05/25/79 American Airlines

Official Accident Report Index Page

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Report Title	American Airlines, Inc. DC-10-10, N110AA, Chicago-O'Hare International Airport Chicago, Illinois, May 25, 1979
Report Date	December 21, 1979
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Keywords	DC-10; loss of control; asymmetrical stall; stall warning system; slat disagreement warning light; structural separation; No. 1 engine and pylon; design criteria; maintenance procedures; FAA regulations; industry communications procedures.
Abstract	About 1504 c.d.t., May 25, 1979, American Airlines, Inc., Flight 191, a McDonnell-Douglas DC-10-10 aircraft, crashed into an open field just short of a trailer park about 4,600 ft northwest of the departure end of runway 32R at Chicago-O'Hare International Airport, Illinois. Flight 191 was taking off from runway 32R. The weather was clear and the visibility was 15 miles. During the takeoff rotation, the left engine and pylon assembly and about 3 ft of the leading edge of the left wing separated from the aircraft and fell to the runway. Flight 191 continued to climb to about 325 ft above the ground and then began to roll to the left. The aircraft continued to roll to the left until the wings were past the vertical position, and during the roll, the aircraft's nose pitched down below the horizon. Flight 191 crashed into the open field and the wreckage scattered into an adjacent trailer park. The aircraft was destroyed in the crash and subsequent fire. Two hundred and seventy-one persons on board Flight 191 were killed; two persons on the ground were killed, and two others were injured. An old aircraft hangar, several automobiles, and a mobile home were destroyed. The National Transportation Safety Board determines that

the probable cause of this accident was the asymmetrical stall and the ensuing roll of the aircraft because of the uncommanded retraction of the left wing outboard leading edge slats and the loss of stall warning and slat disagreement indication systems resulting from maintenance-induced damage leading to the separation of the No. 1 engine and pylon assembly at a critical point during takeoff. The separation resulted from damage by improper maintenance procedures which led to failure of the pylon structure.

Contributing to the cause of the accident were the vulnerability of the design of the pylon attach points to maintenance damage; the vulnerability of the design of the leading edge slat system to the damage which produced asymmetry; deficiencies in Federal Aviation Administration surveillance and reporting systems which failed to detect and prevent the use of improper maintenance procedures; deficiencies in the practices and communications among the operators, the manufacturer, and the FAA which failed to determine and disseminate the particulars regarding previous maintenance damage incidents; and the intolerance of prescribed operational procedures to this unique emergency.

Facts of the Accident

Accident NTSB ID	79-17
Airline	American Airlines
Model aircraft	DC-10-10, N110AA, serial No. 46510
Year shipped	1972
Aircraft manufacturer	McDonnell-Douglas
Engine type	CF6-6D
Engine manuafacturer	General Electric
Date	05/25/79
Time	1504
Location	Chicago-O'Hare International Airport, Chicago, Ill.
Country	USA
IFR or VFR?	VFR
Fatalities	271 plus 2 on ground
Injuries	2 on ground
Fire during flight?	N
Fire on the ground?	Y
Probable cause	The asymmetrical stall and the ensuing roll of the aircraft because of the uncommanded retraction of the left wing outboard leading edge slats and the loss of stall warning and slat disagreement indication systems resulting from maintenance-induced damage leading to the separation of the No. 1 engine and pylon assembly at a critical point during takeoff. The separation resulted from damage by improper maintenance procedures which led to failure of the pylon structure.
Contributing causes	The vulnerability of the design of the pylon attach points to maintenance damage; the vulnerability of the design of the leading edge slat system to the damage which produced asymmetry; deficiencies in Federal Aviation Administration surveillance and reporting systems which failed to detect and prevent the use of improper maintenance procedures; deficiencies in the practices and communications among the operators, the manufacturer, and the FAA which failed to determine and disseminate the particulars regarding previous maintenance damage incidents; and the intolerance of prescribed operational procedures to this unique emergency.
Weather conditions	Clear, visibility 15 miles
Total crew size	13
Cockpit crew size	3
Cabin crew size	10

Passengers	258
Report ID	NTSB-AAR-79-17
Pages	103
Day or night?	Day
Flight number	191
Flight origin	Chicago O'Hare, IL
Flight destination	Los Angeles, CA
Description	During takeoff rotation the left engine and pylon assembly and a portion of a wing separated from the aircraft and fell to the runway. The aircraft continued to climb then began to roll to the left, crashed into an open field and burned. All perished.

Synopsis

About 1504 c.d.t., May 25, 1979, American Airlines, Inc., Flight 191, a McDonnell-Douglas DC-10-10 aircraft, crashed into an open field just short of a trailer park about 4,600 ft northwest of the departure end of runway 32R at Chicago-O'Hare International Airport, Illinois.

Flight 191 was taking off from runway 32R. The weather was clear and the visibility was 15 miles. During the takeoff rotation, the left engine and pylon assembly and about 3 ft of the leading edge of the left wing separated from the aircraft and fell to the runway. Flight 191 continued to climb to about 325 ft above the ground and then began to roll to the left. The aircraft continued to roll to the left until the wings were past the vertical position, and during the roll, the aircraft's nose pitched down below the horizon.

Flight 191 crashed into the open field and the wreckage scattered into an adjacent trailer park. The aircraft was destroyed in the crash and subsequent fire. Two hundred and seventy-one persons on board Flight 191 were killed; two persons on the ground were killed, and two others were injured. An old aircraft hangar, several automobiles, and a mobile home were destroyed.

The National Transportation Safety Board determines that the probable cause of this accident was the asymmetrical stall and the ensuing roll of the aircraft because of the uncommanded retraction of the left wing outboard leading edge slats and the loss of stall warning and slat disagreement indication systems resulting from maintenance-induced damage leading to the separation of the No. 1 engine and pylon assembly at a critical point during takeoff. The separation resulted from damage by improper maintenance procedures which led to failure of the pylon structure.

Contributing to the cause of the accident were the vulnerability of the design of the pylon attach points to maintenance damage; the vulnerability of the design of the leading edge slat system to the damage which produced asymmetry; deficiencies in Federal Aviation Administration surveillance and reporting systems which failed to detect and prevent the use of improper maintenance procedures; deficiencies in the practices and communications among the operators, the manufacturer, and the FAA which failed to determine and disseminate the particulars regarding previous maintenance damage incidents; and the intolerance of prescribed operational procedures to this unique emergency.

1. Factual Information

1.1 History of the Flight

At 1459 c.d.t.,[1] May 25, 1979, American Airlines, Inc., Flight 191, a McDonnell-Douglas DC-10 series 10 aircraft (DC-10-10) (N110AA), taxied from the gate at Chicago-O'Hare International Airport, Illinois. Flight 191, a regularly scheduled passenger flight, was en route to Los Angeles, California, with 258 passengers and 13 crewmembers on board. Maintenance personnel who monitored the flight's engine start, push-back, and start of taxi did not observe anything out of the ordinary.

The weather at the time of departure was clear, and the reported surface wind was 020° at 22 kns. Flight 191 was cleared to taxi to runway 32 right (32R) for takeoff. The company's Takeoff Data Card showed that the stabilizer trim setting was 5° aircraft noseup, the takeoff flap setting was 10°, and the takeoff gross weight was 379,000 lbs. The target low-pressure compressor (N_1) rpm setting was 99.4 percent, critical engine failure speed (V_1) was 139 kns indicated airspeed (KIAS), rotation speed (V_R) was 145 KIAS, and takeoff safety speed (V_2) was 153 KIAS.

Flight 191 was cleared to taxi into position on runway 32R and hold. At 1502:38, the flight was cleared for takeoff, and at 1502:46 the captain acknowledged, "American one ninety-one under way." Company personnel familiar with the flightcrew's voices identified the captain as the person making this call and the ensuing V_1 and V_R speed callouts on the cockpit voice recorder (CVR).

The takeoff roll was normal until just before rotation at which time sections of the left, or No. 1, engine pylon structure came off the aircraft. Witnesses saw white smoke or vapor coming from the vicinity of the No. 1 engine pylon. During rotation the entire No. 1 engine and pylon separated from the aircraft, went over the top of the wing, and fell to the runway.

Flight 191 lifted off about 6,000 ft down runway 32R, climbed out in a wings-level attitude, and reached an altitude of about 300 ft above the ground (a.g.l.) with its wings still level. Shortly thereafter, the aircraft began to turn and roll to the left, the nose pitched down, and the aircraft began to descend. As it descended, it continued to roll left until the wings were past the vertical position.

Flight 191 crashed in an open field and trailer park about 4,600 ft northwest of the departure end of runway 32R. The aircraft was demolished during the impact, explosion, and ground fire. Two hundred and seventy-one persons on board Flight 191 were killed, two persons on the ground were killed, and two persons on the ground sustained second- and third-degree burns.

The aircraft crashed about 1504, during daylight hours; the coordinates of the crash site were 42°00'35"N, 87°55'45"W.

1.2 Injuries To Persons

Injuries	Crew	Passengers	Others
Fatal	13	258	2
Serious	0	0	2
Minor/None	0	0	0

1.3 Damage To Aircraft

The aircraft was destroyed.

1.4 Other Damage

An old aircraft hangar, several automobiles, and a mobile home were destroyed.

1.5 Personnel Information

All flight and cabin personnel were qualified. (See appendix B.)

1.6 Aircraft Information

Flight 191, a McDonnell-Douglas DC-10-10, N110AA, was owned and operated by American Airlines, Inc., and was powered by three General Electric CF6-6D engines. (See appendix C.) According to the manufacturer, the left engine weighed 11,612 lbs, the pylon, 1,865 lbs, for a total engine-pylon assembly weight of 13,477 lbs. With the loss of the engine pylon structure, the aircraft's center of gravity (c.g.) moved aft 2 percent to about 22 percent mean aerodynamic chord (MAC). The resultant c.g. was within the forward (16.4 percent MAC) and aft (30.8 percent MAC) c.g. limits. The lateral c.g. shift was 11.9 inches to the right.

1.7 Meteorological Information

At the time of the accident, the weather at the airport was clear. The surface observations at O'Hare International were as follows:

1451, surface aviation: Clear, visibility--15 mi, weather--none, temperature--63°F, dewpoint--29°F, winds--020° at 22 kns, altimeter--30.00 inHg.

1511, local: Clear, visibility--15 mi, weather--none, temperature--63° F, dewpoint--29° F, winds--020° at 19 kns gusting to 28 kns, altimeter--30.00 inHg., remarks--aircraft mishap.

1.8 Aids To Navigation

No applicable.

1.9 Communications

There were no known communications malfunctions.

1.10 Aerodrome Information

Chicago-O'Hare International Airport is located 16 mi northwest of downtown Chicago, Illinois, and is served by seven runways. Runway 32R is 10,003 ft long and 150 ft wide, and has a concrete surface. The runway elevation is 649 ft mean sea level (m.s.l.) at its southeast end and 652 ft m.s.l. at its northwest end.

1.11 Flight Recorders

The aircraft was equipped with a Fairchild Model A-100 CVR, serial No. 2935. The CVR was recovered and brought to the Safety Board's laboratory where a transcript of the recording was prepared. The recording was incomplete because of the loss of electrical power to the recorder during aircraft rotation. However, the aircraft's gross weight, stabilizer trim setting, V_1, and V_R callouts were recorded.

The aircraft was equipped with a Sundstrand digital flight data recorder (DFDR), serial no. 2298. The recorder had been damaged structurally, but there was no fire or heat damage. The recording tape was broken; upon removal from the recorder the tape was spliced together and a readout was made. Two 6-sec areas of data were damaged because of the breaks in the tapes; however, most of these data was recovered.

The DFDR recorded 50 sec of data during the takeoff roll and 31 sec of airborne data before the recording ended. (See appendix H.) The DFDR readout showed that the stabilizer trim setting for takeoff was 6.5° aircraft noseup. The DFDR's tolerance for this parameter is ± 1°. Because of unusual aircraft attitudes during the last few seconds of the flight, the recorded altitude and airspeed data were not correct. Therefore, the DFDR altitude and indicated airspeed values cited hereafter have been corrected for the position errors resulting from the aircraft's attitudes during the last few seconds of the descending flight.

Correlation of the DFDR and CVR recordings disclosed that the flightcrew had set the flaps and stabilizer trim at 10° and about 5° aircraft noseup, respectively, for takeoff. A rolling takeoff was made, takeoff thrust was stabilized at 80 KIAS, and left rudder and right aileron were used to compensate for the right crosswind. The V_1 and V_R callouts were made about 2 sec after these speeds were recorded by the DFDR. The elevator began to deflect up at V_R. The aircraft began to rotate upward immediately and continued upward at a rate of 1.5° per sec. Flight 191 accelerated through V_2 speed during rotation and before it lifted off the runway. The last stable takeoff thrust on the No. 1 engine was recorded 2 sec before liftoff. One second later, the word "damn" was recorded on the CVR, and then the CVR ceased operating.

