard-timed to 8,000 hours for DC-9s and 16,000 hours for MD-80s.

3.1.3.13.2 Down Trim Relay

The R20-15 down trim (lower) relay (model no. 9207-10296; p/n 258-0039-004-000; s/n MD92425; mfg. date codes 8918, 9608E, 9707E) was manufactured in 1989. No evidence of thermal damage or excessive wear on the contacts was noted. Corrosion was observed on all metal components. No evidence of any pre-impact mechanical malfunction was noted. The wiring harness remained attached to the terminals, and some terminal insulators/separators were missing. Electrical continuity through the coil was verified. Contact continuity of all four contacts was open. The relay functioned when 28 VDC power was applied. Additional testing could not be performed due to extensive physical damage to the relay.

3.1.3.13.3 Up Trim Relay

The R20-16 up trim (upper) relay (model no. 9207-10296; p/n 258-0039-004-000; s/n not available; mfg. date codes not available) exhibited greater damage than the down trim relay. No evidence of thermal damage or excessive wear on the contacts was noted. Corrosion was observed on all metal components. No evidence of any pre-impact mechanical malfunction was noted. The interlock wings of the relay were found loose in the relay mounting assembly; the pins that connect the wings to the relay shaft were sheared. Electrical continuity through the coil was verified. Additional testing could not be performed due to the extensive physical damage to the relay.

3.1.3.13.4 Trim Brake Relay No. 1

A determination of which trim relay was ANU and which was AND could not be made. For the sake of the examination, the relays were designated as no. 1 and no. 2.

Trim brake relay no. 1 (model no. 9274-7813; p/n 242-0018-004-000; date code 9126) was examined first and exhibited corrosion on its exterior surfaces. The base plate had been torn off the relay body. All “normally open” contacts were in the open position, and the “normally closed” contacts were in the closed position. Contacts H and K exhibited evidence of some arcing, which appeared to be “normal”(according to Leach representatives) for a relay of that vintage. No evidence of any pre-impact mechanical malfunction or abnormal wear was noted.

3.1.3.13.5 Trim Brake Relay No. 2

Trim brake relay no. 2 (model no. 9274-7813; p/n 242-0018-004-000; date code 9126) was examined and exhibited corrosion on its exterior surfaces. The base plate had been torn off the relay body. All “normally open” contacts were in the open position, and the “normally closed” contacts were in the closed position. No evidence of any pre-impact mechanical malfunction or abnormal wear was noted.

3.1.3.14 Trim Brake Override Switch

An evaluation of the trim brake override switch (activated by a suitcase handle) revealed no evidence of any pre-impact mechanical malfunctions. The switch serial number was 2761, and the part number was 1616. The date code was “9142” indicating that the switch was manufactured on week 42 of 1991. The switch was a “momentary,” spring-return to neutral, 3-position switch.

According to representatives of the switch manufacturer (Janco), no ADs had ever been issued against the switch. An electrical continuity check revealed that the switch was in the neutral (normally expected) position. An attempt was made to manually rotate the shaft, but the shaft was bound in place. Corrosion was observed on the inside of the switch.

3.1.3.15 Cockpit Trim Switch

Only the first officer’s primary trim thumb switch was recovered. It was found attached to the first officer’s yoke and was substantially damaged. The front cover of the switch and switch knobs were missing. All wires were found properly attached to the switch. Additional disassembly revealed that only leaf springs and a shim were present from the no. 1 contact. Four of the five leaf springs were able to “snap” back, while the fifth did not snap back. When the stack was reassembled, it was able to “snap” with no anomalies noted. No springs or shims were found in the assembly for the other three switch contacts.

According to representatives of the thumb switch manufacturer, the thumb switch is designed to last through 750,000 cycles. The most common failure mode is mechanical wear (typically due to leaf spring fatigue failures), and the manufacturer does not typically send out a “significant” number of spares.

3.1.3.16 Miscellaneous Longitudinal Trim Control Components

The following longitudinal trim system components were identified in the wreckage:

- The actuation link and torque tube for the primary longitudinal trim brake override switch (controlled by the outboard suitcase handle) with cables 127B and 128B attached.
- The horizontal stabilizer electrical shutoff cable drum (which drives the power relays) with cables 129A and 130A attached. The drum torque tube and drive pushrods were missing.
- One of the two alternate trim switches from the pedestal (power switch) with its input crank arm and a portion of the input pushrod still attached.
- The pedestal longitudinal trim indicator (“bug”) sector. It was bent, broken, and separated from the cockpit pedestal assembly.
- A piece of the cockpit trim indicator was found.
- A portion of the elevator variable load feel mechanism acme screw drive. A measurement of 5/8-inch was taken between the aft surface of the rod end and the cable drum end plate. The forward and aft stop gaps were measured to be ¼ and ½ inch respectively. The corresponding horizontal stabilizer position could not be determined.

3.1.4 Elevators

3.1.4.1 Cables and Pulleys

One-hundred percent of the LHS Elevator Control Cables (segments 3 and 4) were recovered and identified. Approximately 90 percent of the RHS Elevator Control Cables (segments 5 and 6) were recovered and identified. Missing from the RHS elevator cables were segments 5A and a portion of 6A, which mount to the first officer’s control column, and small portions of 5B and 6B in the vicinity of the aft pressure bulkhead.

The upper elevator waterfall pulley gang (six metal sheaved pulleys) was recovered with its pivot bolt still intact. Portions of the upper pulley bracket were found between the pulleys. Portions of the lower elevator waterfall pulley bracket (p/n 5937262-1) mounting pads were found on a portion of empennage structure. The lower elevator waterfall pulley gang was not recovered.

3.1.4.2 Elevator Controls in Horizontal Stabilizer

3.1.4.2.1 Elevator Control Tab Sectors and Torque Tubes

Both elevator control tab sectors and torque tubes were attached to the rear spar of the horizontal stabilizer. Both sectors were damaged. Both aircraft nose down (AND) sections of the sectors were fractured and missing from the installation. The aircraft nose up (ANU) section of the sectors was in place but no cables were attached. Both torque tubes were in place with the control sectors affixed at one end and the elevator control “balance beam” components attached at the other. The RHS torque tube's return spring was still attached; the LHS spring was not recovered.

3.1.4.2.2 Torque Tube Inboard Support “A” Frames

The torque tube inboard support “A” frames for both LHS and RHS were found intact in the horizontal stabilizer and appeared largely undamaged except for the disintegration of the large diameter torque tube ball bearings.

3.1.4.2.3 Tab Control Pushrods

Only a small portion of the RHS elevator and control tab were recovered intact with the recovered horizontal stabilizer structure. The RHS tab control pushrod and elevator surface position feedback link were intact and attached to the elevator tab and elevator surface respectively. The surface feedback link remained attached to the sector balance beam mechanism. The RHS tab pushrod was broken at its forward rod end bearing. The broken portion of the rod end bearing was attached at the sector balance beam. The LHS pushrod was not located, although a portion of the forward rod end bearing from this pushrod was still attached to the sector balance beam. The LHS sector balance beam component itself was attached to the LHS sector torque tube.

3.1.4.2.4 Elevator Pulleys and Pulley Brackets

The two pulley brackets forward of the elevator sector mechanism were broken away and not recovered. Portions of the mounting flanges of these brackets were still attached to the lower surface of the horizontal stabilizer.

3.1.4.2.5 Elevator Control Cables

Three of the four elevator control cable segments were identified in the vertical rear spar structure. Cable segment 5C (RHS, AND) from its elevator sector ball termination to its first turnbuckle forward was found, as well as a portion of 5B from the turnbuckle to an upstream fracture point forward of the turnbuckle. This 5C/5B turnbuckle was firmly imbedded in the four-pulley. Cable segment 4C (LHS, AND) was found at the top of the vertical spar with the cable ball still attached to the broken portion of the elevator sector. The opposite end of cable 4C was fractured at a point before its first forward turnbuckle. Cable segment 6C (RHS, AND) was also found at the top of the vertical spar with a portion of its broken elevator sector still attached. This cable terminated at its first forward turnbuckle. Detailed observations of the structure associated with these cables were documented by the Structures Group.

3.1.4.3 Elevator Pitch Augmentation System (EPAS)

Both LHS and RHS valves were in place in the horizontal stabilizer and appeared undamaged. The valve pushrods were found to be intact and connected with the sector control balance beam on both LHS and RHS elevator mechanical installations. The opposite ends of both LHS and RHS push rods were still connected to their respective valve input crank arms. The valve crank arms, however, were fractured and had separated from the valve on both sides. Both LHS and RHS hydraulic cylinders were found in the leading edge of the elevators. Visual examination of the cylinders indicated that the cylinders were in the full extend position; however, the EPAS cylinders could be moved easily by investigator and the position of the elevators and the state of the EPAS system at the time of the impact could not be ascertained.

The manual bypass valve for the EPAS system was recovered and examined. The valve appeared largely undamaged, but the lockwire that secures the valve knob arm in the full counterclockwise open (normal) position was broken. The valve was found in the 90-degree clockwise position from its full open position. A nick was found in the knob arm.

3.1.4.4 Elevator Anti-Float Tab Mechanism

3.1.4.4.1 Drive Cranks and Pushrods

Both LHS and RHS crank arms that extend below the surface of the horizontal stabilizer, aft of the horizontal stabilizer actuation system, were not recovered. The crank support bracket pivots on both sides exhibited evidence of impact damage and tensile overload Of the mating crank pushrods, only the portion inside the horizontal stabilizer remained; these ends were secured to the anti-float tab torque tube drive cranks. The fracture point of the pushrods corresponded with the clearance holes in the bottom of the horizontal stabilizer structure.

