Uncontained engine failure, American Airlines, Inc., Boeing 747-121, N743PA, San Francisco, California, September 18, 1970

Micro-summary: **This Boeing 747-121 experienced an uncontained engine failure of the #1 engine on climbout.**

Event Date: **1970-09-18 at 0851 PDT**

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

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

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SA-None File No. 4-0045

AIRCRAFT INCIDENT REPORT

AMERICAN AIRLINES, INC.,

BOEING 747-121, N743PA

SAN FRANCISCO, CALIFORNIA

SEPTEMBER 18, 1970

Adopted: February 3, 1971

NATIONAL TRANSPORTATION SAFETY BOARD
Washington, D. C. 20591
REPORT NUMBER: NTSB-AAR-71-7

American Airlines, Inc.,
Boeing 747-121, N743PA
San Francisco, California
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SYNOPSIS

American Airlines Flight 14, a Boeing 747-121, N743PA, was a scheduled passenger nonstop flight which originated at San Francisco International Airport at 0830 P.d.t. 1/ on September 18, 1970. Its destination was New York, John F. Kennedy International Airport. At departure from San Francisco, 105 revenue passengers, 12 nonrevenue passengers, and a crew of 15 were aboard the flight.

The flight's departure from the gate was routine with the exception of some difficulties in starting with No. 1 engine. The airplane was cleared for takeoff from San Francisco's International Airport, Runway 1-R, at 0851. During the takeoff, it was necessary for the flight engineer to reduce power on the No. 1 engine by .10 EPR 2/ in order to maintain the EGT 3/ within limits. Approximately 16 seconds after lift-off at an altitude of 525 feet m.s.l. 4/, the No. 1 engine sustained a separation of the second-stage turbine disk rim. The turbine blades and rim fragments penetrated the high-pressure turbine (HPT) case, engine cowling, and adjacent airplane structure. All fluid lines, electrical cables, and pneumatic ducts located in the pylon area were severed and an intense fire ensued. Two fuel tank access plates on the bottom of the wing inboard of No. 1 pylon were also penetrated by turbine fragments.

1/ All times used herein are Pacific daylight based on the 24-hour clock.

2/ Engine pressure ratio (EPR) is indicated as a measure of thrust developed by the engine. This is the ratio of the turbine discharge total pressure to the equivalent compressor inlet total pressure.

3/ Exhaust gas temperature.

4/ M.s.l. - Mean Sea Level

The fire warning for the No. 1 engine came on simultaneously with the engine explosion. Emergency fire control procedures were initiated and executed. The fire, which was observed by the captain, was propagating over the top of the left wing and lasted approximately 3 minutes. As a result of complete failure of the No. 1 hydraulic system, alternate extension of the body main landing gear, nose landing gear, and inboard trailing edge flaps was necessary. A successful landing was accomplished on San Francisco's International Airport. Passengers and crewmembers were deplaned on the taxiway by means of boarding steps. There were no injuries to passengers, crewmembers, or persons on the ground.

The National Transportation Safety Board determines that the probable cause of this incident was a progressive failure in the high-pressure turbine module in the No. 1 JT9D-3A engine. This failure was initiated by the undetected stress rupture fractures of several first-stage turbine blades and culminated in the inflight separation of the second-stage turbine disk rim.

The Safety Board sent a letter to the Federal Aviation Administration (FAA) on September 25, 1970. This letter related some of the problem areas associated with JT9D engine operations at higher than desirable turbine temperatures and made recommendations toward correction of these conditions.

The Administrator's response dated October 1, 1970, indicated that appropriate action had been taken regarding most of the Board's recommendations and that the remaining items were being evaluated. The Administrator's additional response dated December 23, 1970, indicated that further action had been taken to resolve the problems.

INVESTIGATION

American Airlines Flight 14 of September 18, 1970, was a regularly scheduled nonstop passenger flight between San Francisco, California, and New York's John F. Kennedy International Airport. Flight 14 was scheduled to depart San Francisco International Airport at 0830.

The airplane had been serviced with 180,000 pounds of Jet A fuel. The fuel load distribution at departure was 3,350 pounds in No. 1 and No. 4 reserve tanks, 24,600 pounds in No. 1 and No. 4 main tanks, and 62,050 pounds in No. 2 and No. 3 main tanks. The combined airplane, passenger, cargo, and fuel weight was computed at 558,810 pounds for takeoff. The maximum allowable takeoff weight was 676,600 pounds for the existing ambient temperature of +60°F., light and variable winds, and the projected use of Runway 1-R.

Difficulty was experienced in starting the No. 1 engine at the ramps. Two starting attempts had to be terminated because of a rapid rise in EGT. The airplane's APU 5/ system, which normally supplies a minimum of 35 p.s.i. air pressure for starting, was inoperative. In order to obtain sufficient pneumatic pressure for satisfactory engine starting, an additional external ground air unit had to be utilized.

Flight 14 was cleared for takeoff by San Francisco tower local control at 0851. Takeoff power, which was computed to be 1.37 EPR, was set, and the takeoff roll was started. During the takeoff, the EGT on No. 1 engine started to climb, and it became necessary for the flight engineer to reduce power by .10 EPR in order to maintain EGT within specified limits.

Approximately 16 seconds after lift-off while climbing through 525 feet m.s.l., as the landing gear retraction cycle was in progress, the crew heard an explosive sound. This was followed immediately by activation of the fire warning system of the No. 1 engine. Engine emergency fire control procedures were initiated and both containers of fire extinguishing agent were discharged. The presence of an intense fire, with white flames propagating over the top of the left wing, had been visually confirmed by the captain. The first discharge of extinguishing agent did not control the fire; however, the intensity of the fire decreased considerably after the second discharge. The fire continued to burn for approximately 3 minutes.

The initial approach for an emergency landing on Runway 28L had to be abandoned because of the inability of the crew to lower the body

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5/ Auxiliary power unit supplies pneumatic pressure and electrical power for ground operations.

landing gear and extend the inboard leading edge flaps below the 10° (takeoff) position. It was determined that the No. 1 hydraulic system, which is required for body gear and inboard leading edge flap operation, had been rendered inoperative by the explosion and fire in No. 1 engine. The body gear and inboard leading edge flaps were subsequently extended by alternate means, and a second approach was initiated at 0902. With airport emergency equipment standing by, Flight 14 landed safely on Runway 28L at 0906. During the landing roll, the tower controller observed smoke coming from the area of No. 1 engine and so advised the flight.

The airplane was taxied to the far end of Runway 28L and was stopped on the taxiway between Runway 28L and 28R. The left forward, main cabin slide had been deployed but was not utilized because it was determined that the fire in the No. 1 engine had been completely extinguished. All passengers and crewmembers evacuated the airplane by means of portable loading steps. During and subsequent to the evacuation of the passengers, there was fuel spillage from the two punctured fuel tank access plates between wing stations 950 and 1000.

The San Francisco Fire Department, which provides rescue and fire fighting services for the San Francisco International Airport, reported that it used 5,000 gallons of water to wash down spilled fuel to eliminate any further hazard to the airplane and its occupants. The air carrier's personnel transferred fuel from the penetrated tank to adjacent tanks to prevent further spillage.

The overall investigation of this incident was conducted in two phases. Phase I consisted of the immediate on-the-scene investigative activities, while Phase II consisted of the detailed examination of the engine and laboratory analysis of failed parts.

Phase I

On-the-scene examination of the No. 1 engine disclosed that the outer rim of the second-stage turbine disk had separated from the remainder of the turbine hub. One 14.5-inch-long segment of this rim was recovered from the No. 1 pylon structure. This segment was forwarded to the metallurgical laboratories of Pratt & Whitney Aircraft for examination in the presence of the National Transportation Safety Board metallurgist.

