Uncontained engine failure, Air Florida Airlines Inc., McDonnell-Douglas DC-10-30CF, N101TV, Miami International Airport, Miami, Florida, September 22, 1981

Micro-summary: This McDonnell-Douglas DC-10-30CF experienced an uncontained engine failure of the #3 engine on takeoff.

Event Date: 1981-09-22 at 1648 EDT

Investigative Body: National Transportation Safety Board (NTSB), USA

Investigative Body's Web Site: http://www.ntsb.gov/

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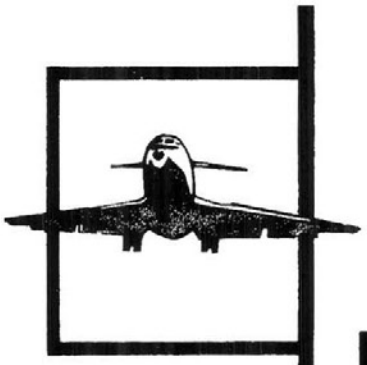

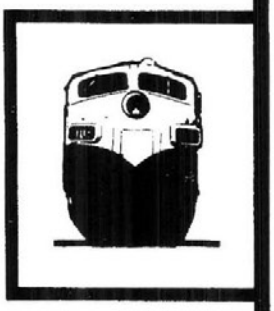

NATIONAL TRANSPORTATION SAFETY BOARD

WASHINGTON, D.C. 20594

AIRCRAFT ACCIDENT REPORT

AIR FLORIDA AIRLINES, INC.
McDONNELL-DOUGLAS, INC.,
DC-10-30CF, N101TV
MIAMI INTERNATIONAL AIRPORT
MIAMI, FLORIDA
SEPTEMBER 22, 1981

NTSB-AAR-82-3

UNITED STATES GOVERNMENT

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AIRCRAFT ACCIDENT REPORT

Adopted: April 6, 1982

AIR FLORIDA AIRLINES INC.
MCDONNELL-DOUGLAS DC-10-30CF, N101TV
MIAMI INTERNATIONAL AIRPORT
MIAMI, FLORIDA
SEPTEMBER 22, 1981

SYNOPSIS

About 1648 e.d.t., September 22, 1981, Air Florida Airlines, Flight 2198, a McDonnell-Douglas, Inc., DC-10-30CF sustained an uncontained failure of its right underwing engine (No. 3) during the takeoff roll at Miami International Airport, Miami, Florida. The engine failure occurred at about 90 knots indicated airspeed; the pilot rejected the takeoff and stopped the aircraft safely.

The aircraft was damaged by the release of high energy engine debris. The resultant damage caused an uncommanded retraction of the right wing outboard leading edge slat. Components of the No. 3 engine control system and fire protection system, the electrical system, and the Nos. 1 and 3 hydraulic systems were also damaged by engine debris.

The National Transportation Safety Board determines that the probable cause of this accident was the failure of quality control inspections to detect the presence of foreign material in the low pressure turbine cavity during the reassembly of the low pressure turbine module after installation of the stage 1 low pressure turbine rotor disk. The foreign material in the low pressure turbine cavity damaged the bolts holding the stage 1 low pressure turbine rotor disk and stage 2 low pressure turbine rotor disk together. The bolts failed at high engine thrust and the stage 1 low pressure turbine disk separated from the low pressure turbine rotor assembly, oversped, and burst.

1. FACTUAL INFORMATION

1.1 History of the Flight

Air Florida Airlines, Inc., Flight 2198 was a regularly scheduled passenger flight from Miami, Florida, to Newark, New Jersey, with an en route stop at Fort Lauderdale, Florida. Flight 2198 departed the terminal gate at Miami International Airport at 1630 e.d.t., 1/ on September 22, 1981, and taxied to runway 9L for takeoff. There were 56 passengers and 15 crewmembers on board.

The computed gross weight for takeoff was 354,921 pounds, the center of gravity was 20.6 percent, M.A.C., and the takeoff flap setting was 19.5 degrees. A reduced thrust setting was to be used for takeoff, and the computed takeoff speeds were: critical engine failure speed (V1) -- 133 KIAS; rotation speed (VR) -- 133 KIAS; and takeoff safety speed (V2) -- 149 KIAS.

1/ All times herein are eastern daylight time based on the 24-hour clock.

According to the flightcrew, all instrument indicators were normal during the start engines and taxi checklists and while taxiing to runway 9L for takeoff. The flight was cleared onto the runway "and hold." While the aircraft was taxiing onto the runway, the flightcrew completed the before takeoff checklist. At 1647:16, Flight 2198 was cleared for takeoff, and the captain, who was flying the aircraft, began the takeoff roll. The airspeed indicators were cross checked at 80 KIAS and no discrepancies were noted.

About 3 seconds after the 80 KIAS check, the flightcrew heard a noise "like rushing air" followed by a "hollow boom." Almost simultaneous with the second sound, the aircraft yawed right and began to vibrate. The captain corrected the yaw and initiated rejected takeoff procedures. The thrust levers were retarded, the wheel brakes were applied, and the spoilers were deployed. Reverse thrust was not selected since it could have adversely affected directional control. At the same time, the first officer informed the tower of their actions and requested that the emergency equipment be dispatched.

At about 60 KIAS, as the aircraft decelerated, the flightcrew noted that the No. 3 engine's N1 indicator read zero, that its N2 indicator read 107.5 percent, that its exhaust gas temperature (EGT) gauge read 657° C, and that the engine failure light had not illuminated. The captain ordered the first officer to shut down the engine, and the first officer did so by placing the fuel lever to the off position.

While the aircraft was decelerating, the tower informed the flightcrew that there appeared to be smoke or vapor coming from the No. 3 engine. The captain directed the flight engineer to pull the fire handle and discharge the extinguishing agent into the engine.

The aircraft exited the runway at taxiway M-9 and was stopped on taxiway M at its intersection with taxiway M-9, about 7,000 feet east of runway 9L's west threshold (25° 47' N latitude, 80° W longitude). After the tower and emergency equipment personnel visually confirmed that there was no fire, the auxiliary power unit (APU) was started, the Nos. 1 and 2 engines were stopped, portable air stairs were positioned at the forward left cabin door (L-1), and the passengers were deplaned. After the passengers and flight attendants deplaned, the flightcrew completed the park aircraft checklist, shut down the APU, secured the aircraft, and then deplaned.

According to the flightcrew, "At no time during this incident, did we receive any engine warning indications other than the N1 gauge on the No. 3 engine." In addition, the flightcrew could not recall seeing the wing slat disagreement warning light during the rejected takeoff; however, they did see that it was illuminated during the secure aircraft checklist when the flap/slat handle was placed in the "up" position. The flight engineer also said that all bus-tie-relays (BTR) remained closed and that the three generator alternating current (AC) buses and the transformer rectifier (TR) buses remained powered.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others	Total
Fatal	0	0	0	0
Serious	0	0	0	0
Minor/None	15	56	0	71

1.3 Damage to Aircraft

The aircraft sustained substantial damage.

1.4 Other Damage

None.

1.5 Personnel Information

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

1.6 Aircraft Information

McDonnell-Douglas, Inc., DC-10-30CF, N101TV, was leased from Trans International Airlines by Air Florida Airlines, Inc., on March 16, 1981, and had been operated continuously by Air Florida Airlines Inc. since that date. The aircraft had been maintained in accordance with prescribed regulations. The aircraft's maintenance records showed that its stall warning system had been modified in accordance with Airworthiness Directive (AD) 80-03-10. (See section 1.17.2.)

The aircraft's gross weight and center of gravity for takeoff were within prescribed limitations. (See appendix C.)

The aircraft was powered by three General Electric CF6-50C2 high bypass turbo fan engines which are rated in the 50,000-lb thrust category.

The review of the maintenance records for the No. 3 engine, serial No. 455-123, disclosed that all inspections and checks were accomplished in accordance wtih applicable inspection procedures and programs. The review also disclosed that all required airworthiness directives had been complied with and that all of the engine's life limited parts were within required limits.

On October 12, 1979, the stage 1 low pressure turbine rotor disk, S/N MPOA 1552, was installed in engine No. 455-123 and the engine was converted to a CF6-50E2 rating. The maintenance was accomplished by United Airlines at their San Francisco, California, maintenance facility. The United Airlines Job Instruction Cards showed that the required sign offs by supervisory personnel had been accomplished in accordance with prescribed procedures. The low pressure turbine module was closed and it was not reopened again. At that time, the turbine rotor disk had accumulated 13,391 hours and 3,427 cycles. The engine was given a test cell run and the data from the test cell run showed that the engine was within the balance limits of 3.0 mils used by United Airlines at that time. The engine was installed on a Boeing 747, and was removed 118 hours later, on July 29, 1980, for reported excessive vibration. The as received vibration level was 2.6 mils at a maximum continuous thrust and 2.3 mils at takeoff thrust. The engine was reworked, placed in a test cell, and was trim balanced to 1.2 mils and 1.3 mils at maximum continuous and takeoff thrust, respectively, and returned to service on September 11, 1980.

In December 1980, engine No. 455-123 was removed from service and converted to a CF6-50C2 rating. On December 18, 1980, it was installed on N101TV in the No. 3 position. On September 22, 1981, before the accident, the stage 1 low pressure

turbine rotor disk had accumulated 16,790 hours and 4,015 cycles. (See appendix C.) At the time of the accident, the service life limit of the stage 1 low pressure turbine rotor disk was 7,300 cycles. On December 12, 1981, as a result of service history analysis, the Federal Aviation Administration (FAA) approved an increase to 12,350 cycles.

The aircraft's log sheets disclosed that there were four pilot reports of engine vibration during the month of September 1981, — on the 4th, 14th, 17th, and 21st. Only two of these reports--the one on the 14th and the one on the 17th --involved the No. 3 engine, and the logs indicated that the following corrective actions were taken:

> September 14; Visually inspected No. 3 engine and removed foreign object from between variable stator vanes (VSV) and VSV actuator arm. Checked engine for visible damage, none found. Checked pylon attach fitting visually and found normal.
>
> September 17; Found No. 3 engine reverser out of rig. Rerig as per S.B. (Service Bulletin) 78-60.

The aircraft engines were not equipped with airborne vibration monitoring systems nor are they required by regulations.

1.7 Meterological Information

The pertinent surface weather observations at the Miami International Airport at 1651 were as follows:

> 3,000 feet scattered (clouds), estimated 12,000 feet broken, 30,000 feet broken; visibility--7 mi; temperature--82° F; dew point--69° F; winds--030° at 8 kns; altimeter setting--29.90 inHg. Cumulonimbus (clouds) northwest through northeast, moving northeast, rain ended 11 minutes after the hour.

1.8 Aids to Navigation

Not applicable.

1.9 Communications

There were no known communications malfunctions.

1.10 Aerodrome Information

Miami International Airport, elevation 10 feet mean sea level (m.s.l.), is located 9 miles northwest of downtown Miami, Florida, and is served by three runways. Runway 9L is 10,500 feet long, 200 feet wide, and has an asphalt surface.

1.11 Flight Recorders

The aircraft was equipped with a Sundstrand V557 Cockpit Voice Recorder (CVR), Serial Number 1994, and a Sundstrand 573, Serial Number 3206, Digital Flight Data Recorder (DFDR). The CVR tape was brought to the Safety Board's laboratory for readout. The tape was distorted, had poor speed control, and oscillated about 25 percent on the slow side. The tape was unreadable and no transcript was made.

