File No. 1-0020

CIVIL AERONAUTICS BOARD

AIRCRAFT ACCIDENT REPORT

ADOPTED: September 28, 1966 **RELEASED:** October 5, 1966

PAN AMERICAN WORLD AIRWAYS, INC.
B-707-321B, N761PA
SAN FRANCISCO, CALIFORNIA
JUNE 28, 1965

SYNPOSIS

A Pan American World Airways, Inc., B-707-321B, N761PA, experienced an explosive disintegration of the third stage turbine disk of the No. 4 engine at approximately 1410 P.d.t., June 28, 1965. The accident occurred shortly after takeoff from San Francisco International Airport, San Francisco, California, at an altitude of about 800 feet above the ground. Disintegration of the turbine disk was followed by a fire in the No. 4 engine area and an explosion in the outboard reserve fuel tank. The No. 4 engine and approximately 25 feet of the right outer wing separated from the aircraft.

The fire was extinguished and a successful emergency landing was accomplished at Travis Air Force Base, California, with no injuries to the 143 passengers or 10 crewmembers aboard the flight.

The Board determines that the probable cause of this accident was the failure of the third stage turbine disk. This failure was caused by a transient loss of operating clearance between the third stage turbine disk and the third stage inner sealing ring. This loss of clearance resulted from a combination of improper turbine rotor positioning during engine assembly, the use of serviceable worn parts, and an operating clearance which was less than predicted in design analysis.

1. INVESTIGATION

1.1 History of Flight

Flight 843, a Pan American World Airways B-707-321B, N761PA, was a regularly scheduled international passenger operation. The flight departed San Francisco International Airport, San Francisco, California, at approximately 1409 P.d.t.[1/] June 28, 1965, for Honolulu, Hawaii.

Witnesses stated that the takeoff and initial climb were normal. Shortly after the climb was initiated, at an altitude of approximately 800 feet, ground witnesses observed a fire around the No. 4 engine. At this time the crew noted heavy vibration, and yawing of the aircraft. The crew initiated shut down procedures for the No. 4 engine until they received an intermittent fire warning signal. They then initiated engine firefighting procedures for the No. 4 engine. During this

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

period the right outboard (No. 4) engine and approximately 25 feet of the right outboard wing separated from the aircraft. After the fire was extinguished, control was assured, and the situation evaluated, the crew elected to proceed to Travis Air Force Base, California. They stated this decision was based on a partial hydraulic failure and the uninhabited approaches to Travis, plus the availability of military firefighting equipment and a longer runway.

After lowering the landing gear by emergency means, the flight landed at Travis at 1434 without further difficulty. Passengers evacuated the aircraft using the inflated emergency slides because no stairs were available.

The accident occurred about 1-1/2 miles west of the departure end of runway 28L at San Francisco International Airport, San Francisco, California, at an altitude estimated to be 800 feet m.s.l. The accident occurred during daylight hours.

1.2 Injuries to Personnel

Injuries	Crew	Passengers	Others
Fatal	0	0	0
Non-fatal	0	0	0
None	10	143	

1.3 Damage to Aircraft

The aircraft received substantial damage. The No. 4 engine separated from the wing, and approximately 25 feet of the right outboard wing separated from the aircraft. The remaining wing section received fire and puncture damage around the area of the wing separation line.

1.4 Other Damage

The main portion of the No. 4 engine struck and damaged a building approximately 1.8 miles from the departure end of runway 28L.

1.5 Crew Information

Captain Charles H. Kimes, age 44, possessed airline transport pilot certificate No. 38522-40 with multi-engine land and sea, DC-4, DC-6/7, C-46, B-377, and B-707 ratings issued May 16, 1958. His last proficiency check was completed May 21, 1965, and his last line check completed on June 17, 1965. Captain Kimes held an FAA first-class certificate issued April 29, 1965, without waivers. He had a total flying time of 17,736 hours of which 2,606 hours had been flown in B-707 aircraft. He had flown 228 hours in the preceding 90 days and 71 hours in the preceding 30 days. He had 5-1/2 days off duty, and had not flown in the 24-hour period preceding the accident.

First Officer Frederick R. Miller, age 48, possessed airline transport pilot certificate No. 34563, with multi-engine land, DC-3, DC-4, DC-7, B-377, and B-707 ratings issued April 18, 1961. He also held navigator certificate No. 1178229. His last proficiency check was accomplished April 23, 1965. He held an FAA first-class medical certificate issued May 14, 1965, with a limitation that "Holder shall possess correcting glasses for near vision while exercising the privileges of his airman certificate." First Officer Miller had a total of 17,027 hours flying time with 2,817 hours in B-707's. He had flown 61 hours in the last 30 days. He had 2-1/2 days off duty and had not flown in the 24 hours preceding the accident.