3.1.4.4.2 Torque Tubes and Cable Control

The torque tubes for both sides were found in the horizontal stabilizer; both had significant damage and did not function. The LHS outboard drive sector was found attached to the torque tube. One pulley bracket assembly (with the pulley still installed) was attached to the horizontal stabilizer structure outboard and forward of the LHS torque tube. The drive sector cable was not attached and not recovered. The LHS anti-float tab cable was found within the outboard-most section of the LHS horizontal stabilizer. The RHS anti-float tab components were not found as the RHS horizontal stabilizer was not recovered.

3.1.4.5 Elevator Position Sensors

3.1.4.5.1 Sensors

Both LHS and RHS position sensors were intact in the horizontal stabilizer. The electrical connectors and a portion of the wire and wire conduits were also intact for both sides. These wire conduits run inboard from the sensors generally along the horizontal stabilizer rear spar to a point several inches from aircraft centerline, where the conduits route upwards to the top of the horizontal stabilizer surface. Approximately a foot of wires extended forward beyond the end of the conduit tubes. The LHS and RHS sensor wires were severed at this point.

3.1.4.5.2 Sensor mounting brackets

The RHS sensor mounting bracket showed indications of fracturing. The LHS mount was intact with no apparent damage.

3.1.4.5.3 Sensor Input Cranks and Pushrods

Both LHS and RHS sensor input cranks were fractured. A portion of the crank remained clamped up against the sensor input shaft on each side. The remaining portions of the broken cranks were attached to their respective input pushrods on both sides. The input pushrods were bent and attached to the control tab mechanisms.

3.1.4.6 Elevator Servo Force Limiter Mechanism

3.1.4.6.1 Augmenter Arm and Pulley Assembly

The augmenter input pushrod was bent upwards almost 90 degrees. The opposite end of this pushrod was attached to a drive crank arm. A portion of the elevator autopilot servo cables (6A, 6D and 6E) was found entangled within the pulley arm.

3.1.4.6.2 Autopilot Servo

The autopilot servo box was impact damaged and found attached to its swing arm. The servo cable drum was also present with portions of servo cables (5E and 6E) still wrapped in the drum grooves. The servo swing arm concentric return springs were also found separated from the mechanism. The spring hooks on the servo side were both distended and broken. The opposite hooks, which mount to the limiter input sector arm, were not damaged.

3.1.4.6.3 Flap Input Drum

The entire flap input drum was present with a portion of its mounting bracket. The upper proximity switch and the lower 26-degree switch were present without wire connections. The switch adjustment turnbuckle was present and connected at both ends, one to the actual switch mounting bracket and one to a piece of structure. Of the two possible cable attachments to this drum, only a 25-inch section of the upper groove cable was present with the cable ball lockwired to the drum. The flap input drum cam roller crank arm was also recovered though separated from the rest of the mechanism. The roller crank arm pushrod was attached to the crank, but the opposite end of the pushrod was broken at the adjustable rod end. The roller crank arm support bracket was basically intact and attached to a piece of structure.

3.1.4.6.4 Sector ("barn door") Crank Arm:

The sector "barn door" crank arm was found in an assembly of several other components of the mechanism. The barn door crank was broken approximately 9 inches from its pivot point (the overall length of this part is 11.6 inches). The cable sector was not attached. Several scuff marks appeared on the outboard surface of the crank approximately 4.5 inches away from the pivot, and a chip was found on the edge of the same surface approximately 4.75 inches from the pivot. According to Boeing, the distance from the pivot to the major axis of the horizontal stabilizer position sensor used by the DFDR is 4.75 inches.

The pedestal indicator cable sector that normally attaches to the top of the barn door was not present. The position sensor drive crank to the first sensor (DFDR) was present and attached to the sensor input crank; however, the input crank was fractured and the position sensor was not present. A portion of the drive link to the second (forward most) sensor was fractured at a point prior to the second sensor input crank. This second sensor was not present either. The aft most sensor installation (for stall warning) was almost completely intact with the sensor, input crank arm, and link all present. The sensor body was bent and the sensor mounting bracket was not present. The servo arm spring attach point was still attached to another crank arm beneath the barn door. This crank arm was severely fractured in multiple places.

3.1.4.6.5 The Barn Door Cable Sector

This sector was found separate from the rest of the mechanism. Although intact, the inboard one-third of the sector (as viewed from above) was bent aft. Both cables were still attached to the sector at the lockwired cable balls. Cable 129C was found complete and intact; it was connected to both the barn door cable sector and to the HS-in-motion cable drum. Only a 4-inch portion of cable 129D was found with the sector.

3.1.4.6.6 Stick Pusher

The output drum of the Stick Pusher Stall Servo with cable 5F still attached was recovered. This component was found with the forward control cables. The body of the actuator was not recovered.

3.1.5 Rudder

The main torque tube of the rudder drive system remained intact. The upper portion of the torque tube was attached to the rudder surface fitting, while the lower portion of the torque tube was attached to the main power control unit (PCU) drive crank. The inner tab torque tube was also still intact. The upper portion of this torque tube was attached to the forward rudder tab pushrod crank; the lower portion was attached to the lower tab drive crank. The forward portion of the tab pushrod remained attached to the forward crank. This pushrod was severed approximately 11.5 inches aft of the torque tube. The lower torque tube support bracket was completely separated from the rest of the mechanism.

The input cable sector for the rudder control mechanism was broken away from its hinge point and was found in two pieces. Each piece of the sector had a cable ball still lockwired into the sector groove.

The yaw damper actuator was heavily impact damaged and had broken away from its structural mounting. The yaw damper output shaft remained connected to the mechanism. The distance from the face of the damper housing to the end of the chrome actuator shaft was measured to be 2.2 inches. According to Boeing, this corresponds to approximately the fully extended position of the yaw damper actuator shaft.[20]

The rudder position sensor mounting bracket was present; however, the sensor body itself was shattered, and only the sensor mounting collar and a portion of the internal rotor remained. The electrical connector was not present. The sensor input crank was broken. The input pushrod was not found.

[20] The yaw damper is authority limited to +/- 2 degrees of rudder (measured hinge-wise) or +/- 0.185 inches of linear actuator travel.

The rudder limiter mechanism "bellows", linkage, and input pushrods were not present in the empennage structure. Only a portion of the linkage support bracket could be identified. The rudder limiter hook was engaged slightly into the PCU actuator cylinder rod. The amount of engagement was estimated to be 5/8ths of an inch (estimated due to the approximate 30-degree axial rotation, clockwise when looking inboard from RHS, of the cylinder rod from its normal position). According to Boeing, this engagement corresponds to approximately 190 knots equivalent airspeed, and 12.5 to 15.5 degrees of rudder authority available.[21] The limiter proximity sensors were still attached to their mounts; however, the proximity sensor target was not present.

A portion of the rudder limiter pitot bellows was recovered consisting of the piston, diaphragm, and part of the mounting bracket. None of the downstream springs or linkages were present.

The PCU was found in the manual (hydraulics off) mode as evidenced by the condition of the hydraulic reversion mechanism (the input link to the control valve was found in a position that engaged the gripper arms). The amount of chrome showing from the cylinder rod (measured from the right end of the cylinder housing to the inboard face of the clamp-up washer) was 4-7/8 inches. The cylinder rod was slightly bent. According to Boeing, this corresponds to a 13.5-degree airplane nose left (ANL) position. When looking up at the reversion gripper arms, measurements of 2-1/4 and 2-3/8 inches (LHS and RHS respectively) were taken from the inside face of each arm tip and the center of the tab input crank bearing. The tab drive linkage could be easily moved by investigators; therefore, these measurements could have changed during salvage and are not useful in determining rudder surface position at the time of impact.

The rudder trim and load/feel actuator was heavily damaged and bent along its major axis. The cable drum portion of the actuator was detached and not found. The output end of the actuator was still attached to its input crank arm. Rudder trim position could not be determined in the as-recovered condition.

The rudder power shutoff sector was found separated from the hydraulic valve. The mounting bracket was fractured. A portion of the cable was attached to the sector. The arm that normally connects to the sector and valve was connected on the sector side and broken off on the valve side.

The rudder and aileron trim control drum assembly, normally mounted in the control pedestal, was recovered separately. Both trim cables were attached to their respective control drums.

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[21] The MD-83 rudder's maximum travel limit at lower airspeeds is 22.25 degrees.

3.1.6 Aft Pressure Bulkhead Interface with Flight Control Cables

The cable pressure seals on the lower LHS (LHS elevator and rudder) and upper RHS (RHS elevator, HS position, rudder trim) were examined and found to be intact on the aft pressure bulkhead. No evidence of ripping or garroting was seen. The pulley brackets for upper and lower RHS "waterfall" cable pulleys were not found. The mounting surface on the bulkhead for the lower "waterfall" bracket was not found either.

Two other pieces of the aft pressure bulkhead were also recovered. These two pieces comprised about an additional 15 percent of the aft pressure bulkhead. The larger of these two pieces contained a section of the rudder power shutoff cables, a negative pressure relief valve, and a portion of the right hand thrust reverser control.

3.1.7 Lubrication of Flight Control Actuators

The Group documented the characteristics of the lubrication on all recovered elevator and aileron actuators. The observations were recorded by the Group on a lubrication work card illustration (attached).

3.2 Fuel System

3.2.1 Left Fuel Tank and Associated Hardware

The forward fuel boost pump (s/n 5992A) was found without the upper portion of the volute. The pump was attached in the fuel tank with its fuel manifold secured to the pump outlet. The aft fuel boost pump (s/n 5394A) was found with a portion of the volute crushed on the outboard side. This pump was attached in the fuel tank with its manifold secured to the pump outlet. A fuel tank vent float valve (s/n 4950) was found damaged and secured to its manifold. All fuel manifolds in the left fuel tank were found crushed, bent, or kinked. A fault isolation junction probe was found detached and damaged with all fuel quantity wiring attached. All other fuel quantity probes were missing.