The high-pressure turbine case had been penetrated by the rim fragments and had sustained massive deformation, both forward and aft of the penetrated areas. Similar massive deformation and tearing of engine components in the immediate area, including that of the engine oil tank, had occurred. Major damage was sustained by portions of the airplane's hydraulic, pneumatic, fuel, and electrical systems which were located in the No. 1 pylon and adjacent wing leading edge areas. All pylon plumbing extending forward of nacelle station 210, with the exception of the engine-driven hydraulic pump case drain line, was severed between nacelle stations 197 and 208 (see photographs Nos. 1 and 2). Both the throttle and thrust reverser control cables were severed at this location. All electrical wiring in the forward portion of the pylon between nacelle stations 165 and 185 was either melted or severely burned. The No. 1 hydraulic system pump supply line was severed between the system reservoir and the firewall shutoff valve (between nacelle stations 197 and 208). The No. 1 hydraulic system pressure line was also severed at this location. All usable fluid supply for the No. 1 hydraulic system was depleted.

The pylon valve-to-duct pressure line, as well as the pneumatic cross-ship manifold in the wing leading edge, was partially severed approximately 24 inches inboard of No. 1 pylon. The No. 1 pylon pneumatic duct was severed between nacelle stations 197 and 208. The pylon pneumatic shutoff valve, however, had been placed in the "off" position by closing of the firewall shutoff valve. The No. 1 engine fuel supply line was severed between nacelle stations 197 and 208. The firewall (fuel) shutoff valve operated normally, and terminated fuel supply when it closed during engine shutdown and fire control procedures.

Major aircraft structural damage was inflicted by failed turbine fragments and the ensuing fire.

The most severe structural damage in the pylon was sustained between nacelle stations 197 and 208, (Photographs Nos. 1 and 2) which was directly in the plane of rotation of the second-stage turbine wheel. The lower spar (firewall), mid-spar outboard cap and web, and inboard stiffeners were severed. The pylon outboard skin was cut from nacelle water

line 136 to 186, while the inboard side skin was cut from nacelle water line 136 to 164. In this same area, the front spar chord lower flange was bent and the lower flange of the inboard stiffener was broken.

Shrapnel and fire damage between nacelle stations 168 and 192, above nacelle water line 154, completely severed the pylon from the outboard lower spar (firewall) chord across the top to the inboard lower spar chord. This area of the pylon interior also exhibited the most intense heat and fire damage. The outboard stiffener was severed at this location and the fire extinguishing agent container mounting brackets were burned to the degree that agent containers had fallen onto the lower spar web.

Two holes were burned through the pylon outboard skin between nacelle stations 236 and 265 and nacelle water line 136 and 154. Much of the pylon outboard skin was discolored and buckled by heat. Although nacelle station 265.94 bulkhead remained otherwise intact, it also was discolored and buckled by heat.

The forward portion of the pylon was found drooping approximately 6 to 8 inches with the engine installed.

Varying degrees of fire (Photograph No. 3) and/or shrapnel damage were sustained by the left outboard flap, outboard aileron, No. 1 spoiler, flap track fairings, leading edge panels/fairings, wing leading edge support structure, and trailing edge panels.

The most severe damage was sustained by the underside of the left wing, both inboard and outboard of No. 1 pylon. The first and third fuel tank access plates outboard of the No. 1 pylon exhibited evidence of heat discoloration. Two fuel tank access plates between wing stations 950 and 1000 and 975 and 1000 were punctured and were the source of profuse fuel leakage. There was no ignition of the fuel which was leaking from these two access plates.

Gouges in the lower wing skin, inboard of No. 1 pylon, formed a pattern which ran diagonally inboard and rearward between wing stations 1070 and 940 and from the front spar to an area slightly aft of the fuel tank access plates. There were approximately 100 such gouges. Six relatively deep gouges were concentrated in an approximately 1-square-foot area at wing station 1035 just forward of the mid-spar. The deepest of these six gouges measured 0.187 inches in depth. Lower wing skin thickness at this point is .40 inches. Another concentration of gouges was located immediately forward of the fuel tank access plate between wing stations 975 and 1000. The deepest of these gouges measured 0.218 inches in depth. Lower wing skin thickness at this point is .326 inches.

Phase II

The rubbing of the second-stage turbine stator shroud assembly against the front surface of the first-stage turbine hub was initiated after the stress rupture failure of at least four first-stage turbine blades. The failure of these blades imposed unusually severe rearward loads upon the rear inner shroud feet of 32 second-stage turbine vanes, (Photograph No. 4). Consequent fatigue failures of these rear inner shroud vane feet allowed the second-stage turbine stator shroud assembly to shift rearward, under gas path pressures, and contact the rotating first-stage hub.

Initial disassembly and examination of the No. 1 engine had confirmed failure of the front turbine hub (first-stage) by fracturing circumferentially through the web adjacent to the web rim radius. The entire rim portion was consequently released through the high-pressure turbine case. Metallurgical examination of the failed hub assembly disclosed that a series of concentric grooves had been worn into the front face of the web from contact with the second-stage turbine stator shroud assembly. This assembly is located directly to the rear of the rotating first-stage turbine assembly. The fracture had occurred through the outermost groove. The failed hub was found to conform to specification mechanical property requirements, except in the rubbed areas where low hardness was evident due to rub-induced, localized overtemperatures conditions.

Detailed laboratory examination of seven P/N 674331 first-stage turbine blades disclosed transverse fractures through the airfoil sections approximately 1 inch above the blade root platforms. Thirty additional P/N 674331 first-stage blades exhibited cracks in the leading edges at locations similar to those found in the fractured blades (Photograph No. 5). The complete fractures, as well as the fractures seen through the cracks, exhibited oxide discolorated dendritic surfaces. There was no evidence of fatigue. Spectographic analysis of the turbine blade material disclosed that the material fully conformed to required specifications. None of the failed blades, nor 16 additional blades sectioned at random, showed any evidence of blocked cooling passages.

Metallographic examination of the fractured blades disclosed intergranular, oxidized fracture surfaces with associated alloy depletion. Evidence of sulfidation attach was also present along the fracture surfaces. This examination also provided evidence of metal temperatures of approximately 2,050° F. to 2,200° F. at the first-stage blade leading edges, and 2,000° F. to 2,050° F. at the blade trailing edges. Normal leading edge temperature of first-stage blades at takeoff thrust, operating on a standard 80° F. day, average 1,970° F. Under the same conditions, the maximum permissible temperatures at this point are between 2,025°F. and 2,050° F.

Thirty-two P/N 654352 second-stage turbine vanes had transverse fatigue fractures through the vane feet. The fractures, as well as cracks in other additional vanes, originated from the base of the front face of the vane feet near the convex side. Laboratory analyses disclosed that the mechanical and chemical properties of the vanes, as well as their dimensions, conformed to existing specifications.

Tests were also conducted during this investigation to determine the effect of one broken first-stage turbine blade on the vibratory loads which are normally placed upon the second-stage vanes. These tests disclosed that the breakage of one-half of a first-stage turbine blade results in an increase of vibratory loads from 3,800 to 4,000 p.s.i. range to the 12,000 to 16,000 p.s.i. range.

The wear patterns exhibited by the second-stage turbine shroud assembly confirmed direct contact of its rear inboard face with the front web portion of the second-stage disk portion of the front turbine hub.

A review of the operating history of this engine for the 7 days and for approximately 55 flight-hours immediately preceding the turbine failure reflected several mechanical discrepancies. Dates, discrepancies, and corrective action pertaining to No. 1 engine were as follows:

DATE	DISCREPANCY		CORRECTIVE ACTION
9/15/70	(24)	No. 1 and No. 2 throttles very far out of rig, unable T. O. power on No. 1 1.21 EPR No. 2 approximately 3" back of No. 3 and No. 4. At cruise. Unable use throttle bar account No. 2 too far back, No. 1 approximately 3 inches ahead of No. 3 and No. 4.	Checked No. 2 thrust lever rigging and adjusted idle and part power stops. Swapped No. 1 and No. 2 EPR indicators.
9/16/70	(30)	Throttle out of alignment at T. O. power.	Checked and Deferred.
9/16/70	(35)	Repeat Items Nos. 24 and 30.	