One second before liftoff and simultaneous with the loss of the CVR and the No. 1 engine's parameters, the DFDR ceased recording the positions of the left inboard aileron, left inboard elevator, lower rudder, and Nos. 2 and 4 left wing leading edge slats. The DFDR continued to record all other parameters including the position of the upper rudder, the outboard aileron, the outboard elevator, and the No. 4 leading edge slats on the right side of the aircraft. The electrical power for the CVR and the sensors for the lost DFDR functions were all derived from the aircraft's No. 1 a.c. generator bus.

Flight 191 became airborne about 6,000 ft from the start of the takeoff roll and remained airborne for 31 sec. It lifted off at V_2 + 6 KIAS and at 10° pitch attitude. Two seconds after liftoff, the DFDR reading for the No. 1 engine's N_1 was zero, the No. 2 engine's N_1 speed was increasing through 101 percent, and the No. 3 engine's N_1 was essentially at the takeoff setting.

The flight lifted off in a slight left wing-down attitude. Application of right wing-down aileron and right rudder restored the flight to a wings-level attitude and the heading was stabilized between 325° and 327°. The flight maintained a steady climb about 1,150 feet per minute (fpm) at a 14° noseup pitch attitude--the target pitch attitude displayed by the flight director for a two-engine climb. During the climb, the No. 2 engine N_1 speed increased gradually from 101 percent to a final value of 107 percent; the No. 3 engine N_1 speed did not change appreciably from the takeoff setting. During the initial part of the climb, the aircraft accelerated to a maximum speed of 172 KIAS; it reached this value about 9 sec after liftoff and about 140 ft a.g.l.

Flight 191 continued to climb about 1,100 fpm. The pitch attitude and heading were relatively stable. Right wing-down aileron and right rudder were used to control and maintain the heading and the roll attitude during the climb in the gusty right crosswind.

During the climb, the aircraft began to decelerate from 172 KIAS at an average rate of about 1 kn per second. At 20 sec after liftoff, at 325 ft a.g.l. and 159 KIAS, the flight began to roll to the left and passed through 5° left wing down. The left roll was accompanied by increasing right-wing-down aileron deflection. At this point, the previously stabilized right rudder deflected suddenly to zero, remained at zero for 1 sec, and then moved toward its previous deflection. The flight began to turn to the left, and the left roll increased even though increasing right rudder and right-wing-down aileron deflections were being applied. At 325 ft a.g.l. the flight had turned through the runway heading and was rolling to the left at 4° per second. The right rudder deflection increased during the turn. The previously stable pitch attitude began to decrease from 14° even though the elevator was being increased to the full aircraft noseup deflection. The maximum pitch rate of about 12° per second was reached just before the crash.

Flight 191 continued to roll and turn to the left despite increasing right rudder and right-wing-down aileron deflections. Three seconds before the end of the DFDR tape, the aircraft was in a 90° left bank and at a 0° pitch attitude. The DFDR recording ended with the aircraft in a 112° left roll and a 21° nosedown pitch attitude with full counter aileron and rudder controls and nearly full up elevator being applied.

DFDR longitudinal and vertical acceleration data were integrated to determine the headwind components at points where the aircraft attained certain speeds and where it lifted off; to establish an altitude profile; and to determine the location where the DFDR stopped. These data showed that the DFDR ceased operation 14,370 ft from the southeast end of runway 32R and 820 ft left of the runway's extended centerline. Examination of the crash site showed that the first point of impact was 14,450 ft beyond the southeast end of runway 32R and 1,100 ft left of its extended centerline. Based on these data and the corrected altitudes, the DFDR ceased operating at impact. The flight reached a maximum altitude of 350 ft a.g.l.

1.12 Wreckage and Impact Information

Flight 191 struck the ground in a left wing-down and nosedown attitude. The left wingtip hit first, and the aircraft exploded, broke apart, and was scattered into an open field and a trailer park. The disintegration of the aircraft structure was so extensive that little useful data were obtained from postimpact examination of the wreckage with the exception of the No. 1 pylon, which was found off the right side of runway 32R.(See figure 1.)

Investigators located and documented identifiable aircraft components. Except for the No. 1 engine and pylon, portions of the engine cowling, and a part of the leading edge of the wing directly above the pylon, the aircraft wreckage came to rest in the open field and trailer park. (See appendix D.)

The first marks made by engine contact of the No. 1 engine and pylon with the runway began about 19 ft to the right of the centerline lights and about 6,953 ft beyond the southeast end of runway 32R. Other parts of engine and pylon structure were located in this area; however, no spoiler actuators or hydraulic lines were found.

The pylon is attached to the wing using spherical ball joints in three different structural elements. Two of the spherical joints are aligned vertically in a forward bulkhead which is attached to structure in the wing forward of the front spar. Another spherical joint behind the forward bulkhead transmits thrust loads from pylon structure into a thrust link which in turn is connected through another spherical joint to structure on the lower surface of the wing. The third attachment point is a spherical joint in the pylon aft bulkhead which attaches to a clevis mounted on the underside of the wing. The pylon forward bulkhead and portions of the flange from the pylon aft bulkhead either remained with the separated No. 1 pylon or were scattered along the runway. (See figure 2 and figure 3.) The No. 1 pylon's aft clevis attach assembly and portions of the pylon aft bulkhead, wing thrust angle assembly and thrust link, and pylon forward bulkhead attach assembly remained with the wing.

The pylon forward bulkhead was bent forward about 30° and most of the bolts which held the bulkhead upper plates were missing. The upper 12 inches of the forward plate were bent forward an additional 10° to 15°. The aft plate was broken below the thrust fitting connection, and a large piece of the upper left corner was missing.

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Figure 1. Accident site.

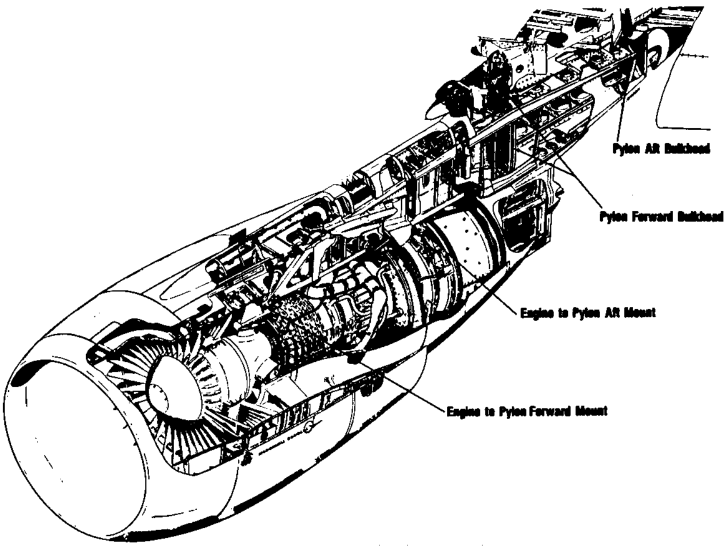

Figure 2. Engine and pylon assembly.

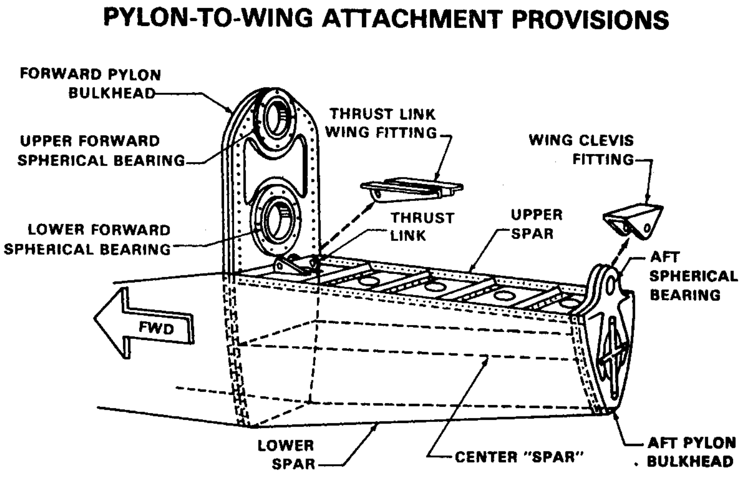

Figure 3. Pylon assembly.

The wing's forward support fitting, which attached the pylon forward bulkhead to the wing at the upper and lower plugs and spherical bearings, was found at the main wreckage site. The upper and lower plugs and their attaching hardware were intact, and the upper and lower spherical bearings were attached to the fitting.

The pylon thrust fitting remained attached to the forward portion of the pylon's aft upper spar web. The pylon thrust link, which attached the pylon thrust fitting to the wing thrust angles, was found at the main wreckage site attached to a portion of the wing thrust angles. Its forward spherical bearing was cocked to the extreme left, and a segment of the bearing which had broken away was found on the runway.

The thrust bushing bolt had broken in two parts, both of which were found in the grass adjacent to the runway. The bolt nut was attached to one of the broken pieces, and the faces of the nut were gouged severely. Except for one lubrication retainer washer, which was not found, the remaining portions of the thrust bushing bolt assembly were found along the runway. One shim spacer from the assembly was crushed severely while the other was relatively undamaged.

The upper two-thirds of the pylon aft bulkhead separated from the flanges around its periphery and was found in the wreckage. The top two pieces of its attach lugs had separated from the bulkhead, and the aft side of the bulkhead was gouged heavily near the lower edge of the wing clevis lug, which attached the aft bulkhead to the wing. The wing clevis was attached to the wing. The aft bulkhead's spherical bearing was attached to the clevis, and the separated pieces of the aft bulkhead's attach lugs were found on top of the spherical bearing.

The Nos. 2 and 3 engines were located in the main wreckage. The damage to the engines indicated that they were operating at high rpm at impact. All three engines were taken to the American Airline's Maintenance Facility at Tulsa, Oklahoma, where they were torn down and examined. There was no evidence of any preimpact malfunctions.

The examinations of the main and nose landing gears and actuators indicated that the gear was down and locked at impact. The left and right stabilizer jackscrews were recovered and the distance between the upper surfaces of the jackscrews' drive nuts and the lower surfaces of the actuators' upper stops was measured. These measurements indicated that the stabilizer was positioned at 5.71° aircraft noseup.

Examination of the hydraulic system components did not reveal any evidence of internal operating distress. The control valve of the 2-1 nonreversible motor pump was in the open position, indicating that the No. 2 hydraulic system was driving the No. 1 hydraulic system's pump.

All eight flap actuators were recovered, and investigators attempted to verify the position of the trailing edge flaps by measuring the extension of the flap actuator pistons. The piston extensions were compared to those of another aircraft with flaps extended to 10°. Based on this comparison, some degree of flap extension was probable, but the actual position could not be established. However, the DFDR data showed that the flaps were set at 10°.

A 3-ft section of the left wing's leading edge, just forward of the point where the forward part of the pylon joined the wing, was torn away when the engine pylon assembly separated from the aircraft. The No. 1 and No. 3 hydraulic system's extension and retraction lines and the followup cables for the left wing's outboard slat drive actuators were severed. Thirty-five of the 36 leading edge slat tracks were examined. The examination disclosed that at impact the left wing's outboard slats were retracted, while the left wing's inboard slats and the right wing's inboard and outboard slats were extended to the takeoff position.

The examination of the cockpit instruments did not disclose any usable information.

1.13 Medical and Pathological Information

A review of the autopsies and toxicological examinations of the flightcrew disclosed no evidence of preexisting physiological problems which could have affected their performance.

1.14 Fire

The aircraft was subjected to severe ground fire.

1.15 Survival Aspects

This accident was not survivable because impact forces exceeded human tolerances.

1.16 Tests and Research

1.16.1 Study of Photographs

Five photographs taken of Flight 191's departure by two cameras--one in the terminal and one onboard a DC-10 on final approach to runway 9R--were sent to Lockheed's Palo Alto Research Laboratories for a Photo-Image Enhancement Study to determine the position of the flight controls. The process produced black and white images containing expanded variations of gray shading which, in the absence of the enhancement process, would be too subtle for the eye to distinguish. Based on the study of these photo-images, the following observations were made: (1) The tail assembly was not damaged; (2) the nose gear was down during the initial climbout and before the onset of roll; (3) spoilers Nos. 1, 3, and 5 were extended on the right wing; and (4) the trailing edge of the right wing inboard aileron was up. Although the position of the slats was difficult to determine, the left wing inboard slats appeared to be extended, and the position of all other control surfaces appeared to be the same as recorded by the DFDR. The pitch and roll attitudes of the aircraft were extrapolated from the photographs, and extrapolations agreed closely with those recorded by the DFDR.