3.2.2 Center Fuel Tank and Associated Hardware

The fuel boost pump unit (two pumps in one container) was found with the container crushed around one pump. One pump was missing and the remaining pump (s/n 5937A) was attached inside the container, which was detached from the center tank. The fuel manifold, fuel quantity, and other associated fuel components were missing.

3.2.3 Right Fuel Tank and Associated Hardware

Three fuel fill valves were found to be damaged on the front spar, next to the no. 6 slat track. A fuel line (p/n 7939010-503) was found crushed. The fuel manifold, fuel quantity, and other associated fuel components were missing.

3.2.4 Forward Auxiliary Tank

Three fuel quantity probes were found detached inside the ruptured tank. Fuel manifolds were found crushed, bent, or kinked. The fuel boost pumps were missing. The fuel tank vent float valve was secured with its manifold. A 16-foot section of the manifold (p/n 5954715-1) connected to the center tank was found crushed.

3.2.5 Aft Auxiliary Tank

A fuel boost pump (s/n 3209A) was found damaged and detached inside the ruptured auxiliary tank. The remaining fuel pump was found to be secured with its associated manifold. All fuel manifolds were found to be kinked or dented. All three fuel quantity probes were detached from their mounts inside the tank.

3.2.6 Miscellaneous Fuel System Components

A fuel boost pump (s/n 5887A) was found damaged; its installation location could not be determined. A 29-foot section of engine fuel manifold section (p/n 5937268-501) was found crushed, kinked, or bent. A fuel check valve (Imco Industries Inc. p/n 71208-1, s/n 2270) was found detached and damaged at both ends; its installation location could not be determined.

3.3 Air Conditioning Systems

Both air conditioning system water separators were found flattened and attached together to their mounting brackets but without air conditioning components. The recirculation fan was found separated from its associated components; the fan case was split in half with four blades broken. The remaining components from the left air conditioning system were not recovered.

From the right air conditioning system, the ground cooling fan with cage was found detached from its components and ducting. The cage was crushed into the fan and approximately 25 percent of the mounting flange was broken off. The primary heat exchanger was found separated from its associated components. The outer case was bent and distorted. The secondary heat exchanger was found separated from its associated components and ducting. The air cycle machine (ACM) primary heat exchanger and secondary heat exchanger were found attached as an assembly. The ACM case was cracked. The flow control valve was detached from its associated components and the valve was found twisted along its length. The pressure regulator valve was found separated from its associated components and its diaphragm housing cracked.

3.4 Cabin Pressure Control

The cabin pressure controller nozzle valve was found separated from the fuselage. The butterfly valve was present and still functional. The valve drive assembly was present and corroded. The electromechanical actuator was found separated from the drive mechanism. The negative pressure relief valve was identified on the aft pressure bulkhead. The cabin pressure controller was also identified.

Two cabin pressure control units (AirResearch Electronics; p/n 2118490-2; s/n 17-8021; p/n 2118490-1, s/n 90-420) were recovered. Both were intact, but deformed. The outflow valve breakers (H-2 and J-2) were not recovered. Pieces of the panel where these breakers would normally be located were missing. The pressurization control circuit breakers (U-22, W-22) were not visible because the edge of the breaker panel was folded over their location.

3.5 Auxiliary Power Unit (APU) System

The APU was found contained inside the APU compartment. All accessories were found attached to the APU. The APU inlet door was closed. The generator cooling housing was pushed aft. Approximately 30 inches of the lower fuselage section aft of the APU was found attached to the APU compartment; the tailskid and shock absorber were still attached to this fuselage section. The pneumatic ground conditioned air duct and aft air stair control valve were missing. Approximately eight feet of bleed air duct from the APU load control valve to the pneumatic crossfeed valve was still attached to the APU assembly. The APU exhaust elbow was attached and pushed aft. The other section of the APU exhaust was detached from the APU exhaust and found crushed.

3.6 Recovered Avionics Components

- Display head of the Digital Flight Guidance Computer (only one of two identified). Honeywell; p/n 4034235, s/n 94052761.
- Collins Radio Altimeter no. 1, s/n 7550.
- VHF Transceiver, p/n 622-1181-001, s/n 9641.
- Erection Control Monitor, p/n 2588089, s/n 5121.
- Teledyne Controller p/n R6357-550, s/n 0122.
- TCAS Computer p/n 4066010-904, s/n 0327.
- Digital Computer Symbol Generator, p/n 4055900-905, s/n 0595.
- Generator Control p/n 947F945-3, s/n 3775.
- AC Bus Control Panel p/n 947F946-3, s/n 1393.
- Aural Warning Central Unit p/n H05A0035-4, s/n1258.
- Air Data Computer #2, p/n unknown, s/n unknown.

- Directional Gyro (2 ea), p/n unknown, s/n unknown.
- Collins ADF-1, p/n unknown, s/n unknown.
- LH Generator Control Units, p/n 947F945-3, s/n 3442.
- RH Generator Control Units, p/n 947F945-3, s/n 3766.
- VHF Nav-1 and –2, p/n unknown, s/n unknown.
- Honeywell Transponder –1, p/n 4061400-94, s/n 0576.
- Honeywell Transponder –2, p/n 4061400-94, s/n 0625.
- King Radio DME-1 and –2, p/n unknown, s/n unknown.
- Digital Air Data Computer #1, p/n HG280D80, s/n 2402.
- Air Data Computer Switching Unit, p/n 001048-101, s/n 1041.
- Ground Proximity Computer, p/n unknown, s/n unknown.
- Stall Warning Computer, Sundstrand Data Control; p/n 995-0449-002; s/n 698.
- Flight Data Acquisition Unit; Teledyne p/n 2222601-6, model 70-201G, s/n 2478.
- Unknown component p/n 1013893-0001, s/n unknown.
- Unknown component p/n 7892551-011, s/n 0958.
- Unknown component p/n 4034241-971, s/n 1755.
- Unknown Collins p/n 622-5129-205, s/n unknown.

An attempt was made to locate the digital flight guidance computers (DFGC) in order to determine if there were any non-volatile memory chips. The units were not identified; however, all loose circuit boards from the recovered wreckage were collected and boxed for an examination at Honeywell. A review of all circuit cards revealed that no DFGC cards were recovered.

3.7 Cockpit Documentation

3.7 1 Circuit Breakers

3.7.1.1 Right and Left Generator Main Panels (lower bulkhead behind left seat)

Left Generator Bus: The protection bar on this panel was deformed and pushed over on top of the panel; the bar was contacting all CBs on the left side of the panel. All circuit breakers (CBs) were found in the "opened" (or "tripped) position except for the following:

- All three "Primary Longitudinal Trim:" The left primary longitudinal trim CB was found halfway into the opened position; the other two CBs were out.[22]
- The CBs for the "Left AC Bus Sensing" were as follows: left - opened; middle - missing; right - opened.

Right Generator Bus: Overwing heat CB missing; two of three of the right AC Bus sensing were in; the third was missing. Of the six Right "AC Bus" breakers, one

[22] The vertically-oriented protection bar on this panel was deformed and pushed over on top of the panel; the bar was contacting all CBs on the left edge of the panel.

was missing, one was broken, one was in, and the other three were opened. All three of the ground service bus power CBs were missing. The top three rows of the CBs on this panel were missing or destroyed, except for the upper left galley power no. 2 CB, which was opened.

3.7.1.2 Upper Electrical Power Center Panel (upper bulkhead behind left seat)

This panel was broken in half. The majority of the CBs that were present were found in the opened position.

The “Autopilot and Alternate Longitudinal Trim” CBs were on this panel and the following observations were recorded:
D9 was out & bent; D10 was in & bent; D11 was out & bent.

The “Primary trim control/brake” CBs were also on this panel and the following observations were recorded:
G22 was missing; G23 was in.

3.7.1.3 Lower Electrical Power Center (EPC) Panel (mid to lower bulkhead behind left seat)

This panel was in one piece. The bottom half was more damaged (crushed inward) than the top half. The majority of the breakers in the top half were in the opened position. All the breakers were present except for the spoiler control.

3.7.1.4 Forward Flight Attendant Panel

This panel is located just outside the cockpit door. The Emergency Light Switch was jammed in the off position; its guard was missing.

3.7.1.5 Ground Service Bus Breaker Panel (aft left cockpit wall)

This panel was deformed (bent). About one-third of the breakers were broken off. The following breakers were found in the opened position: lower right sidewall lights; fwd & mid cargo light; aft left lavatory breakers; forward cabin utility outlet; charger transfer bus. Both of the Captain’s headset plugs were in their sockets with the wires separated from them. The Interphone switch was jammed in the “Interphone” (versus “mask”) position.

3.7.2 Center Pedestal and Throttle Quadrant

The center pedestal was severely deformed and twisted clockwise (when viewed from above). The speed brake handle assembly was separated and found retained in the ½ flight spoiler position notch. When viewed from above, the throttle quadrant was severely deformed and twisted clockwise. The cabin pressurization switch was in the automatic position. The cabin outflow valve indicator was jammed in the ½-closed position. The fuel crossfeed lever was closed. The left fuel shutoff lever was missing. The right fuel shutoff lever was deformed and in the "on" position. The rudder power lever was broken and hanging loosely from the pedestal. The alternate trim switch levers were both deformed; the right lever was slightly forward of the left lever. The top half of the Flap/Slat handle was broken, while the bottom half was attached to the pedestal and free to move. Both throttles were deformed; the right throttle was about 3 inches aft of the left throttle, consistent with clockwise rotation. The rudder trim knob was separated and the indicator was indicating 3 units. The aileron trim knob was missing. The red guarded "Stabilizer Trim Power Shutoff Switch" was missing.

The no. 1 radio control head displayed 126.52 on the in-use window and 135.5 on the standby window.[23] The no. 2 radio control head displayed 133.80[24] on the in-use window and 131.20[25] on the standby window.