DATE	DISCREPANCY	CORRECTIVE ACTION
9/16/70	(36) Throttle alignment is very poor. Note items 24, 30 and 35.	Trim checked accomplished on No. 1 engine.
9/17/70	NOTE: 3 abort starts No. 1 engine and 1 on No. 4 (rapid rise exceeding N2).	Noted Starts.
9/17/70	(44) No. 1 engine EGT limited on T.O. and climb (.09 EPR less than others to hold 775° EGT) (other 766° at 1.31 EPR).	Replaced EPR transmitter.
9/18/70	(46) No. 1 engine blew up shortly after T.O. - fire warn-came on - fired both bottles (Wps).	

Airplane records disclosed that tailpipe inspections were accomplished per existing requirements after every flight.

American Airlines' computerized JT9D engine conditioning monitoring program provided for teletype inputs of engine operating data, which is manually recorded in flight by the flight engineer. The computer is programmed to correct the data for varying flight operating conditions. In order to ensure current availability of data, a computer run printout was made three times a week, although American's engineering specifications only required a weekly review of engine monitor logs. The purpose of this program is the detection of incipient engine problems by interpreting parameter trends, such as progressive or sudden changes in vital performance parameters.

A computer run and analysis of data relative to No. 1 engine on N743PA on September 23, 1970, disclosed a progressive increase in EGT and fuel flow, and progressive decrease in N2. 6/ Data points on this run, however, were not available prior to the failure on September 18, 1970.

6/ N2 -- r.p.m. of the high-pressure compressor.

Since the September 18 incident, American Airlines has implemented improved data transmittal procedures in order to ensure a more expedited analysis of engine operating parameters and to reduce the time lag between the development of trends and the initiation of corrective activity.

ANALYSIS AND CONCLUSIONS

One of the primary factors relative to the total analysis of this occurrence is the temperature sensitivity of the basic turbojet engine. Some of the parts, particularly those closest to the combustion chamber outlet or turbine inlet, are continuously subjected to high temperatures. They are also being subjected to rapid changes in temperatures. Maximum temperature transients are reached mainly during the takeoff phase of flight. The parts, which are subjected to these high temperatures, were carefully designed and tested to withstand them.

In the case of the JT9D-3 or -3A, which can operate at 43,500 or 45,000 pounds of thrust, respectively, the average metal temperature of the first-stage turbine blades is approximately 1,970° F. during takeoff thrust conditions. The solution heat treat temperature of these blades is 2,125° ± 25° F. If for some reason the operating temperature of the blade closely approaches or exceeds this temperature, the blade's designed high-temperature strength and resistance to failure by stress rupture may be seriously impaired.

The laboratory analysis of the four failed first-stage blades disclosed relative depths of oxide scale and alloy depletion layers that suggest these blades had been fractured for some time prior to the turbine disk rupture. Fresh fracture surfaces of similar blades were statically exposed in air at various controlled temperatures to determine the time element required to attain certain alloy depletion depths. The results of these laboratory tests were conclusive in establishing that several hours of exposure to temperatures in excess of 2,050° F. were required to attain the degree of alloy depletion seen in four of the failed first-stage blades; however, engine overheating conditions of only a few seconds duration may be sufficient to alter significantly the alloy microstructure.

Once the failure of one or more of the first-stage blades had taken place, severe and unusual vibratory loads were imposed upon the second-stage turbine vanes, inducing progressive fatigue failures of approximately 30 percent of the rear inner shroud feet of the vanes. The minimum thickness of the vane feet utilized in the No. 1 engine of N743PA was .083 inches. These were not designed to withstand the nearly 4.1 increase in vibratory loads which occurred when the initial failure of the first-stage blades occurred.

The fatigue failures of these inner shroud near vane feet allowed the second-stage turbine stator shroud assembly to shift rearward and rub against the front web surface of the second-stage turbine hub. This continued rubbing caused grooves to be worn into the disk portion of the front turbine hub. Finally, the front turbine hub fractured circumferentially in a rapid tensile manner at the grooved and weakened points, relating the entire rim portion of the disk.

In reviewing the operating history of the engine (S/N 662274), it becomes quite apparent that the laboratory findings relative to the pre-existing failures of at least four first-stage turbine blades is quite accurate.

Flight discrepancies reported on three separate occasions, indicating "poor throttle alignment" and, in one case, the No. 1 throttle's being as much as 3 inches forward of other engines, reflected an apparent deterioration in engine performance. However, due to the chronic throttle alignment problems which are, according to the carrier, normally encountered on the 747, as well as on other airplanes, this symptom was improperly diagnosed. The corrective action, consisting of rig check and adjustment of No. 2 thrust lever as well as swapping of No. 1 and No. 2 EPR indicators, was however, responsive to the flight discrepancy as reported. The flight engineer had obviously attempted to describe the cause of the problem by stating "No. 1 and No. 2 throttles very far out of rig," rather than to describe accurately the symptoms which possibly would have required more intense troubleshooting on the ground.

A subsequent flight discrepancy reported on September 17, 1 day prior to the final failure, showed the No. 1 engine to be EGT limited on takeoff and climb. It was necessary to operate No. 1 engine at .09 EPR less than others to maintain an EGT of 775° F. Other engines on the airplane, during this phase of the flight, operated at 1.31 EPR, maintaining 766° F. A thrust reduction of .09 EPR at takeoff under standard conditions can be translated into approximately 6,500 pounds of thrust and 105° F. of EGT.

Here again, it is apparent that the corrective action in replacing the EPR transmitter was based upon an erroneous assumption that an instrument error was responsible for the low-thrust indication on the No. 1 engine. An effective troubleshooting program at this point would, in all probability, have determined the reason for the high EGT and low EPR. Although tailpipe inspections were performed after every flight, the evidence of an incipient failure was either not present or was not recognized.

The inputs into the computerized JT9D condition monitoring program for this engine from September 10 through 16 likewise reflected adverse changes in trends of vital operating parameters which were indicative of the possibility of a serious engine malfunction. These data points showed a progressive increase of EGT and fuel flow while N2 showed a progressive decrease over the same period of time. Due to the time lag between data acquisition, the computer printout, and analysis of this above data, the trends shown were not available until after the failure had occurred and, consequently, could not be used effectively in diagnosing the problem.

The difficulty in starting the No. 1 engine, which was experienced at the origination of the flight, is not necessarily considered a major factor in this particular failure. It is noted that the airplane's APU was inoperative when N743PA was released for flight.

While the APU is needed for ground operations only, its functions, such as supplying pneumatic pressure for engine starting, are vital. In order to obtain a satisfactory engine start without the risk of approaching overtemperature conditions, a minimum of 30 p.s.i. pneumatic duct pressure should be available. The APU is capable of supplying between 40 and 45 p.s.i. pneumatic pressure; on the other hand, ground air supply units, which must be used when the APU is inoperative, generally do not have this capability unless the dual or triple external air connections on the airplane can be supplied by the ground air units.

In summary, the probability of an overtemperature condition during engine starting is considered higher when the airplane's APU system is inoperative and marginal capability ground units are utilized.

EFFECTS OF ENGINE STRUCTURAL FAILURE UPON THE AIRCRAFT

In view of the compounded and direct effects of a turbine failure such as this upon the continued safe flight of the airplane, the Safety Board finds a need for reviewing this aspect of the occurrence.