1.16.2 Metallurgical Examinations and Postaccident Inspections of the DC-10 Fleet

N110AA's pylon aft bulkhead was examined at the Safety Board's metallurgical laboratory. The examination disclosed a fracture of the upper forward flange. (See figure 4.) The larger part of this fracture was just forward of the radius between the flange and forward bulkhead plane and was about 10 inches long in the inboard-outboard direction. (See figure 5.) The fracture characteristics were typical of an overload separation. Chevron and tear marks on the fracture indicated that the rupture progressed downward at the center of the flange, then in inboard and outboard directions on the flange. The bottom portion of the fracture exhibited smearing consistent with the compression portion of a bending fracture. The smear was more prevalent--about 6 inches long--in the thinner center portion of the upper flange structure, but became less prevalent at the outer ends of the fracture. The 10-inch-long fracture resulted from overstress. The overstress was initiated by the application of a downward bending moment at the center section of the flange just forward of the fracture plane. The surface of the fracture appeared to be relatively free of oxidation and dirt.

Fatigue cracking was evident at both ends of the fracture. At the inboard end, the fatigue progressed inboard and aft; then, it progressed downward and inboard to the upper inboard fastener that attached the forward section of the bulkhead to the aft section. The fatigue progressed past the fastener a short distance before exhibiting rapid overstress characteristics in the downward direction as it proceeded along the inboard side of the side flange radius of the forward flange section. At the outboard end of the fracture, the fatigue propagated forward and slightly outboard toward the most forward outboard hole in the upper flange. The total length of the overstress fracture and fatigue cracks was about 13 inches. The remainder of the fractures on the bulkhead and within the pylon structure resulted from overload.

The examination also disclosed that three shims were installed on the upper surface of the forward upper flange. Two shims (Part No. AUB-7034-25) were installed, one on the inboard top shoulder of the upper flange and one on the outboard top shoulder. These shims are about 2 inches long, 1 inch wide, and .063 inch thick. A 10-inch-long, .050-inch-thick shim was installed during production to fill a gap between the upper flange and upper spar web. (See figure 6.) The manufacturer's drawings specify that the AUB-7034-25 shim may be required along the side of the bulkhead; however, they do not indicate that shims may be required on the upper surface of the flange. The fatigue propagation on the inboard and outboard ends of the overstress fracture began in the area underneath the .063-inch-thick shims.

The aft fracture surface of the upper flange contained a crescentshaped deformation which matched the shape of the lower end of the wing clevis. This deformation was in line with the vertical centerline of the aft bulkhead attachment hole as indicated by arrow "d" in figure 7. A deformation was noted in the lower surface of the aft wing support fitting's forward clevis lug in the area indicated by the brackets in figure 9. A small shallow gouge was apparent in the area of the arrow in figure 10. This gouge was in a position which would conform to a fastener location on the top flange assembly of the aft bulkhead. The gouge appeared to be produced by a fastener head, hitting the clevis with a sliding movement. The upper flange aft fracture surface and radius appeared to have been deformed by the wing clevis' striking these surfaces in the downward direction.

Figure 4. Overall view of the wing pylon aft bulkhead installation portions.

Pieces are numbered for identification purposes and placed in relative locations as if intact.

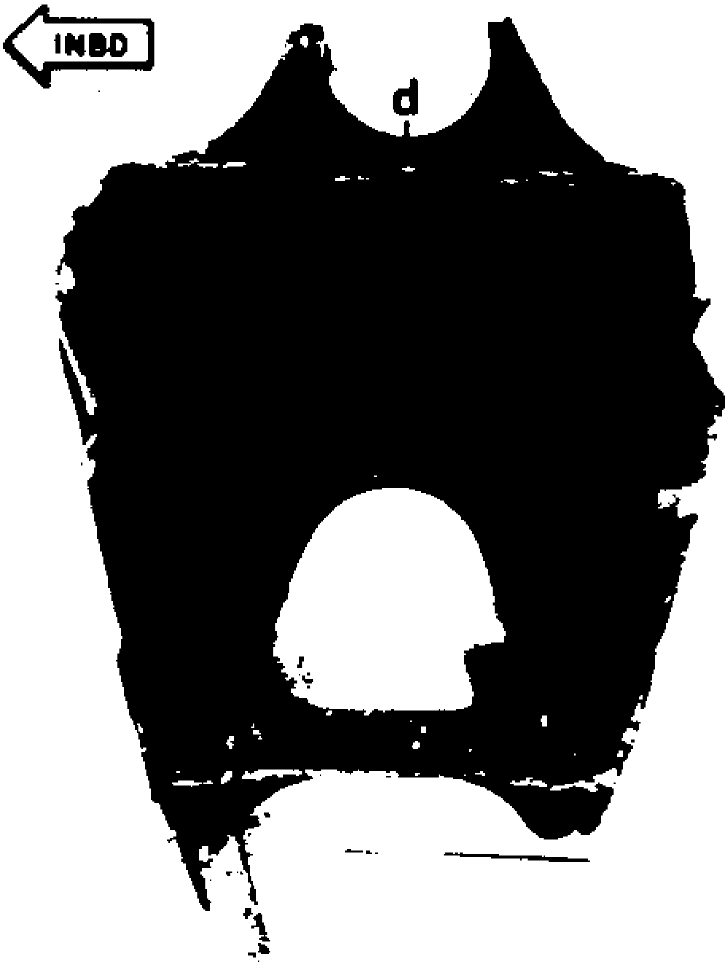

Figure 5. View of the aft bulkhead piece indicated as Item 1 in Figure 4.

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Figure 6. View looking forward and slightly down on the No. 1 pylon before disassembly of the flange pieces indicated as items 4 through 9 in Figure 4.

Arrow "x" shows the shim on the outboard side between the angle and the top surface of the upper flange piece No. 7 held in place by the fastener arrowed "y". Arrow "s" shows the location for the shim on the inboard side between the angle and upper surface of the upper bulkhead flange.

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Figure 7. Closer view of fracture on upper flange in the area of deformation (arrow "d", see Figure 5).

This image is not available at this time.

Figure 8. Detail of deformation denoted by arrows "d" in Figure 5 and 7.

This image is not available at this time.

Figure 9. Overall view looking up on the wing mounted aft support fitting with spherical bearing attached.

This image is not available at this time.

Figure 10. Close up view of damage in the area between the brackets of Figure 9.

The clearances between the upper flange surface and the bottom surface of the wing clevis were examined using the aft wing support fitting from N110AA and the aft bulkheads of another DC-10-10 (N119AA). (See figure 11 and figure 12.) With the aft spherical bearing and bushing in place, the vertical distance from the bottom of the clevis to the surface of the flange is about 0.5 inch. (See figure 11.) When the bushing was removed and the aft bulkhead moved up against the far inside portion of the wing fitting, the flange was displaced about 0.6 inch above its previous position. (See figure 12.) In this position, the lower portion of the clevis was about 0.1 inch below the fracture on N119AA's bulkhead. The 0.1 inch between the upper flange fracture's upper surface and the lower portion of the wing fitting clevis was the same as the vertical depth of the deformation found on N110AA's aft bulkhead.

Taking into account the stackup on the forward flange created by the spar web, doubler, and fasteners, the clearance between the bottom of the clevis and the top of the web fasteners could be about .005 to .045 inch. The addition of a shim would narrow the clearance, and taking into account all tolerances in the spherical bearing assembly, there could be an interference. A postaccident survey of the DC-10 fleet revealed seven pylons with such interference. McDonnell-Douglas had not established a standard minimum clearance between the bottom of the clevis and the top of the fastener.

Despite numerous searches of the runway and adjacent areas after the accident, investigators were not able to find one of the forward thrust bushing attachment's retainer washers. However, measurements between the mating portions on the fracture and the undersides of the thrust bolt head and nut as well as the physical evidence produced by the separation of the parts indicated that the missing washer was in place when the pylon separated and that the thrust bushing assembly had been installed properly.

After the accident, the Federal Aviation Administration (FAA) required a fleetwide inspection of the DC-10. During these inspections, discrepancies were found in the pylon assemblies. Among these discrepancies were variances in the clearances on the spherical bearing's fore and aft faces; variances in the clearance between the bottom of the aft wing clevis and the fasteners on the upper spar web; interferences between the bottom of the aft clevis and the upper spar web fasteners; pylons with either loose, failed, or missing spar web fasteners; and aft pylon bulkheads with upper flange fractures. The fractured flanges were found only on the DC-10-10 series aircraft.

During postaccident inspections, six DC-10's were found to have fractured upper flanges on the pylon aft bulkheads: Four American Airlines DC-10's--N106AA, N107AA, N118AA, N119AA-- and two Continental Airlines DC-10's--N68050, N68047.

The failure modes on the Continental Airlines' aircraft that were examined by metallurgists were similar to those found on the American Airlines' DC-10's. Of the two Continental fractures discovered during the postaccident inspections, one crack was 6 inches long, and the other 3 inches long; neither crack showed any evidence of fatigue propagation.

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Figure 11. Wing pylon aft bulkhead from N119AA assembled to wing mounted aft support fitting of N110AA showing normal position of wing fitting clevis with respect to the upper forward flange of the aft bulkhead.

Note: The wing fitting on this figure is canted relative to the bulkhead to simulate the dihedral of the left wing relative to the bulkhead.

This image is not available at this time.

Figure 12. Same as Figure 11 except the attachment bushing was removed and the aft bulkhead was moved up against the far inside portion of the wing fitting.

Note the location of the bottom portion of the wing fitting clevis with respect to the fracture on the bulkhead.

The investigation also disclosed that two other Continental Airlines DC-10's--N68041, N68049--had had fractures on their upper flanges. These two aircraft were damaged on December 19, 1978, and February 22, 1979, respectively. The damage was repaired and both aircraft were returned to service. In addition, a United Airlines' DC-10, N1827U, was discovered to have a cracked upper spar web on its No. 3 pylon and 26 damaged fasteners.

The damaged pylon aft bulkheads of the four other American Airlines' DC-10's were also examined at the Safety Board's metallurgical laboratory. Each of these aft bulkheads contained visible cracks and obvious downward deformations along their upper flanges. The shortest crack appeared to be on the N107AA bulkhead and the longest crack--about 6 inches--was on the N119AA bulkhead. The crack on the N119AA bulkhead was the only one in which fatigue had propagated; the fatigue area was about .03 inch long at each end of the overstress fracture.

Of the nine DC-10's with fractured flanges, only the accident aircraft had shims installed on the upper surface of the flange.

1.16.3 Stress Testing of the Pylon Aft Bulkhead

As a result of the discovery of the damaged upper flange on the accident aircraft, laboratory tests were conducted in an attempt to reproduce the 10-inch overload crack. The testing involved both static and dynamic loading with and without the .050-inch-thick shim installed on the flange. Static load tests conducted by American Airlines involved the use of a Tinnius Olsen universal test machine 2 and a shimmed (.050-inch) spar web. The results showed that the flange cracked under a 6,400-lb load and when deflected .122 inch. The initial crack was 1.1 inch long. The crack progressed through the flange when the flange was deflected 0.2 inch after loading of 7,850 lbs; its length was 2.8 inches. Once the flange was penetrated, it required lighter loads to produce greater deflections. A crack 7.4 inches long was produced with a 5,175-lb load at a 0.6-inch deflection. At this point, the ends of the crack had disappeared under the spar web.

Additional static and dynamic load tests were conducted at McDonnell-Douglas. A test specimen, consisting of the aft bulkhead, connecting spar web, and a .050-inch shim installed between the bulkhead flange and the spar web, was used in one static test. A jackscrew was used to apply load to the specimen. Cracking began when the flange was deflected 0.1 inch with an applied load of about 6,900 lbs. At 11,500 lbs, the flange was deflected 0.2 inch, and the crack propagated to about 2 inches. Increasing the deflection to 0.6 inch lengthened the crack to 7.8 inches; however, the required load was only 8,600 lbs.

The evidence indicated that the maximum interference that would result from the insertion of the .050-inch-thick shim was .024 inch. The static tests conducted by American Airlines and McDonnell-Douglas showed that a crack would begin at a deflection of about 0.1 inch; thus, in the worst case, an additional deflection would be required to crack the flange.

During dynamic testing, seven specimens were subjected to impact loads of varying energy levels and numbers of strikes. Specimens struck at high energy levels (6,000 inch-pounds) failed in unrelated modes. Specimens struck at low energy levels (1,500 to 2,500 inch-pounds) required seven to eight strikes to create an 8- to 10-inch-long crack. The total absorbed energy required to produce a 10-inch crack in an unshimmed specimen was 16,000 inch-pounds. The absorbed energy required to create a 10-inch crack in a shimmed specimen (.050-inch thick) was about 18,000 inch-pounds. In one test, a 10-inch crack was produced on an unshimmed specimen after two blows; the total absorbed energy was 5,200 inchpounds.

In another test, conducted by American Airlines, an aft bulkhead, in which a 6-inch crack had been produced in the flange by forcing a simulated wing clevis vertically down on pylon web bolts, was subsequently subjected to a thrust load. With a thrust load of 11,625 pounds, the 6-inch crack extended to 10 inches, at which point the thrust load was relieved.