3.7.3 First Officer's Instrument Panel

The only gauge present on the first officer's instrument panel was the left hydraulic quantity gauge, which indicated 12 quarts. The auxiliary pump switch was broken off, and its shaft was oriented in the off position. The right engine pump switch was broken off and its shaft was oriented in the high position. The left engine pump switch was broken off and its shaft was oriented in the low position. The transfer pump switch was missing.

3.7.4 Center Instrument Cluster Panel

The flap gauge indicated: left flap (needle loose) at -2 degrees, right flap (needle rigid) at +2 degrees. The remaining recovered gauges from this panel had broken or missing needles: N1 (right); N1 (left); EGT (right); N2 (right); Fuel flow gauge (left); Oil quantity (left); EPR (left or right position unknown) digital reading 1.62, limit window reading 1.98.

3.7.5 Captain's Instrument Cluster

All primary flight instruments on the Captain's side were missing or broken. The standby altimeter was broken and its needle was missing; this instrument was set to 1022 millibars. The inches window was damaged and unreadable.

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[23] These frequencies are the same as the frequencies used by the Los Angeles Air Route Traffic Control Center.

[24] This frequency is the same at the frequency used by the Los Angeles International Airport Automated Terminal Information Service (ATIS) for arriving aircraft.

[25] This frequency is used by Alaska Airlines as a "company frequency."

3.7.6 Forward Overhead Panel

The AC voltmeter selector switch was found in the "Battery Amp" position. The AC cross tie switch was jammed in "auto." The DC cross tie switch was jammed in "open." The engine ignition switch was jammed in "ground start and continuous." The emergency light switch was jammed in the "on" position. The no smoking switch, seat belt switch, and battery switch were all jammed in the "on" position. The APU master switch was broken off and its shaft was oriented in the "off" position. The APU fire control switch was jammed in "normal." The APU door guard was missing. The APU door guard switch was in the "auto" position. The engine sync switch was missing. The wind shear test switch was missing.

The radio rack switch was jammed in the "fan" position. The air conditioner shut-off switch was broken off and its shaft was in the "auto" position. The ram air switch was broken off and its shaft was oriented in the "off" position. The cabin altitude and cabin rate of climb gauges were missing. The emergency power selector switch was present, out, and detached. The left pack temperature control was in the 10:00 o'clock position and the right pack temperature control was in the 1:00 o'clock position. The right and left pack pressure gauges were missing. The pneumatic cross feed pressure gauge read 5. The AC and DC load meters were damaged and had no needles. The APU AC load meter was damaged and indicated "1" (needle loose). The left constant speed (CSD) drive temperature gauge was missing. The right CSD read 120 degrees C. Both AC volt and frequency gauges were missing. The DC/Volt/Amps gauge was broken and had no needle. The APU EGT gauge was broken and had no needle. The APU RPM gauge was broken; its big needle was missing and its little needle was pointed to 9. The cabin temp gauge was missing. The cockpit voice recorder and flight data recorder control heads were missing.

3.7.7 Miscellaneous/Separated Instruments

An altimeter (position unknown) was recovered; its needles were missing and its digital reading was 660 feet at 1022 MB and 30.19 inches Hg. An airspeed indicator was recovered; its needle indicated zero and the barber pole was pointing to 357 knots. The standby attitude indicator indicated a pitch-down attitude and steep right roll.

3.8 Oxygen Generators

The Group searched for all oxygen generators and masks that were recovered in an effort to ascertain if they had been activated before impact. Only ten of the approximately 64 installed passenger service units (PSUs) were found. None of the recovered oxygen masks showed evidence that the head straps had been cinched. No evidence was found to indicate that any of the ten recovered PSUs had been activated before impact.

4.0 Powerplants

4.1 Left Engine (no 1): JT8D-217C, s/n: 728068

4.1.1 General Findings

No sign of fire was observed. No signs of engine uncontainment were observed. The engine separated into two sections at the high compressor, which was not recovered. The two sections, forward and aft of the high compressor, were recovered. None of the engine nacelle components were attached to the recovered engine. A significant portion of the engine nacelle hardware was recovered and found to be fragmented.

4.1.2 Front Accessory Drive Group

The front accessory drive housing, the no. 1 bearing scavenge oil pump, and the N1 tachometer drive were missing.

4.1.3 Compressor Inlet Group

Two pieces of the outer diameter of the fan inlet structure were recovered. Each piece spanned the circumference of 10 fan inlet guide vanes. The remaining section of this diameter was not recovered. The inner diameter of the fan guide vane structure split axially and unraveled; it remained attached to the inlet guide vanes. The flanges of the outer diameter of the no. 1 bearing support structure fractured; however, the bearing support remained intact. No signs of rotational distress were observed on the number 1 bearing front face.

4.1.4 Front Compressor Group

The fan front case was split in two pieces, consistent with the fan inlet outer diameter case. The acoustic treatment material in the fan front case had imprints similar to fan blade or inlet guide vane profiles. The entire fan rear case was recovered. The fan case rub strip exhibited circumferential rub marks around the entire case. All 34 fan blades were fractured just above the blade platform. No signs of distress or rupture were observed on the integral fan hub and fan disk.

All six stages of low compressor blades were bent opposite the direction of engine rotation, and the low compressor vanes were bent toward the direction of rotation. A section of the fan air splitter, between 9:00 o'clock[26] and 11:00 o'clock, was missing, while the remaining fan air splitter was pushed radially inward on the

[26] The clock convention for position is used by viewing the engine from the rear and referencing in the direction a clock would move.

right side of the engine. The fan exit guide vane outer diameter case was intact and pushed radially inward on the right side of the engine.

4.1.5 Compressor Intermediate Group

The fan discharge front compressor outer duct fractured circumferentially roughly 13 inches aft of the E flange. This duct was also crushed axially, with the most significant deformation occurring around the 9:00 o'clock position. A 3-inch wide section of the fan discharge inner duct remained attached to the intermediate case forward flange. The intermediate case secondary flow struts around the 3:00 o'clock position were bent over against the inner diameter of the fan discharge flow path. The compressor intermediate case outer diameter, between flanges F and G from 12:00 o'clock to 6:00 o'clock, was missing. The fan discharge rear compressor outer duct and inner duct were missing. The engine separated just aft of the intermediate case. The low rotor shaft was separated in line with the tower shaft. The no. 3 bearing housing did not exhibit any signs of heat or rotational distress.

4.1.6 Rear Compressor Group:

Only the rear hub of the rear compressor rotor assembly was recovered. All other hardware was missing from the recovered engine. The hub did not exhibit any signs of rotational distress or rupture. The engine separated at the forward flange of the high compressor rear hub.

4.1.7 Diffuser Group

The compressor exit outer diffuser duct was intact. Fifteen of the 99 13th stage stators exhibited minor impact damage to the leading edges. These stators were not adjacent to one another. The engine separated at the forward flange of the compressor exit outer diffuser duct.

4.1.8 Diffuser Outer Fan Duct Group

A section of the diffuser outer fan duct, between 4:00 o'clock and 7:00 o'clock, was recovered.

4.1.9 Fan Discharge Section

A section of the fan exhaust outer duct, between 3:00 o'clock and 7:00 o'clock, remained attached to the engine. The fan exhaust inner duct, between 9:00 o'clock and 2:00 o'clock, was crushed inward to the High Pressure Turbine (HPT) containment ring. The HPT containment ring was intact but pushed aft 2 inches at the 12:00 o'clock position. The fan exhaust inner duct segments were in place.

4.1.10 Combustion and Number 5 Bearing Section

The combustion chamber outer case was crushed radially inward from 8:00 o'clock to 12:00 o'clock. No distress was observed within the no. 5 bearing compartment (visible around the compressor rear hub). No metal splatter was observed on the forward surfaces of the combustion chambers (visible through the 13^{th} stage stator vanes). No other hardware from this section was visible during the on-scene inspection.

4.1.11 Turbine Nozzle Group

The forward flange of the turbine front case, from 10:00 o'clock to 3:00 o'clock, separated from the combustion chamber outer case. No other hardware from this section was visible during the on-scene inspection.

4.1.12 Low Pressure Turbine Group

The low-pressure turbine case was intact; however, the rear flange was pushed radially inward at the 12:00 o'clock position. All the 4^{th} stage blades were fractured just above the attachment platform. The 4^{th} stage vanes were intact and in place.

4.1.13 Engine Exhaust Case Group

The fan exhaust outer duct separated from the engine at the forward flange and split open axially. About 20 percent of the circumference was not recovered. The fan exhaust inner duct exhibited impact damage between 8:00 o'clock and 2:00 o'clock. The exhaust case was intact, but bent radially inward between 7:00 o'clock and 11:00 o'clock.

4.1.14 Fan and Turbine Integrated Exhaust Group

The turbine exhaust duct was crushed radially inward around the 3:00 o'clock position. The exhaust cone was missing. The exhaust mixer was bent to the left, consistent with the crushed turbine exhaust duct. Both the front and rear fan exhaust outer ducts were missing.

4.1.15 Thrust Reverser

The thrust reverser separated from the fan exhaust outer duct at the forward flange of the outer duct. The thrust reverser was crushed radially inward at the 3:00 o'clock and 9:00 o'clock positions until nearly flat. The thrust reverser was found in the stowed position.

4.1.16 Externals

The main accessory gearbox was missing from the recovered engine. The main fuel pump was recovered and showed no signs of pre-impact distress. Neither the front nor the rear engine mount remained attached to the engine.

4.2 Right Engine (no. 2): JT8D-217C, s/n: 726852

4.2.1 General Findings

No sign of fire was observed. No signs of engine uncontainment were observed. None of the engine nacelle components were attached to the recovered engine. A significant portion of the engine nacelle hardware was recovered and found to be fragmented.