Second Officer Max A. Webb, age 46, held airline transport pilot certificate No. 211105 with multi-engine land and DC-6/7 ratings, issued January 4, 1961. He also held navigator certificate No. 1263016. Mr. Webb had a total of 13,826 hours flying time of which 35 were in the B-707. He had flown 72 hours in the last 30 days before the accident. His FAA second-class medical examination was completed October 21, 1964, with no waivers required. He had been off duty 2-1/2 days and had not flown in the 24 hours preceding the accident.

Flight Engineer Fitch Robertson, age 44, held flight engineer certificate No. 575029, and private pilot certificate No. 509935. His last FAA second-class physical examination was dated October 29, 1964, and no medical waivers were entered. Mr. Robertson had 17,753 hours flying time including 3,901 in B-707. He had flown 82 hours in the preceding 90 days and 69 hours in the 30-day period preceding the accident. He had been off duty 5-1/2 days and had not flown for 24 hours preceding the accident.

The two pursers and four stewardesses were regularly employed by the company and their emergency training was current.

1.6 Aircraft Information

N761PA was a Boeing 707-321B, manufactured in June 1962, with serial No. 18336. A current airworthiness certificate was issued June 15, 1962. The aircraft had flown a total of 12,789 hours at the time of the accident. The aircraft was powered by four Pratt and Whitney JT3D-3B engines.

Position	Date of Mfg.	S/N	Total Time	Time Since Overhaul	Date Instl.	Place Instled.	Time at Instl.
1	4/18/63	P6440 50B	5515	1363	2/15/65	NYC	0
2	12/16/63	P6442 33B	4149	4149	5/8/65	NYC	3652
3	9/16/64	P6446 67B	1817	1817	6/16/65	SFO	1665
4	1/2/62	P6432 50B	8204	39	6/25/65	NYC	0

The flight logs were reviewed for the period June 25-28, 1965, and no performance writeups were entered regarding the No. 4 engine. This engine had been removed from another aircraft due to a scheduled third stage turbine disk change and was given a full overhaul at this time. The overhaul included removal and replacement2/ of turbine blades, and replacing of the Nos. 2, 3, and 4 turbine disks. The three replacement disks were new Pratt and Whitney items with zero operating time. A number of nozzle guide vanes were removed and replaced in all

2/ All turbine blades were supplied from PAWA serviceable parts pool. If parts are pooled at disassembly there is no history of these parts, they are not serialed, or time controlled.

four stages of the turbine section. Ninety-five guide vanes, from the PAA serviceable vane pool, were installed in the third stage. The first and second stage turbine inner air seals were removed, inspected, the No. 2 seal reworked, and both seals were reinstalled in the engine. The third and fourth stage inner air turbine seals were removed, reworked, and the third stage seal replaced with a new part while the fourth stage seal was replaced with a reworked part. The front compressor drive turbine rotor shaft was replaced. All recorded measurements of clearances in the reassembled engine were within the manufacturer's prescribed limits. The engine was tested in an engine test stand, inspected, and approved for installation.

Following its installation on N761PA the engine operated for slightly over 39 hours with no reported discrepancies.

The center of gravity (c.g.) limits for this flight were 16-35 percent. The takeoff c.g. was computed to 30.2 percent MAC at a weight of 266,631 pounds. According to Boeing Company computations, after engine separation the gross weight was 258,159 pounds and the c.g. was 29.3 percent MAC. Following the loss of the outer wing section the weight was 255,154 pounds and the c.g. was 28.3. Landing weight was 249,538 pounds with a c.g. of 28.2 percent MAC. The weight and c.g. were within limits throughout the flight.

The aircraft had been serviced with 88,000 pounds of Jet A turbine fuel before takeoff. Approximately 82,000 pounds of fuel was aboard after landing.

1.7 Meteorological Information

The weather was clear and sunny, visibility 8 miles, temperature 77°F., and the wind from 310° at 15 knots.

1.8 Aids to Navigation

Not applicable.

1.9 Communications

Air to ground communcation was continuous although it was necessary for other aircraft to relay messages during part of the flight.

1.10 Aerodrome and Ground Facilities

Not applicable.