As indicated by the crew of Flight 14, the natural and most immediate concern was the control of the fire in flight and the safe return to the airport. Although the extinguishing agent was discharged by the crew, the agent's effectiveness in controlling or extinguishing the fire seems quite questionable. The most serious impairment of the system's effectiveness occurred when the engine and pylon enclosures were penetrated during the turbine failure, allowing a substantial portion of the extinguishing agent to escape into the atmosphere. The fire inside of the pylon continued with such intensity that both of the agent containers became physically detached from their mountings and fell to the bottom of the pylon structure. This fact alone can leave little doubt that the fire continued for some time after the agent was discharged. It is the opinion of the Board that the fire terminated only when the sources of flammable materials became exhausted. In this respect, termination of fuel supply was effective since the fuel line was severed downstream of the firewall shutoff valve which was closed by the timely action of the flightcrew. Flame propagation over both the top and bottom of the wing was such that there was danger of ignition of the fuel which was leaking out of the punctured fuel tank access plates.

In the case of the No. 1 hydraulic system supply, line severance occurred between the reservoir and the shutoff valve, allowing depletion and leakage into the fire area of the total fluid supply for the No. 1 system. Fluid supply was then no longer available to the No. 1 system's air-driven hydraulic pump. This pump normally provides a backup pressure source for the No. 1 system in case engine pump pressure is either lost or demands upon it become excessive. Of further significance is the puncture of the left wing pneumatic duct which supplies pressure for all of the pneumatically operated units in the left wing. Consequently, the operation of the No. 2 air-driven hydraulic pump would have been impaired by greatly reduced pneumatic pressure, if such operation became a requirement.

The loss of No. 1 hydraulic system, requiring alternate extension of the body landing gear and leading edge flaps, caused a delay in returning to San Francisco Airport and placed an additional burden upon the flightcrew during the already existing fire emergency. Although other, less vital systems were either fully or partially deprived of normal hydraulic and/or pneumatic pressure, there appeared to be no further adverse effects upon the operation of the airplane.

CONCLUSIONS

From the investigation of this incident, the Safety Board concludes the following:

1. There were no material deficiencies of the HPT module which either caused or contributed to the failure.

2. The engine had been allowed to attain operating temperatures which were sufficiently in excess of design limits to initiate stress rupture failures of first-stage turbine blades.

3. The engine normally operates at relatively high turbine temperatures and therefore requires most precise monitoring of all vital operating parameters and effective analysis of any confirmed deviations from normal parameters.

4. Vibratory stresses in excess of four times their normal level were imposed upon the second-stage vane feet after stress rupture of first-stage turbine blades.

5. Multiple failures of second-stage vane feet and resultant rearward shift of the nozzle inner support caused rubbing of the support against the second-stage turbine disk until separation of the disk rim occurred.

6. Deterioration of vital engine operating parameters was evident on both the narrative portion of the flight log and the computerized engine condition monitor log.

7. Maintenance actions taken by the carrier in attempting to correct the in-flight discrepancies as reported on the No. 1 engine were not responsive to the problem that existed.

8. While the computerized engine monitor log used by the carrier was effective in accurately identifying the progressive decrease of N2 and increase in EGT and fuel flow, the data was not available for use in time to effect corrective action prior to severe engine failure.

9. The fire which resulted from the turbine failure was terminated by the immediate response of the flightcrew in successfully shutting off fuel supply to the No. 1 pylon. The fire extinguishing agent appeared to have little effect in combating the fire.

10. The results of the primary failure affected other systems critical to the landing phase and compounded the already existing emergency by placing an additional burden upon the flightcrew.

PROBABLE CAUSE

The National Transportation Safety Board determines that the probable cause of this incident was a progressive failure in the high-pressure turbine module in the No. 1 JT9D-3A engine. This failure was initiated by the undetected stress rupture fractures of several first-stage turbine blades and culminated in in-flight separation of the second-stage turbine disk rim.

RECOMMENDATIONS AND CORRECTIVE ACTION

On September 25, 1970, the Board sent the following letter to the Administrator of the Federal Aviation Administration:

"The National Transportation Safety Board is now investigating the JT9D-3 engine failure and in-flight fire involving American Airlines, Boeing 747, N743PA, which occurred during takeoff from the San Francisco International Airport on September 18, 1970. A failure occurred in the No. 1 engine 13 seconds after lift-off, followed by a fire warning. The flight returned to the airport after shutdown of the engine and extinguishing of the engine fire.

"During the return to the airport, the flightcrew experienced difficulty in extending the landing gear and the wing flaps after parts of the failed engine severed the hydraulic and pneumatic systems' supply lines. The captain elected to "go around" and extended the landing gear by the alternate system. The aircraft made a successful landing, and there were no injuries to the 15 crewmembers or the 127 passengers.

"Our preliminary investigation of the engine failure revealed that a separation occurred to the rim portion of the second-stage turbine disk. It has been confirmed that failures of at least four of seven first-stage turbine blades contributed to the fracture of numerous second-stage turbine vane feet. As a result of the cumulative effect of the broken vane feet, an aft deflection of the nozzle support resulted, causing interference with and rubbing of the second-stage turbine disk. Progressive weakening of the disk rim area resulted in the in-flight failure of the rim. We have also confirmed that although failure mode of this second-stage turbine disk rim was similar to that of the Air France JT9D-3A engine failure of August 17, 1970, the failure mechanism was entirely different.

"As a result of our investigation and meeting with Pratt & Whitney engineering staff personnel and your Eastern Region Flight Standards personnel, immediate inspection action was initiated. This was considered fully responsive to the immediate needs of this situation. The Safety Board commends the Administrator's formalizing this corrective action in the form of your engineering alerts of September 19 and 23, 1970.

"In view of the potentially catastrophic results of the failure such as experienced by American Airlines, the Board remains concerned about this matter in the longer range sense and would urge the Administrator to initiate further expeditious actions in order to preclude recurrence of similar failures. Accordingly, the Board offers the following observations.

"It is generally recognized that the JT9D engine is normally operating near critical turbine temperature conditions. This is particularly true when operating in high ambient temperatures. Several JT9D engines have recently been removed from service and returned to Pratt & Whitney for overhaul, because of failed first-stage turbine blades as well as broken second-stage vane feet. There is evidence that these failures had occurred as the result of operation at higher-than-desirable temperatures.

"In the case of the most recent American Airlines turbine disk rim separation, there was evidence that at least six first-stage turbine blades had sustained varying degrees of fractures some time prior to the final failure. Our technical staff finds it most difficult to reconcile the fact that the airborne vibration monitoring equipment installed in the aircraft was either inadequate or was not effectively utilized in detecting this condition. We also feel that other engine instrumentation, namely: fuel flow, engine pressure ratio, and exhaust gas temperature should have been capable of collectively reflecting appropriate changes in the engine's operating parameters, if such instrumentation were properly calibrated and the respective readings were recorded and closely analyzed.

"In this area, we recommend the following be considered.

1. Initiate appropriate action toward the operators' maintaining a program of current engine condition monitoring.

2. Review engine instrumentation calibration and existing instrument tolerances to assure the most precise engine operating parameter indications.

"Further, it appears that the reliability of the Boeing 747 auxiliary power units is somewhat marginal. When engine starts must be accomplished by the use of ground units, pneumatic duct pressures may often be less than what is required, even when multiple units are used. The result is usually a start that may involve a temperature rise, approaching the "recoverable stall" condition. Since exhaust gas temperature, although above normal under these conditions often do not exceed the published limits, no record is made of these occurrences, and there is no possible way to determine how many times an engine hot section has been exposed to higher-than-normal temperatures. The effects of thermal transients are known to be cumulative and conceivably affect turbine blade reliability.

"As another measure toward improving the service reliability of first-stage turbine blades, it is recommended that appropriate action be initiated to:

1. Improve the reliability of auxiliary power units in order to reduce the probability of high thermal transients while starting engines with marginal air supply.

2. Ensure that flightcrews maintain adequate pneumatic air duct pressure during engine starts.

3. Record any abnormal starts when an approach to a "recoverable stall" is experienced.

4. Establish precise limitations regarding the number of "approaches to recoverable stall" conditions which may be tolerated without cumulative adverse effects upon turbine blade durability.