4.2.2 Front Accessory Drive Group

The front accessory drive housing, the no. 1 bearing scavenge oil pump, and the N1 tachometer drive were missing.

4.2.3 Compressor Inlet Group

The outer diameter of the fan inlet and all 23 fan inlet vanes were missing. The outer diameter of the no. 1 bearing support structure remained intact. No signs of rotational distress were observed on the no. 1 bearing front face. The inlet flow guide, located aft of the inlet inner diameter, was buckled radially inward at the 6:00 o'clock position.

4.2.4 Front Compressor Group

The entire fan front case and the entire fan rear case were missing. One fan blade was fractured 5 inches above the platform; the adjacent blade, in the direction of engine rotation, was fractured 12 inches above the platform. Ten fan blades, adjacent and ahead of the two fractured blades, were bent to varying degrees opposite the direction of rotation. Seven fan blades, following the fractured blades, were bent slightly opposite the direction of engine rotation. The remaining 15 fan blades exhibited nicks and local deformations, but were not bent. All 34 blade roots remained in the fan hub. None of the fan blades exhibited significant blade tip rubbing. No signs of distress or rupture were observed on the integral fan hub and fan disk.

All observed compressor blades and vanes did not exhibit any signs of distress from engine rotation. The stage 1 compressor stators were liberated from the inner diameter. The fan air splitter was intact. The fan exit guide vane outer diameter case was intact and pushed rearward.

4.2.5 Compressor Intermediate Group

The fan-discharge front compressor outer duct, between 12:00 o'clock and 6:00 o'clock, was missing. The fan-discharge inner duct exhibited impact damage around the entire circumference. The intermediate case secondary-flow struts were fractured between 1 inch and 5 inches above the secondary-flow inner diameter. The compressor intermediate case outer diameter, between flanges F and G from 12:00 o'clock to 6:00 o'clock, was missing. The fan-discharge rear compressor outer duct, from 12:00 o'clock to 6:00 o'clock, was missing. The fan discharge rear compressor inner duct was intact.

4.2.6 Diffuser Group

The compressor exit outer diffuser duct was fractured from 11:00 o'clock to 7:00 o'clock, 6 inches aft of the forward flange. The fracture opened to a maximum width of roughly 2 inches at the 3:00 o'clock position. No damage was observed on the 13th stage stators that were visible through the diffuser duct fracture.

4.2.7 Diffuser Outer Fan Duct Group

This hardware was missing from the recovered engine.

4.2.8 Fan Discharge Section

Only a segment of the fan exhaust inner duct, between 8:00 o'clock and 10:00 o'clock, remained attached to the engine. The HPT containment ring was intact and in-place. The remaining hardware of this section was missing on the recovered engine.

4.2.9 Combustion and Number 5 Bearing Section

No metal splatter or distress was observed on the burner hardware that was visible through the diffuser case fracture. The combustion chamber outer case was crushed radially inward between the 12:00 o'clock and 6:00 o'clock position.

4.2.10 Turbine Nozzle Group

The forward flange of the turbine front case was crushed radially inward, consistent with the combustion chamber outer case. No metal splatter or distress was observed on the 1st stage turbine nozzle guide vanes, visible through the damage openings in the turbine front case.

4.2.11 Low Pressure Turbine Group

The low-pressure turbine case was fractured from 11:00 o'clock to 7:00 o'clock, 4 inches aft of the forward flange. The fracture opened to a maximum width of roughly 2.5 inches at the 3:00 o'clock position. No rotational distress was observed on either the 2nd stage vanes or 2nd stage blades, visible through the fractured low-pressure turbine case. No rotational distress was observed on the 4th stage blades; however, a section of 13 adjacent blades was fractured between 1 inch and 7 inches above the inner platform, and a section of 2 adjacent blades was fractured at and 1 inch above the inner platform.

4.2.12 Engine Exhaust Case Group

The fan exhaust outer duct separated from the engine at the forward, or K, flange over roughly one-third the diameter, and separated at the forward rail for another third of the diameter. The remaining separation occurred between the rails and centered about the 3:00 o'clock position. This duct was also bent rearward and radially inward around the 3:00 o'clock position. The fan exhaust inner duct exhibited impact damage between 12:00 o'clock and 6:00 o'clock. The exhaust case was intact, but bent radially inward between 1:00 o'clock and 5:00 o'clock.

4.2.13 Fan and Turbine Integrated Exhaust Group

The turbine exhaust duct was crushed radially inward around the 3:00 o'clock position. The exhaust cone was missing. The exhaust mixer was bent to the left, consistent with the crushed turbine exhaust duct. Both the front and rear fan exhaust outer ducts were missing.

4.2.14 Thrust Reverser

The thrust reverser separated from the fan exhaust outer duct at the forward flange of the outer duct. The thrust reverser was crushed radially inward at the 3:00 o'clock and 9:00 o'clock positions until nearly flat. The thrust reverser was found in the stowed position.

4.2.15 Externals

The main accessory gearbox was missing from the recovered engine. The main fuel pump and the oil tank were recovered and showed no signs of pre-impact distress.

The front engine mount and a section of the pylon structure remained attached to the engine. The rear engine mount hardware was missing from the recovered engine.

5.0 Tulsa Ground Testing and Examinations of MD-80 Flight Controls

5.1 Purpose and Background of Tests

The Systems Group met in Tulsa, Oklahoma on February 29 and March 1 2000, to perform research of the MD-80 series flight control system. The following activities were performed:

- Observation/Familiarization of horizontal stabilizer acme screw and nut endplay and free-play check procedures.
- Observation/Familiarization of longitudinal trim actuation system operation while operating in various positions and with primary and alternate motors.
- Observation of stabilizer and elevator cable runs in the vicinity of the flap/slat actuation hardware.
- Detailed physical measurements and observations of the stabilizer acme screw in various positions.
- Observations, both visually and with the use of the DFDR, of elevator position and elevator position sensors in various positions, including those beyond the normal range of motion of the sensor, and with electrical disconnects.
- Observations, both visually and with the use of the DFDR, of stabilizer position, stabilizer actuation hardware, and stabilizer position sensor in various positions, including those beyond the normal range of motion of stabilizer cable actuation hardware, and with electrical disconnects.

Plots of the DFDR information obtained from these tests are attached.

The airplane that was used in Tulsa was a McDonnell Douglas MD-82. It had accumulated 25,544 flight hours and 12,889 cycles at the time. The acme nut and screw assembly were original equipment (Peacock s/n P 2742). The elevator position sensors were manufactured by Honeywell (p/n 4034233-901). The horizontal stabilizer position sensors were also manufactured by Honeywell (p/n 4034233-901).

5.2 Elevators and Sensor Range and Limit Observations

The right and left elevators were moved throughout their entire nominal range of motion while being recorded by the DFDR. The position sensor input link was then disconnected at two different locations and the sensor was manually driven to its internal stops or to the limits of structural contact, whichever came first. The elevator position sensors themselves were then rotated throughout their entire range while their output was recorded on the DFDR. The electrical connectors were then removed for DFDR recording, and the excitation voltage was shut off via the cockpit circuit breaker. This was done to simulate various modes of disruption of electrical power to the sensors. All of these tests were done to simulate any possible severing of the elevator and/or sensing mechanisms.

5.3 Longitudinal Trim Actuation Operation and Documentation

The horizontal stabilizer trim was operated throughout its full range of nominal motion with the use of both primary and alternate trim motors. Digital video of the operation was taken. The Materials Group recorded various physical dimensions of the acme screw and surrounding structure in various positions.

5.4 Control Cable Routing Examination and Observations

The passenger seats and floorboards above the flap/slat actuation hardware were removed to afford a view of the cable routing in this area. Digital video and still photographs were taken of this area. The horizontal stabilizer position cables (cables 129/130) and the right elevator position cables (cables 5/6) were observed to be about 4 inches directly above the flap/slat sequencing mechanism. The purpose of this examination was to determine if cable sagging due to a cable or hardware failure could interfere with flap/slat actuation. The Group did not ascertain any probable command-side cable jam or fouling due to obvious potential failures in the noted cables.

5.5 Horizontal Stabilizer Hardware and Sensor Range and Limit Observations

The longitudinal trim system was operated throughout its entire nominal range in increments of one unit (as measured on the cockpit indicator) while physical dimensions of the acme screw and sensor hardware locations were measured for baseline data. The trim system was moved through its entire range (electrical stop airplane nose up to the electrical stop of airplane nose down).

The longitudinal trim system position cable (cables 129/130) follow-up system was then detached from the crank arm (at the stabilizer drive unit) and manually moved beyond the nominal limit of the electrical stop in the airplane nose down (AND) direction. Digital and still photo documentation was made of the crank arm cable movement, electric stop sensor, and position sensor hardware. It was noted that the range of travel comes to a physical stop initially at the relay shutoff mechanism located in the forward cargo compartment.

The relay shutoff mechanism was then taken out of the cable loop and the loop was driven further to determine the next physical stop. This next stop was determined to be the elevator servo force mechanism input arm contacting the horizontal stabilizer position sensor body about 4.5 inches away from the pivot. While moving to the second stop, the pedestal horizontal trim indicator and the elevator variable load feel mechanism did not reach their respective physical stop in the test aircraft.

The horizontal stabilizer position sensor itself was then rotated throughout its entire range while its output was recorded on the DFDR. The electrical connector was then removed for recording, and the excitation voltage was shut off via the cockpit circuit breaker. (Detailed steps for this process are attached.) This was done to simulate various modes of disruption of electrical power to the sensor.

6.0 Seattle Flap/Slat Sequencing Ground Test

On March 7, 2000, some of the Systems Group members participated in a ground test (test plan and detailed results attached) of an Alaska Airlines MD-82 in Seattle, Washington, under the supervision of an NTSB investigator from the Seattle Regional Office. The purpose of the test was to obtain additional information related to the flap/slat sequence.