1.11 Flight Recorder

The flight recorder installed in this aircraft was a Lockheed model 109C. The flight recorder tape was examined and read out for the entire flight, approximately 24 minutes and 30 seconds. Approximately 39 seconds after the indicated lift-off time the heading trace shows a sharp yaw, with a 1 "g" excursion of the acceleration trace. About 63 seconds after lift-off momentary excursions of the acceleration trace from 0.2 to 1.75 ≠ "g" were noted with variations of the heading, airspeed, and altitude traces occurring at the same time. No other significant variations occurred. There was no mechanical damage to the recorder medium.

1.12 Wreckage

The engine components that separated from the aircraft were recovered from an area beginning 1.5 miles from the departure end of runway 28L, to 1.8 miles from the runway end. The wing components that separated from the aircraft were recovered from an area beginning 1.5 miles from the end of the runway to a point 8.1 miles from the runway.

Parts of the No. 4 engine turbine case, the third stage turbine disk, turbine blades, guide vanes and a piece of gap cover were found 1.5 miles from the end of the runway. The fourth stage turbine disk and blade assembly with hub attached were found 1.7 miles from the runway. Nearby was the aft reverser sleeve. Next, at 1.8 miles was the main portion of the engine, the forward part of the No. 4 strut, turbocompressor, and part of the cowling including the entire nose cowl.

Three miles from the end of the runway were two outboard aileron balance panel access covers. One-half mile farther, at 3.5 miles was the major portion of the right outer wing panel. At 3.7 and 3.9 miles were two more outboard aileron balance panel access covers. At 5.0 miles was the largest piece of lower wing skin (Sta. 866-948) with a 12-inch section of the aft spar bottom chord and web. From 6.2 to 8.1 miles were scattered pieces of burned lower wing structure.

The right outer wing panel separated near wing station 748 through the inboard end of the reserve fuel tank. The wing inboard of the fracture received only minor mechanical damage while the separated panel was extensively damaged.

The front spar separated at station 748. This fracture and those of the forward upper stringers were generally sharp, with 45-degree edges, and had the appearance of tensile failure. The rear spar and upper rear stringers were bent, twisted, and cracked spanwise in places. The rear spar was fractured at station 779 and its bottom chord was bent up and aft and was twisted clockwise, looking outboard.

The entire skin panel over the reserve tank was buckled upward from its normal position, with a maximum deflection of five inches. Every interspar rib between the tank ends was buckled up and outboard, and the rib-to-bottom stringer connections were all failed.

The fuel shutoff valve for the No. 4 engine was found in the closed position. It was bench-checked and operated properly without leaks.

On inspection of the recovered engine it was found that the fan inlet case had separated from the fan exhaust case. The fourth stage of the front compressor through to the turbine nozzle case remained together as an assembly. The rear thrust reverser remained attached to the exhaust case which had separated from the engine at the exhaust case front flange. The front compressor case and stages were generally bent and crushed at the top.

The rear compressor rotor was intact. In general, there was evidence of stator assembly rub from the tenth through the fifteenth stages. The rear compressor case was generally in satisfactory condition although dented.

The combustion chamber cases were intact but buckled in toward the top. The combustion chambers were in satisfactory condition except that the top chambers were collapsed in the rear. The turbine nozzle inner case was crushed inward through the top 90-degree segment. There was no evidence of overtemperature in the first stage nozzle guide vanes. The high pressure turbine shaft was fractured circumferentially through the forward breather holes to the forward spline rear fillet.

The first stage turbine disk was still intact and attached to the shaft. The disk was rubbed on the inner bore, the front web and rim, and the rear rim. A 90-degree segment of the first stage turbine blades, severly bent opposite to rotation and lacking tip shrouds, was heavily rubbed and notched on the leading edge. Twenty-four blades were missing and the remainder were broken off above the platform. The low pressure compressor turbine drive shaft was slightly bent approximately 17 inches from the forward end and circumferentially rubbed in the planes of the high pressure turbine shaft locknut, the No. 5 bearing, and the first stage turbine disk.

The front and rear face of the outer rail of the second stage inner sealing ring assembly was heavily rubbed and partially crushed. The second stage nozzle guide vanes were severely damaged along the trailing edges with no evidence of overtemperature.

The second stage turbine disk was intact except that half the rear mount flange was broken off. The tip shrouds were missing from all the blades and the fractures varied along the span of the blades. The front rim of the disk and leading edges of the blades were heavily rubbed.