The major elements of the pylon structure were also examined to determine primary and fail-safe 3 load paths. Normally the vertical and side forces, as well as torque or rolling moments in the plane of the bulkhead, are transmitted from the pylon structure through the spherical joints in the forward and aft bulkheads and into the wing structure. All of the thrust load from the pylon is intended to be transmitted through the thrust link.

The capability of the forward and aft bulkheads to serve as alternate fail-safe load paths in the event of a thrust link failure was assessed during the postaccident investigations. Therefore, in addition to the tests of the upper flange, a full-scale wing pylon test was conducted to evaluate load distributions and flexibility of the pylon-mounted bulkheads both with and without a thrust link installed.

The design gap between the forward and aft faces of the aft spherical bearing and the respective faces of the clevis is .080 inch. With this clearance, minimal thrust loads of about 600 lbs are experienced at the aft bulkhead. However, during the postaccident investigation, this gap was measured throughout the DC-10 fleet, and the smallest gap found was .047 inch. With that size gap and the engine at maximum thrust, thrust loads of about 6,650 lbs are experienced at the aft bulkhead; this load is still within the bulkhead's strength capability.

The failed thrust link tests showed that the thrust load was distributed between the aft and forward bulkheads--75 percent of the load (30,000 lbs) at the aft bulkhead and 25 percent at the forward bulkhead. The imposition of 75 percent of the thrust load on the aft bulkhead will shorten its service life. According to the evidence for the worst case, which is a DC-10-40 with the largest available engines, the estimated life of the aft bulkhead would be greater than 3,000 flight-hours. The bulkhead of the DC-10 series 10, 30, and 40 aircraft are essentially identical. Further analysis based on the DC-10-10 thrust showed that an undamaged aft bulkhead would support the entire thrust load.

During the postaccident investigation, McDonnell-Douglas conducted flight tests to measure the wing pylon's relative deflection at the aft pylon mount and the stress created at selected nearby structural members throughout the normal flight regime. The flight regime investigated included, in part, taxi; takeoff including normal and rapid rotations; 2-G turns; moderate turbulence encounters; 2-G pullups, 0.2-G pushovers, landings, and rollouts; and the effects of maximum reverse thrust. The highest stresses measured on the aft bulkhead were less than 10 percent of the static strength of the material in the bulkhead.

Other tests were conducted at McDonnell-Douglas to determine the stress distribution and residual strength of the aft bulkhead under various load conditions with cracks in the forward flange. The aft bulkhead was mounted in a cantilevered structure that simulated the aft 3 ft of the pylon. Loads were applied to the bulkhead through the lug of the aft pylon at the wing attachment joint. The damage to the bulkhead was imposed by saw cuts, the ends of which were further cracked by the application of cyclic loads. Photo-stress and strain gage data were taken with the flange cracked 6 inches, 10.5 inches, and 13 inches; the latter condition was intended to replicate the crack and fatigue damage evident on the accident aircraft. It was determined that even a 6-inch crack would extend by fatigue progression with the application of cyclic loads representative of those encountered in service. The vertical and side loads representing those for a takeoff rotation with gusty crosswinds were applied to the bulkhead with the 13-inch crack without producing failure. A thrust component load was then added and increased to 9,000 lbs. at which time the bulkhead failed. The ends of the 13-inch crack, however, progressed to fastener holes, whereas the crack in the accident bulkhead did not. A theoretical analysis by a McDonnell-Douglas stress engineer showed that vertical and side loads alone could fail the bulkhead completely with a 13-inch preexistent crack in the forward flange.

During the reassessment of the fail-safe analysis of the aft bulkhead, the effect of a 6-inch fracture on the bulkhead's forward upper flange was further analyzed and tested. The crack location was similar to the locations of those found during postaccident inspections. The analysis and tests showed that the damaged structure could carry the fail-safe design loads for the worst case--the aircraft with the largest engine.

During ground operation of the aircraft--taxiing, landing, and takeoff rolls--the aft bulkhead is subjected to compression loads and the aft end of the pylon is forced upward. During rotation, the loading changes and the aft bulkhead is subjected to tension-type loads. Those loads were found to be significantly lower than the fail-safe design loads.

1.16.4 Wind Tunnel and Simulator Tests

The wind tunnel at the National Aeronautics and Space Administration's Langley Research Center was used to determine the aerodynamic characteristics of a DC-10 wing with the left engine and pylon missing, left wing leading edge damaged, and the left wing's outboard leading edge slats retracted. In this configuration, the aircraft's stall speed, minimum control speeds with the critical engine inoperative (V_{MC}), and controllability were calculated. The effects that the loss of the No. 1 hydraulic system and the possible loss of the No. 3 hydraulic system would have on the aircraft's control authority were also investigated and calculated.

The DFDR data, aerodynamic data derived from wind tunnel tests, and the atmospheric conditions on the day of the accident were integrated into the Douglas Motion Base Simulator. The following conditions were simulated: (1) The separation of the No. 1 engine and pylon and the aerodynamic effects of the separation and resultant damage, such as changes in the aircraft's gross weight and lateral and longitudinal c.g.; (2) the uncommanded retraction of the left wing's outboard leading edge slats; (3) the loss of the No. 1 and No. 3 hydraulic systems; (4) the loss of power from the No. 1 a.c. electrical bus and resultant loss of the captain's flight instruments; and (4) both the loss and retention of the stall warning system and its stickshaker function.

The wind tunnel data for the damaged aircraft were correlated with the DFDR data so that the simulator data reflected those derived from Flight 191's DRDR. With the slats extended, the all-engine-operating stall speed was 124 KIAS; the asymmetric slat-retracted stall speed for the left wing was 159 KIAS; and the estimated wings-level V_{MC} for the damaged aircraft was 128 KIAS. With a 4° left bank-- a bank into the missing engine — 159 KIAS was the minimum speed at which directional control could be maintained with the engines operating at takeoff thrust.

Each of the thirteen pilots who participated in the simulation was thoroughly briefed on the flight profile of Flight 191. In the simulator the No. 1 engine and pylon assembly was programmed to separate at 10° of rotation on all takeoffs with simultaneous loss of the No. 1 hydraulic system. On some test runs the No. 3 hydraulic system was also programmed to fail. Generally, slats began to retract about 1 sec after the engine and pylon separated and were fully closed in about 2 sec. Some test runs were conducted with the slat retraction beginning 10 to 20 sec after the engine and pylon separated. Speed control guidance from the flight director was available for all runs, and the stickshaker, programmed for the slat-retracted-airspeed schedule, was operational on some runs.

During the tests, about 70 takeoffs and 2 simulated landings were conducted. In all cases where the pilots duplicated the control inputs and pitch attitudes shown on the Flight 191's DFDR, control of the aircraft was lost and Flight 191's flight profile was duplicated. Those pilots who attempted to track the flight director's pitch command bars also duplicated Flight 191's DFDR profile.

According to American Airline's procedures, the standard rate of rotation is between 3° to 4° per second, whereas Flight 191 rotated at only about 1.5° per second. In those simulations in which the standard rate was used, the aircraft lifted off at a lower airspeed, and the airspeed did not increase to the levels recorded by Flight 191's DFDR. The left roll began at 159 KIAS; however, because of the lesser amount of excess airspeed, the roll started below 100 ft a.g.l. In those cases where slat retraction was delayed, the left roll started at a higher altitude but its characteristics remained the same. In all cases, however, the roll began at 159 KIAS.

In many cases, the pilots, upon recognizing the start of the roll at a constant pitch attitude, lowered the nose, increased airspeed, recovered, and continued flight. The roll angles were less than 30°, and about 80 percent right rudder and 70 percent right-wing-down aileron were required for recovery. In those cases where the pilot attempted to regain the 14° pitch attitude commanded by the flight director command bars, the aircraft reentered the left roll.

On those test runs with an operative stickshaker programmed to begin at the slat-retracted-airspeed schedule, the stickshaker activated 7 sec after liftoff and the pilot flew the aircraft at the stickshaker boundary speed of 167 to 168 KIAS (V_2 + 15). Also, when V_2 + 10 was obtained and the pilot disregarded the pitch command bars, a stable climb was readily achieved. Attempts to duplicate the 1-sec interval of zero rudder displacement did not have any noticeable effect on the flight profile.

Based on the probable electrical configuration existing after the takeoff of Flight 191, pilots and test pilots who testified at the Safety Board's public hearing believed that the stall warning system and the slat disagreement warning light were inoperative. They stated that the flightcrew cannot see the No. 1 engine and left wing from the cockpit and, therefore, the first warning the flightcrew would have received of the stall was the beginning of the roll. Under these circumstances, none of these pilots believed that it was reasonable to expect the flightcrew of Flight 191 to react in the same manner as did the simulator pilots who were aware of Flight 191's profile and were able to recover from the stall.

The FAA conducted a second series of tests to determine the takeoff and landing characteristics of the DC-10 with an asymmetrical leading edge slat configuration. The slat configuration which existed on Flight 191 before impact was duplicated during about 84 simulated takeoffs and 28 simulated landings. Takeoffs were performed at both normal and slow rotation rates, at normal V_R speeds, at V_R -5 kn, and with thrust reduced to simulate a limiting weight condition during a second-segment climb.

The "slat disagree" light, takeoff warning system, and stall warning system were programmed to operate properly for both the normal and asymmetric outboard slat configuration.

Landings were performed at the maximum landing weight, 50° of flap, and a normal approach speed. The simulator was programmed so that a left outboard slat failure would cause the slat to fully retract at altitudes as low as 30 ft a.g.l. The FAA concluded that "The speed margins during the final portion of the landing approach are also very small; however, the landing situation is considered less critical since powered slat retraction from the landing configuration requires 18 seconds and an additional thrust is readily available to adjust the flight path." 4

During these tests, none of the pilots experienced problems with aircraft controllability. In many of the test runs, the stickshaker activated at or just after liftoff, and the pilots altered the aircraft's attitude and airspeed in response to the warning. A loss of thrust from an engine during the takeoff roll was not simulated during any of the tests. Based on a study performed by the J. H. Wiggins Company 5, the best estimates of the probabilities of an uncommanded slat retraction during takeoff ranged from one chance in one hundred million (1×10^{-8}) to two chances in a billion (2×10^{-9}) per flight.

1.17 Other Information

1.17.1 Air Carrier Maintenance Procedures

On May 31, 1975, and February 1, 1978, the McDonnell-Douglas issued DC-10 Service Bulletins 54-48 and 54-59, respectively. Both bulletins were issued to correct service-related unsatisfactory conditions. Service Bulletin 54-59 called for the replacement of the pylon forward bulkhead's upper and lower spherical bearings and contained procedures for accomplishing the maintenance. Compliance was recommended at the "operator's convenience."

Service Bulletin 54-48 called for the replacement of the pylon aft bulkhead's spherical bearing, and compliance with the modification was "optional, based on operator's experience." The procedures for accomplishing the modification contained the following note: "It is recommended that this procedure be accomplished during engine removal." The Service Bulletin later reiterated the recommendation and then stated, "The following instructions assume that engines 1 and 3 are removed." However, the vice president for maintenance and engineering, American Airlines, testified at the Safety Board's public hearing that the manufacturer's consideration for maintenance timing is not necessarily consistent with air carrier operations. For example, American Airlines' maintenance cannot forecast "with any great accuracy" when or where an engine would have to be changed. Since "it has to be scheduled," it would have been impractical to try to carry out the procedures of Service Bulletin 54-48 in that manner, and the aircraft would have to be scheduled to undergo the modification.

Service Bulletin 54-48 directed that the pylons were to be removed in accordance with the procedures contained in Chapter 54-00-00 of the DC-10 Maintenance Manual. Chapter 54-00-00 called for, first, removal of the engine and then removal of the pylon. The pylon alone weighs about 1,865 lbs. and its c.g. is located about 3 ft forward of the forward attachment points whereas the pylon and engine together weigh about 13,477 lbs. and the c.g. of the assembly is located about 9 ft forward of the forward attachment points. According to the manual, the sequence shown for the removal of the attach fittings was: The forward upper attach assembly, the forward lower attach assembly, the thrust link, and the aft bolt and bushing.

American Airlines decided to comply with Service Bulletins 54-48 and 54-59 and to perform the work during a maintenance "C" check at its Tulsa maintenance facility. (See appendix C.) On July 28, 1978, American Airlines issued Engineering Change Order (ECO) R-2693 establishing the maintenance procedures for accomplishing the modifications contained in the service bulletins.

The ECO was developed from the company's experiences during modifications on four DC-10-30's during the spring and fall of 1977, at Los Angeles, California. American Airlines, in accordance with a contract with a foreign carrier, modified four of the foreign carrier's DC-10-30's. The carrier also requested that American Airlines perform the spherical bearing replacement program contained in Service Bulletins 54-48 and 54-59. While establishing the maintenance procedures for the four DC-10-30's, American's maintenance and engineering personnel evaluated the feasibility of raising and lowering the engine and pylon assembly as a single unit using a forklift-type supporting device. This technique would save about 200 man-hours per aircraft, but more importantly from a safety standpoint, it would reduce the number of disconnects (i.e., hydraulic and fuel lines, electrical cables, and wiring) from 79 to 27. American personnel knew that United Airlines was using an overhead hoist to lower and raise the engine and pylon assembly as a single unit.