The results of the flap/slat sequence test revealed that the DFDR recorded flap movement about six seconds prior to slat movement when the flap/slat handle in the cockpit was moved to the flaps 11 position in one continuous movement.[27] This recorded sequence was opposite to the actual movement of the flaps and slats that was visually observed on the ground during the test. The recorded ground test sequence data was consistent with the DFDR data recorded during the accident flight, with the exception that the flap movement on the accident flight was recorded about three seconds prior to the slat movement.

7.0 Longitudinal Trim System Acme Screw and Nut Wear Checks

7.1 Revised Maintenance Procedure

On April 13, 2000, as a result of feedback received from airlines, Boeing issued a temporary revision to the longitudinal trim system acme screw and nut wear check procedure[28] that clarified and augmented several steps. They also issued several all-base messages that addressed this procedure (attached).

7.2 Observations of Wear Checks

During the course of the investigation, some members of the Systems Group observed several horizontal stabilizer actuator endplay checks performed by two MD-80 operators on four airplanes, and also observed a check performed by the supplier[29] of an acme screw and nut assembly on a test bench. Additionally, certain members of the Systems Group reviewed the results of hundreds of endplay checks that were

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[27] Whenever the flap/slat handle is moved to the flaps 11 position from the retracted position in one continuous movement, the system design is such that the slats will extend to the mid-slat position while the flaps simultaneously move to the 11-degree position.
[28] This procedure is described in detail in the Maintenance Records Group Chairman's Factual Report.
[29] The current supplier of the acme screw and nut assembly is now Integrated Aerospace, Inc. (formerly known as Trig Aerospace, Inc.), and is located in Santa Ana, California. The assembly on the accident aircraft was supplied to McDonnell Douglas by the Peacock Company of Norwalk, California on June 28, 1990. In late 1994, Peacock was purchased by the Derlan Company, which in turn was purchased by Trig Aerospace in July 1999, which then became Integrated Aerospace, Inc., in October 2000.

submitted to the FAA as a result of an AD (attached) that was issued immediately after the accident.[30]

It was noted that accurate results[31] of the endplay check procedure are dependent on the use of properly calibrated dial indicators, the configuration of the dial indicator set-up, the correct application of specified torque to the restraining fixture, a correctly fabricated[32] and maintained restraining fixture, and the lack of rotation of the acme screw within its gearbox.

7.3 Alaska Airlines Oakland Maintenance Facility Visit

On March 10, 2000, some members of the Group traveled to an Alaska Airlines heavy maintenance facility in Oakland, California, to gather data regarding horizontal stabilizer endplay checks and lubrication of the acme screw. The following information was noted:

The facility had only one Horizontal Stabilizer Restraining Fixture (p/n 0-1301-0-0169, de no. 070812), Dial Indicator set (no. 0-1301-0-1193, CE no. 0814), and Go-No-go Gage (ce no. 70868, p/n 804605).

The facility has many different types of grease stored in various locations and containers. Two unmarked "grease pots" were located on the tail stand that was being used for the endplay check procedure. The grease was red in these pots. Tubes of Aeroshell 33 were kept in stores, and also loaded in individual mechanics' grease guns.

The airplane used, N932AS, was in for a 17-day C-check. The airplane had logged 45,521 hours and 29,586 cycles. The last endplay was recorded as 0.032 when Phase II of the AD was performed on Feb. 11, 2000. The airplane also had reported brass-colored shavings at that time. Two brass-colored shavings measuring about 1/16-inch were found in the grease droppings by the Group.

An Acme Screw and Nut Wear check was performed by two sets of mechanics. The procedure was observed by the Group members from start to finish, including panel removal, which took about 2 hours.

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[30] On February 11, 2000, the FAA issued AD 2000-03-51 which requires all U.S. operators of Douglas DC-9 series, McDonnell Douglas MD-80/90 series, and Boeing B-717 series airplanes to check the wear of stabilizer trim acme screw and nut assemblies every 2000 flight hours.

[31] Endplay checks were performed on-wing by Alaska Airlines under the supervision of NTBS and System Group members. The observations of these checks are contained in this report. In many instances, the on-wing results were compared with more accurate bench test results conducted by the Materials Group and the manufacturer of the acme screw and nut assembly.

[32] The investigation has revealed that Alaska Airlines used a company-fabricated restraining fixture for all endplay checks that they performed prior to the accident. It was discovered that other airlines were also using fabricated tools. Some of the fabricated tools did not meet the drawing specifications provided by Boeing. The Systems Group is currently investigating the significance of fabricated fixtures versus the Boeing-supplied fixture.

Mechanic no. 1 stated that he last performed an endplay check about six months before the Group's arrival. He chose 275 in-lbs. for torque on the fixture and initially received a 0.000-inch endplay for several attempts after torqueing the fixture. The inspector that was paired up with him stated that the 0.000-inch reading would be what he would use for the final reading; however, the mechanic realized a few seconds after the inspector made this statement that he had torqued the fixture in the wrong direction (i.e. lengthened it, instead of shortening it, as stipulated in the procedure.). After this correction, the endplay was consistently measured to be 0.032 inch.

At the Group's direction, mechanic no. 1 then performed the check at 250 in-lbs. and 300 in-lbs. The results were 0.033 in. at 250 in-lbs. and 0.032 in. for 300 in-lbs.

Close observation revealed that the acme screw rotated slightly about two to three degrees during the torqueing and untorqueing of the restraining fixture, and it appeared that the rotation occurred due to gearbox backlash. The Group asked Mechanic no. 1 to repeat the process with the same set-up, except to restrain the acme screw with a large wrench. (Restraining the jackscrew was not specified in the procedure at that time.) The readings at 250 in-lbs. came up consistently at 0.036 in., and at 300 in-lbs., the reading was 0.0365 in.

Mechanic no. 1, under direction, dismantled the dial indicators, and invited Mechanic no. 2[33] to set them up again and repeat the process. Mechanic no. 2 chose 275 in-lbs. on the restraining fixture to perform the check; the results were 0.031 in. on the first two checks, and 0.033 in. on the third. He stated that he would have been satisfied with three attempts and that the acme screw and nut passed the endplay as the "average would be 0.032 inches." Again, the acme screw rotated during each attempt. Mechanic no. 2 was asked to perform the check using 300 in-lbs. (he got 0.033 in.), and 250 in-lbs. (0.032 in.). He then restrained the jackscrew while he performed the check at 250 and 300 in-lbs.; he consistently observed 0.035 in. for all attempts.

Mechanic no. 2 was then instructed by the Group to completely dismantle the fixture and dial clamps, reposition the stabilizer, and start the procedure from the first step again. For the first attempt (acme screw unrestrained) at 275 in-lbs., he measured an endplay of 0.035 in. on his first attempt, and 0.032, 0.032, and 0.031 in. on three more attempts. He then measured 0.032 and 0.033 in. during two attempts at 250 in-lbs., and 0.032 and 0.031 in during two more attempts at 300 in-lbs. Mechanic no. 2 then restrained the acme screw and measured 0.036 in. at 300 in-lbs., and 0.035 at 250 in-lbs.

Mechanic no. 2 was instructed to lubricate the acme screw and then perform an endplay measurement (acme screw unrestrained) at 250, 275, and 300 in-lbs. on the restraining fixture. He measured 0.034 inches for all attempts.

[33] Mechanic no. 2 said that it had been about 3 or 4 months since his last endplay check experience.

Additional endplay readings were taken by the Group by placing the dial indicator plunger at about a 35-degree angle instead of perpendicular to the acme nut. At 250 in-lbs. (unrestrained), the mechanic measured 0.034, 0.032, and 0.032 in. on three attempts respectively. At 300 in-lbs., he measured 0.030, 0.030, and 0.0315 in. on three attempts. At 300 in-lbs. of torque applied, and with the acme screw restrained, he measured 0.037 in. on all three attempts.

Mechanic no. 2 was then instructed to mount the dial indicator perpendicular again (normal position), but as far out from the centerline of the jackscrew as possible on the c-clamp. (According to representatives at Trig Aerospace, endplay readings are sensitive to radial play movement: the further away from the centerline of the acme screw the greater the sensitivity.) The mechanic measured 0.035 at 250 in-lbs. (unrestrained), and 0.035 in. at 250 in-lbs. (restrained.)

7.4 Wear Check of Oakland Hardware at Trig Aerospace

After the acme screw was measured on the airplane, the Group ordered that the acme screw be removed and sent to Trig Aerospace for another endplay measurement utilizing the facility and processes of the manufacturer in an effort to obtain further information about endplay measurements and their meaning.

The acme screw (p/n 1558) assembly was placed on a bench, and a magnetic dial indicator was affixed to the screw. The plunger of the indicator was made to contact the acme nut. The acme nut was manually moved (as per Trig procedures) in an attempt to obtain an endplay measurement. For the first two attempts, the endplay was 0.001 in. After the grease was distributed, the endplay was consistently reading from 0.025 in. and 0.027 in. (Note: Representatives from Trig indicated that they normally perform endplay checks on overhaul-bound acme screws after all of the grease has been steam cleaned from them.)

At the request of the Group, the assembly was then affixed to a Boeing fixture that is normally used to perform acceptance tests.[34] The screw support assembly was bolted to the fixture, without the gearbox attached, so that there would be no rotation of the screw due to gearbox backlash. The magnetic dial indicator was then affixed to the screw and a 2,076-pound weight was placed on the fixture to simulate the weight of the horizontal stabilizer. During the first attempt, the acme nut was about three inches from the top of the screw. The dial indicator was set to zero while it was under full load, and then the load was slowly pulled off the fixture via a hydraulic hoist. (This simulated what the Alaska Airline's maintenance procedures call for during an endplay check; i.e. torque the horizontal stabilizer to shorten the fixture, and thus relieve the load of the stabilizer while watching the dial indicators.) When the load was alleviated,

[34] Trig does not perform endplay checks by using this fixture during their normal course of operation. They only use the bench check mentioned above and only after the grease has been cleaned off of the assembly.

the endplay was measured to be 0.036 in. The load was then applied and the dial indicator went back to zero. The load was then relieved and the endplay was measured at 0.035 in. The load was then applied again and the dial indicator read zero.