The third stage turbine disk, installed new at the last overhaul, was notched 0.020 inch deep into the front web in the circumferential plane of the stationary third stage inner sealing ring. Apart from the missing web rim segment the remainder of the disk was intact, with the faces randomly rubbed and the tie rod holes slightly distended. The third to fourth stage spacer was axially split at a radial buckle and tie rod holes were distended and cracked. The fourth stage inner seal was intact with the rims severely distorted and the knife edges heavily rubbed. The fourth stage turbine disk was intact with slight rubbing on the rear web and rim. The blades were bent axially forward and opposite to rotation. The shrouds were missing as were nine of the blades and the leading edges of the blades were notched.

The third stage inner seal was axially and circumferentially split between the forward flange and the first knife edge, which was unrubbed, and partially split aft of the rear knife edge. The third stage inner sealing ring assembly, installed new at the last assembly, was distorted into three damaged pieces with the inner sealing ring separated from the support through the rivets. The outer rails of the support showed normal guide vane patterns and no abnormal wear was found. The inner sealing ring was torn apart, severely distorted, and gouged. The complete rear lip was particularly damaged and showed evidence of an overtemperature condition. Several small sections of the edge of the lip that were not obliterated did not display visible evidence of wear.

The third stage guide vanes were damaged mostly on the trailing edges with no evidence of overtemperature. The vane outer feet were bent outward and worn 0.008

inch. Normal contact patterns were evident in the inner center tang of the vanes. Several vanes were buckled and missing the inner platform. The fourth stage inner seal ring assembly was severely torn and twisted. Normal vane patterns were evident on the rail section. The separated inner sealing ring was torn apart and severely rubbed. The fourth stage nozzle guide vanes were damaged slightly on both the leading and trailing edges, most of the outer forward feet were broken, and several vanes were broken through the airfoil. There were varied temperature patterns visible on the guide vanes but no evidence of overtemperature. The lock rings for the nozzle guide vanes were recovered, severely distorted, as well as the outer seals which were also severely rubbed and distorted.

The turbine housing assembly was circumferentially gouged around the inner surface and the turbine exhaust case was collapsed in the horizontal plane and the interior of the case was severely scored and peened. The upper rear half of the turbine nozzle case was split open from the plane of the third stage turbine disk aft to the rear flange, and forward to the first stage turbine rotor.

1.13 Fire

There was evidence of fire on the separated wing section, the wing around the point of separation, and on the No. 4 engine. Fire was observed by ground witnesses, passengers and crewmembers, and photographed, in color, from the ground and by a passenger.

The flight crew was alerted to the fire when an intermittent fire warning was observed while they were going through the engine shutdown procedure following the failure of the No. 4 engine. The first officer then actuated the fire selector lever for the No. 4 engine and discharged both fire extinguisher bottles to the engine. The fire was observed to go out and did not recur although fluid was observed streaming from the right wing for the duration of the flight. Fuel was still streaming from the right wing, after landing, from the No. 4 tank area until the fire department plugged the hole in the bottom of the tank. The area around the fuel spill and the wing stub were foamed as a preventive measure while the passengers were debarking from the aircraft.

Examination of the fire extinguisher bottles in the right wing revealed that they were both empty and the discharge indicating disk was missing while the thermal discharge indicating disks were intact. The fire extinguisher line which runs along the forward face of the right front spar was severely chafed and worn through at about front spar station 653. The chafed area was on the bottom of the line where it contacts the front spar bottom cap, and was 6-1/2 inches long and 5/16 inche wide at its widest point. A hole in the chafed area was 1 inch long and 1/8 inch wide. Aside from this chafed area the line was intact to a point 45 inches beyond the No. 4 pylon firewall where it separated from the engine plumbing. The No. 4 engine fire detector wiring was in the No. 4 pylon wire bundle and was charred in the upper pylon area, aft of the pylon firewall. The extremity of the wire was stripped of insulation and exposed to the airstream.

Fire damage to the No. 4 engine consisted of sooting of the accessory drive gear and a heat pattern in the left side cowl panel beginning 12 inches from the top of the cowl, at the leading edge, extending diagonally down and aft to a point 8 inches above the surge bleed door. This heat pattern was about 12 inches wide and consisted of molten metal on the inside of the cowl and soot on the outside.