"The Safety Board is aware that the manufacturer has developed an improved type first-stage turbine blade (vented) which is expected to provide improved cooling characteristics and be more reliable when operating at high temperatures.

"With respect to the improved first-stage turbine blades, the Safety Board recommends:

1. Incorporation of the "vented" first-stage turbine blade in all JT9D series engines be the subject of regulatory action as soon as sufficient production is assured and service bulletins and engineering orders are formulated by the manufacturer.

"Water injection is presently being used on an optional basis by individual operators. Since water injection allows utilization of 45,000 pounds of thrust versus 43,500 pounds for take-off, some operators elect to use water only when takeoff weight, runway lengths, and ambient temperature conditions require the maximum thrust rating of 45,000 pounds. We believe that the use of water injection on those aircraft so equipped would be beneficial in providing for turbine blade cooling. The Safety Board recognizes that there are some operators whose engines are not equipped for water injection at this time, and to require use of water injection for all takeoffs would constitute an economic burden. However, we believe that the benefits may justify the expense.

"The Board, therefore, recommends the following:

1. Consideration should be given to require the use of water injection for all takeoffs regardless of takeoff thrust requirements.

2. Upon installation of the improved, "vented" turbine blades in all engines, the mandatory use of water injection could be rescinded.

"Technical details of the items outlined above have been discussed by members of both your Eastern and Western Region engineering staffs and our Bureau of Aviation Safety investigative personnel. Our staff members will be available for further discussions, if desired."

The Administrator's response was received on October 1, 1970.

"This is in reply to your letter of 25 September 1970 in which you offer recommendations as a result of your continuing investigation of the American Airlines engine failure at San Francisco on 18 September 1970.

"We appreciate receiving your commendation on the effectiveness of the immediate inspection actions initiated by the Federal Aviation Administration. As you know, appropriate actions had already been instituted for all items except the APU engine starting procedures and required use of water injection. These items are currently being evaluated.

"We will appreciate receiving any additional information developed from your continuing investigation of the American Airlines engine failure."

The Administrator's additional response dated December 23, 1970, was as follows and indicated that further action had been taken to resolve the problems:

"This will supplement our letter of 1 October 1970 regarding the investigation of the American Airlines Pratt & Whitney JT9D engine failure on 18 September 1970 at San Francisco.

"With regard to the Boeing 747 auxiliary power units, we find that, when the recent significant improvements have been accomplished, auxiliary power unit reliability is excellent. Boeing has updated their ground-starting information for both auxiliary power units

and ground-starting equipment in a customer letter and in the Boeing 747 Facilities Planning Document. Instructions on engine starting have been reviewed with the carriers at several of our industry meetings. Adherence to these procedures should prevent hot starts.

"The suggested mandatory use of water injection on all aircraft on takeoffs is not viewed as a panacea to the turbine blade-cracking problem as turbine blade problems have not been confined to "dry" engines. The use of water offers a reduction in turbine gas inlet temperature only for moderate ambient temperature takeoff conditions. More effective in this area is the procedure of using reduced thrust levels where possible to lower the turbine gas temperature.

"The incorporation of improved turbine blades is considered to be the best solution for improving the durability of these parts. First-service use of "vented" turbine blades has begun. Limited quantities of the new type "vented" turbine blades are available now and are being installed as rapidly as practicable. The carriers are estimating completion of retrofit on their fleet engines in the latter half of 1971.

"Pratt & Whitney Aircraft is developing further improved turbine blades of a more heat-resistant material. These should be available in the near future.

"It was agreed on 2 October 1970 that borescope inspection frequency of the combustion section would be established at 100-hour intervals on engines with more than 500 hours or more than 250 cycles. This, we believe, will effectively detect blades cracked from heat distress before they progress to failure."

CORRECTIVE ACTION

Manufacturer

The manufacturer, in conjunction with FAA advisories, has required:

1. Radioisotope inspection program for all JT9D engines which is in effect to inspect for vane lug failures.

2. In addition, the following inspections are required:

   a. Inspect engine tailpipe for metal after arrival at each station;

   b. Monitor aircraft airborne vibration monitoring equipment;

   c. If there is an indication as a result of inspection (a) or (b) above, borescope or chamberscope the turbine area for failed first-stage turbine blades; and

   d. Continue borescope inspection of the turbine area on a scheduled basis, but in no case exceeding 200-hour intervals.

In addition to the above field inspections, the following engineering change improvements are either being incorporated in the JT9D engine now or are scheduled for incorporation within the next several months:

1. Revised first-stage turbine blade leading edge cooling plus improved blade material.

2. Impingement-cooled first-stage turbine blade.

3. Second-stage vane rear inner lug thickness increased from .083 to .110 inches.

4. Increased cooling flow to second-stage inner seal and support, and second-stage vane lug.

5. Second-stage vane rear inner lug thickness increased from .110 to .145 inches.

AIR CARRIERS

American Airlines has revised their computerized engine condition monitoring program to provide for more rapid transmittal and review of data, greatly reducing the time lag between data acquisition and analysis of engine trends.

Other carriers have elected to utilize water injection for takeoff operations. Water injection used on a JT9D-3A engine operated at 43,500 pounds of thrust results in a 100° F. reduction in turbine inlet temperature on an 80° F. day. The same -3A engine operated with water injection at 45,000 pounds of thrust on an 80° F. day realizes a 60° F. reduction in turbine inlet temperatures. All JT9D engines may be equipped for water operation. Approximately 60 percent of the Boeing 747 aircraft in service are equipped for water operation.

AIRCRAFT INFORMATION

The airplane, a 747-121, N743PA manufacturer's serial No. 19638, is owned by Pan American World Airways and was being leased to and operated by American Airlines.

N743PA was delivered on March 3, 1970, and had accumulated a total of 1599:09 flight hours.

Pratt & Whitney JT9D-3A engines were installed on the airplane as follows:

POSITION	SERIAL NUMBER	TOTAL TIME	T. S. O.
No. 1	P-662274	1601:00	New
No. 2	P-662513	1114:00	New
No. 3	P-662455	611:00	New
No. 4	P-662287	1032:00	New

Airplane records disclosed that required inspections and line maintenance operations had been performed at specified time intervals.

Prior recorded inflight mechanical discrepancies related to this incident are outlined under the investigation portion of this report.

SUPPLEMENTAL DATA

Crew Information

The pilot in command, Captain Walter P. Steiner, holds a valid FAA Airline Transport Pilot Certificate No. 22991, as well as a current first-class FAA medical certificate. Captain Steiner holds type ratings for Boeing 707, 720, and 747 aircraft. His total flying time as of September 18, 1970, was 32,850.00 hours, 197:57 hours of which were accumulated in the Boeing 747.

The first officer, Joseph H. Martin, holds a valid FAA Commercial Pilot's Certificate No. 1600395, with multiengine, single-engine, and instrument ratings as well as a current first-class medical certificate. His total flying time as of September 18, 1970, was 5400:00 hours, 52:05 hours of which were accumulated in the Boeing 747.

The flight engineer, Marion H. Kilborn, holds a valid FAA Flight Engineer's Certificate No. 1211416 for reciprocating as well as turbojet-powered aircraft. He also holds a valid FAA Airframe and Powerplant Mechanic's Certificate No. 469023 and a current FAA second-class medical certificate. His total flying time as of September 18, 1970, was 14,290:00 hours, 60:37 hours of which were accumulated in the Boeing 747.

BY THE NATIONAL TRANSPORTATION SAFETY BOARD:

/s/ JOHN H. REED
Chairman

/s/ OSCAR M. LAUREL
Member

/s/ LOUIS M. THAYER
Member

/s/ ISABEL A. BURGESS
Member

Francis H. McAdams, Member, did not participate in the adoption of this report.