American Airlines personnel contacted McDonnell-Douglas personnel about this procedure. According to the American Airlines' manager of production for the Boeing 747 and DC-10 in Tulsa, Oklahoma, who participated in the development of the maintenance procedures, a McDonnell-Douglas field service representative stated that McDonnell-Douglas did not know of any carrier that was removing the engine and pylon as single unit. He said that the field service representative conveyed concern "in reference to clearances to me." However, he assumed that these clearances involved those between the clevis and the fore and aft faces of the aft pylon bulkhead's spherical bearing.

The McDonnell-Douglas field service representative who was contacted by American's personnel stated that he conveyed American's intentions to his superiors. According to him, "Douglas would not encourage this procedure due to the element of risk involved in the remating of the combined engine and pylon assembly to the wing attach points" and that American Airlines' personnel were so advised.

McDonnell-Douglas does not have the authority to either approve or disapprove the maintenance procedures of its customers. American Airlines decided to lower the engine pylon assembly as a single unit and requested that McDonnell-Douglas provide it information concerning the c.g. of the engine and pylon, including the nose cowl and both fan cowl and core cowl thrust reversers, as a single unit. The single unit was to be lowered by a forklift. On March 31, 1977, the McDonnell-Douglas field service representative informed his company that American Airlines "proposes to drop the wing engines, pylon ... as a single unit package directly on to an engine stand by means of a (forklift)" and then asked for the "C.G. of the pylon in the above described condition." On April 8, 1977, McDonnell-Douglas furnished the data to American.

The evidence showed that, during the time the procedure was in use, several McDonnell-Douglas employees saw the engine and pylon assembly after it was lowered from the wing; however, none of them observed either the actual mating, separating, raising, or lowering of the unit. Those who stated that they had seen the unit resting on the floor of a hangar also stated that they attached no significance to what they saw.

American Airlines used the newly developed removal method to modify the four foreign DC-10-30's. While working on the first aircraft, the maintenance personnel had difficulty removing the forward bulkhead's attach assemblies before removing the aft bearing bolt and bushing. They reversed the procedure and found that removing the aft bolt and bushing first expedited the removal of the forward attach assemblies and the thrust link. The reversed procedure was followed on the remaining three aircraft, and the modification program was completed. The fleet inspection conducted after the accident did not disclose any damage to the upper flanges on these four aircraft. However, the DC-10-30's aft bulkhead design affords more clearance between the bottom of the clevis and the upper spar web fasteners than the DC-10-10's design.

When the decision was made to modify American Airline's DC-10-10 fleet, the procedures used during the DC-10-30 modification program were adopted and incorporated in ECO R-2693. A Hyster forklift, Model 460B, American Airlines No. 3145, was used to raise and lower the engine and pylon assembly. The forklift has a design load capacity of 42,500 lb. In addition to vertical movement, the lifting forks can be moved in several directions: They can be yawed from left to right, tilted from left to right, tilted forward and aft, or moved laterally by moving the mast in the desired direction. A new c.g. for the DC-10-10's engine and pylon assembly was computed by American Airlines and instructions for centering the forklift at the c.g. were incorporated in the ECO. The operator was directed to insert the forks into an engine shipping stand and attach the supported stand to the engine. The ECO stated, "Adjust the engine support adapter aft so that the centerline of the lifting forks are centered with the center hinge on the (engine's) thrust reverser." The lifting forks are 5 ft apart. There was no mark on the forklift denoting the midpoint between the forks; therefore, alignment was a visual estimate. (See figure 13.)

The engine shipping stand, which can be used to support either a JT9D or CF6-6D engine, was used to support the engine and pylon assembly on the lifting forks. The stand can be adjusted for the different c.g.'s of the two engines, which are denoted by an arrow. The stand has a movable top cradle to which the engine is affixed; the cradle can be moved about 12 inches horizontally. There is also an arrow on the cradle's frame. The arrow on the frame of the cradle must be aligned with the arrow denoting the type engine to be loaded before the engine is placed on the cradle. Eight clamps secure the cradle in position on the stand. However, the cradle can be moved on the stand after the engine has been affixed to it.

American Airlines' maintenance personnel testified about their experiences with the forklift while handling the engine and pylon assembly. Directions were transmitted to the lift operator either by voice, hand signals, or both. The testimony varied regarding the capability to raise or lower the lifting forks a finite distance. One mechanic said it could be limited to .001 inch; the estimates of others ranged from .25 to .06 inch.

This image is not available at this time.

Figure 13. View of forklift and engine stand.

When the full weight of the engine and pylon assembly was on the lifting forks, the pressure gauge reading was 18,000 lbs. Maintenance personnel stated that a 2,000-lb to 3,000-lb pressure bleedoff on the pressure gauge was common; however, they all stated that the lifting forks did not move. Supervisory personnel stated that it was normal for the gauge reading to bleed off 2,000 to 3,000 lbs during a 15-min period without any perceptible load movement. The load remains fixed because of the frictional load on the mast and rollers. Although mechanics testified that the load did not move, they also said that they would manipulate the controls to restore the original reading on the pressure gauge. One mechanic stated that the pylon and engine assembly would "jump" as lowering began. He said the "jerking" motion moved the forks about 1 or 2 inches.

On October 5, 1979, McDonnell-Douglas tested the capability of its Hyster 460B. An 18,000-lb load was placed on the forks, and the equipment was tested for drift down and control capability. The tests showed that an experienced operator was able to move the load in both directions vertically in steps of .187 to .250 inch consistently. When the load was stopped the peak dynamic deflections were (±) .03 to (±) .06 inch about the final rest value. A sink rate of about 1.25 inches per hour was measured during the drift down test.

From March 29 through 31, 1979, the accident aircraft underwent the spherical bearing modification. On April 19, 1979, the forklift's maintenance log contained a writeup which noted, in part, "trouble shooting, forks creeping down under load." There was no record that any corrective action was taken. On May 17, 1979, the log showed "Inspect lift cylinder--per Engineering." There was no record of any findings. On June 20, 1979, the forklift was tested for drift down. An engine-pylon assembly was placed on the lifting forks, and the forks drifted down 1 inch in 30 min. A lift cylinder check valve was found to be defective and was replaced.

The ECO's procedures for detaching the pylon from the wing were as follows: Item F of the ECO called for the removal of the lower attach plug and attaching parts; item G called for removal of the upper attach plug and attaching parts; item H called for removal of the thrust link; and item I--called for removal of the aft bolt and bushing. The ECO did not caution or advise that items F through I must be performed in the sequence listed. According to American Airlines maintenance and supervisory maintenance personnel, since the ECO did not contain such advice, it did not require that items F through I be performed in sequential order. Rather, it merely provided a checklist and signoff sheet to insure that all the steps were performed. Consequently, maintenance personnel saw no harm in performing the modification by first removing the aft spherical bearing's bolt and bushing. Engineering personnel who drafted the ECO were not informed formally of the difficulties experienced in removing the fittings as prescribed.

Mechanics and the inspector who performed the spherical bearing modification on the accident aircraft recounted the operation for the Safety Board. The midnight shift started the modification and removed the aft spherical bearing's bolt and bushing before going off duty on March 30. When the day shift reported for duty, two of the mechanics saw the upper lug of the aft bulkhead come in contact with the bolts attaching the clevis to the wing. These bolts are located at the top of the clevis. The forklift's engine was running at the time, and the pressure gauge reading was 18,000 lbs. When the crew could not remove the forward attach assemblies, they discovered that the engine stand was misaligned. The clamps holding the cradle to the stand were loosened, and the lifting forks and engine stand were shifted to the left--forward on the engine--until the cradle was properly aligned on the stand, The clamps were then affixed. According to one mechanic, the stand was moved forward about 12 inches. After the stand was realigned, the forward upper and lower attach assemblies were removed, and the engine and pylon assembly was lowered to the hangar floor.

The testimony of the mechanics disclosed that the mechanics' training for this modification was limited to on-the-job training. The inspector had not received any training with regard to this particular modification.

The work cards used to accomplish the modification on the accident aircraft were examined. The inspector's signoff blocks on the ECO's work cards did not contain any requirement for the inspector to inspect the forward or aft attach assemblies after the pylon and engine had been reinstalled on the wing. The work cards included in the ECO showed that after the inspector cleared the pylon for installation, his only inspection requirements were to inspect the connections for integrity and to check for fuel and hydraulic leaks. The work cards also disclosed that there was a nick on the top surface of the pylon aft bulkhead's attach lug, and some of the mechanics recalled seeing the nick.

The inspector stated that chronic problems in the maintenance procedures should be reported on a significant item form. This form is then channeled through maintenance supervision to engineering for action. He said that he thought that the out-of-sequence performance of the tasks in an ECO should be reported to those who formulate the ECO's.

In summary, an overall assessment of the manner in which American Airlines' Engineering developed and then monitored the two ECO's used to replace the pylon's spherical bearings showed they had evaluated the capabilities of the forklift before the decision was made to use the equipment. The engineer who wrote the procedures knew that the forklift was capable of applying high forces. He believed that the movement of the lifting forks could be controlled within "very small fractions" of an inch, but he did not know the resultant rate of movement of these forks in response to a control input. However, since the maintenance personnel were familiar with the forklift, he believed that its use would be more suitable for "our operation."

According to the engineer the procedures of the ECO's and the capabilities of the forklift were analyzed for safety of operation and personnel informally. However, they did not use or perform a formal fault analysis to evaluate the effect on the structure that might result from either personnel error or equipment malfunction. Procedures of this nature, according to the engineer, had never been used to evaluate ECO's. Members of the engineering department observed the prototype procedure on the first two DC-10-30's. However, they only observed the lowering and raising of the engine and pylon as a single unit. They did not witness the removal of the wing to pylon attach assemblies; consequently they were not aware of the difficulties that were encountered, and the subsequent departure from the sequence contained in the ECO.

The maintenance procedures used by Continental Airlines to accomplish SB 54-48 were similar to those of American Airlines. The same type forklift was used to raised and lower the pylon and engine assembly.

On December 19, 1978, the upper flange of the No. 1 pylon aft bulkhead on Continental Airlines DC-10-10, N68041, sustained a crack which penetrated the flange. The upper and lower forward attach assemblies had been removed, and the aft spherical bearing's bolt and bushing had been removed and a pin inserted in its place. When the pin was removed, the aft end of the pylon moved up slightly and a "loud pop described as a pistol shot" was heard. The fracture was discovered, the upper flange repaired, and the aircraft was returned to service.

On February 22, 1979, the upper flange of the No. 3 pylon aft bulkhead in Continental Airlines DC-10-10, N68049, sustained a crack which penetrated the flange. In this case the pylon had been disconnected and the lead mechanic was attempting to clear the aft bulkhead lug from the clevis. He instructed the forklift operator to raise the nose of the engine in order to lower the aft end of the pylon. The forklift operator either misunderstood or inadvertently moved the wrong lever and lowered the nose. The aft end of the pylon was raised with the same results and noise effects described above.

Continental Airlines' investigation concluded that both mishaps were maintenance errors and neither was reported to the FAA.

The forklift was checked for drift down after the December mishap and "nothing was found." Several months later the unit was rechecked. Downward drift was found, the malfunction was corrected, and no further difficulties were encountered.

During the postaccident investigation, the maintenance procedures of all United States carriers operating DC-10 series aircrafts were inspected. The evidence disclosed that United States carriers had removed and reinstalled 175 pylon and engine assemblies. Eighty-eight of these operations involved the lowering and raising of the pylon and engine as a single unit. Of these 88, 12 were lowered and raised with an overhead crane. The remaining 76 were lowered and raised with a forklift. The nine situations wherein impact damage was sustained and cracks found involved the use of the forklift.

1.17.2 Federal Aviation Administration Reporting and Surveillance Procedures

Air carrier reporting requirements are established in 14 CFR Part 121, and are basically contained in two regulations. 14 CFR 121.703 establishes the mechanical reliability report (MRR) system. The regulation requires a certificate holder to report "the occurrence or detection of each failure, malfunction, or defect concerning ...." The regulation contains 16 paragraphs setting forth the conditions that must be reported. 14 CFR 121.703(14) requires the carrier to report "Aircraft structure requiring major repair" and paragraph (15) requires the carrier to report "cracks, permanent deformation, or corrosion of aircraft structures, if more than the maximum acceptable to the manufacturer or the FAA." According to the FAA, the MRR system is, for the most part, limited to service-related problems and to failures and malfunctions which have occurred after the aircraft's engines are started with the intent for flight and while they are running. In response to a question as to whether paragraph (15) would apply to the December 1978, and February 1979, upper flange cracks at Continental Airlines, a FAA air carrier maintenance specialist stated that historically and traditionally the MRR procedures have always dealt with service-related problems. He said that under the MRR concept "we would not consider it because it was not a service related problem."