There were three distinct fire patterns on the part of the wing which remained on the aircraft. (See Attachment No. 1.) One originated on the inboard gap cover area, followed the outboard nacelle upper surface support outboard to the trailing edge of the upper surface, and inboard to the inboard end of the outboard flap. The wing skin over all three fuel tank vent lines in this area was burned away or cracked for lenghts ranging from 44 inches over the forward vent line to 26 inches over the aft vent line. In this same area there were three holes, located in buckled skin, through the upper skin over the No. 4 main fuel tank.

The second fire pattern was in the area of the wing separation outboard of station 733. This fire consumed much of the portion of the aft spar remaining outboard of the production break and the centers of the access doors remaining with the attached lower skin. The fracture surfaces inboard of this area were extensively burned and metal sprayed while those on the separated panel had little heat damage and no soot deposits.

The third fire area was defined by heavy sooting and/or consumption of much of the outboard skin and ribs along the entire length of the nacelle strut trailing edge fairing. Finally, metal droplets were found on the right horizontal stabilizer and on the outboard three wing vortex generators in the aft row.

The outboard wing section that separated from the aircraft was extensively burnt and sooted. There was a ground fire where the wing section came to rest and the fire department reported using 250 gallons of water to extinguish it.

1.14 Survival Aspects

Prior to takeoff the stewardesses gave a passenger briefing and demonstration of the lifevests and oxygen masks and indicated the location of the emergency exits.

Following the loss of the engine and the outboard wing the passengers were briefed on ditching procedures and instructed to don their lifevests. After the decision was made to land at Travis AFB, the passengers were briefed by the cabin crew on preparation for an emergency landing including instructions on removal of shoes, sharp objects, etc., and the "braced position" to be assumed for the landing. Male passengers were placed at the emergency window exits and others chosen to assist with the evacuation slides.

The passengers assumed the braced position for the landing on instructions from the second officer who took charge of the cabin. The second officer and the cabin attendants opened the main entry doors and the galley service doors and, due to a lack of loading stairs, extended the evacuation slides. The captain stated that, to insure an orderly debarkation, he delayed the movement of the passengers until he was certain all stations were prepared. The debarkation was reported as orderly and took about four minutes from the time the aircraft stopped.

1.15 Tests and Research

Portions of the third stage turbine disk from the No. 4 engine were examined by the National Bureau of Standards and the Board's metallurgist. This examination revealed a circumferential fracture approximately two inches from the bottoms of the blade retention slots. This fracture occurred through a groove approximately

0.020/-0.025 inch deep. Radial fractures occurred at each end of both the pieces examined. There were numerous cracks in the groove that appeared to be typical of brittle, intergranular cracks that occur while metal is at a high temperature. The surfaces of the groove and a portion of the circumferential fractures, about 0.1 inch, adjacent to the groove were coated with a black, high temperature oxide scale. Distinct temper colors on the circumferential fracture started at the edge of the black scale and extended to the backface of the disk. Temper colors on the radial fractures extended 0.6-0.9 inch from the circumferential fracture. These temper colors indicate that an abnormal temperature gradient existed at the time the fractures occurred. The maximum temperature was at the groove and the gradient extended through the disk to the backface and to about one inch radially from the groove.

The circumferential fracture surface, near the groove, showed evidence of a brittle intergranular type separation, while the fracture adjacent to the backface of the disk showed evidence of a tensile shear failure. There was no evidence of fatigue or corrosion.

Examination of the microstructure of the metal on a section through the groove, and the circumferential fracture, revealed evidence of a heat affected zone in the metal adjacent to the groove. The temperature in this zone had been high enough to change the etching characteristics and reduce the hardness of the metal. The evidence indicated that the metal had been heated to $1,700^{\circ}$F. or higher. Hardness testing of the metal 0.1 inch from the groove was found to be about 149-184 on the Brinnell[3/] scale. Hardness testing of disk material remote from the groove area gave Brinnell readings of 336-362 compared to the specification of 302-388 Brinnell.

The engine, less exhaust case, was returned to the manufacturer where it was disassembled and examined under Board supervision. Special attention was given to the operating position of the low pressure turbine assembly. The wear pattern was measured on the low pressure compressor turbine drive shaft and found to be 3.233-3.257 inches long. Measurements of similar drive shafts in the Pan American jet overhaul base ranged from 3.120-3.212 inches with an average of 3.159 inches. Comparative measurements of the wear pattern on 3 drive shafts in Pratt and Whitney production engines ranged from 3.151-3.170 and averaged 3.162 inches.