The DFDR recording was read out at the Safety Board's laboratory in Washington, D.C., and the following performance parameters were plotted for the attempted takeoff: the low compressor speed (N1 rpm) of each engine (the DFDR samples each engine's N1 rpm once every 4 seconds); aircraft heading; KIAS; and the position of the right wing's outboard leading edge slat. The last three parameters are sampled once every second. (See appendix G.)

The DFDR is powered by the No. 3 generator's AC bus. The DFDR readout showed that the recorder's synchronization was interrupted twice during the attempted takeoff and data were not recorded during these interruptions. The length of the interruptions could only be estimated.

At 01:04 (DFDR Elapsed Time), the N1 rpm on all three engines began to increase and the takeoff (TO) was started. (All times hereafter are stated in seconds after the start of takeoff.) The first recorder interruption began at TO+19 seconds and was estimated to have ended at TO+21 seconds. The second interruption began at TO+25 seconds and was estimated to have ended at TO+30 seconds.

According to the DFDR, at TO+14 seconds, all three engines had stabilized between 106 and 109 percent N1 rpm. At TO+22 seconds, after the end of the first recorder interruption, the N1 rpm of the No. 3 engine was 1.19 percent and the N1 rpm's of engines Nos. 1 and 2 were decreasing.

At TO+24 seconds, the airspeed was 104 KIAS — the maximum recorded value. At TO+30 seconds, after the end of the second recorder interruption, the airpseed was about 96 KIAS, and thereafter, it began decreasing at a rate of about 2.25 KIAS/seconds.

At TO+23 seconds, the recorder data showed that the right wing's outboard leading edge slat was "in transit," and this indication remained constant throughout the remainder of the readout.

1.12 Aircraft Damage

The No. 3 engine's left core cowling, the aft two-thirds of the right core cowling, and the turbine rear frame and reverser assembly separated from the engine and fell to the runway. (See figure 1.)

The low pressure turbine (LPT) casing, the stage 1 low pressure turbine rotor and the exhaust cone also separated from the engine. The exhaust cone was found on the runway. The low pressure turbine rotor had fragmented, and rotor fragments were found lodged in the right wing leading edge slat area, the left wing trailing edge flap area, and on the airport surface south of the centerline of runway 9L. One piece of the turbine rotor was found about 2,500 feet south of the runway centerline. The aircraft sustained about 30 penetrating strikes from high energy engine debris. Virtually all of these strikes were inboard of the No. 3 engine and aft of the plane of rotation of the failed stage 1 low pressure turbine rotor. (See figure 2.)

The forward flange of the right wing's front spar's lower cap was cracked, twisted, and bent upwards along a 24-inch span centered on right wing station (WS)-XORS 231. The rest of the damage incurred from the engine debris was for the most part limited to dents and punctures in the wing and fuselage. The empennage was not damaged.

The left and right sides of the pylon outer skin at zone 438--above the engine tail cone--had numerous scratches, pits, and gouges. The forward and aft sections of

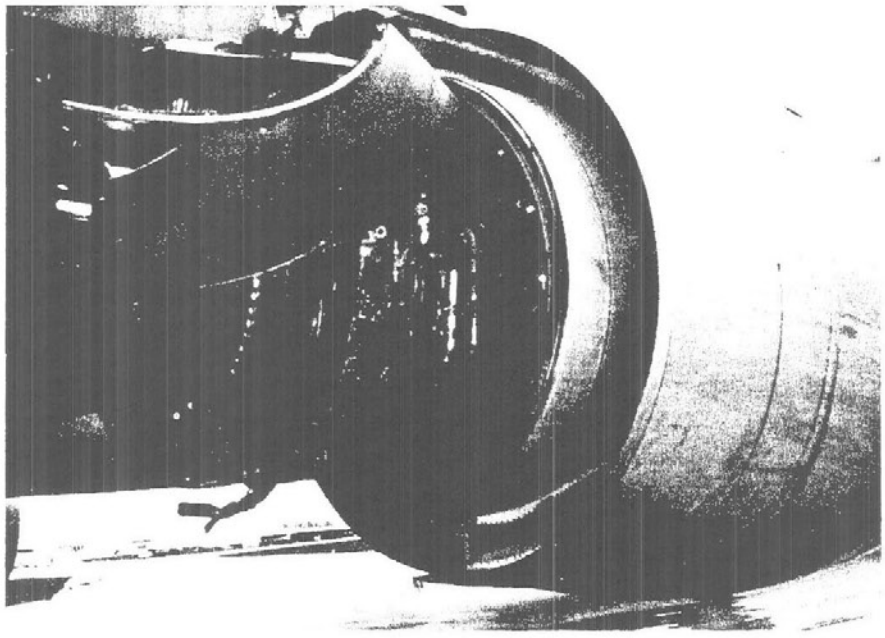

Figure 1.--Damaged No. 3 engine.

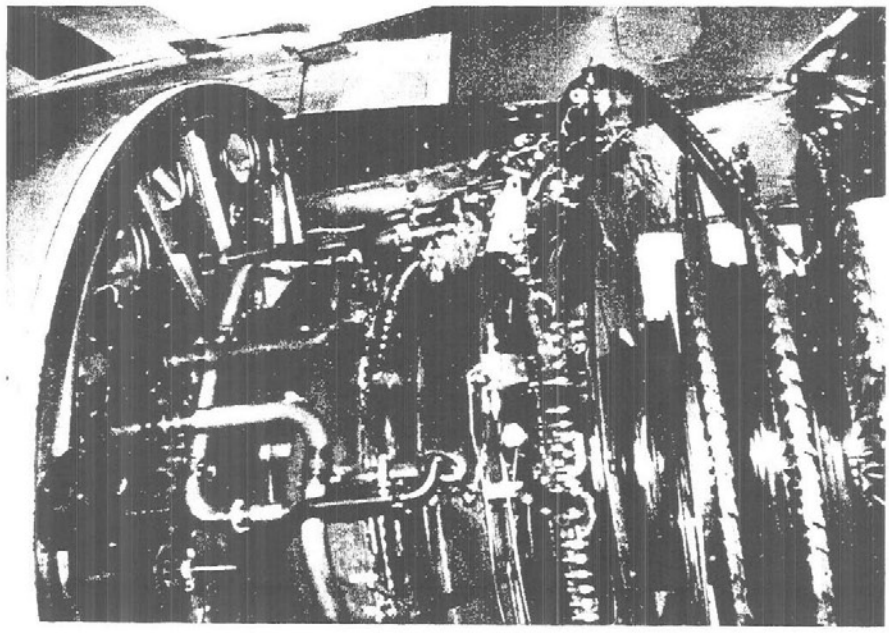

Figure 2.--Low pressure turbine section and missing stage 1 low pressure turbine disk.

blanket installation below the horizontal firewall and aft of the rear engine mount attach point were torn away. Portions of the pylon just above the engine cowling and in the area of the hinge assemblies of the engine's core cowl doors' were either damaged by impact or torn away.

Six of the left and right main landing gear wheel and tire assemblies were damaged by engine debris; however, of these six wheel and tire assemblies, only the tire of the right main landing gear forward outboard wheel (No. 4) had blown out.

Examination of the air frame revealed that components of the aircraft's electrical system, hydraulic systems, and wing leading edge flight control systems had been damaged by penetration of the engine debris.

The Nos. 1 and 2 electrical systems were not damaged. Two wire bundles at right wing station (WS)-XORS 230 were cut about half way through. The fuel-quantity wiring to the indicating system on the refueling panel, located at right WS-XORS 522, was severed inboard of the panel at right WS-XORS 230. Terminal strip S3-457 at right WS-XORS 240 was damaged by impact. One "A" phase aluminum generator feeder cable from the No. 3 generator was cut through about two-thirds of its diameter. There are two cables for each of the generator's three phases; only the outer insulation on the other five feeder cables were damaged. However, the No. 3 bus tie relay remained closed and the No. 3 generator AC bus remained powered.

The slat retract line of the No. 1 hydraulic system and the auto/prime bleed valve line of the No. 3 hydraulic system were both severed at right WS-XORS 230. The hydraulic fluid in each system's main reservoirs was lost, and both systems and their associated systems were inoperable. The No. 2 hydraulic system was not damaged.

The No. 3 engine's fuel feed system and associated plumbing was not damaged. The firewall fuel shutoff valve drive and fuel shutoff valve were in the open position. The firewall fuel shutoff valve cables and the No. 3 engine's thrust lever control cable were severed, and the discharge line for the fire extinguishing agent was separated between the bottles and the No. 3 engine. The thrust lever control cable inboard of the No. 3 engine pylon--at right WS-XORS 230--was severed. Based on this damage, the firewall fuel shutoff capability was lost and the fire extinguishing agent could not be discharged. The No. 3 engine was shut down by moving its fuel lever to cut off.

Examination of the leading edge slats after the aircraft had been secured revealed the following: the left outboard slats were extended to the 30-degree position; the left and right inboard slats were extended to the 19-degree position; and the right outboard slats were partially retracted to about the 5-degree position.

Examination of the flight control mechanism in the center fuselage disclosed that the right hand outboard slat actuator position follow-up cables were slack and had been severed at right WS-XORS 513.9. The outboard slat follow-up cable drum input to the summing mechanism for the control valve was bottomed against the "extend" over-travel stop. With the exception of the broken follow-up cables to the slat control valve, none of the slat actuating mechanisms or controls had been damaged.

1.12.1 Associated Aircraft Damage

As a result of the severed hydraulic lines and severed wires within the damaged wiring bundles, several associated aircraft systems were either rendered inoperable or their effectiveness was impaired.

The loss of the Nos. 1 and 3 hydraulic systems affected the functions of several aircraft systems. The landing gear could not be retracted; however, the wheel brakes continued to function with residual fluid in the pressurized hydraulic accumulator. After the aircraft had been stopped and secured, the brake accumulator indicators of the Nos. 1 and 3 hydraulic systems were still within their green bands.

Although the leading edge slat system was inoperable, the trailing edge flaps were operational.

Four of the 10 spoiler panels --2 on each wing-- remained operable; 3 of the 4 ailerons and 3 of the 4 elevators remained operable and both the upper and lower rudder were operational. The rudder and horizontal stabilizer have an additional standby power system. When this system senses a fluid loss in the No. 1 or No. 2 hydraulic system, it closes the return lines to the affected main hydraulic system and seals hydraulic fluid within the rudder and stabilizer activating systems. The system incorporates two non-reversible motor-pump assemblies and, in this instance, the No. 2 hydraulic system would have powered its pump assembly and provided pressure to the sealed off portion of the No. 1 system to operate the upper rudder and the horizontal stabilizer. In addition, during certification the manufacturer demonstrated that the aircraft could be flown safely with any one of its three hydraulic systems operable.