February 3, 1971.

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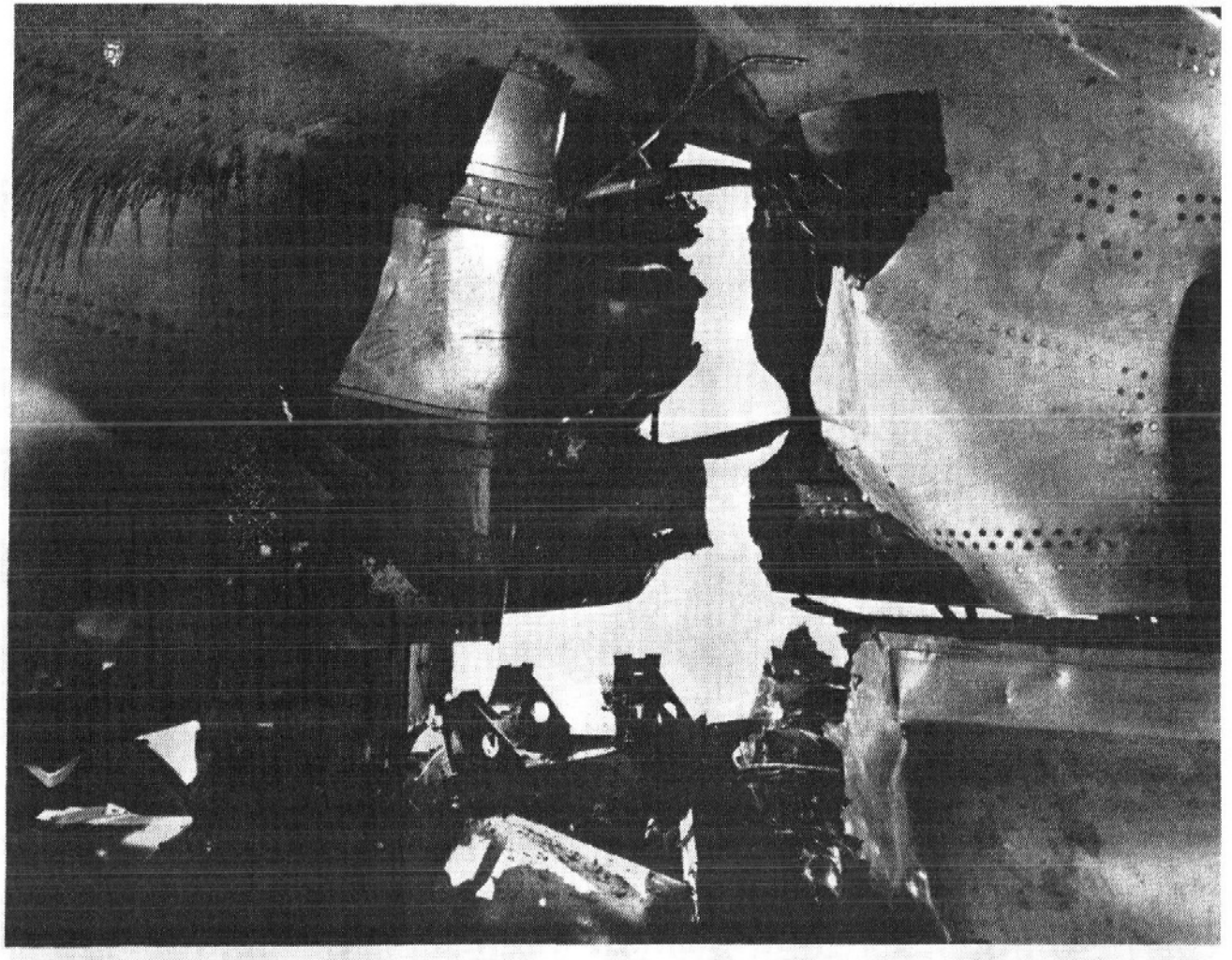
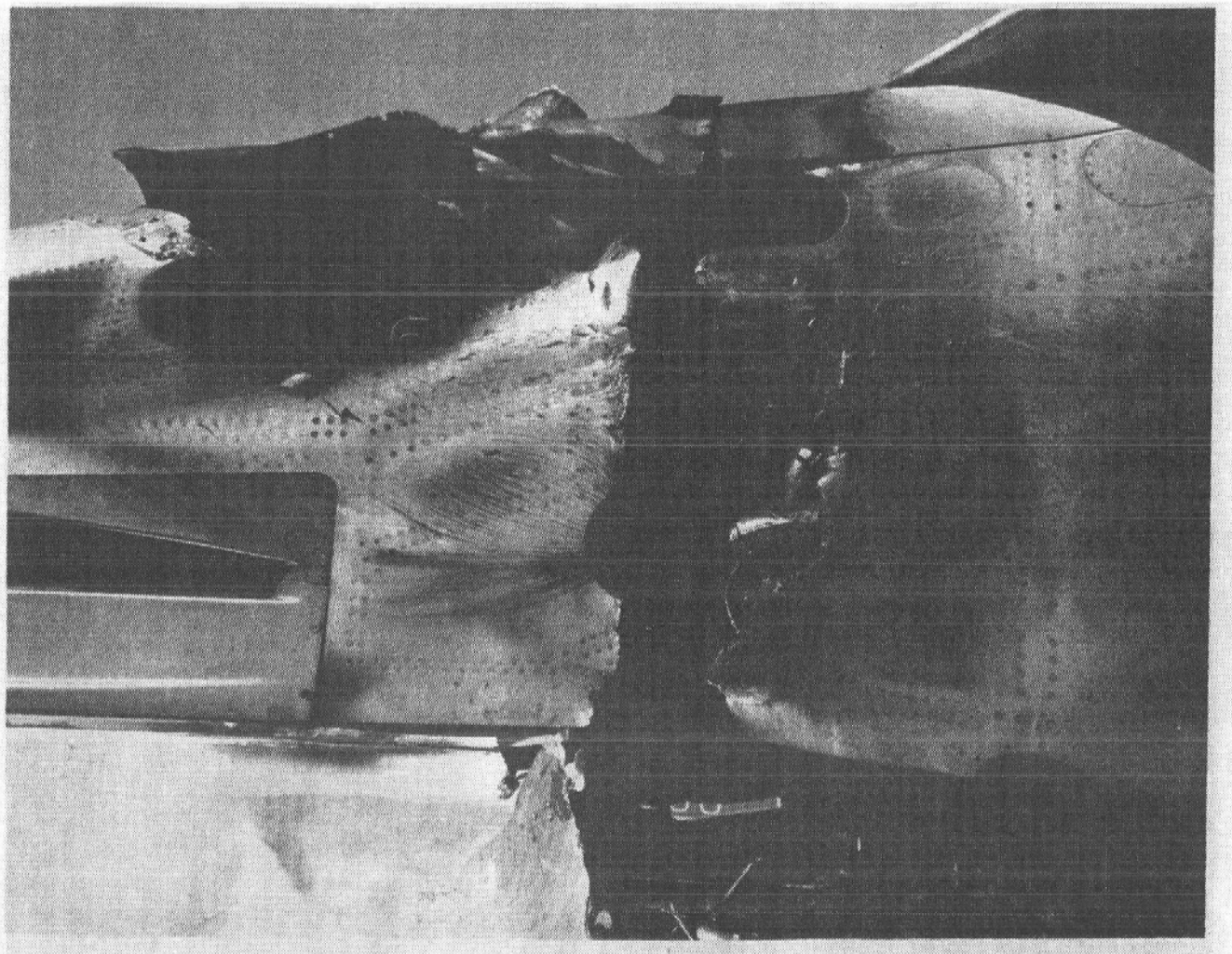
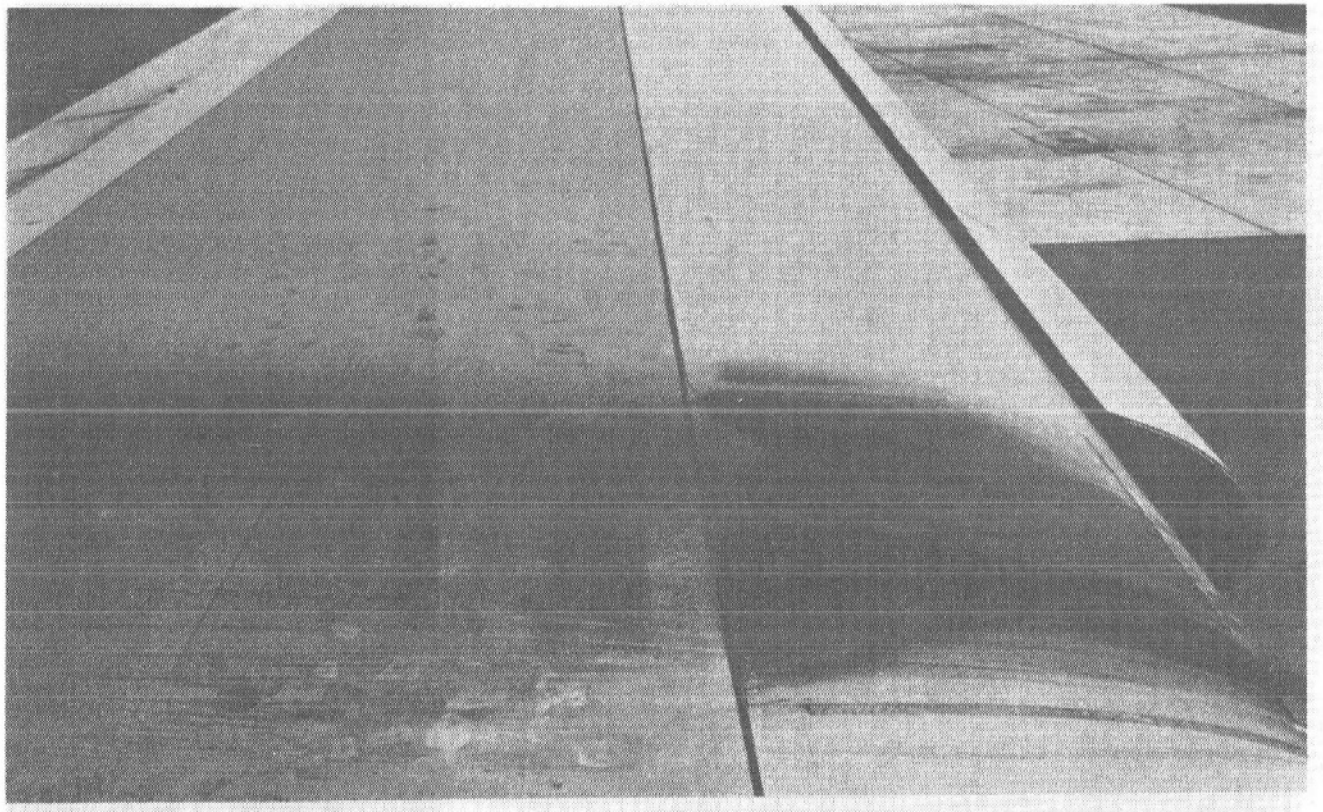