The three knife edges on the second stage turbine seal left only two tracks on the shoulder of the low pressure compressor turbine drive shaft beginning 0.350 inch aft of the front shoulder. A survey of similar drive shafts revealed three tracts on the shoulder. Tracks on the No. 4-1/2 seal liner of the accident engine were compared to a production engine and estimated to be 0.250±0.040 inch forward of similar marks on the production engine. The third and fourth stage inner sealing ring track marks were comparable to similar marks on other engines.

A review of the carrier's jet engine buildup procedures for this engine was conducted, and depositions taken from the personnel who assembled the engine. The low pressure turbine shaft was installed twice in the course of determining the correct size shaft positioning spacer. This spacer determines the actual

3/ Brinnell hardness numbers are determined by pressing a hard steel or carbide ball of specified diameter into the material being tested. The applied load in kilograms is divided by the surface area of the resulting indentation expressed in millimeters. Therefore, the larger the Brinnell reading the harder the material tested.

position of the low turbine shaft assembly. Measurements and calculations were performed to determine the correct size of this spacer. The calculations for determining the turbine shaft positioning spacer were not recorded, nor was the size of the installed spacer. The mechanic who did this work testified that the spacer was within tolerance but on the low side. Measurement after the accident showed this spacer was 0.258. The blueprint specifications for the spacer was 0.256 to 0.366 inch.

The following shift removed the turbine rotor to adjust the position of the second stage turbine outer air seal, and then reinstalled the same rotor. The second stage turbine outer air seal was installed using a pusher tool with ratchet adapters rather then knurled nuts. The clearance from the rear of the second stage turbine disk to the rear flange of the turbine nozzle case was not measured and the third stage nozzle vanes were assembled. The shop work card used for this operation stated in item 21C "Repeat the position measurement, if the rotor was removed and reinstalled." This was not done by the second shift.

A measurement was taken from the turbine nozzle case rear flange to the inner sealing ring to establish that adequate clearance existed between the third stage nozzle vanes/inner sealing ring assembly and the third stage disk. It was performed after the inner sealing ring was pulled rearward by use of "hand force" in accordance with the approved overhaul manual.

The inspector on duty during the reinstallation of the turbine rotor testified that he signed off work he had not inspected. He also lined-out a mechanic's initials placed on the work card to indicate that work had been accomplished on the previous shift. He later voided this line-out when the work was reaccomplished by a different mechanic.

The engine was run on a test stand without any reported discrepancies and was installed on N761PA two days later, operating 39.21 hours without a reported discrepancy until the accident occurred.

Laboratory tests of fuel samples taken from the six remaining fuel tanks on the aircraft revealed no significant deviation from the specification established for a Jet A turbine engine fuel. It was estimated that the fuel temperature in the tanks of N761PA at the time of the accident was between 70° and 80°F. The flammability limit of Jet A fuel was reported by the FAA to be from 90-170°F. This figure does not take into account variations to the flammability limits of fuel in a tank due to volume, size, shape, agitation, and other factors that affect a fuel tank vapor space when the aircraft is in flight.

2. ANALYSIS AND CONCLUSIONS

2.1 Analysis

The number three turbine disk in the No. 4 engine of N761PA failed due to a localized reduction in its cross-sectional area and overheated conditions due to rubbing between the turbine disk and the third stage turbine inner sealing ring immediately forward of the disk. This rubbing was the result of a transient loss of clearance between these parts on takeoff. The maximum difference in the thermal expansion rates of the rotating assembly and the outer turbine cases which support the inner sealing ring occurs 1-2 minutes after application of takeoff power.

The disk failure resulted in an explosive failure of the No. 4 engine and its separation from the wing due to high vibration and out of balance oscillation of the rotating parts of the engine. The right outer wing received so much damage to the lower load bearing skin and structure that the capability of the wing to sustain in-flight loads was reduced below the loads imposed and the outer wing panel separated from the wing. Fuel from the engine fuel line was then being pumped directly into the airstream. This fuel was ignited by an undetermined source shortly after the engine separated and resulted in an explosive separation of a portion of the lower wing skin. The fire was sustained by the continued supply of fuel through the engine fuel line until the flight engineer or the first officer shut off the main fuel supply either by activating the fuel shutoff valve to the closed position or actuating the fire selector handle.

There was no evidence of foreign object damage to the compressor before the engine failed. The damage incurred by the compressor section of the engine was caused by impact following separation from the aircraft or was the result of turbine disintegration. There is no evidence in the combustion chambers, or on the turbine vanes or blades, that would reflect a critical overtemperature operation of the engine.