The Safety Board and McDonnell-Douglas identified the damaged wires in the wire bundles and the systems that would have been affected by the damage. The possible effects that damaged wires may have on an associated system is a function of the system configuration when the wires were damaged and the type of damage to the wires -- severed or short circuited. Based on both the open and short circuit cases, the systems affected by the damaged wires were the fire detectors on the No. 3 engine and detector loop B; the fuel valves and fuel indicators and lights in the Nos. 1, 2, and 3 fuel tanks; the right wing and No. 3 engine anti-ice systems; the navigation lights; the pneumatic system temperature sensors, indicators, and detection lights; the No. 3 engine failure light and its engine pressure ratio (EPR) gauges, tachometers, oil pressure indicators, and starter valve functions; and the No. 3 generator functions.

Although four wires of the No. 3 generator current transformer wires were severed, the DFDR which is powered from the No. 3 generator AC bus, continued to run, indicating that the bus remained powered through the bus tie relay.

1.13 Medical and Pathological Information

There was no evidence of preexisting physiological problems which could have affected the flightcrew's performance.

1.14 Fire

There was no fire.

1.15 Survival Aspects

The incident was survivable. The escape slides were not used. The passengers deplaned through the aircraft's L-1 door via portable airstairs.

1.16 Tests and Research

1.16.1 Engine Teardown and Metallurgical Examination

Engine No. 455-123 was shipped to the General Electric Company, Evendale, Ohio, facility where it was disassembled. The engine components were examined visually and subjected to metallurgical examination and analysis. These examinations were conducted under the supervision of Safety Board personnel.

The examination showed that the major damage to the engine was incurred aft of the high pressure turbine section. The low pressure turbine stator assembly case had separated circumferentially along the flange attaching its forward end to the turbine mid-frame housing. This separation, located at engine station (ES) 294, was just forward of the plane of rotation of the stage 1 low pressure turbine rotor. The stator case separated fore and aft along the top of the engine; it tore horizontally above its outboard split flange and the forward one-third of the inboard split flange had separated. The stator assembly case had split into three large segments and numerous smaller segments which separated from the engine and fell to the runway. (See figure 3.)

Examination of the low pressure turbine rotor showed that the stage 1 low pressure turbine rotor had separated from the stage 2 low pressure turbine rotor and was missing. The stage 1 and 2 low pressure turbine rotors are attached to each other by 60 bolts. These bolts attached the aft spacer arm of the stage 1 low pressure turbine rotor disk to the rotating air seal and forward spacer arm of the stage 2 low pressure turbine rotor disk. The rotating air seal is a circular metal seal with 60 bolt tabs and fits between the forward and aft spacer arms of the two disks when the bolts are in place. All the bolt tabs on the rotating air seal of the stage 2 low pressure turbine disk through which the stage 1 to 2 connect bolts are inserted and fastened were bent forward. Five bolt hole tabs were broken off through the bolt holes and the flange was cracked in four places. The forward face of the seal's flange exhibited severe heat discoloration and there was a circumferential band of metal smears outside of the bolt holes. These smears varied in width from 0.189 to 0.375 inch. Two bolt holes had heavy side fretting on their forward face. The right hand side of the bolt tabs--viewed from the aft of the bolt tabs looking forward--had varying amounts of impact damage, some with smooth impressions.

Thirty-four pieces of the stage 1 to stage 2 low pressure turbine rotor nuts and bolts were recovered. Nearly all these pieces exhibited tensile fracture surfaces and were gouged and damaged heavily by impact. Many of the gouges had a rusty brown color. Only two of the stage 1 to stage 2 connecting bolts remained in the stage 2 low pressure turbine rotor disk's forward spacer arm attach flange. A portion of the thread was visible on both bolts. The bolt head surfaces had impact damage and had a smooth smeared appearance on one-half of the impact surface. One bolt had a deep concave impact gouge and the gouge had a rusty brown color. When the stage 2 turbine rotor disk was removed from the forward shaft, stage 1 to stage 2 nuts and bolts were found lodged in the assembly. Except for 1 bolt which had a small area of fatigue, the other 33 recovered bolts showed varying combinations of tensile and shear type failures.

Examination of the stage 2 low pressure turbine rotor disk disclosed that its forward spacer arm had bulged outward and that there were numerous indentations around the circumference of its inner diameter surface. The forward spacer arm's bottom hole tabs were dented and some were missing the cross-section material.

All the stage 2 blades were present and had sustained severe impact damage. Three shrouds were missing. The rotor disk's aft spacer arm's inner diameter had impact damage; however, these marks were not shiny and some corresponded to protrusions on the spacer arm's outer diameter.

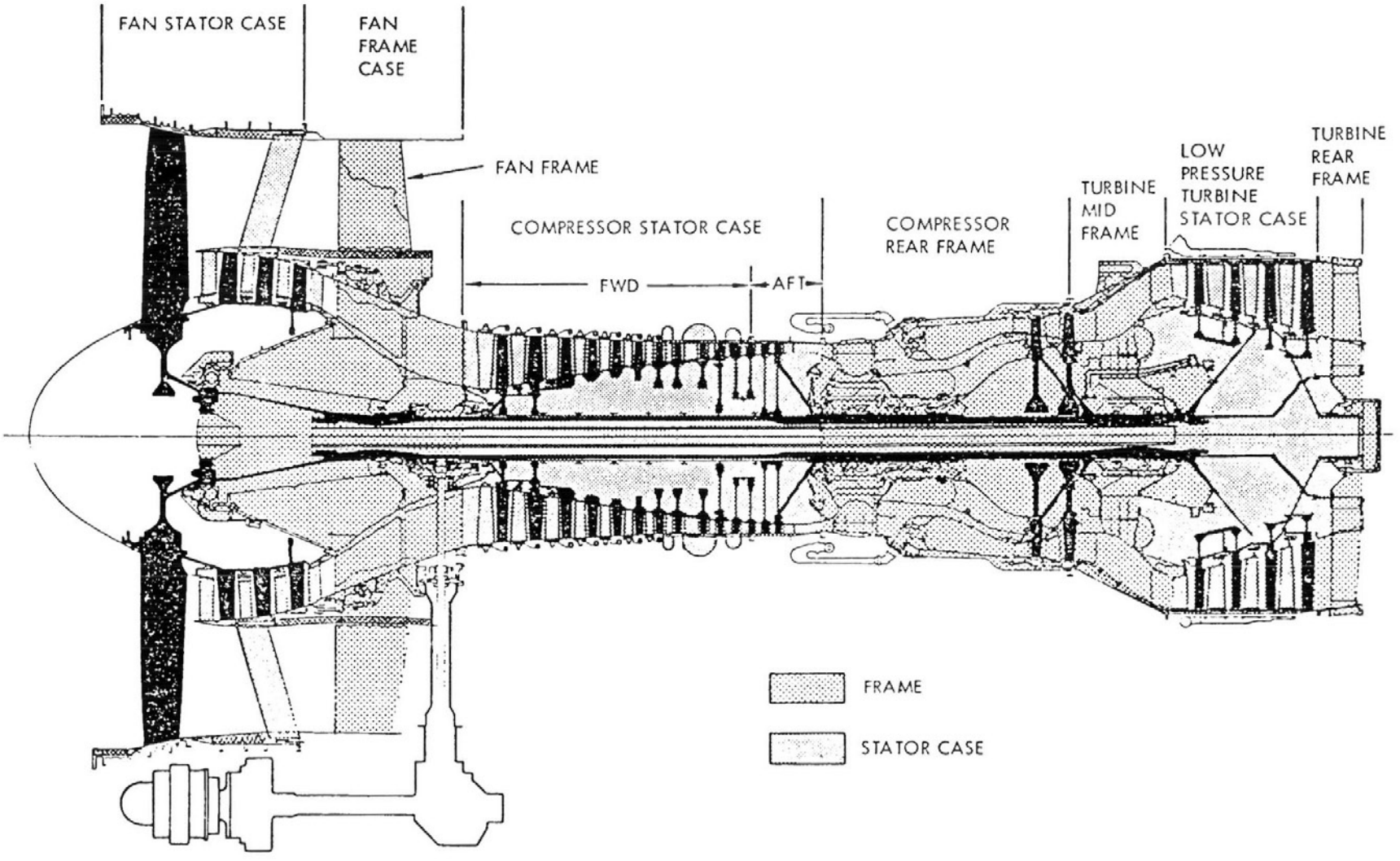

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Figure 3.--CF6-50/-45 frames and stator assemblies.

The missing stage 1 low pressure turbine rotor disk was fragmented into five major sections containing varying amounts of bore and rim sections. Additional pieces of disk rim and web were recovered. Each of the recovered pieces was numbered 1 through 22 for identification and replication purposes. As of this date, two disk fragments consisting of 5 inches of inside bore diameter and five dovetail posts have not been recovered. All the fracture surfaces on the turbine rotor disk were consistent with overstress separations. No evidence of fatigue or other type of preexisting cracking was found.

Measurements were made of the disk's bore thickness on two of the disk pieces; in all, 13 measurements of thickness were made. The measurements ranged from 1.058 inches in thickness to 1.115 inches in thickness; the manufactured thickness was 1.128 inches. The surface of one of these disk pieces exhibited signs of localized heating and was blue and gold in appearance.

Examination of the reconstructed stage 1 low pressure turbine rotor disclosed numerous impact gouges on the various surfaces of its disk. Many of these gouges contained a rust colored material similar to that found on the rotor bolts. This material was removed from the gouges in the bolts and disks and submitted to Energy Dispersive Analysis via X-rays (EDAX) for analysis. The resulting x-ray spectrum showed a high iron content thus indicating that the rust colored material was iron oxide.

The surfaces of the web and bore of the failed disk contained circumferential bands of smeared material. EDAX analysis of the material indicated that the smears were caused by low alloy steel having the characteristics of M-50 alloy steel, similar to that removed from the gouges.