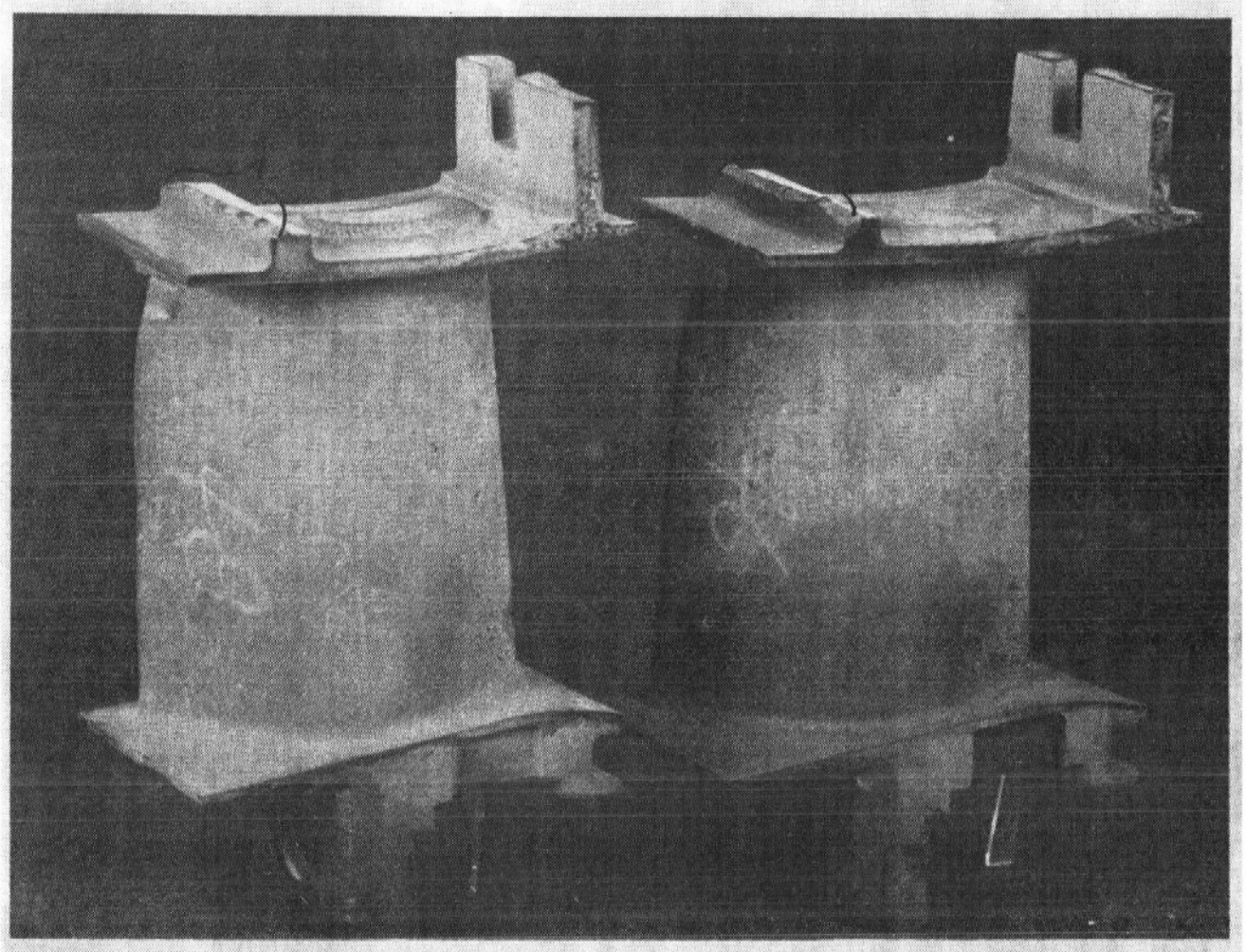
Photograph No. 1 Inboard side of No. 1 pylon - N743PA showing penetration damage between nacelle stations 196 and 208. (Note: Deflection of pneumatic duct and massive deformation of adjacent structure).