The notch on the face of the third stage turbine disk was caused by the third stage turbine inner sealing ring, which rubbed the disk during takeoff due to insufficient clearance. The minimum clearance occurs during takeoff thermal transients approximately 1-2 minutes after applying takeoff thrust.

The small depth of the notch in the turbine disk indicates that the interference was small and did not occur continuously through the 39 hours of engine operation prior to engine failure.

The third stage turbine inner sealing ring was rubbing the disk immediately prior to the disk failure. The .020 inch notch and the .125 inch annealed zone were not sufficient to cause the disk to fail at normal operating takeoff metal temperatures. The metal temperature through the web of the disk adjacent to the annealed zone had to be higher than the normal operating temperatures to allow the elevated temperature tensile shear failure that occurred. Therefore, the sealing ring was rubbing and generating heat through the web of the disk at the time of the disk failure. The disk had been manufactured to the proper specifications, dimensions, and hardness. There was no evidence of fatigue or corrosion in the fracture surfaces. Further, there was no evidence of an overspeed. The failure of the disk was associated directly with the notch, the adjacent annealed zone, and the increased temperature associated with the rubbing of the sealing ring against the disk web, which caused the disk to rupture. The entire outer web-rim section did not separate from the disk because the notch did not provide a continuous plane of maximum stress. As sections of the web-rim separated the induced bending stresses shifted the propagating crack out of the notch toward the rim.

The low pressure turbine was running too far forward based on the wear evidence observed on the spline, the second seal track, and the positioning spacer dimension of 0.258 inch measured after the accident. This spacer size is on the low side when compared with the computed spacer requirements taken after the accident and calculations performed by the manufacturer.

The cause for this apparent mispositioning lies in the assembly of the engine during overhaul. Certain of the overhaul shop procedures could have resulted in the mispositioning of the turbine rotor although the individual procedure causing it cannot be accurately pinpointed. The shop practices in question involved reinstalling the turbine rotor without position measurements and installing the turbine inner sealing ring without depth measurements. Furthermore, the use of a rachet wrench to position the outer turbine air seal can crimp the damping spring and cause improper seating of the seal and nozzle guide vanes. The inspector signoff of the work sheets for the low pressure turbine was improper and precluded any assurance that the engine was properly assembled in that section.

In addition to the loss of clearance due to the apparent mispositioning of the turbine rotor, some loss of clearance may also have been introduced by adherence to the assembly procedure specified in the approved engine manufacturer's overhaul manual. To insure correct positioning of the inner sealing ring a depth measurement was taken from the turbine nozzle case rear flange to the inner sealing ring. In taking this measurement, the third stage vanes and inner sealing ring were pulled rearward by "hand force" as specified in the approved overhaul manual for turbine rotors installed in the horizontal position. Tests conducted after the accident have shown that the "hand force" is not always adequate to insure seating of the most adverse combination of new and worn guide vanes. Lack of seating could result in a measurement which was not indicative of the true position of the inner ring.

Also, laboratory deflection tests performed after the accident showed that the deflection of the inner sealing ring caused by the use of vanes worn 0.007 inch or more was greater than previous analytical studies predicted by the engine manufacturer.

Another factor determined by tests conducted after the accident was that the actual operational clearance between these parts was less than that predicted analytically for a properly assembled engine.

The exact cause of the loss of operating clearance could not be defined, but apparently the cumulative effects of the normal actual clearance which was less than that predicted in the design analysis, worn vanes and the resultant added deflection of the inner sealing ring, and a forward positioned turbine rotor associated with assembly procedures resulted in the loss of sufficient transient operating clearance.

The disintegration of the third stage turbine disk cut the engine in two pieces and threw turbine chunks into the wing inboard of the engine pylon. The two engine sections, each supported by only one mount on the strut, began to oscillate and separated from the wing in approximately four seconds. The strut failures were caused by the oscillation, possibly coupled with mechanical damage from flying engine parts. The engine fuel line pulled from the strut closure rib when the engine separated from the wing. Fuel was pumped through this line for an estimated 99 seconds at a rate of approximately 30,000 pounds per hour, until the fuel valve was shut off by the action of either the first officer or the flight engineer. A second fuel source was the fuel line on the forward face of the main spar which had a loosened fitting that leaked and supplied fuel for a fire over the strut center spar between the front spar and the nacelle closure rib. A third possible fuel source was the ruptured slat hydraulic line in the inboard gap cover area.