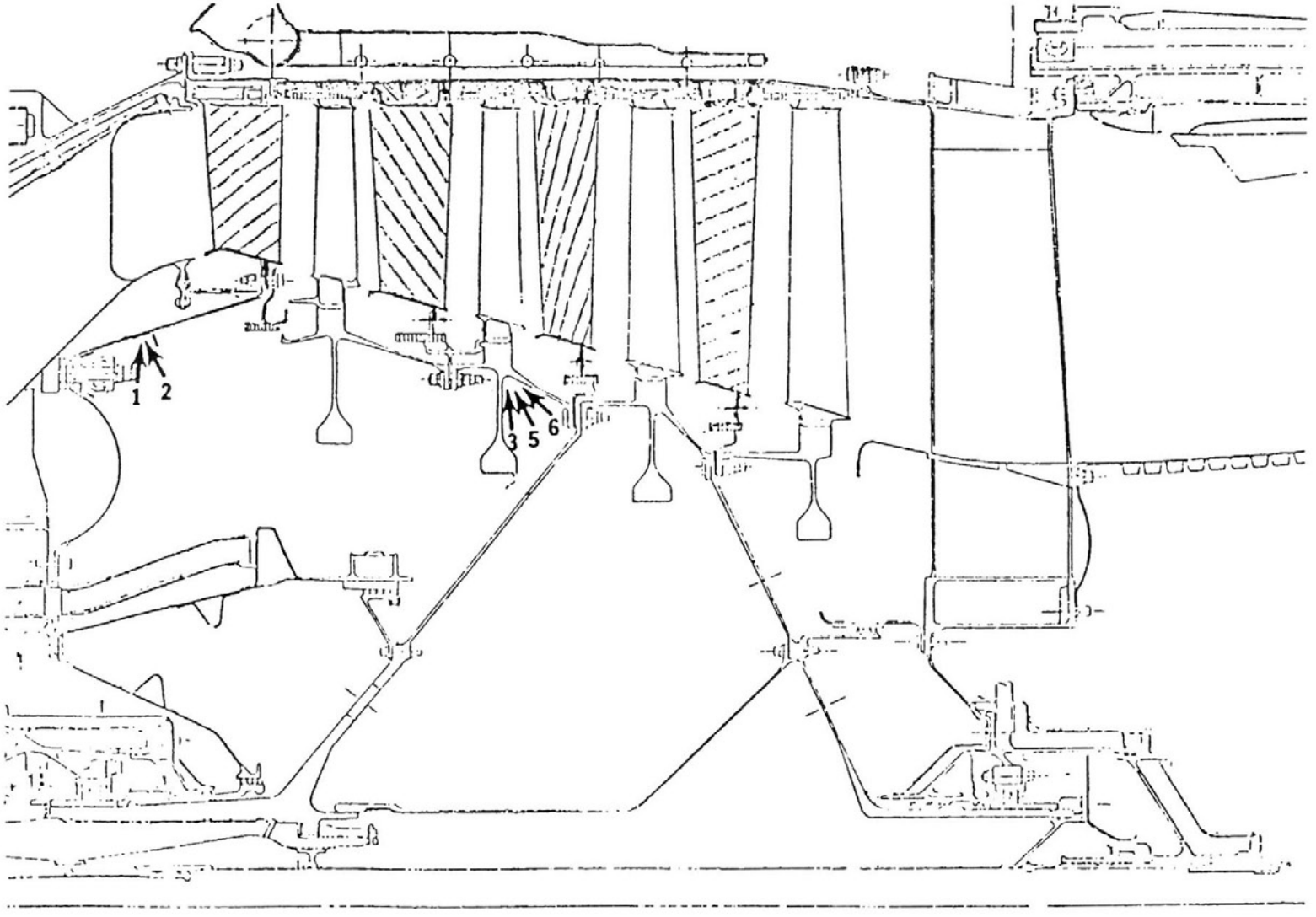
M-50 alloy steel material is used in machine tools and was also used in the seven CF6 main engine bearings. Six of these main engine bearings are located forward of the low pressure turbine section and they were not damaged.

The No. 7 main engine bearing, located aft of the low pressure turbine section, was impact damaged. The damage was substantial and the engine bearing's outer race was broken. This was the only main engine bearing which exhibited a broken bearing race.

Six pieces of magnetic material were found on the runway and submitted for identification. These pieces were clean, were not oxidized, and were not battered. There was no evidence of fatigue propogation on the fracture surfaces, and all fractures were tensile in mode. EDAX analysis identified the material as M-50 alloy steel. The appearance of the pieces indicated that they were fragments from the shoulders of the outer race of the No. 7 main engine bearing.

Six additional pieces of ferro magnetic material were found: one piece was found on the runway; three pieces were found in the low pressure turbine stage 2 rotor cavity; and two pieces were found in the low pressure turbine's stage 1 toroid cavity. EDAX analysis identified the material as M-50 alloy steel. All six pieces were severely battered and oxidized. Five of the six pieces were measured for thickness. One piece was too battered for thickness determination. The thickness of one piece ranged from 0.140 to 0.150 inches; one piece was 0.148 inches thick; and three pieces were 0.149 inches thick. (See figure 4.)

These six pieces were compared to a typical CF6-50 No. 5 roller bearing inner race which is manufactured from M-50 alloy steel. The specified thickness for this inner race is 0.146 inches to 0.148 inches. In addition, two of these six pieces had



Sketch of the low pressure turbine portion of CF6-50 engine S/N 455-123. After the low pressure turbine failure of this engine, six pieces of foreign material were recovered and submitted for examination. For identification purposes these six pieces were arbitrarily assigned numbers - #1 thru #6. Pieces #1 and #2 were found near the turbine mid frame stiffener (see arrows #1 and #2). Pieces #3, #5, and #6 (see arrows #3, #5, and #6) were found in the cavity formed by the Stage 2 low pressure turbine disk and the forward low pressure turbine shaft. Piece #4 was recovered from the runway near the location where the engine failure occurred.

Figure 4

retained evidence of beveling and the geometry appeared similar to that of a typical No. 5 engine bearing inner race.

1.17 Other Information

1.17.1 Engine Containment

The DC-10-30 was certified on November 21, 1972, in accordance with the applicable provisions of 14 CFR 25, effective February 1, 1965, as amended and certain Special Conditions.

The engines were certified in accordance with provisions of 14 CFR 33. The provisions relating to damage containment were contained in 14 CFR 33.19, which provided that "Engine design and construction must minimize the development of an unsafe condition of the engine between overhaul periods. The design of the compressor and turbine rotor cases must provide for the containment of damage from rotor blade failure." There was no regulatory requirement that provided that damage from a rotor or turbine disk failure be contained within the engine.

The special propulsion condition regarding engine installation was contained in Special Condition No. 25-18-WE-7(1), dated January 7, 1970.

> No. 25-18-WE-7(1) provided that: In lieu of the requirements of Section 25.903(d)(1), the airplane must incorporate design features to minimize hazardous damage to the aircraft in the event of an engine rotor failure or of a fire which burns through the engine case as a result of an internal engine failure.

To satisfy the requirements of the above Special Condition for the DC-10-30, McDonnell-Douglas provided data to the FAA in a letter dated November 1, 1972. 2/ On November 8, 1972, the FAA accepted the data as establishing compliance with the Special Condition. The data showed that the redundancy and separation of critical systems minimized the possibility that damage from the debris of an uncontained engine failure would jeopardize the safe operation of the aircraft, and that the probability of an uncontained engine failure was extremely remote.

After this accident, the FAA requested McDonnell-Douglas to conduct a study concerning the following: the probability of experiencing an uncontained disk failure; the probability of such a failure causing damage that would cause slat retraction; and the probability of such a combined failure occurring at a time which would cause a significant control problem with the aircraft. The study considered all CF6-6 and CF6-50 failures known to General Electric and McDonnell-Douglas that have resulted in segments of turbine engine rotors leaving the engine. There have been 10 occurrences of uncontained rotor failures in the history of the CF6 engines, and although these failures were not limited to the DC-10, all 10 occurrences were accounted for in the McDonnell-Douglas study. Of these 10 occurrences, only 4 were of the type which released sufficient energy to inflict significant secondary damage.

The study was conducted to produce the most conservative results; thus, since the angular fraction of the circle of engine rotation within which disk segments could cut the slat follow-up control cables includes the angle within which the segments would cut

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2/ McDonnell-Douglas Letter No. MDC C1-25-7362

both hydraulic extension lines, the larger angle was used. The trajectory angles include the angles required for segments from both the high and low pressure turbine rotors to strike the cables or both hydraulic lines. In all cases, the largest possible segment, one-half of a rotor disk, was used as the projectile, and this segment was always assumed to have rotated such that the largest dimension was normal to the trajectory. These assumptions produced the largest cutting swath. Finally, no credit was taken for pilot corrective action that could be assumed to have taken place for those failures that occurred near the end of the critical exposure period.

Based on the worst-case combination, the study concluded that the probability of an uncontained turbine rotor disk failure during the critical phase of the takeoff--between V1 and V2 + 10 KIAS--was 0.72×10^{-9}, or less than one in a billion.

The McDonnell-Douglas study was limited to the CF6 engines and the DC-10 aircraft series; the data did not include other high bypass engines and other wide-body aircraft. As of March 2, 1982, data compiled by the Safety Board concerning uncontained rotor disk failures on high bypass ratio engines disclosed the following statistics: the General Electric CF6 has had 10 failures; the Pratt and Whitney JT9D has had 9 failures; and the Rolls Royce RB-211 has had 4 failures. (See appendix F.)

The regulations and special conditions which were in effect when the DC-10 series aircraft were certified have been amended and amplified. In April 1970, Special Condition No. 25-18-WE-7(1) was incorporated into 14 CFR 25.903. In addition, on December 1, 1978, 14 CFR 25.571(e)(3) was also added to the certification regulations. This section provided for a damage tolerance assessment to ensure that the aircraft was "....capable of successfully completing a flight during which likely structural damage occurs a result of--(3) Uncontained engine failure,..."

Further guidance material for showing compliance with the regulations is contained in advisory circular (AC) 25.571-1; Damage Tolerance and Fatigue Evaluation of Structure. Paragraph 4(g)(2) of the AC states, in part, "...in the case of uncontained engine failures, the fragments and paths to be considered should be consistent with those showing compliance with 14 CFR 25.903(d)(1) of the FAR's (Federal Aviation Regulations), and with typical damage experienced in service."

In addition, the FAA is trying to formulate additional guidance for demonstrating compliance with these regulations. On January 8, 1981, the FAA solicited comments on a draft AC, file No. AC 25-903-X. The draft AC is intended to "...provide guidance for demonstrating compliance with the design requirements of the Federal Aviation Regulations to minimize the hazards caused by uncontained turbine engine and auxiliary power unit rotor and blade failures." The AC also defines, in part, the segments and fragments that must be considered; the critical components to be considered; the fragment trajectories that should be considered; the design considerations that would satisfy the regulations; and the maximum fragment energy. However, the draft AC has not been adopted and the FAA is still evaluating the comments of the engine and aircraft manufacturers.

1.17.2 DC-10 Leading Edge Wing Slat System

During the Safety Board's investigation of this accident, the uncommanded retraction of the right wing's outboard slat group caused concern even though the retraction ocurred long before the flightcrew was committed to continuing the takeoff.

This concern originally arose during the course of the Safety Board's investigation of the American Airline's DC-10 accident at Chicago-O'Hare International Airport (O'Hare), Illinois, May 25, 1979, 3/ and the operation and certification of the DC-10's leading edge slat system was examined in detail during that investigation. The leading edge slat system was certified in accordance with the applicable sections of 14 CFR 25. The flap control requirements of 14 CFR 25.701(a) were applied to the certification of the leading edge slat system. Paragraph (a) states:

> The motion of 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 since 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 position -- 260 KIAS -- 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, that the slats are in transit, or that the slats have been extended automatically.

The commanded slat position is held by trapped hydraulic fluid in the actuators; therefore, no consideration was given to an alternate locking mechanism. The slat's 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 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 are correctable by the flightcrew. However, McDonnell-Douglas did consider one multiple failure in its failure mode and effect analysis (FMEA)--erroneous motion transmitted to the right hand outboard slats and a failure of the No. 3 engine. 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 $1x10^{-10}$." The FMEA did not discuss the type engine failure nor the cause of the erroneous slat motion.

The first analysis to calculate the probability of engine debris causing an uncommanded retraction of the outboard leading edge slats was conducted after the American Airlines DC-10 accident. 4/ According to this study, the best estimate of

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3/ Aircraft Accident Report--"American Airlines, Inc., DC-10-10, N110AA, Chicago-O'Hare International Airport, Chicago, Illinois, May 25, 1979" (NTSB-AAR-79-17).

4/ Estimating the Probability of Asymmetric Deployment of the Landing Edge Slat System on the DC-10 Aircraft J.H. Wiggins Co. Technical Report No. 79-1365, August 7, 1979.

a chance for an uncommanded slat retraction during takeoff was $2x10^{-9}$. For the purpose of this study, takeoff included the period of flight from the start of the takeoff roll to the intentional retraction of the slats after the takeoff climb.