14 CFR 121.707, Alteration and Repair Reports reads as follows:

a. Each certificate holder shall, promptly upon its completion, prepare a report of each major alteration or major repair of an airframe, aircraft engine, propeller, or appliance of an aircraft operated by it.

b. The certificate holder shall submit a copy of each report of a major alteration to, and shall keep a copy of each report of a major repair available for inspection by, the representative of the Administrator assigned to it.

The authority for an air carrier to perform maintenance is derived from several sources. Pursuant to the provisions of 14 CFR 21, Subpart M, an air carrier may be certified by the FAA as a Designated Alteration Station (DAS), as were American and Continental Airlines. In accordance with this certification, either carrier could issue supplemental type certificates and perform its own alterations without prior FAA approval; however, the required reports must be submitted to the FAA.

14 CFR 121.379 also contains authorization for a Part 121 certificate holder to perform maintenance and alterations. This section reads, in part, as follows:

(a) A certificate holder may perform, or it may make arrangements with other persons to perform maintenance, preventive maintenance, and alterations as provided in its continuous airworthiness maintenance program and its maintenance manual....

(b) A certificate holder may approve any aircraft, airframe, aircraft engine, propeller, or appliance for return to service after maintenance, preventive maintenance, or alternations that are performed under paragraph(a) of this section. However, in the case of a major repair or alteration, the work must have been done in accordance with technical data approved by the Administrator.

The investigation showed that there were large differences in the interpretation of what constituted a major alteration or repair despite the guidelines contained in the Federal regulations. 14 CFR 1.1 defines a major repair and alteration as follows:

A major alteration means an alteration not listed in the aircraft, aircraft engine, or propeller specifications--(1) That might appreciably affect weight, balance, structural strength, performance, powerplant operation, flight characteristics or other qualities affecting airworthiness; or (2) That is not done according to accepted practices or cannot be done by elementary operations.

'Major Repair' means a repair: (1) That, if improperly done might appreciably affect weight, balance, structural strength, performance, powerplant operation, flight characteristics, or other qualities affecting airworthiness; or (2) That is not done according to accepted practices or cannot be done by elementary operation.

The FAA air carrier maintenance specialist stated that the classification of major alteration or repair related to the requirement that either or both be accomplished in accordance with approved data. It is a method of protecting the type certificate design and of assuring that the repair or alteration does not change or modify a design feature.

Continental Airlines' principal maintenance inspector stated that there are no "clear cut rules" for interpreting the regulation. "It has been argumentative for 30 years that I know (sic) it." Although it was his opinion that the major part of the bulkhead was a structurally significant item, he did not consider the upper flange part of the bulkhead.

The FAA team investigating maintenance and airworthiness procedures after the accident found that FAA regulations and guidance did not adequately define what constitutes a major repair. The team found that the repairs made to five pylons, including the two upper flanges at Continental Airlines, constituted major repairs since critical structure was involved. Therefore, the team concluded these repairs should have been submitted to the FAA for approval.6

The FAA principal maintenance inspectors are responsible for the surveillance of the maintenance activities and procedures of those air carriers assigned to their office. The principal inspector for Continental Airlines was not aware of the cracks sustained in the upper flanges of the two Continental aircraft during the modification procedure, nor did he know when the carrier began the modifications contained in Service Bulletins 54-48 and 54-59.

The principal maintenance inspector at American Airlines' Tulsa Maintenance Base was also the chief of the Tulsa Air Carrier District Office, and had served 7 years as chief. However, he had been principal inspector for American Airlines since January 15, 1979. The principal maintenance inspector did not know that American Airlines was removing the pylon and engine assembly as a single unit until May 30, 1979. In accordance with a request from his office sometime before May 23, 1977, American Airlines had been requested to revise its ECO distribution to the Tulsa Air Carrier District Office to "include cover sheets only, without the detailed technical data." Thus, the FAA received only the cover sheet of ECO R-2693. The material containing the maintenance procedures was retained by the carrier, and the Tulsa Air Carrier District Office did not conduct any checks on the pylon maintenance.

The cover sheet of ECO R-2693 classified the repairs as minor. The principal inspector said that the cover sheet also contained the FAA-approved Service Bulletins 54-48 and 54-59. Therefore, he had no reason to either doubt the classification or the carrier's capability to carry out the repair. In his opinion, there was no reason to expend manpower in surveillance of a minor repair.

Evidence developed during the investigation showed that FAA approval of a service bulletin indicates to the operator that the change in design included in the bulletin has been approved by the FAA, thereby relieving the operator of the necessity of obtaining his own design approval. However, the FAA approval does not apply to the maintenance procedures incorporated in the service bulletin.

1.17.3 DC-10 Certification

The DC-10's pylon structure, flight controls, hydraulic system, and electrical system were certificated in accordance with the applicable provisions of 14 CFR Part 25 effective February 1, 1965, as amended, and Special Condition No. 25-18-WE-7, January 7, 1970, as amended. (See appendix E.)

Special Condition No. 25-18-WE-7, Docket No. 10058, was issued pursuant to 14 CFR 21.16 because the airworthiness regulations of Part 25 did not contain adequate or appropriate safety standards for the aircraft because of a novel or unusual design feature. In the case of the DC-10, this feature was the fully powered flight control system.

The function of assessing compliance with certain aspects of the type certification was delegated to FAA Designated Engineering Representatives who were employed by McDonnell-Douglas. Such representatives are designated by the FAA to represent the Administrator pursuant to Section 314 of the Federal Aviation Act of 1958 and 14 CFR 183.29. According to FAA and McDonnell-Douglas witnesses, the workload involved in the certification process far exceeds the FAA's manpower resources.

The chief of the FAA's Western Region Aircraft Engineering Division stated that during the type certification process the review of the basic data and the most critical tests are reserved to the FAA itself. The fault analysis data are reviewed and approved by FAA engineering personnel. He also said that little delegation is done in the flight test area. The chief of the FAA's Western Region Flight Test Branch stated that the DC-10's type certification required 500 hrs of flight testing, and 90 percent of that time was flown by FAA test pilots.

The principle underlying the regulations concerning the certification the aircraft's systems was redundancy. This principle contemplates that, while each critical component of a system is required to perform functions within the design envelope of the aircraft, its failure will nevertheless be assumed. Accordingly, appropriate analyses and tests are required to insure that sufficient redundancy exists so that after a single failure of any component or element its functions will be distributed to other components capable of assuming them safely.

The criteria for the certification of the aircraft's pylon and its components were contained in 14 CFR 25.571, "Fatigue Evaluation of Flight Structure". (See appendix E.) This regulation required the manufacturer to show, by analysis, tests, or both, that those parts of the structure whose failure could result in catastrophic failure of the aircraft would be able to withstand the repeated loads of variable magnitude expected in flight, that catastrophic failure or excessive structural deformation that could adversely affect the flight characteristics of the aircraft are not probable after fatigue failure or obvious failure of a single principal structural element, and that after this type of failure of a single principal structural element, the remaining structure must be able to provide an alternate load path. The regulation only required that fatigue damage be evaluated. The chief of the FAA's Western Region Aircraft Engineering Division testified that under normal loading there was "extremely low stress" on the upper flange and "the possibility of fatigue was believed to be extremely low, low enough that you would not consider fatigue failure.

Because all flight controls were hydraulically actuated and the basic regulations did not cover this configuration, Special Condition No. 25-18-WE-7 was formulated. However, the trailing edge flap and leading edge slat systems were certified under the basic regulations.

The leading edge slat system was certified in accordance with 14 CFR 25.671--general control system requirements, 14 CFR 25.675--control system stops, 14 CFR 25.685--detailed design requirements for flight control systems, and 14 CFR 25.689--cable system design. The chief program engineer at McDonnell-Douglas said that the flap control requirements of 14 CFR 25.701(a) were also applied to the slats. Paragraph (a) states:

> *The motion on the flaps on opposite sides of the plane of symmetry must be synchronized unless the aircraft has safe characteristics with the flaps retracted on one side and extended on the other.*

Since the left and right inboard slats are controlled by a single valve and actuated by a common drum and the left and right outboard slats receive their command from mechanically linked control valves which are "slaved" to the inboard slats by the followup cable, the synchronization requirement was satisfied. However, since the cable drum actuating mechanisms of the left and right outboard slats were independent of each other, the possibility existed that one outboard slat might fail to respond to a commanded movement. Therefore, the safe flight characteristics of the aircraft with asymmetrical outboard slats were demonstrated by test flight. These flight characteristics were investigated within an airspeed range bounded by the limiting airspeed for the takeoff slat positions --260 kns--and the stall warning speed; the flight test did not investigate these characteristics under takeoff conditions. In addition, a slat disagree warning light system was installed which, when illuminated, indicated that the slat handle and slat position disagree, or the slats are in transit, or the slats have been extended automatically.

The program engineer stated that the commanded slat position is held by trapped fluid in the actuating cylinder, and that no consideration was given to an alternate locking mechanism. The slats' hydraulic lines and followup cables were routed as close as possible to primary structure for protection; however, routing them behind the wing's front spar was not considered because of interference with other systems.

The branch chief of the Reliability and Safety Engineering Organization of the Douglas Aircraft Company described the failure mode and effects analysis (FMEA) and fault analysis. The witness indicated that the FMEA was a basic working document in which rational failure modes were postulated and analyzed; vendors and subcontractors were requested to perform similar analyses on equipment they supplied to McDonnell-Douglas. Previous design and service experience was incorporated in the initial DC-10-10's FMEA's and analyses were modified as the design progressed. The FMEA's were synthesized to make fault analyses, which were system-oriented summary documents submitted to the FAA to satisfy 14 CFR 25.1309. The FAA could have requested and could have reviewed the FMEA's.

The basic regulations under which the slats were certified did not require accountability for multiple failures. The slat fault analysis submitted to the FAA listed 11 faults or failures, all of which were correctable by the flight-crew. However, one multiple failure--erroneous motion transmitted to the right-hand outboard slats and an engine failure on the appropriate side--was considered by McDonnell-Douglas in its FMEA. The FMEA noted that the "failure increases the amount of yaw but would be critical only under the most adverse flight or takeoff conditions. The probability of both failures occurring is less than 1×10^{-10}." The evidence indicated that this FMEA was not given to the FAA formally but was available for review.

Special Condition No. 25-18-WE-7 requires the applicant to show that the aircraft is capable of continued flight and landing after "any combination of failures not shown to be extremely improbable." According to FAA witnesses, the definition for extremely improbable that they have been using and have been accepting for a number of years is one chance in a billion, or 1×10^{-9}.

The regulation, 14 CFR 25.207, requires that "Stall warning with sufficient margin to prevent inadvertent stalling with the flaps and landing gear in any normal position must be clear and distinctive to the pilot in straight and turning flight." The warning can be furnished through the inherent aerodynamic qualities of the aircraft or by a mechanical or electronic device. A visual warning device is unacceptable. The warning must begin at a speed exceeding the stall speed or the minimum speed demonstrated "....by seven percent or at any lesser margin if the stall warning has enough clarity and duration, distinctiveness, or similar properties." The flight testing of the DC-10 disclosed that the inherent aerodynamic stall warning exceeded the required regulatory margin in all flap configurations until the landing flap configuration (50°) was reached. According to the chief of the FAA's Flight Test Branch, with 50° flaps the stall buffet still precedes stall onset, "but it occurs quite close, within just a few knots of the aerodynamic stall." Since the margin did not meet the regulatory criteria, a stall warning system was installed.

The initial DC-10 design incorporated the left (No. 1) and right (No. 2) autothrottle speed computers (AT/SC) as stall warning computers. The No. 1 and No. 2 AT/SC's were powered by the No. 1 and No. 3 a.c. buses, respectively. The No. 1 AT/SC received inputs from the left inboard flap position transmitter, from a position sensor on the left outboard slat section, and the left angle-of-attack sensor. The No. 2 AT/SC received its inputs from counterpart sensors and components on the right side of the aircraft. The stickshaker motor was mounted on the captain's control column and was powered by the No. 1 d.c. bus. A stall signal from either computer would actuate the stickshaker motor. The design contained provisions for a second stickshaker motor to be mounted on the first officer's control column; however, the second stickshaker was a customer designated option. The accident aircraft's stall warning system did not incorporate the second stickshaker described above.

The December 1, 1978, revision of 14 CFR 25.571 retitled the regulation "Damage-Tolerance and Fatigue Evaluation of Structure." The fail-safe evaluation must now include damage modes due to fatigue, corrosion, and accidental damage. According to the manufacturer, the consideration for accidental damage was limited to damage which can be inflicted during routine maintenance and aircraft servicing.