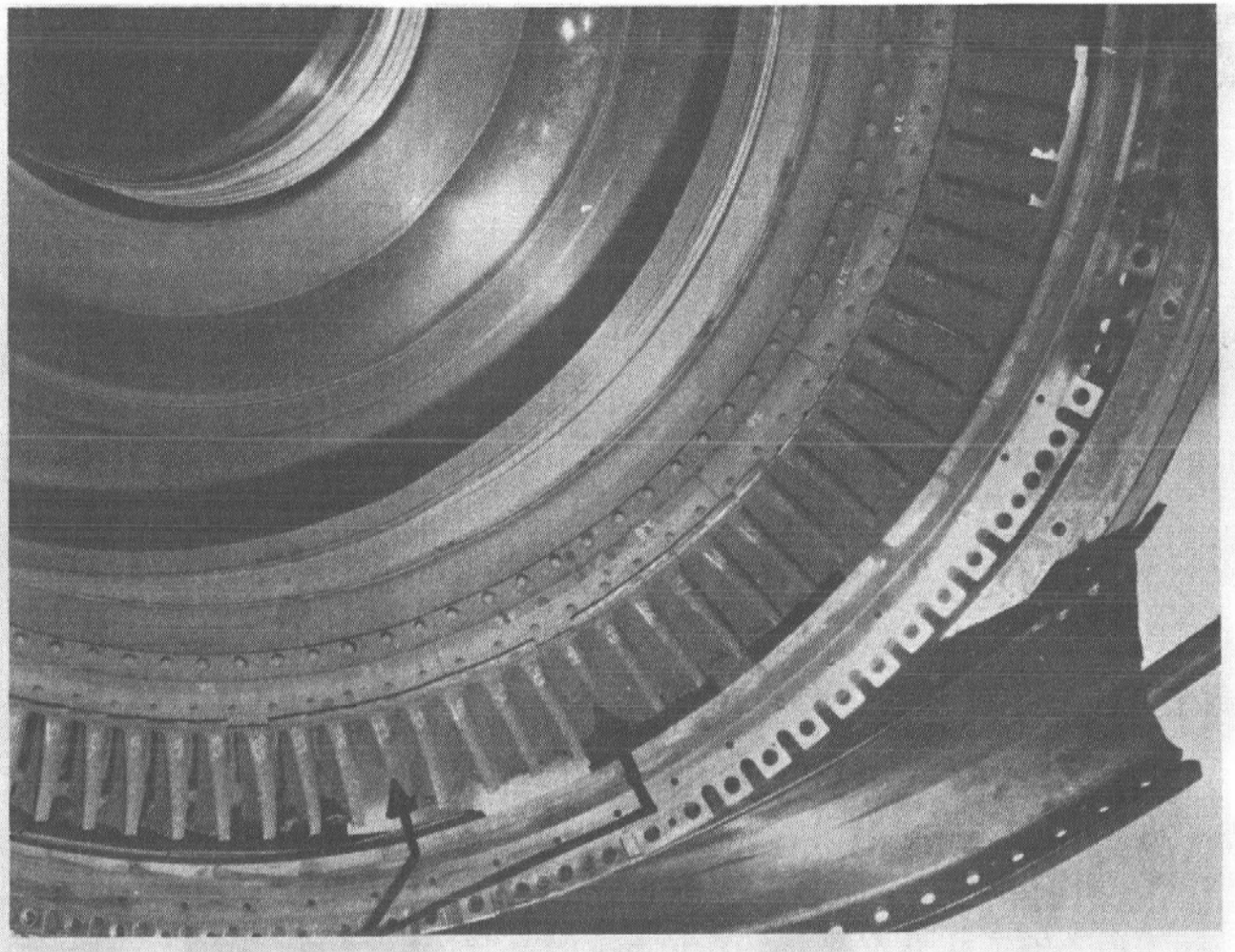
Photograph No. 2 Outboard side of No. 1 pylon, N743PA, showing penetration and fire damage between nacelle stations 196 and 208.



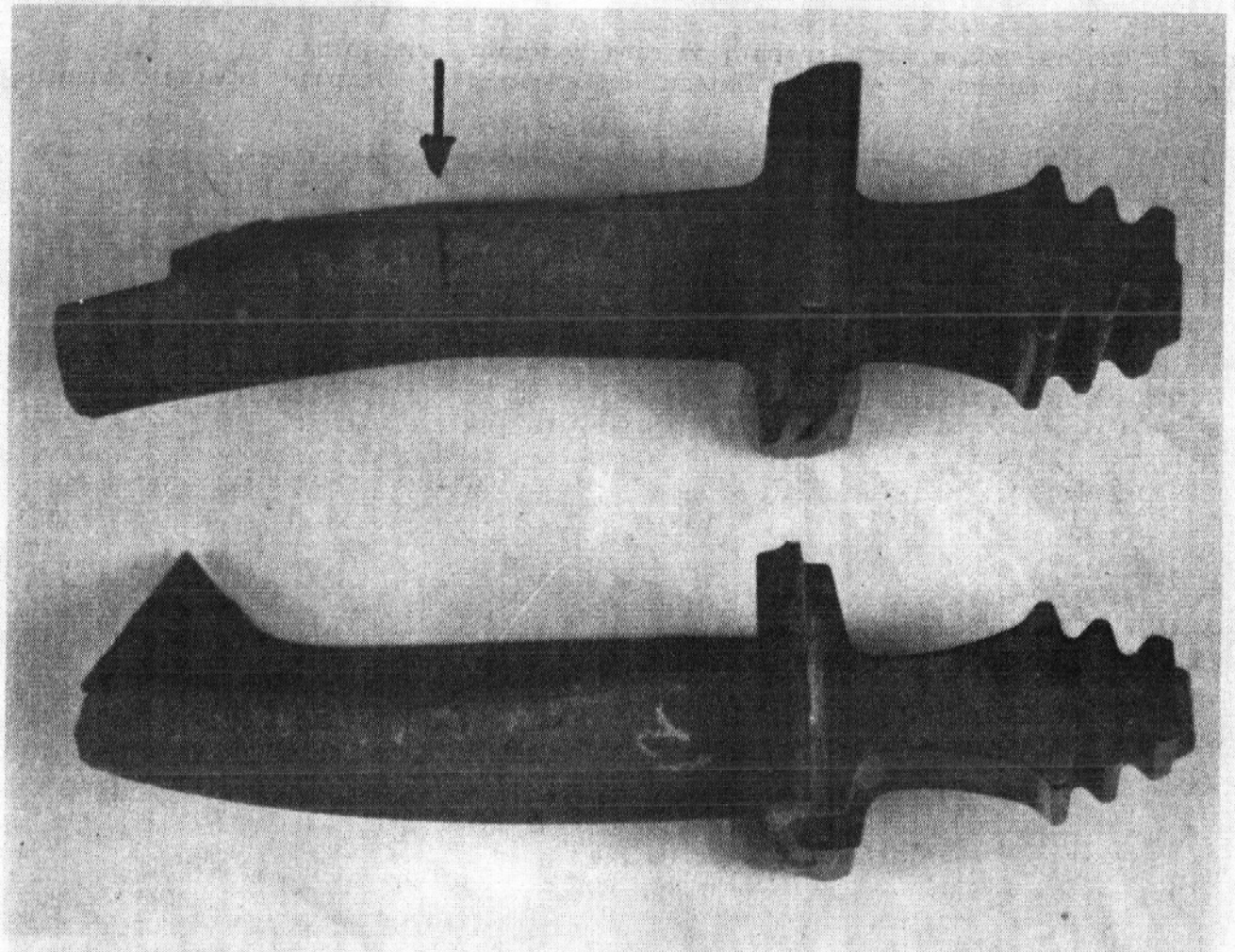
Photograph No. 3 Forward area of top of left wing. Aft of No. 1 engine and pylon showing fire damage. (Note: Wrinkling of leading edge panel).



Photograph No. 4 Second-stage turbine vane feet showing locations of vane foot fractures. (Note: Bracketed areas of vane feet at fracture surfaces).



Photograph No. 5 Quadrant of first-stage turbine wheel No. 1 engine N743PA showing two of the fractured turbine blades. (See arrows at top of photo).



Photograph No. 6 Two first-stage turbine blades showing airfoil bending and stress rupture crack of one blade. Blade on top of photograph shows partial crack in the same general area where complete fractures occurred.

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