During the investigation of the O'Hare DC-10 accident, the takeoff performance of the DC-10 series aircraft with a retracted outboard leading edge slat group was explored. The investigation noted that in the worst case liftoff occurred 2 knots above the 1G stalling speed — the minimum speed at which the wing is capable of generating lift equal to the aircraft weight. However, "the analysis did not attempt to show the aircraft would be controllable in the event of an engine failure in addition to the slat retraction and, in fact, the speed margin could be negative in this case; however, this combination of failures has been shown to be extremely improbable." 5/

As a result of the investigation of the O'Hare DC-10 accident, the Safety Board delivered several recommendations to the FAA concerning the DC-10, two of which are relevant to this accident. Recommendation A-79-99 requested that the FAA:

> Insure that the design of transport category aircraft provides positive protection against asymmetry of lift devices during critical phases of flight; or, if certification is based upon demonstrated controllability of the aircraft under conditions of asymmetry, insure that asymmetric warning systems, stall warning systems, or other critical systems needed to provide the pilot with information essential to safe flight are completely redundant.

Recommendation A-79-105 requested that the FAA:

> Revise operational procedures and instrumentation to increase stall margin during secondary emergencies by:
>
> (a) Evaluating the takeoff-climb airspeed schedules prescribed for an engine failure to determine whether a continued climb at speeds in excess of V2, up to V2 + 10 knots, is an acceptable means of increasing stall margin without significantly degrading obstacle clearance.
>
> (b) Amending applicable regulations and approved flight manuals to prescribe optimum takeoff-climb airspeed schedules; and
>
> (c) Evaluating and modifying as necessary the logic of flight director systems to insure that pitch commands in the takeoff and go-around modes correspond to optimum airspeed schedules as determined by (a) and (b) above.

Since the Safety Board issued Recommendation A-79-105, all DC-10 pilot flight manuals have been modified and they require that the pilots operate their aircraft in accordance with the speed schedules contained in paragraph (a) therein. McDonnell-Douglas issued Service Bulletin 22-107 which modified the DC-10 flight director to provide the following speed commands to a pilot:

5/ Report to the Administrator on the Investigation of Compliance of the DC-10 Series Aircraft With Type Certification Under Asymmetric Slat Conditions, July 9, 1979.

1. If an engine fails before attaining V2 speed, the flight director will command the pilot to fly V2.

2. If an engine fails between V2 and V2 + 10 KIAS, the flight director will command the pilot to fly the speed that existed at the time of engine failure.

3. If an engine failure occurs at or above V2 + 10 KIAS, the flight director will command the pilot to maintain V2 + 10 KIAS.

The provisions of this service bulletin had not been accomplished on N101TV; however, the Air Florida operational procedures required its DC-10 pilots to adhere to the airspeed schedules cited above.

As a result of recommendation A-79-99, the FAA issued AD 80-03-10, effective February 21, 1980. This AD required the operators to install two Auto Throttle Speed Command Systems (ATSC), stall warning stick shakers on the captain's and the first officer's control columns, and to revise these systems so that slat position information from the inboard slat group and both the left and right wing's outboard slat group position sensors are provided independently to each ATSC. The ATSC, in turn, provides the logic which activates the stall warning devices. The same slat position sensors which supply slat position logic to the ATSC's also supply information to the slat position indicator (slat disagree light). If a slat retraction is sensed on any slat group, the stall warning provided by the ATSC is based on the slat retracted stall speed. However, based on the signals from the ATSC, the flight director will command the pilot to fly a speed about 20 percent faster than the slat retracted stall speed. In addition, the logic of the system is such that, in the event of a sensor malfunction or a lost signal from a sensor, the ATSC will provide slat retracted aerodynamic logic to the stall warning system and flight director. N101TV had been modified in accordance with this AD.

Recommendation A-79-99 also requested that the lift devices provide "positive protection against asymmetry" during critical phases of flight. However, since the certification of the DC-10 was also based upon demonstrated controllability under conditions of asymmetry, the recommendation, as proposed, could have been satisfied alternatively by the corrective actions contained in AD 80-03-10. The Board believed that these actions, combined with the low probability that a dual failure would occur during the small time period that could be considered a critical phase of flight, satisfied the intent of this recommendation.

Although the study conducted after this accident concluded that the probability of an uncontained rotor failure causing an uncommanded slat retraction during the critical portion of the takeoff roll was still extremely remote — 0.72×10^{-9} -- the fact that there have been three uncommanded slat retractions was significant to the Safety Board, the FAA, and McDonnell-Douglas. 6/ Even though the three uncommanded slat retractions did not occur during the critical phase of takeoff, on February 19, 1982, McDonnell-Douglas issued DC-10 Service Bulletins 27-187 and 27-189 which modified and changed the leading edge slat control system. The first change modifies the hydraulic valves at the slat actuator to insure that in the event of a broken hydraulic line or lines, the fluid in the actuator would not be ported overboard and that the slats would remain in

6/ In Addition to the two accidents cited herein, on April 14, 1977, an overseas air carrier's DC-10-30 experienced an uncontained failure of its No. 3 engine's stage 1 low pressure turbine rotor followed by an asymmetric leading edge slat condition. The incident occurred between 400 feet to 600 feet altitude and thereafter the aircraft landed safely.

the commanded position. The second change modifies the slat control system's followup cables to insure that in the event they were subjected to impact damage they would not impart an unwanted command signal to the hydraulic valves in the system. These changes were incorporated into AD 82-03-03. (See appendix D.) The AD became effective February 25, 1982, and compliance is required "...on or before January 31, 1983, or in accordance with a schedule of accomplishment approved by the Chief, Los Angeles Area Aircraft Certification Office, FAA, Northwest Region."

1.17.3 Takeoff Performance Data

Although the engine failure occurred well below V1 speed and before the aircraft entered any phase of operation that could be categorized as critical, the Safety Board constructed a performance summary depicting the aircraft's capabilities had the engine failure occurred at V1 and the flightcrew continued the takeoff.

The takeoff performance data cited below were based on the aircraft's takeoff flap setting, takeoff thrust settings, the existing weather conditions, and the following assumptions:

1. The No. 3 engine failed 1 second before V1.
2. The outboard slat group began retracting 1 second after engine failure and was fully retracted 11.5 seconds later.

Since V1 = VR, the engine failure would occur 1 second before VR (VR-1 second) at 130 KIAS. At VR, the right outboard slat would start to retract. At this moment, the aircraft's 1G stall speed is 118 KIAS and this stall speed will continue to increase as the slat moves to its fully retracted position. At VR + 4 seconds, at 146 KIAS, liftoff occurs and the 1G stall speed is 130 KIAS. At VR + 5 seconds, at 148 KIAS, stall warning occurs, the slat is still in transit, and the 1G stall speed has increased to 133 KIAS. (The stall warning system incorporates a delay circuit and the system cannot operate until 5 seconds after nosewheel liftoff.) At VR + 7 seconds, V2, 149 KIAS, is reached. The slat is in transit and the 1G stall speed is 138 KIAS. At VR + 11.5 seconds, at 149 KIAS, the right outboard slat has retracted and the 1G stall speed is 143 KIAS.

Analysis of the takeoff performance data showed that the aircraft's V2 speed was well above 1.2 Vs. According to 14 CFR 25.107(b)(1) and (3), minimum V2 speed can never be less than 1.10 times the minimum control speed (Vmc). In this instance, because of the aircraft's light takeoff weight, the Vmc was faster than 1.2 Vs and it was the governing factor for establishing the V2 speed. In addition, in this case, V2 was also 6 KIAS faster than the slat retractd 1G stall speed.

On May 25, 1979, an American Airlines DC-10 series 10 aircraft, Flight 191, had its left engine (No. 1) and pylon and about 3 feet of the leading edge of the left wing separate from the aircraft during the takeoff. The separation occurred during rotation and the pilot continued the takeoff. Subsequent to the loss of the engine and pylon, the left wing's outboard slat group retracted, the aircraft stalled, rolled to the left, and crashed. Although the sequence of events on Flight 191 concerning the loss of the engine and retraction of the wing's outboard slat group would have been similar to that which could have occurred with Air Florida Flight 2198 had the engine failure occurred at V1, an examination of the factual data concerning Flight 191 showed significant differences.

Flight 191's takeoff weight was much heavier than that of Flight 2198, consequently its V2 speed was based on its stall speed (1.2 Vs). V2 for Flight 191 was 153 KIAS and the stall speed for the aircraft with the slat retracted was 159 KIAS.

Although Flight 191 reached 172 KIAS after takeoff, the pilot, in accordance with the operating procedures in effect at that time, decelerated his aircraft toward V2, and at 159 KIAS the aircraft stalled. After this accident, the operating procedures were changed. If the circumstances of May 25, 1979, were to recur today, pursuant to current procedures, the pilot would decelerate his aircraft to V2 + 10, 163 KIAS, and maintain 163 KIAS; thus the aircraft would not stall.

Although Flight 191 had two stall warning computers (Nos. 1 and 2 ATSC's), the ATSC's did not receive cross-over information from the slat sensors. The No. 1 ATSC received slat position information from the inboard slat sensors and from the left wing's outboard slat sensor only. The No. 1 ATSC was powered by the No. 1 generator's AC bus and this bus was lost when the pylon separated. Consequently, the stall warning system receiving information from the retracted slat was inoperative, the pilot's flight director was inoperative, and the slat disagreement warning function was inoperative. In accordance with AD 80-03-10, the DC-10's stall warning system has been modified since the Flight 191 accident.

2. ANALYSIS

The aircraft was maintained in accordance with prescribed regulations and procedures. The flightcrew was qualified in accordance with prescribed regulations. The aircraft's maintenance and flight logs disclosed no discrepancies that would have indicated a significant problem with the No. 3 engine.

The evidence showed that the incident was precipitated by the uncontained failure of the stage 1 low pressure turbine disk in the No. 3 engine. Since the failure occurred almost 40 KIAS below V1 speed, the flightcrew was able to stop the aircraft safely.

2.1 Engine Failure

Examination of the engine components showed that the failure sequence began with the failure of the stage 1 low pressure turbine rotor. The localized heating and smearing on the outer circumference of the rotor's aft spacer arm indicated that the rotor had oversped and then failed. This was supported by the uniform tensile type fracture surfaces of the stage 1 to stage 2 bolt heads and the fact that the stage 1 low pressure turbine rotor disk bore thickness when measured after the accident was 0.020 inches less than its original manufactured thickness. This cross sectional area would "neck down" during a turbine disk overspeed. In addition, the metallurgical examination of the pieces of the failed disk showed no evidence of preexisting fatigue or cracking and that the fracture surfaces were consistent with overstress separations.