The FAA's Aircraft Engineering Division chief also stated that while the recertification process disclosed a deficiency in design data on file with the FAA it did not disclose any deficiency in the pylon's design. In some cases, the manufacturer had the data on file. In one instance, the data concerning the alternate load paths for thrust loads following a thrust-link failure were questioned. The manufacturer's analysis assumed the loads would be carried by the forward bulkhead. The manufacturer also stated that the thrust loads could be carried out by the aft bulkhead. The FAA asked McDonnell-Douglas to substantiate this claim, and they did so successfully.

As a result of the postaccident simulator tests, an AD was issued which required, as a condition for reinstatement of the type certificate, that the aircraft be operated either with both AT/SC's installed and operating, or with a modified single AT/SC that would receive slat information from both sides of the aircraft. (See appendix F).

On July 30, 1979, a Notice of Proposed Rule Making (NPRM), docket No. 79WE-17AD, was issued. (See appendix F.) The NPRM contained an AD which will require that the stall warning system incorporate two AT/SC's and two stickshaker motors, and that the AT/SC's be modified to receive position information from both outboard wing leading edge slat groups.

1.17.4 DC-10 Maintenance and Inspection Programs

During the investigation, the development of the DC-10 maintenance program was studied to determine the methods used to establish the aircraft's maintenance program and the inspection requirements for the wing pylons. The program guidelines were embodied in the "Airline Manufacturer Maintenance Program Planning Document, MSG-2." The document was formulated by a working group composed of representatives of user air carriers, McDonnell-Douglas, and one or more FAA observers. The document was then submitted to the FAA Maintenance Review Board where FAA observers and engineers met to evaluate the proposals. The review board issued a report which prescribed the minimum maintenance program for DC-10 operators and required a review of the specific work programs of each operator by its FAA principal maintenance inspector to assure conformance with the program.

When an aircraft is delivered to an operator, the manufacturer must, by regulation, furnish the operator with a maintenance manual (14 CFR 25.1529). The manual must contain the essential information and procedures necessary to maintain the aircraft.

The maintenance programs for modern aircraft are comprised generally of three primary maintenance processes known throughout the industry as "hard-time," "condition-monitoring," and "on-condition." Hard-time is a preventive maintenance process which requires that an appliance or part be overhauled or replaced after a specific period of service. This process is generally applied to parts which are subject to predictable wear, such as engines or engine components. Condition-monitoring is a process which applies to components, the output of which can be monitored to detect degradation in performance indicating that the maintenance is required. When applying the condition-monitoring process, the potential effect of an unpredicted failure of the part is also considered.

The airworthiness of most of the structural elements of the aircraft is maintained by the on-condition maintenance process. This process requires that a part be periodically inspected against some physical standard to determine whether it can continue in service. Thus, the maintenance program established for the aircraft includes specified inspection requirements for each structural element. The inspection interval depends upon an analysis which considers the susceptibility of the part to fatigue damage, corrosion, and crack propagation. The degree of redundancy and the accessibility for inspection are also considered.

The on-condition process also incorporates the principle that similar parts behave in similar ways. Thus, if a part is analyzed to be relatively resistant to damage throughout the anticipated life span of the aircraft, an inspection of that part on every aircraft--a 100-percent inspection--may not be required; the part will be placed in a sampling inspection program and a statistically representative sample of the parts on the entire fleet of aircraft will be inspected. If a problem is detected during the sampling inspection program, the FAA's service difficulty reporting program incorporates the mechanism whereby revised inspection requirements can be evaluated and levied on the operators for application to the entire fleet of aircraft. The on-condition maintenance program, thus, is intended to be a conservative method to verify the design resistance to fatigue or corrosion damage during the aircraft's service life. However, the maintenance programs are not designed to detect damage resulting from improper manufacturing processes or maintenance.

During the investigation, the Safety Board examined closely the sampling inspection program for the wing pylon. The program, sampling base, and inspection frequency were based upon factors, including projected aircraft life as well as structurally significant items and their resistance to fatigue and corrosion.

The maintenance document (MSG-2) defined structurally significant items as "those local areas of primary structure which are judged by the manufacturer to be relatively the most important from a fatigue or corrosion vulnerability or from a failure defect standpoint," and it required that these items be classified as to relative importance. The classification and ratings of these items were based upon the fatigue, corrosion, and crack-propagation resistance properties of the structure. These properties were analyzed on the basis of fatigue testing, special tests for crack growth rates, and the company's previous experience with the aircraft structure.

The structurally significant items with a classification or rating number of 1 or 2 (indicative of a lower overall level of structural integrity) would probably be placed on a 100-percent inspection program. The 100-percent program would require that these items be inspected on every aircraft at an interval which is determined by testing and analysis. Structures classified as 3 or 4 would probably be sampled. The sampling program required the inspection of some structurally significant items on only a specified fraction of an operator's fleet and at a predetermined interval.

The initial DC-10 program required 100-percent inspection for some items and placed others on a fractional sampling program. The inspection frequency for some items on the 100-percent program was based on their classification. Structurally significant items (SSI), classified as a Class-1 SSI, were to be inspected on all aircraft every 4,000 hrs, Class-2 SSI's every 8,000 hrs, and a Class-5 SSI every 20,000 hrs. In the fractional sampling program, only a certain proportion of a carrier's aircraft was to be inspected to monitor the condition of a structurally significant item. Thus, only 1/5 of a carrier's aircraft population was to be inspected at a 20,000-hr interval to monitor a Class-1 SSI, whereas 1/12 of its aircraft population was to be inspected at a similar interval to monitor a Class-5 SSI. For example, under the sampling program, the upper attach lug of the pylon aft bulkhead was on a 100-percent inspection program, while the aft bulkhead's upper flange and other portions of the bulkhead were on a fractional sampling program. The upper attach lug is designed to fail in the event of a wheels-up landing and thus prevent fuel tank rupture; accordingly, the lug was not overdesigned and is subjected to significant stresses which places it in a class requiring 100-percent inspection every 4,000 hrs. In contrast, the rest of the bulkhead is subjected to relatively low stresses; therefore, it is considered to be less susceptible to difficulty in service and suitable for sample-type inspection.

1.17.5 Manufacturer's Service Bulletin and Customer Service Programs

The FAA's service difficulty reports and McDonnell-Douglas service bulletins were reviewed to determine if any chronic difficulties related to aft bulkhead cracking had existed before the accident. The service difficulty reports indicated that some problems existed with wing spherical bearing attach fittings. These problems were not anticipated during design and did not develop until the aircraft was placed into service. As a result, programs were launched to replace the old spherical bearings with stronger and more efficient bearings through Service Bulletin 54-48 and 54-59.

McDonnell-Douglas maintains a customer support program. Under this program, the company maintains field service representatives at the operators' maintenance facilities and receives reports from operators concerning service difficulties encountered by its aircraft.

During December 1978, when Continental Airlines cracked the forward flange of an aft bulkhead during its bearing modification program, McDonnell-Douglas provided the operator with a engineer product specialist to assist it in repairing the flange. The product engineer specialist testified that he was responsible for investigating, analyzing, and interpreting customer reports regarding unsatisfactory performance and service failure of the aircraft structure. He was also responsible for supplying any necessary corrective procedures. At the Safety Board's public hearing the engineer specialist testified that he did not see the pylon and engine assemblies raised or lowered, that he assisted the carrier in making the required repairs, and that he was told that the carrier "cracked the part while lowering the pylon. And that was the extent of the discussion."

According to the engineer, about 1 week later he wrote a short paragraph describing the problem and its disposition for inclusion into a company Operational Occurrences Report. This was published on January 5, 1979, as part of Report No. 10-7901 and read as follows:

An operator has reported a case of damage to the wing pylon aft monoball (spherical) bearing support bulkhead, P/N AUB7002-1. This apparently occurred when the pylon shifted while it was being lowered. The aft end of the pylon rotated up, and the forward lug of the wing clevis fitting contacted the upper horizontal flange of the support bulkhead. The flange on the support bulkhead was sheared off for most of its length; necessitating removal of the support bulkhead from the pylon for repairs.

Operational Occurrence Reports are distributed to all DC-10 operators. American Airlines did not recall receiving this Operational Occurrence Report, but Continental Airlines found it in its service library after the accident. The Operational Occurrence Report contained reports concerning all types of mishaps, system malfunctions, and structural defects that the manufacturer believed would be of interest to his customers. The report which contained the description of the bulkhead damage also contained reports of an air conditioning pack malfunction, a lightning strike, collapse of a passenger loading stand, and a flight attendant injury suffered in the gallery cart lift.

14 CFR 21.3 establishes the responsibility of the holder of a type certificate to report failures, malfunctions, or defects to the FAA. The regulation requires a certificate holder to report any defect in any product or part it manufactures and that it has determined resulted in any of the occurrences set forth in the regulation. The primary structural defects the certificate holder is required to report are limited to those caused by "any autogenous condition (fatigue, understrength, corrosion, etc.)." Further, 14 CFR 21.3 (d)(i) states that the reporting requirements do not apply to failures, malfunctions, or defects that the certificate holder "determines were caused by improper maintenance."

1.17.6 Manufacturer's Production Line Procedures

The production line procedures of the facilities producing the wing pylon assembly were investigated, including the installation of shims on the upper surface of the horizontal flange on the accident aircraft. According to the McDonnell-Douglas' Vice President for Quality Assurance, the .063-inch-thick shims installed on the upper shoulders of the upper flange were standard shims. He said that these shims can be installed any place they are needed to reduce a clearance. No approval is needed since the procedure is authorized by Douglas Process Standard 2.70.2 (DPS 2.70.2).

The 10-inch-long, .050-inch-thick shim installed on the accident aircraft was not a standard shim and, according to McDonnell-Douglas engineers who testified at the Safety Board's public hearing, written authorization was required to use it. Such an authorization is processed through the company's engineering liaison group and reviewed by stress liaison personnel of the structural analysis group. Rejection and Disposition Item AO81757 had been issued authorizing the insertion of the shim, and had been signed by an engineer in the liaison group. Although a McDonnell-Douglas engineer assumed that the proper stress analysis had been performed before the issuance of the Rejection and Disposition Item, there was no signature to indicate specifically that the analysis had been done, nor was space provided for such a signature.

The evidence disclosed that 23 pylons were placed into service with shims on the top of the upper flange. The clearance problem on the upper flange began with fuselage No. 15 and continued through fuselage No. 36 (the accident aircraft was fuselage No. 22). A McDonnell-Douglas investigation disclosed that the clearance problem was the result of a tooling malfunction, and it was resolved by repositioning locator pins on the tooling jigs.

In October 1974, the pylon production line was transferred from McDonnell-Douglas' Santa Monica, California, location to the Huntington Beach, California, facility. The transfer was made at fuselage No. 208. During an inspection conducted after the accident, 31 aircraft were found to have had wing pylons with loose, failed, or missing fasteners. Fifteen of these aircraft were between fuselage No. 170 and 208. Six of these 15 aircraft had more than 5 loose or missing fasteners. Of the other 16 aircraft, 1 had 7 and another had 5 loose or missing fasteners; the remaining 14 aircraft had less than 5 loose or missing fasteners. McDonnell-Douglas personnel believed that one of the causes of this production breakdown was the effect the impending transfer of the production line had upon worker experience, morale, and productivity.

The investigation of the upper spar web cracks and fasteners found on United Airlines DC-10, N1827U, fuselage No. 196, also showed that its problems probably were traceable to production line procedures at McDonnell-Douglas. The damage on the United Airlines DC-10 was limited to the cracking of the upper spar web and failure of 26 fasteners. There was no damage to the aft bulkhead flange. An examination of the aircraft's history showed that it had not been exposed to any hard landings; however, it had experienced an engine failure and had been subjected to vibrations resulting from the windmilling of an unbalanced engine during 1 hr 20 min of flight.

Engine vibration testing was conducted at the General Electric facility at Peebles, Ohio, to investigate the possibility that a significant imbalance accompanied with windmilling for 80 min was a possible or plausible explanation of the damage. The results were negative.

A metallurgical examination of the spar and fasteners showed evidence of high-cycle, low-stress fatigue along the majority of the upper spar web fractures as well as fatigue cracking in 26 of the 29 fasteners. Only one fastener had failed due to overload. Evaluation of the data indicated that there was no similarity to the damage noted on the accident aircraft; that no single event explains the damage on the United DC-10's upper spar web; and that the damage occurred over a long period of time and was likely to have initiated from manufacturing discrepancies. Fuselage No. 196 was among those manufactured at the Santa Monica plant where the greatest frequency and number of production discrepancies to the fasteners occurred.

1.17.7 DC-10 Hydraulic and Electrical Systems

Hydraulic power is provided by three hydraulic systems. Each system is powered by two engine-driven hydraulic pumps. Additionally, two electric auxiliary pumps are provided in system No. 3. Emergency hydraulic power is available from one of these auxiliary pumps when powered by the air-driven generator. Two reversible motor-pumps can transfer power from an operating system to an unpressurized system if an engine fails. In addition, two nonreversible motorpumps can provide a similar transfer of power to certain components of the flight control system.