The acme nut was then repositioned to about the middle of the acme screw. The load was applied and the dial indicator was set to zero. The load was then removed and the endplay was measured to be 0.036 in.

The acme screw/nut assembly was then attached to the motor/gearbox assembly on the test fixture, and the endplay was taken again as the fixture was unloaded. Again, the endplay was measured to be 0.036 in. The acme screw was then manually rotated by hand to observe the slack in the gearbox. As the acme screw was rotated to its limits, the dial indicator went from 0.036 to 0.039.[35]

The acme nut was then removed from the acme screw. The threads inside the acme nut were rough, nicked, and had sharp edges. The acme screw was also examined in detail. The non-wearing sides of the threads had evidence of manufacturing tooling marks that had a “tiger stripe” pattern. The acme screw and nut were then steam cleaned, as per the overhaul process, to remove all of the grease in order to prepare for the next endplay measurement.

Following a thorough cleaning, the acme nut was screwed onto the acme screw, and several endplay bench checks were performed as the assembly was exposed to manual force by Group members and Trig personnel. The endplay was consistently measured to be 0.032 in. at various nut positions along the acme screw.

The nut was then taken off the screw and placed back on the screw via the incorrect thread start. The incorrectness of the start was verified when the nut stop improperly struck the lower stop nut target. The endplay was then measured at 0.032 in. and 0.033 in. during several attempts. The nut was then placed back on the screw via its correct start thread for the next endplay check on the fixture.

The assembly was then bolted to the test fixture (without connection to the motor gear box, so that gearbox slack rotation was not a factor). The magnetic dial indicator was placed on the acme screw, and numerous endplay readings were recorded as the fixture was unloaded and as the acme nut was repositioned to various locations along the acme screw. The readings were consistently indicating 0.032 in., or about 0.002 less than the bench test.

The acme nut and dial indicators were then repositioned to determine if the use of the opposite flat of the acme nut would have an effect on the endplay measurement.

[35] According to the DC-9/MD80 Overhaul Manual (Acme screw section), the backlash may be as high as 2.6 degrees of rotation.

The measurements consistently indicated 0.034 in. as the weight was relieved from the fixture.

7.5 Alaska Airlines MD-83; N982AS; Seattle, Washington

On February 9, 2000, metal shavings were found in an Alaska Airlines MD-83, N982AS, during a fleet inspection in Seattle, Washington. An NTSB field investigator from the Seattle Regional Office was dispatched to investigate. Metal shavings, some measuring 1/8-inch in length, were found in the well directly beneath the jackscrew and in the grease on the jackscrew. A semicircular shaving measuring about 2.5 inches in length was also found. Grease samples were taken and sent to the NTSB Materials Laboratory. An on-wing endplay check was performed. The check revealed that the endplay was 0.051, 0.0525, and 0.0525 inches for the first three attempts at 275 in-lbs. and 250 in-lbs. At a torque of 250 in-lbs., the endplay was 0.082 on three successive attempts. The jackscrew and gimbal nut assembly was removed and sent to the NTSB Materials Laboratory. The endplay reading as performed by the NTSB Materials Group, after the assembly was cleaned, was 0.046 inches in a loaded condition.[36]

7.6 Alaska Airlines MD-83; N981AS; Portland, Oregon

On February 9, 2000, metal shavings were found in an Alaska Airlines MD-83 during a fleet inspection in Portland, Oregon. An NTSB field investigator from the Seattle Regional Office was dispatched to investigate. Metal shavings, some measuring $^{1}/_{8}$-inch to ¼-inch in length, were found in the well directly beneath the jackscrew and in the grease on the jackscrew. Several semicircular shavings measuring about 2.5 inches in length were also found. Grease samples were taken and sent for analysis. An endplay check was performed. An on-wing endplay check revealed that the endplay was 0.055 inches. The jackscrew and acme nut assembly was removed and sent to the NTSB Materials Laboratory. The endplay reading as performed by the NTSB Materials Group, after the assembly was cleaned, was 0.053 inches in a loaded condition.

7.7 Tulsa, Oklahoma

During the ground tests conducted by the Group in Tulsa on February 29, the Group observed maintenance personnel from another airline perform the acme screw endplay and free-play procedure on that airlines' airplane. The airplane used was a McDonnell Douglas MD-82 with the following data: Operating Hours: 25,544, Cycles: 12,889. The airline personnel installed the dial indicators on the upper and lower stops, and installed the fixture needed to provide torque force on the jackscrew. Two different sets of mechanics performed the procedure, and three different torque values were used (250 in-lbs., 275 in-lbs., and 300 in-lbs.). The personnel restrained the acme screw

[36] Details regarding endplay checks that were performed by the Materials Group are documented separately in the Group's laboratory reports.

without instruction by the group, and as per that airline's maintenance procedure. The following results were recorded by the first set of mechanics:

Torque Applied	Free-Play	Endplay
250 in-lbs.	0.004	0.012
275 in-lbs.	0.004	0.012
300 in-lbs.	0.004	0.012

The endplay readings that were taken by the second set of mechanics did not vary from these by more than 0.001 inches.

8.0 Pertinent Reported Endplay Data

8.1 FAA Report of Acme Screws with Excessive[37] Endplay

On February 11, 2000, the FAA issued AD 2000-03-51 (later superseded by AD 2000-15-15 – both ADs attached), which requires all U.S. operators of DC-9, MD-80/90, and 717 series airplanes to inspect and lubricate the longitudinal trim system acme screw and nut assembly every 650 flight hours and to perform a check of the acme screw and nut wear every 2,000 flight hours.

The following information was reported to the NTSB by the FAA in May 2000. This listing includes all "positive findings" involving endplay that were reported to the FAA by all US airlines and most foreign carriers as of May 4, 2000:

Reporting Airline	Fleet Size	# of Positive Endplay Findings
Alaska Airlines	34	6[38]
Airline A	44	1
Airline B	172	3
Airline C	69	3
Airline D	136	2
Airline E	10	1
Airline F	15	1
South/Central America	90	2
Europe/North Africa	430	3
Totals	1000	22

8.2 Alaska Airlines

The following acme screw and nut assemblies were removed and replaced by Alaska Airlines from November 1999 to the present:[39]

[37] For the purposes of this report, "excessive end-play" refers to end-play readings of 0.040 and higher.
[38] This includes the acme screw and nut from the accident airplane.

Airline Aircraft no.	Flight Hours at Endplay[40]	Date of Action	OnWing Endplay	Bench Endplay (in.)	Wear Rate[41] (inches/ 000 hrs)	Remarks[42]
931	46,834	5-17-00	0.036	N / A	0.00062	S/n 1279
932	45,521	3-15-00	0.036	0.032 (Trig)	0.00055[43]	OAK/Trig chk
933	46,499	4-6-00	0.033	0.031 (Trig)	0.00052	S/n 1365
934	45,170	6-11-00	0.037	0.031 (Trig)	0.00053	S/n DCA1400
935	45,860	2-22-00	0.041	0.040 (Trig)	0.00072	Reno Incident
947	27,860	6-12-99	N/A	N/A	N/A	Up Stop broke
973	20,640	2-13-00	0.023	0.036	0.0014	None
951	52,894	11-26-99	0.046	N/A	0.00074	S/n 0951
953	43,566	8-4-00	0.034	N/A	0.00062	S/n 0950
954	42,192	2-17-00	0.043	N/A	0.00085	None
955	N/A	3-1-00	0.018	0.010 (Trig)	N/A	Fail freeplay
960	27,502	5-13-00	0.039	0.032 (Trig)	0.00091	S/n DCA2163
961	27,322	2-22-00	0.055	No record	0.0018	S/n DCA2184
964	26,409	2-17-00	0.039	0.023 (Trig)	0.0006	Overhaul unit
968	27,551	2-25-00	0.024	0.050 (PHX)	0.0016	AMS bench
972	23,123	8-4-00	0.031	0.028 (Trig)	0.0009	S/n DCA2251
974	22,307	5-17-00	0.044	N/A	0.0017	S/n 0974
975	19,960	11-17-99	0.042	0.030 (Trig)	0.0018	overhaul unit[44]
975[45]	1,387	3-30-00	0.0345	N/A	0.0047	0.028 start

[39] According to Alaska Airlines, the record system only tracked acme screw replacements beginning in January 1999.

[40] These are the total flight hours reported for the airframe, and do not necessarily reflect the total flight hours of the acme screw and nut assembly.

[41] The wear rates in this column are based on the bench check endplay readings, if available. If there is no bench check data available, then the on-wing data is used. Additionally, the wear rates computed here assume a starting endplay of 0.007 unless otherwise noted. The 0.007-inch figure is used because it is in the mean between the 0.003 to 0.010 manufacturer specification for a newly manufactured unit. Additionally, these wear rates assume that the acme screw and nut assembly have accrued the same number of flight hours as the airplane that they were installed in.

[42] The serial numbers (S/n) in this column are for the acme screw and nut assembly, or the acme nut serial number.

[43] This wear rate was computed using the bench endplay reading of 0.032 inches.

[44] This unit was s/n 2272. It was overhauled and placed on the airplane with an endplay of 0.028 before it was replaced later due to the on-wing endplay shown here at 0.042 inches.