In order for the stage 1 low pressure turbine rotor to have accelerated to the overspeed condition, it would had to have been free of the torque load of its rotor assembly. This freedom was accomplished by the failure of some of the 60 bolts which attached the stage 1 rotor to the stage 2 rotor at the bolted flanges of their spacer arms. Five pieces of foreign material--M-50 alloy steel-- were found in the engine. Three of these pieces were found in the low pressure turbine's stage 2 rotor cavity; two pieces were found in the engine's stage 1 toroid cavity. In addition, the stage 1 to stage 2 bolt heads were smeared with M-50 alloy steel. This condition indicated that, over a relatively long period, the impingement of this material damaged a sufficient number of these 60 bolts to destroy the bolts' capability to hold the stage 1 rotor to the stage 2 rotor. The bolts failed, the rotors separated, and the stage 1 rotor was free of its rotor

assembly load. As a result, the stage 1 rotor disk oversped during the high takeoff power demand on the engine. The overspeed resulted in the fragmentation of the rotor disk and exit of the fragments through the low pressure turbine casing. Consequently, the Safety Board concludes that the failure of the stage 1 low pressure turbine rotor disk was caused by the foreign material in the engine.

The five fragments of M-50 alloy steel found inside the low pressure turbine module were severely oxidized and battered indicating that they had been exposed to heat and impact damage for a considerable period. The main engine bearings of the CF6 engine fleet are made with M-50 alloy steel; however, all the main bearings forward of the low pressure turbine section were undamaged and all the engine bearings' inner races were intact. In addition, the part of the low pressure turbine section where these fragments were found is sealed from the path of gas flow through the engine. Consequently, this material could not have migrated either aft or forward into the low pressure turbine cavity while the engine was in operation. The evidence appears conclusive that either these fragments were inside the low pressure turbine module when it was installed on the engine or they got inside the turbine module the last time it was disassembled and were not detected during maintenance. Further, the battered condition of these fragments precluded all attempts to match them with the tools used in the engine build-up procedure, or with any other type M-50 alloy steel tool, consequently the Safety Board could not identify the source of these M-50 alloy steel fragments.

Examination of the damaged low pressure turbine indicated that the initial exit of the rotor disk fragments occurred just outboard of the top of the engine. Thereafter, the fragments left the engine around the circumference of the engine and produced the air frame damage described previously.

2.2 Slat Retraction and Takeoff Performance

The No. 1 and No. 3 hydraulic reservoirs were empty and the No. 1 hydraulic system's slat retract line was separated. The No. 3 systems pump suction line was separated and the system was slowly depleted by reservoir "head pressure." The right outboard slats were found in an intermediate position but almost fully retracted. This probably occurred for the following reasons:

1. The rotor fragments cut the slat follow-up cable. This cable drives the slat servo valve to the closed position when the slat reaches the position commanded by the pilot. When the cable was cut the slat servo valve was then free to go to any position and, in this case, it traveled to a position wherein the hydraulic fluid from systems Nos. 1 and 3 was ported to the retract side of the slat actuator cylinder.

2. The hydraulic fluid in system No. 1 was pumped overboard at normal system pressure.

3. The hydraulic fluid in system No. 3 was depleted by reservoir "head pressure."

Since the engine failure and slat retraction did not occur during the critical portion of flight between V1 and V2 speeds, speculation on what might have occurred had the takeoff been continued does not constitute any part of the cause and effect of the accident. However, given the background of slat problems in this area, the Safety Board believes that some of the facts relating to a continued takeoff should be discussed.

The No. 2 hydraulic system was intact; therefore, the aircraft was capable of continued flight although with some flight controls at a reduced level of effectiveness.

All three generator buses were operative. The evidence was conclusive that the right wing's outboard slat sensor was providing accurate position information to the DFDR; therefore, this same information would have been provided to the ATSC's and the slat disagreement warning light system. Consequently, the Safety Board concludes that the slat disagreement warning light operated properly and that the flightcrew did not notice that this light was illuminated during the rejected takeoff because they were preoccupied with higher priority crew tasks.

Because of the light takeoff weight, V1 and VR were the same; thus, even if the engine had failed 1 second before V1, the aircraft would have continued to accelerate during rotation and would have lifted off at 142.5 KIAS. The takeoff summary shows that in this instance, all takeoff V speeds would have been faster than the slat-retracted 1G stall speed. Finally, in accordance with prescribed procedures, the pilot would accelerate the aircraft to and then maintain V2 -- 149 KIAS. The data showed that slat retraction takes 11.5 seconds and that V2 was 6 KIAS above the slat-retracted stall speed. Therefore, since the aircraft would have attained V2 within 3 seconds after liftoff, the aircraft's speed would always have been faster than its slat-retracted 1G stall speed.

Further, since the flight directors, the ATSC's, and the slat sensors would have been functioning properly, the flightcrew would have received additional electronic assistance. The slat disagreement light would have illuminated, and since AD 80-03-10 had been accomplished on this aircraft, the flightcrew would have received a stick shaker warning based on the slat retracted stall speed. Moreover, if the flightcrew were using the flight director for the takeoff, the flight director command bars would have commanded a pitch attitude that would have produced a speed about 20 percent faster than the slat retracted stall speed. Thus, the Safety Board believes that had the engine failure occurred after V1 the flightcrew would have been able to continue the takeoff and could have landed the aircraft safely.

2.3 Aircraft Certification

The evidence showed that the engines, their installation on the DC-10-30, and the DC-10-30's leading edge wing slats were certified in accordance with the applicable regulations and Special Conditions. Although the regulations and Special Conditions did not require damage tolerance evaluations to ensure continued safe flight following an uncontained engine failure, the evidence indicates that McDonnell-Douglas did perform such an evaluation in preparation for United Kingdom certification of the DC-10-30, and that the FAA was in possession of the evaluation before the DC-10-30 was certificated. Further, during the DC-10 certification process, and since certification of the DC-10 series aircraft, 14 CFR 25 and 14 CFR 33 have been amended and amplified. The engine manufacturers are governed by the provisions of 14 CFR 33; however, 14 CFR 33 only requires that the compressor and rotor cases provide "...for the containment of damage from rotor blade failure;" it does not levy any requirement to contain a rotor disk fragment. Part 33 addresses the latter problem by requiring the engine manufacturers to guard against rotor disk burst by establishing service life limitations for the engine rotors, and to demonstrate that the rotors can sustain structural integrity up to, and including, established overspeed and overtemperature limitations.

However, industry-wide studies of all turbine engine experience have indicated consistently that the rotor failure problem, while not statistically alarming (.66 failures

per million engine hours during 1962-1975 and a factor in 0.22 percent of all fatalities 7/), had the potential for causing serious aircraft damage. Part 25 attempts to address this problem by requiring the manufacturer to take design precautions to minimize damage which could occur following an uncontained engine rotor burst. In addition, 14 CFR 25 also has been amended and amplified since 1970 and now reflects the requirements which were contained in Special Condition No. 25-18-WE 7(1).

14 CFR 25.903(d)(1) now states, in part, "Design precautions must be taken to minimize the hazards to the aircraft in the event of an engine rotor failure...." However, the regulation is general and there is no published guidance material setting forth acceptable methods for demonstrating compliance with the requirement. Currently, each applicant must show the design precautions which have been taken to minimize the hazards of rotor failure on his aircraft, and a determination of acceptance is based on a subjective evaluation of the adequacy of the analysis and the effectiveness of the designed precautions.

Consideration of the effect of an uncontained rotor failure is now provided for in 14 CFR 25.571(a) which states, in part, "an evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to....accidental damage will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of paragraphs (b) and (e) of this section." 14 CFR 25.571(b) requires the applicant to determine the probable location and modes of damage which could result from accidental damage and 14 CFR 25.571(e) states, in part, "The airplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of.... (3) uncontained engine failure."

The only published guidance material for showing compliance with this regulation is contained in Advisory Circular (AC) 25.571-1, Damage Tolerance and Fatigue Evaluation of Structure. Paragraph 4(g)(2) of the AC states, in part, "....in the case of uncontained engine failures, the fragments and paths to be considered should be consistent with those in showing compliance with 14 CFR 25.903(d)(1) of the FAR's, and with typical damage experienced in service."

Thus, the regulations now require an aircraft manufacturer to consider the possibility of an uncontained rotor failure and to minimize the effect the rotor fragments will have on the capability of the aircraft to continue safe flight. Based on information provided by the applicant, the FAA then determines whether the standards contained in 14 CFR 25 have been satisfied. Although the regulations do not establish uniform guidelines or standards for determining rotor fragment size, path, energy, or a method of documenting these rotor burst analyses, FAA Order 8110.11, issued on November 19, 1975, provided information on design considerations for minimizing uncontained rotor failure damage. The information was for use in regional flight standards offices and the Aircraft Engineering Division in the Western Region. However, the information did not include analytical methods or probability calculations.

Although engine rotor failures are relatively rare, their potential for causing serious damage to the aircraft or its systems is quite high. With regard to high bypass ratio engines, there have been 23 turbine or compressor rotor disk failures during the past 12 years, many of which have damaged the aircraft or its systems substantially.

7/ Society of Automotive Engineers Aerospace Information Report AIR 1537 Report on Aircraft Engine Containment 1977.

The rate of these failures have been relatively constant, and moreover, current materials and fabrications technology is such that containment of rotor disk fragments is not foreseeable in the near future. Given the technological limitations on containment, the Safety Board has no reason to believe that these failures will cease in the future, either on the current high bypass ratio engines or their successors. Therefore, the Safety Board is concerned that adequate aircraft design precautions are taken to minimize the hazards associated with uncontained rotor disk fragments. Although the Safety Board realizes that applicants for type certificates are required to take such precaution by current federal regulations, 14 CFR 25.903(d)(1) and 25.571(e)(3), the Board believes that the FAA must formulate and publish additional guidance for, and demand increased attention to, the adequacy of methods used to demonstrate compliance with these regulations.

Finally, since the design of the leading edge wing slat system was such that a malfunction could occur during slat operation, which could permit an outboard group of slats to either extend or retract asymmetrically, certification of this system was also based on test flight data showing that the aircraft could be flown safely with one outboard group of wing slats retracted and the other in the takeoff position within an airspeed range bounded by the stall warning speed and 260 KIAS -- the limiting airspeed for the takeoff slat position.

During the recertification tests conducted after the O'Hare DC-10 accident, the data acquired showed that the aircraft could takeoff safely with all engines operating and the outboard wing slat group retracted on one wing. Although analysis of this indicated that the DC-10, in the event of an engine failure in addition to slat retraction, might not be controllable under certain conditions, it also showed that this particular combination of failures was extremely improbable. As a result, the aircraft was recertified. The analysis conducted by McDonnell-Douglas after this accident further verified the data that this combination of failures was extremely improbable. However, despite this, the decision was made to modify the wing's leading edge slat system as provided for in AD 82-03-03. The Safety Board supports this decision and believes that where possible and economically feasible, designs should incorporate maximum safeguards regardless of the probability of occurrence.

3. CONCLUSIONS

3.1 Findings

1. The engine failure occurred below V1 speed. The flightcrew's decision to reject the takeoff was correct.

2. The stage 1 low pressure turbine rotor disk on the No. 3 engine fragmented and separated from the low pressure turbine rotor assembly. The fragments penetrated the engine causing damage to the leading edge of the right wing, and rendered aircraft systems and components inoperable.

3. There was no evidence of preexisting fatigue or cracking on any of the rotor nuts and bolts or pieces of the failed disk. All the fractured surfaces were consistent with overstress separations.

4. Several pieces of foreign material, made of M-50 alloy steel, were found in the engine and engine debris. The pieces were battered severely. Three of these pieces were found in the low pressure turbine's stage 2 rotor cavity; and two of these pieces were found in the engine's stage 1 toroid cavity.