The three hydraulic systems normally operate independently of each other and are pressurized by their respective engine-driven pumps. The systems power the flight controls, horizontal stabilizer, landing gear, brakes, and nosewheel steering. The two electric auxiliary pumps in hydraulic system No. 3 are primarily for ground use when the engines are shut down; however, auxiliary hydraulic pump No. 1 can be used as an emergency pressure source for the flight controls if all three engines are lost. This can be done inflight by deploying the air-driven generator which will provide electrical power to operate the pump.

The system 1-3 and system 2-3 reversible motor pumps are installed to transfer pressure from an operating hydraulic system to an unpressurized hydraulic system; pressure can be transferred in either direction. No fluid transfer takes place--the transfer of energy is mechanical. Control switches for these pumps are provided on the flight engineer's panel. If the fluid in the reservoir of either the operating system or the system being pressurized falls below a preset minimum, that motor-pump combination will automatically stop operating. Two nonreversible motor pumps are installed in the stabilizer and rudder hydraulic systems to provide backup hydraulic power should the normal power source fail.

Under normal operating conditions, hydraulic power is provided by the two engine-driven pumps in each system. The reversible motor pump controls are in the "arm" position to provide automatic operation in the event of engine failure, and the rudder standby power control switch is in the "arm" position to provide automatic standby power for the rudders if the No. 1 or No. 2 hydraulic system fails.

Except for the spoilers and the upper and lower rudders, each flight control surface is powered by two hydraulic systems. Hydraulic system No. 1 powers the No. 2 and No. 4 spoiler panels on each wing; hydraulic system No. 2 powers the No. 1 and No. 5 spoiler panel on each wing, and hydraulic system No. 3 powers the No. 3 spoiler panel in each wing. The landing gear is powered by the No. 3 hydraulic system.

The lower rudder is powered by hydraulic system No. 2, and its backup power is provided by the 3-2 nonreversible motor pump. The upper rudder is powered by hydraulic system No. 1. Backup power is provided by the 2-1 nonreversible motor pump. Each backup power system has its own independent reservoir and fluid. Consequently, a complete loss of hydraulic fluid in system No. 1 will not affect the operation of the backup system.

The 2-1 nonreversible motor pump also supplies backup power to the horizontal stabilizer, and the operation of the stabilizer trim reduces the fluid flow and pressure available to operate the upper rudder. However, the check valves in the rudder actuator will prevent a drop in hydraulic system pressure from causing a loss of any rudder deflection being held.

When the No. 3 hydraulic system's lines to the outboard slat actuator were severed, during pylon separation, hydraulic fluid began to be lost. The rate of loss was dependent upon the positioning of the slat control valve, and the amount of pinching of the hydraulic lines at the point of severance. According to the chief program engineer for DC-10 design, under the worst case--the control valve wide open and no pinching of the lines--it would require 4 minutes to deplete the reservoir. He further estimated that over a 30-sec to 40-sec period after the rupture there would be no pressure loss and that the retraction of the landing gear would not create significant pressure drain during the time the system remained operable. The witness testified that the hydraulic system was certified in accordance with the existing regulations and compliance with 14 CFR 25.1309 was shown by FMEA and flight testing.

During the early service history of the aircraft, some difficulties with the nonreversible motor pumps were encountered. The pumps were of a new design, and the FMEA's did not predict the in-service difficulties which occurred early in the aircraft's service history. The pumps were redesigned, and the malfunction has not recurred.

According to the witness, there has been only one incident of dual hydraulic system failure. That failure resulted from a tire failure; however, the aircraft was landed safely with one hydraulic system.

The DC-10 electrical system is normally powered by three engine-driven generators. Portions of the system may be powered by a battery and an air-driven generator. The electrical generating system is a.c. with necessary d.c. power provided by transformer rectifier units or a battery. The generators will function either paralleled, unparalleled, or isolated, and each generator can supply enough power to operate all essential electrical systems.

A battery and static inverter combination can provide about 30 min of emergency a.c. and d.c. bus power for the captain's flight instruments, essential communication, and navigation equipment when normal sources are inoperative. The battery and static inverter operations can be obtained by rotating the emergency power switch on the pilot's overhead panel to the "on" position.

Three independent a.c. channels provide power to associated generator buses, which feed associated main a.c. buses. The channels are paralleled through the a.c. tie buses which permits assumption of electrical loads by any functioning generator or generators. The a.c. system is operated normally in parallel with the bus tie relays closed. Two emergency a.c. buses are powered by a.c. buses 1 and 3.

The four transformer rectifier units, which are powered from designated a.c. buses, are the primary sources of d.c. power. Except for transformer rectifier No. 2B, which is powered from the a.c. ground service bus during ground operation, the other three transformer rectifier systems are similar to their counterpart a.c. systems. However, the d.c. buses are electrically isolated during normal operation.

Protective circuitry automatically isolates faulted buses or components from the other parts of the system. If the protective circuitry senses a generator fault, such as an under voltage condition, the generator relay will open and isolate the generator from its bus; the rest of the system will be powered by the remaining generators. However, if a bus fault is sensed, such as a differential current, the bus tie relay will open and isolate the generator and its associated a.c. buses from the a.c. tie bus. If this occurs, the protective circuitry will also engage a lockout mechanism to protect the remaining buses from damage. The lockout mechanism can be released and power restored to the bus, provided the fault has been cleared by appropriate actions by the flight engineer on his electrical and generator reset panel located at the top of the upper main circuit breaker panel. When he is positioned for takeoff, the flight engineer cannot reach this panel. He must reposition his seat to face his panel, release his safety belt, and get out of his seat to reach the switches. Company procedures only authorize one attempt to restore power. This procedure is not classified as an emergency procedure; it is an "abnormal procedure." The procedure does not contain any immediate action items which must be done without a checklist.

The loss of the No. 1 engine and its associated generator causes a loss of many aircraft systems and instruments. Among these are: The captain's flight instruments, the left stall warning computer, the stickshaker motor, No. 1 engine's instruments, the slat disagree warning light system, portions of the flight control indicating system, portions of the DFDR sensors, and the CVR. In addition to these losses, the flightcrew would be presented with numerous warning lights. The caution and master warning lights on the glareshield would be illuminated. Hydraulic and electrical malfunction lights would be illuminated on the annunciator panel and on the flight engineer's panel. Power to the left a.c. and d.c. emergency buses could have been restored by rotation of the emergency power switch to the "on" position. This action would have powered the left a.c. and d.c. emergency buses and restored the operation of the captain's instruments as well as some of the engine instruments.

1.17.8 Flightcrew Procedures

American Airline's Operating Manual contains the recommended procedures for operating the DC-10 aircraft and its personnel are required to comply with the procedures set forth therein. Since the failure of the pylon and engine did not occur until after V_1, only those company procedures relating to continued flight were examined. These procedures are contained in the Emergency Procedures Section of the Operating Manual.

The Emergency Procedures Section is prefaced with the following guideline:

The procedures on the Emergency Checklist are those where immediate and precise action on the part of the crew will substantially reduce the possibility of personal injury or loss of life.

The emergency procedures in this section are presented as the best way to handle these specific situations. They represent the safest, most practical manner of coping with emergencies, based on the judgment of the most experienced Pilots and F/E's, the FAA approved procedures, and the best available information. If an emergency arises for which these procedures are not adequate or do not apply, the crew's best judgment should prevail.

The manual also provides guidelines as to how the flightcrew will use the emergency checklist. The manual states, in part:

The checklist is a tool provided to minimize usually hasty and perhaps improper action. Though all checklist procedures are not required to be committed to memory it is expected that all crewmen understand fully each and every procedure.

The nature and seriousness of any given emergency cannot always be immediately and accurately determined. As a professional you will always fly the aircraft and/or immediately correct the obvious prior to any specific reference to the cockpit checklist. Some of the items which fall into the category of attending to the obvious are donning of O_2 masks and goggles, establishing interphone communications, resetting the fire aural warning, etc.

The emergency procedure for a takeoff engine failure, flaps 15° or less or 22°, states, in part:

This procedure assumes indication of engine failure where the takeoff is continued. Each takeoff should be planned for the possibility of an engine failure. Normal takeoff procedures ensure the ability to handle an engine failure successfully at any point.

If an engine failure occurs when making a Standard Thrust takeoff, Standard Thrust on the remaining engines will produce the required takeoff performance. If deemed necessary, the remaining engines may be advanced to Maximum Take-Off Thrust.

Speed . . . . CLIMB OUT AT V2 UNTIL REACHING 800 FEET AGL OR OBSTACLE CLEARANCE ALTITUDE, WHICHEVER IS HIGHER THEN LOWER NOSE AND ACCELERATE

The Operating Manual's discussion of the procedure contained an annotated profile drawing of the takeoff.(See figure 14.) The annotations accompanying the profile sketch state (after the aircraft is airborne), "Continue rotation to V_2 (Deck angle 12°-20°)." Over the next picture of the aircraft is the note, "Positive rate-Gear up." The next picture shows the aircraft level at 800 ft AGL and contains the accelerate instructions noted above.

Take-Off Engine Failure Flaps 15° Or Less Or 22°

This procedure assumes indication of engine failure where the take-off is continued. Each take-off should be planned for the possibility of an engine failure. Normal take-off procedures ensure the ability to handle an engine failure successfully at any point.

If an engine failure occurs when making a Standard Thrust take-off, Standard Thrust on the remaining engines will produce the required take-off performance. If deemed necessary, the remaining engines may be advance to Maximum Take-Off Thrust.

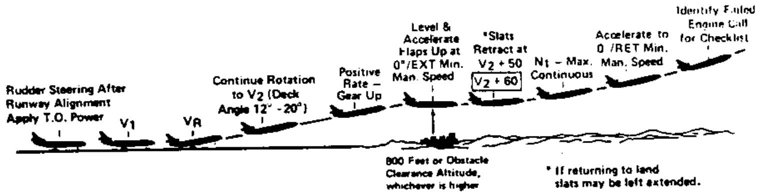

Figure 14. Diagram of AAL emergency procedure.

On July 23, 1979, American Airlines issued Operations Bulletin No. DC-10-73 which amended the procedure. The bulletin states, in part:

The following climb speeds will be utilized to obstacle clearance altitude when an engine failure occurs after V1 on takeoff:

- If engine failure occurs after V1 but not above V2, maintain V2 to obstacle clearance altitude.
- If engine failure occurs after V2, maintain speed attained at time of failure but not above V2 + 10 to obstacle clearance altitude.
- If engine failure occurs at a speed higher than V2 + 10, reduce speed to and maintain V2 + 10 to obstacle clearance altitude.
- If the FD Take-Off mode is engaged at the time of engine failure the Pitch Command Bar (and the Fast/Slow Indicator) will command V_2. Therefore, if the failure occurs above V_2, disregard these indications and fly the speed called for in the above procedure.

1.17.10 Suspension and Restoration of the DC-10 Type Certificate

On June 6, 1979, after a series of postaccident inspections disclosed damaged aft bulkheads in the wing to engine pylons, the Administrator of the FAA issued an Emergency Order of Suspension. The Order suspended the DC-10 series aircraft type certificate "until such time as it can be ascertained that the DC-10 aircraft meets the certification criteria of Part 25 of the FAR and is eligible for a Type Certificate."

On June 26, 1979, the FAA issued Special Federal Aviation Regulation 40 which prohibited the "operation of any Model DC-10 aircraft within the airspace of the United States."

On July 13, 1979, after a series of formal investigations, the Administrator found that the DC-10 met the requirements for issuance of a type certificate. Accordingly, the Emergency Order of Suspension was terminated. (See appendix G.)

2. Analysis

The facts developed during the investigation disclosed that the initial event in the accident sequence was the structural separation of the No. 1 engine and pylon assembly from the aircraft's left wing. Witness accounts, flight data recorder parameters, and the distribution of the major structural elements of the aircraft following the accident provided indisputable evidence that the engine and pylon assembly separated either at or immediately after rotation and about the same time the aircraft became airborne. At that time, the flightcrew was committed to take off, and their decision not to attempt to discontinue takeoff was in accordance with prescribed procedures and was logical and proper in light of information available to them.

The investigation and analysis were concentrated primarily in two major areas. First, the investigation sought to identify the structural failure which led to the engine-pylon separation and to determine its cause; second, the investigation attempted to determine the effects the structural failure had on the aircraft's performance and essential systems, and the operational difficulties which led to the loss of control. In addition, the investigation went beyond these primary areas and probed such areas as the vulnerability of the DC-10's design to maintenance damage, the adequacy of the DC-10's systems to cope with unique emergencies, the quality control exercised during DC-10 manufacturing and aircraft assembly, the adequacy of operator maintenance practices, the adequacy of industry communications of service and maintenance difficulties, the extent of FAA's surveillance of overall industry practices, and the adequacy of an accepted operational procedure.