981	10,201	2-11-00	0.055	0.053-NTSB	0.0045	S/n 3008
982	9,980	2-12-00	0.082	0.082-NTSB	0.0039	S/n 3000
963 (to '97)	17,700	9-27-97	0.033[46]	accident unit	0.0015	S/n P-2663
963 ('97 on)	8,900	1-31-00	>0.080	accident unit	0.0053	9/97 - accident
963 (total)	26,600	1-31-00	>0.080	accident unit	0.0027	Total Wear

8.3 Endplay Data from Other Airlines

8.3.1 Airline B

The following acme screw and nut assemblies were removed by a U.S. operator (designated as Airline "B" by the Group) due to excessive endplay from February 2000 to the present:

Acme Screw s/n	Flight Hours	Endplay (inches)	Previous Endplay Readings for Same Assembly
00945 (DC-9)	61,138	0.048	0.010 in Sept. 98 ; 0.026 in Sept.94
00214 (DC-9)	65,670	0.053	0.002 in Nov.98 ; Overhaul in Nov 95
09612 (DC-9)	60,126	0.058	0.007 in July 98 ; 0.004 in Mar. 95

8.3.2 Airline D

The following acme screw and nut assemblies were removed by a U.S. operator (designated as Airline "D" by the Group) due to excessive , or near excessive, endplay:

[45] The acme screw and nut refereed to in this row was another overhauled unit, s/n 1171, that was placed into Aircraft 975 after s/n 2272 was removed due to the endplay of 0.034. This unit (1171) was sent to Trig for another overhaul and then placed into Aircraft 933.

[46] This figure that was chosen for the purposes of a wear rate here is the more conservative 0.033-inch reading that was recorded 3 days after the initial record that indicated a 0.040-inch reading.

Post-Accident Removals

Acme Screw s/n	Flight Hours	Endplay[47] (inches)	Wear Rate (in./1,000 hrs)	Endplay Noted by Airline During On-Wing Check
0009 (MD-90)	13,840	0.046[48]	0.0033	0.041 (Submitted to NTSB Lab)
1710 (MD-88)	31,765	0.042	0.0011	0.060
1737 (MD-88)	30,673	0.038	0.0010	0.055
2260 (MD-88)	18,494	0.036	0.0016	0.037

Pre-Accident Removals

Acme Screw s/n	Flight Hours	Wear Rate	Endplay Information
1958 (MD-88)	18,231	N/A	Unknown / scrapped
2165 (MD-88)	18,819	0.0016	0.037
1519 (MD-88)	32,934	0.0009	0.037

8.3.3 Airline F

The following acme screw and nut assembly were removed by a U.S. operator (designated as Airline "F" by the Group) due to excessive endplay from November 1999 to the present. (This assembly is currently undergoing additional investigation.)

Acme Screw s/n	Flight Hours	Wear Rate[49]	Endplay Information / Remarks
DCA110	2,581 hours since last overhaul	0.0159 in./1000 hr	0.043 reading on-wing by airline 0.048 on bench after cleaning

8.3.4 Foreign Carriers

The following acme screw and nut assemblies were removed by foreign operators due to excessive endplay:

Acme Screw s/n	Flight Hours	Wear Rate	Endplay Information
AZ004 (DC-9)	16,390	0.0021 (using 0.041)	0.072(on-wing) ; 0.041(bench test at overhaul)

Fuselage No.	Flight Hours[50]	Wear Rate (in/1000hr)	Endplay
2025 (MD-82)	17,747	0.0023	0.048 inches

[47] All of the endplay results in the column were obtained on a bench after the assemblies were removed from the airplane.
[48] This endplay result was obtained by the Materials Group in a loaded, clean configuration, after the assembly was removed from the airline and sent to the NTSB.
[49] The wear rate was computed using the higher of the two endplay readings (0.048 inches).
[50] The flight hours were obtained from the Boeing Fleet Statistics Group.

8.4 History of Excessive Wear of Acme Screw Wear on the DC-9

On February 24, 1967, Douglas issued Design Memorandum C1-250-DES-DC9-683 documenting "Jackscrew Assembly Excessive Wear Rate." Six acme screw and nut assemblies were reported by operators that indicated "a wear rate considerably in excess of that predicted." The following information was summarized in the memorandum:

No.	Flight Hours	Endplay (inches)	Wear Rate (in/1,000hr)
1	4,086	0.030	0.0056
2	3,300	0.0263	0.0058
3	3,322	0.026	0.0057
4	3,390	0.029	0.0065
5	3,400	0.028	0.0062
6	2,200	0.023	0.00073

The memorandum also stated that the wear rate reflected by these reports, and confirmed by laboratory tests, was 0.004 per 1,000 flight hours. This rate was in contrast with the expected life of the acme screw of 30,000 flight hours, at which time Douglas predicted an endplay of 0.0265 inches. As a result, Douglas increased the 0.0265-inch endplay limit to 0.040 inch after consulting with the Dynamics Group to ensure that the 0.040 limit would not create a problem with aerodynamic flutter or other structural limits. Douglas also changed some of the material specifications and manufacturing processes of the acme screw[51] and suggested that operators be advised of the importance of lubrication.

In 1984, one airline removed two prematurely worn stabilizer trim system acme nuts from two Douglas DC-9 airplanes and submitted them to the Douglas Aircraft Company (DAC) for analysis. Flight hour and wear information from the acme screws were reported as follows:

Acme Screw s/n	Flight Hours	Endplay (in)	Wear Rate (in/1,000 hrs)
P1288	10,558	0.043	0.0034
P1531	5,347	0.040	0.0062

According to DAC report of the investigation, the cause of the premature wear was a lack of adequate lubrication, and DAC issued All Operator's Letter (AOL) No. 9-1526 (attached) on May 29, 1984, recommending that all DC-9 operators lubricate the acme screw at an interval of 600 hours.

In 1990, another airline reported that one of their DC-9 stabilizer trim acme nuts was prematurely worn to about the 0.040 inch limit after accumulating 5,780 flight hours. (This results in a wear rate of 0.0057 inches per 1,000 flight hours). Again,

[51] Details of these changes are documented by the NTSB Materials Group for this investigation.

McDonnell Douglas[52] concluded that the cause of the premature wear was a lack of adequate lubrication[53] and issued AOL No. 9-2120A (attached) on September 5, 1991, again recommending the 600-hour lubrication interval.

AOL 9-2120A also provided the results of a survey conducted by McDonnell Douglas to obtain information regarding the lubrication interval, mean time between removal (MTBR), mean time between unscheduled removal (MTBUR), and average wear rate per 1000 flight hours. The results, which were based on responses from ten DC-9 operators and eleven MD-90 operators, were as follows:

Model	Lube Interval (hours)	MTBR (hours)	MTBUR(hours)	Wear Rate
DC-9	1,329	34,054	34,395	0.0011 in./1,000 hrs
MD-80	804	24,397	28,397	0.0013 in./1,000 hrs

9.0 MD-11 Horizontal Stabilizer Acme Nut Wear History

In 1995, numerous operators of McDonnell Douglas MD-11 airplanes reported premature wear of acme nuts from the horizontal stabilizer trim actuating system.[54] Subsequent investigations by McDonnell Douglas and the operators revealed that the premature wear was attributed to improper surface finish of the acme screw thread surface.[55] The investigation resulted in the release of McDonnell Douglas Service Bulletin (SB) MD11-27-067 on July 31, 1997, which provided instructions for measurement, inspection of excessive wear on the actuator assemblies, and repair and replacement instructions, if necessary.

In 1998, another MD-11 operator reported that the acme nut threads on one acme screw and nut assembly had completely worn away from one of its MD-11 airplanes[56] due to an improper acme screw surface finish.[57] The failure of the shear pins installed in the drive mechanism that connects the two acme nuts rendered the trim

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[52] The Douglas Aircraft Company became McDonnell Douglas Aircraft Company in 1967.

[53] Unlike the first AOL issued in 1984, McDonnell Douglas did not examine the acme screw that was reported by the operator because it was not made available to them. Douglas based their conclusion on analysis of in-service data.

[54] The MD-11horizontal stabilizer trim actuation system is very similar to the DC-8 in that it incorporates two separate acme screw and nut assemblies. The design and operating principles of the DC-8 and MD-11 acme screw and nut assemblies is similar to the design for the DC-9, MD-80/90, and Boeing B-717 series airplanes, except that the assembly in the DC-8 and MD-11 is larger to accommodate the larger tail. Additionally, the acme screws installed in DC-8 and MD-11 airplanes do not have a torque tube inside of them.

[55] No obvious evidence of improper surface finish was found on the Alaska Airlines accident acme screw. Details of surface finish were documented by the Materials Group.

[56] This airplane fell under the effectivity of SB MD11-27-067. It is unknown at this time if the airplane underwent the actions of the SB.

[57] The MD-11 is equipped with two acme screw and nut assemblies that drive the horizontal stabilizer. A hydraulic motor drives a chain drive mechanism, which in turn rotate the two acme nuts. Shear pins are installed in the chain drive sprocket that connect the two assemblies and can fail due to the asymmetric loads applied as the acme screw from one side translates through its nut at a different rate than the other.

system inoperative. As a result of this incident, McDonnell Douglas escalated SB MD11-27-067 to an "Alert" status[58] and addressed the incident in the SB's summary section. The summary section of the SB also stated: "If not corrected, this condition, in conjunction with a failure of the opposite side jackscrew assembly, could result in a free horizontal stabilizer that would cause loss of airplane pitch control." On July 30, 1998, the FAA issued an Airworthiness Directive (AD) that reiterated this concern and required MD-11 operators to comply with the SB. Copies of the Alert SB and AD are attached. Additional investigation of this issue is continuing.

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11/22/00

[58] Alert Service Bulletin No. MD-11-27A067, revision 5, was issued on November 19, 1999.