5. The stage 1 to stage 2 rotor bolt heads were smeared with this M-50 alloy steel.

6. The failure of the stage 1 to stage 2 rotor bolts and the subsequent failure of the stage 1 low pressure turbine rotor was caused by the impingement of the pieces of M-50 alloy steel on the rotor bolts.

7. The engine's main bearings are made of M-50 alloy steel. All of the main engine bearings forward of the low pressure turbine section were undamaged.

8. The part of the low pressure turbine cavity where the M-50 fragments were found is sealed from the path of gas flow through the engine. The M-50 fragments could not have migrated either forward or aft into the low pressure turbine cavity while the engine was in operation.

9. The M-50 fragments found inside the low pressure turbine rotor cavities were not part of this engine.

10. Debris from the stage 1 low pressure turbine rotor disk cut the slat follow-up cable which connects the inboard slat actuator mechanism to the servo valve. The right wing outboard slat group was almost completely retracted as a result of this damage.

11. The Nos. 1 and 3 hydraulic systems were inoperative from lack of fluid.

12. The firewall fuel shut off valve actuating cables and the fire extinguishing discharge line from the two agent bottles for the No. 3 engine were severed. The No. 3 engine's thrust lever control cable was severed.

13. All three generator AC buses and all TR buses remained powered. The stall warning systems and flight director systems remained operative.

14. V2 speed was 6 KIAS faster than the slats retracted stall speed. Had the engine failure occurred at or after V1 speed, the flightcrew should have been able to continue the takeoff safely.

15. The DC-10-30 engine installation configuration and its leading edge wing slat system were certified in accordance with the applicable regulations and Special Conditions.

16. The present certification regulations do not establish precise standards and guidelines for analyzing the effects of an uncontained rotor burst.

3.2 Probable Cause

The National Transportation Safety Board determines that the probable cause of this accident was the failure of quality control inspections to detect the presence of foreign material in the low pressure turbine cavity during the reassembly of the low pressure turbine module after installation of the stage 1 low pressure turbine rotor disk. The foreign material in the low pressure turbine cavity damaged the bolts holding the stage 1 low pressure turbine rotor disk and stage 2 low pressure turbine rotor disk together. The bolts failed at high engine thrust and the stage 1 low pressure turbine disk separated from the low pressure turbine rotor assembly, oversped, and burst.

4. RECOMMENDATIONS

As a result of its investigation of this accident, the National Transportation Safety Board recommended that the Federal Aviation Administration:

> Expedite the publication of guidance material for acceptable means of compliance with 14 CFR 25.903(d)(1), which includes compliance documentation by failure mode and effect analysis, provides for rotor fragment energy levels and paths based on cases of severe in-service damage, and reflects advances in analytical techniques and concepts which have taken place since certification programs of the early 1970's. (Class II, Priority Action) (A-82-38)
>
> Actively encourage research and development in containment technology and engine reliability, including basic design concepts, manufacturing processes, and maintenance factors to detect and prevent impending failures. (Class II, Priority Action) (A-82-39)

BY THE NATIONAL TRANSPORTATION SAFETY BOARD

/s/ JAMES E. BURNETT, JR.
Chairman

/s/ FRANCIS H. McADAMS
Member

/s/ PATRICIA A. GOLDMAN
Member

G. H. PATRICK BURSLEY, Member, did not participate.

April 6, 1982

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5. APPENDIXES

APPENDIX A

INVESTIGATION AND HEARING

1. Investigation

The National Transportation Safety Board was notified of the incident about 1740 e.d.t. on September 22, 1981, and immediately dispatched a partial investigative team to the scene. Investigative groups were formed for operations; aircraft structures, systems and powerplants; cockpit voice recorder; flight data recorder; and maintenance records.

Parties to the investigation were the Federal Aviation Administration, Air Florida Airlines, Inc., McDonnell-Douglas, Inc., and General Electric, Inc.

2. Public Hearing

A public hearing was not held and depositions were not taken.

APPENDIX B

PERSONNEL INFORMATION

Captain Ulrich

Captain Raymond Ulrich, 31, was employed by Air Florida on September 17, 1972. He held Airline Transport Certificate No. 263926227 with an aircraft multiengine land rating and commercial privileges in aircraft single engine land. He was type rated in Boeing 737, and McDonnell-Douglas DC-9 and DC-10 aircraft. His first-class medical certificate was issued May 29, 1981, with no waivers or limitations.

Captain Ulrich qualified as captain on DC-10 aircraft on March 20, 1980. He passed his last proficiency check July 9, 1981 and his last line check on May 10, 1980. The captain had flown 7,100 hours of which 800 hours were in the DC-10. During the last 30 days and 24 hours before the accident, he had flown 63 hours and 0 hours, respectively. At the time of the accident, the captain had been on duty about 2 hours. He had been off duty more than 24 hours before reporting for duty on the day of the accident.

First Officer Warde

First Officer Michael B. Warde, 38, was employed by Air Florida on January 30, 1980. He held Airline Transport Certificate No. 2195197 with an aircraft multiengine land rating and commercial privileges in aircraft single engine land. His first-class medical certificate was issued September 10, 1980, with no waivers or limitations. His medical certification had been issued more than 6 months before the flight, therefore, he was exercising the commercial privileges of his Airline Transport Certificate.

First Officer Warde qualified as first officer on DC-10 aircraft on March 13, 1981. The first officer had flown 3,950 hours, of which 460 hours were in the DC-10. During the last 30 days and 24 hours before the accident, he had flown 57 hours and 4 hours, respectively. At the time of the accident, he had been on duty 2 hours. The first officer had been off duty 16 hours before reporting for duty on the day of the accident.

Flight Engineer McDonald

Flight Engineer Richard A. McDonald, 32, was employed by Air Florida on November 17, 1980. He held Flight Engineer Certificate No. 268-481-749 with a turbojet engine rating. His second-class medical certificate was issued September 9, 1981 with no waivers or limitations.

Flight Engineer McDonald, qualified as flight engineer on DC-10 aircraft on December 10, 1980. The flight engineer had flown 2,200 hours, of which 500 hours were in the DC-10. During the last 30 days and 24 hours before the accident, he had flown 55 hours and 0 hours, respectively. At the time of the accident, the flight engineer had been on duty 2 hours. He had been off duty more than 24 hours before reporting for duty on the day of the accident.

APPENDIX C

AIRCRAFT INFORMATION

McDonnell-Douglas DC-10-30 CF, N101TV

The aircraft was leased from Trans International Airlines, Inc., by Air Florida, Inc., on March 16, 1981 and had been operated continuously thereafter by Air Florida. The review of the aircraft's flight logs and maintenance logs disclosed no data which the maintenance review group characterized as other than routine.

The following statistical data was completed:

Aircraft

Total hours	25,824
Total cycles	6,214
Last "C" Check	December 21, 1980
Last "A" Check	September 21, 1981

Powerplants

Engine	No. 1	No. 2	No. 3
Serial No.	455-272	455-214	455-123
Date of Installation	September 1, 1981	December 17, 1980	December 18, 1980
Time since Installation	171 hrs	6,958 hrs	2,682 hrs
Total Time	14,738 hrs	21,548 hrs	18,668 hrs
Total Cycles	3,455	5,033	4,611

APPENDIX D

FAA AIRWORTHINESS DIRECTIVE CONCERNING DC-10 SLAT CONTROL SYSTEM

DEPARTMENT OF TRANSPORTATION

Federal Aviation Administration

14 CFR Part 39

[Docket No. 82-NM-03-AD; Amdt. 39-4306]

Airworthiness Directives: McDonnell Douglas Model DC-10 Series Airplanes

AGENCY: Federal Aviation Administration (FAA), DOT.

ACTION: Final rule.

SUMMARY: This amendment adds a new Airworthiness Directive (AD) that would require modification of the wing leading edge slat control system on McDonnell Douglas Model DC-10 series airplanes. The modification would consist of the installation of two balance spring assemblies on the slat control mechanism for the left and right outboard slat control valves as well as installation of balanced pressure relief valves in hydraulic systems No. 1 and No. 3 slat extend lines of the left and right outboard slat control systems. This AD will improve the capability of these airplanes to continue safe flight and landing by assuring that uncommanded outboard slat retraction does not occur as a result of a failure event during critical flight phases.

DATE: Effective date February 25, 1982. Compliance schedule as prescribed in the body of the AD, unless already accomplished.

ADDRESSES: The applicable service information may be obtained from: McDonnell Douglas Corporation, 3855 Lakewood Boulevard, Long Beach, California 90846, Attention: Director, Publications and Training C1-750 (54-60). This information also may be examined at FAA Northwest Mountain Region, 9010 East Marginal Way South, Seattle, Washington 98108, or 4344 Donald Douglas Drive, Long Beach, California 90808.

FOR FURTHER INFORMATION CONTACT. Gilbert L. Thompson, Aerospace Engineer, Systems and Equipment Branch, ANM-130L, Federal Aviation Administration, Northwest Mountain Region, Los Angeles Area Aircraft Certification Office, 4344 Donald Douglas Drive, Long Beach, California 90808, telephone (213) 548-2833.

SUPPLEMENTARY INFORMATION: On September 22, 1981, a DC-10-30F airplane experienced a failure of the No. 3 engine during takeoff resulting in subsequent rejection of the takeoff with no injuries to passengers or crew. During the investigation of this incident by the FAA, it was learned that [illegible]ied failure of the No. 3 engine first stage low pressure turbine disk resulted in uncommanded retraction of the right wing outboard slats due to failure of the associated outboard slat follow-up cable. It was further learned during this investigation that a similar No. 3 engine failure in 1977 on a foreign operated DC-10-30 may have also resulted in uncommanded retraction of the right hand outboard slats. In the latter case, the engine failure occurred at approximately 400-600 feet altitude with subsequent execution of a safe landing. These events prompted a re-evaluation by the FAA of the ability of the DC-10 airplane to be capable of continued safe flight and landing after the occurrence of a critical engine failure combined with outboard slat retraction, considering such occurrence as a single event. This re-evaluation centered around an analysis of the probability of occurrence of such an event during critical phases of flight in conjunction with a controllability analysis of the airplane under these conditions.

The results of analytical determination of the probability of occurrence of a critical engine failure combined with slat retraction during critical phases of flight (including takeoff and landing) show that the occurrence of such an event is considered extremely improbable (a likelihood of occurrence of less than one in a billion). Though such analysis may be used as evidence to indicate that a specific failure event is extremely improbable and, therefore, not warranting further consideration, it is the FAA's position that probability analyses alone do not determine acceptability of a given design. As with most probability analyses, certain assumptions must be made, based upon historical data were possible, which structure the bounds within which the results remain meaningful. In determining the probability of the event noted, sufficient room for judgment exists in establishing the bounds for some of the assumptions therein postulated to preclude acceptability of the design based on probability analyses alone.

Having considered the DC-10 service experience to date, the existing data concerning turbine engine failures, and the results of the above noted probability analysis, it is FAA's determination that sufficient area exists for judgment in the interpretation of this data to warrant a determination that design changes should be incorporated to improve safety.

Since this condition is likely to exist or develop in other airplanes of the same type design, an Airworthiness Directive is being issued which requires modifications to the DC-10 slat control systems which would assure that the wing slats remain extended even if the systems that actuate them sustain severe damage. This action is in full accord with FAA policy to utilize AD procedures to make changes to the approved type design when appropriate in the interest of aviation safety.

It is expected that kit parts for the above modifications will be obtainable beginning April 1982 with worldwide DC-10 fleet parts availability completed by October 1982.

Since a situation exists that requires immediate adoption of this regulation, it is found that notice and public procedure herein are impracticable and good cause exists for making this amendment effective in less than 30 days.

Adoption of the Amendment

Accordingly, pursuant to the authority delegated to me by the Administrator, § 39.13 of Part 39 of the Federal Aviation Regulations (14 CFR 39.13) is amended by adding the following new Airworthiness Directive:

McDonnell Douglas: Applies to McDonnell Douglas Model DC-10-10, -10F, -15, -30, -30F, and -40 series airplanes certificated in all categories.

To assure that the DC-10 wing slats remain extended even if the slat control system sustains severe damage, accomplish the following:

Unless already accomplished compliance is required with paragraphs A and B on or before January 31, 1983, or in accordance with a schedule of accomplishment approved by the Chief, Los Angeles Area Aircraft Certification Office, FAA Northwest Mountain Region.

A. Modify the leading edge slat servo system and replace the outboard slat system follow-up cables as outlined in the Accomplishment Instructions of McDonnell Douglas DC-10 Service Bulletin 27-187, original issue, or later revisions approved by the Chief, Los Angeles Area Aircraft Certification Office, FAA Northwest Mountain Region.

B. Install balanced pressure relief valves in hydraulic systems No. 1 and No. 3 slat extend lines, left and right wing, as outlined in the Accomplishment Instructions of McDonnell Douglas DC-10 Service Bulletin 27-189, original issue, or later revisions approved by the Chief, Los Angeles Area Aircraft Certification Office, FAA Northwest Mountain Region.

C. Within the next 2000 flight hours after accomplishment of the modifications noted in paragraph A above, and at intervals not to exceed 4000 flight hours thereafter, visually inspect the balance spring assemblies and outboard slat follow-up cables, left and right wing, for integrity of installation.

D. Within the next 4,000 flight hours after accomplishment of the modifications noted in paragraph B above, and at intervals not to exceed 4,000 flight hours thereafter, functionally check for proper operation of the outboard slat relief valves as outlined in the Accomplishment Instructions, paragraph (E) of McDonnell Douglas DC-10 Service Bulletin 27-109, original issue, or later revisions approved by the Chief, Los Angeles Area Aircraft Certification Office, FAA Northwest Mountain Region.

E. Upon the request of an operator, an FAA maintenance inspector, subject to prior approval by the Chief, Los Angeles Area Aircraft Certification Office, FAA Northwest Mountain Region, may adjust the repetitive intervals specified in paragraphs C and D of this AD to permit compliance at an established inspection period of that operator if the request contains substantiating data to justify the change for that operator.

F. Alternate means of compliance with this AD which provide an equivalent level of safety may be used when approved by the Chief, Los Angeles Area Aircraft Certification Office, FAA Northwest Mountain Region.

G. Special flight permits may be issued in accordance with FAR 21.197 and 21.199 to operate airplanes to a base in order to comply with the requirements of this AD.

The manufacturer's specifications and procedures identified and described in this directive are incorporated herein and made a part hereof pursuant to 5-U.S.C. 552(a)(1).

All persons affected by this directive who have not already received these documents from the manufacturer may obtain copies upon request to the McDonnell Douglas Corporation, 3855 Lakewood Boulevard, Long Beach, California 90846, Attention: Director, Publications and Training, C1-750 (54-60). These documents also may be examined at FAA Northwest Mountain Region, 9010 East Marginal Way South, Seattle, Washington 98108, or 4344 Donald Douglas Drive, Long Beach, California 90808.

This amendment becomes effective February 25, 1982

(Secs. 313(a), 601, and 603 Federal Aviation Act of 1958, as amended (49 U.S.C. 1354(a), 1421, and 1423); Sec. 6(c), Department of Transportation Act (49 U.S.C. 1655(c)); and 14 CFR 11.89)

Note.— The FAA has determined that this regulation is an emergency regulation that is not major under Executive Order 12291. It has been further determined that this document involves an emergency regulation under DOT Regulatory Policies and Procedures (44 FR 11034; February 26, 1979). If this action is subsequently determined to involve a significant regulation, a final regulatory evaluation or analysis, as appropriate, will be prepared and placed in the regulatory docket (otherwise, an evaluation is not required). A copy of it, when filed, may be obtained by contacting the person identified under the caption "FOR FURTHER INFORMATION CONTACT."

This rule is a final order of the Administrator. Under Section 1006(a) of the Federal Aviation Act of 1958, as amended, (49 U.S.C. 1486(a)), it is subject to review by the courts of appeals of the United States, or the United States Court of Appeals for the District of Columbia.

Issued in Seattle, Wash., on January 21, 1982.

Charles R. Foster,

Director, Northwest Mountain Region.

[FR Doc. 82-1999 Filed 1-25-82; 8:45 am]
BILLING CODE 4910-13-M

Federal Register, January, 26 1982

APPENDIX E

UNCONTAINED TURBINE OR ROTOR DISK FAILURES IN HIGH BYPASS RATIO ENGINES

RB 211

Date	Aircraft	Engine No. and Failed Part
2/28/72	L1011	No. 3, fan disc
1/11/73	L1011	No. 1, fan disc
11/25/76	L1011	No. 1, high pressure turbine disc
1/28/80	L1011	No. 2, high pressure compressor disc

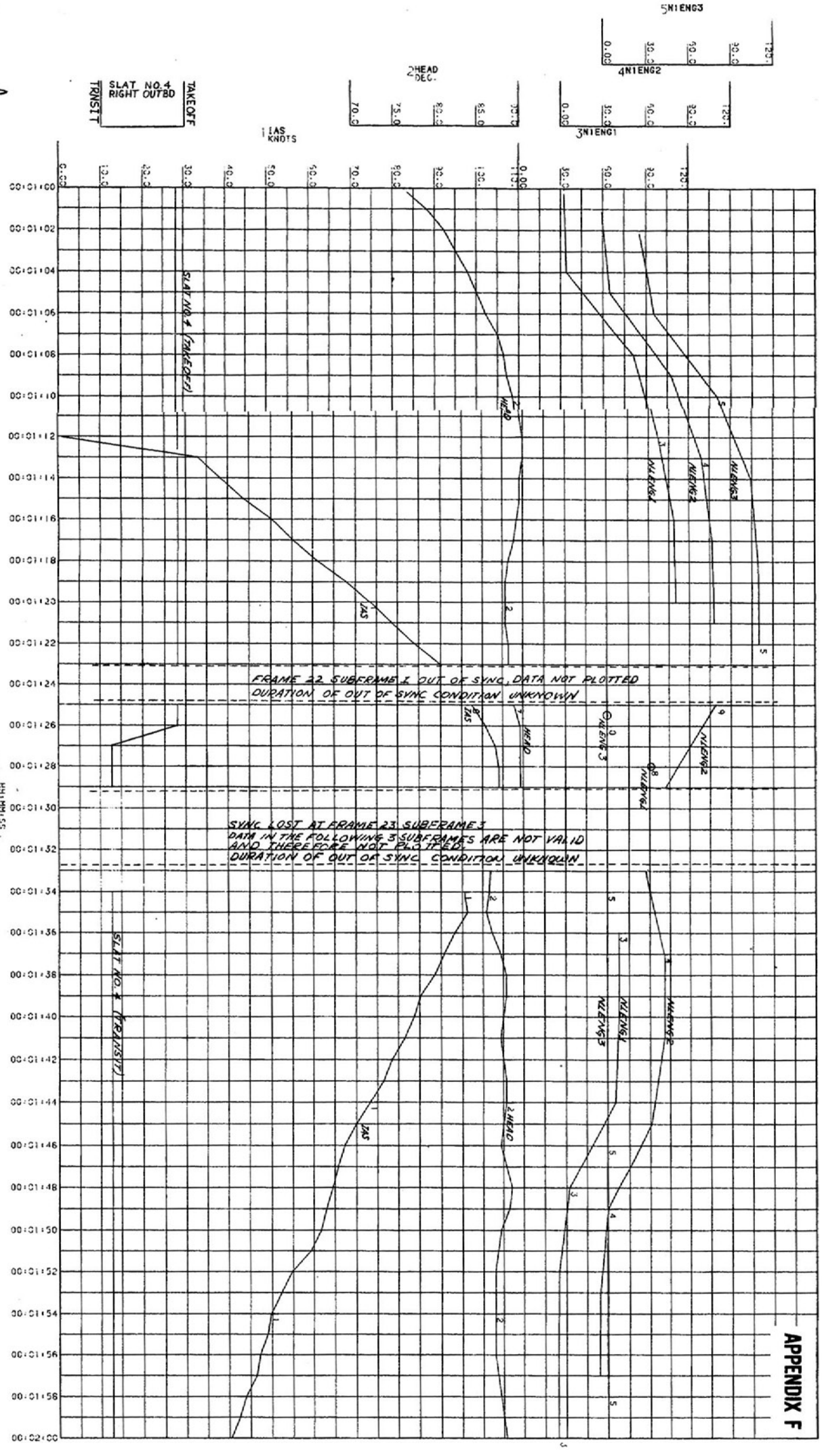
CF6

Date	Aircraft	Engine No. and Failed Part
8/9/74	DC-10	No. 1, high pressure compressor rotor
8/1/76	DC-10	No. 3, high pressure compressor rotor
10/26/76	DC-10	No. 3, high pressure compressor rotor
4/14/77	DC-10	No. 3, low pressure turbine rotor
6/12/77	DC-10	No. 1, high pressure compressor rotor
7/28/77	DC-10	No. 2, high pressure compressor rotor
3/14/79	DC-10	No. 3, high pressure compressor rotor
6/9/80	DC-10	No. 3, high pressure turbine disc
7/3/81	A-300	No. 1, high pressure turbine disc
9/22/81	DC-10	No. 3, low pressure turbine disc

JT9D

Date	Aircraft	Engine No. and Failed Part
8/17/70	B747	No. 3, second turbine disc
9/18/70	B747	No. 1, second turbine disc
5/3/71	B747	No. 3, second turbine disc
5/20/71	B747	No. 4, second turbine disc
4/1/73	B747	No. 3, second turbine disc
5/25/75	B747	No. 3, seventh high pressure compressor disc
11/7/75	B747	No. 2, sixth low pressure turbine disc
3/16/77	B747	No. 4, sixth turbine disc
3/7/78	B747	No. 4, fifth turbine disc

☆ U. S. GOVERNMENT PRINTING OFFICE : 1982 361-828/301



APPENDIX F

NATIONAL TRANSPORTATION SAFETY BOARD
BUREAU OF TECHNOLOGY
WASHINGTON, D. C.

LOCATION:	MIAMI FLORIDA	RECORDER M/M:	SUNDSTRAND 573
DATE :	9-22-81	RECORDER S/N:	3206
AIRCRAFT:	DC-10-10 N1011V	IDENT. NO. :	MIA81IA121
OPERATOR:	AIR FLORIDA	REPORT NO. :	61-40
FLT. NO.:			

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