The source of ignition cannot be determined but the possible sources included, the engine exhaust, hot turbine parts, or arcing from exposed electrical leads. The latter is the most probable source because there was an appreciable time lapse between observation of the fuel spray and ignition. These fuel sources wetted much of the upper wing surface before ignition occurred.

The fact that the No. 4 main tank was full of fuel probably prevented more extensive fire damage to that area of the upper wing surface because the fuel acted as a heat sink. The fire in this area reached temperatures ranging from approximately 870°-1165°F., based on the damage incurred to the metal.

The damage to the right outboard wing panel top and bottom skin and ribs could only have been caused by an over-pressure in the reserve tank. This is demonstrated particularly by the manner in which the lower skin separated from the aircraft. The entire panel was forced straight down, taking the attaching flanges of both spars with it. This is plainly the result of a low order[4/] explosion. The source of ignition for this explosion could not be determined but could have been autoignition, burn-through, or hot point ignition from a localized hot spot.

The final separation of the wing followed the explosion in the reserve tank. The wing separation is not believed to have been simultaneous with the explosion. The indications of yaw and vertical oscillation on the flight recorder readout and the location of the wreckage on the ground indicate that the wing panel remained on the aircraft approximately 10-11 seconds after the separation of the lower skin panel.

The heat damage to the wing structure was not considered to have been a major factor in the wing failure. Rather, the loss of the lower skin panel, stringer, and spar chord flanges reduced the load carrying capability of the wing below that required to support a 1 "g" condition, thus leading to the failure.

2.2 Conclusions

(a) Findings

(1) Weather was not a factor in this accident.

(2) Air Traffic Control procedures were not a factor in this accident.

(3) The aircraft and crew were properly certificated for the flight.

4/ Used in this sense, low order indicates that the pressure wave moved subsonic velocity.

(4) Except for the failure of the captain to sign the flight plan and clearance form the flight was properly dispatched.

(5) Weight and balance was not a factor in this accident.

(6) Approximately 39 seconds after takeoff the No. 4 engine experienced a catastrophic failure resulting in separation of the engine from the wing.

(7) The loss of the engine resulted in mechanical damage to the wing and a severe fire.

(8) The fire triggered a low order explosion in the No. 4 reserve tank which resulted in the loss of the lower wing skin, lower stringers, and spar chord flanges.

(9) The loss of these components resulted in a loss of wing integrity which allowed the outer wing panel to fail and separate from the wing.

(10) The ensuing fire was extinguished by the closing of the main fuel shutoff valve either by the first officer or the flight engineer.

(11) The engine was assembled without proper measurements being made to insure correct axial positioning of the low pressure turbine rotor assembly.

(12) The low pressure turbine was positioned too far forward decreasing the clearance between the third stage turbine disk and the third stage turbine inner sealing ring.

(13) The "hand force" assembly procedure and the use of a combination of new and used vanes could result in a measurement which is not indicative of the true, inner sealing ring position. Thus, the added deflection of the used vanes could result in a more rearward running position of the inner sealing ring than was anticipated.

(14) Laboratory tests conducted since the accident have shown that the actual running clearance was less than the engine manufacturer's analytical design prediction.

(15) The cumulative results of the assembly mispositioning of the rotor, the reduced actual operating clearance, and the additional rearward deflection of the inner sealing ring caused by the use of worn vanes resulted in the sealing ring contacting the third stage turbine disk.

(16) The rub occurred only during the period of maximum thermal transient 1-2 minutes after the application of takeoff thrust.

(17) The rub induced a notch in the face of the turbine disk which caused a localized reduction in the cross-sectional area, and also produced a heat weakened annealed zone.

(18) During takeoff operation the rub was occurring and the disk ruptured causing a catastrophic failure of the engine.

(b) Probable Cause

The Board determines that the probable cause of this accident was the failure of the third stage turbine disk. This failure was caused by a transient loss of operating clearance between the third stage turbine disk and the third stage inner sealing ring. This loss of clearance resulted from a combination of improper turbine rotor positioning during engine assembly, the use of serviceable worn parts, and an operating clearance which was less than predicted in design analysis.

Corrective Action

Following this accident PAA personnel testified that inspection procedures ha been changed to require duplicate measuring and calculations of spacer sizes and clearances by the mechanic and the inspector, working separately. Additionall the PAA maintenance manual and work card was changed to reflect this requirement.

A Pratt and Whitney witness testified that a revision in the procedures for the assembly of their engines was made to preclude the possibility of future disk failures.

BY THE CIVIL AERONAUTICS BOARD:

/s/ CHARLES S. MURPHY
Chairman

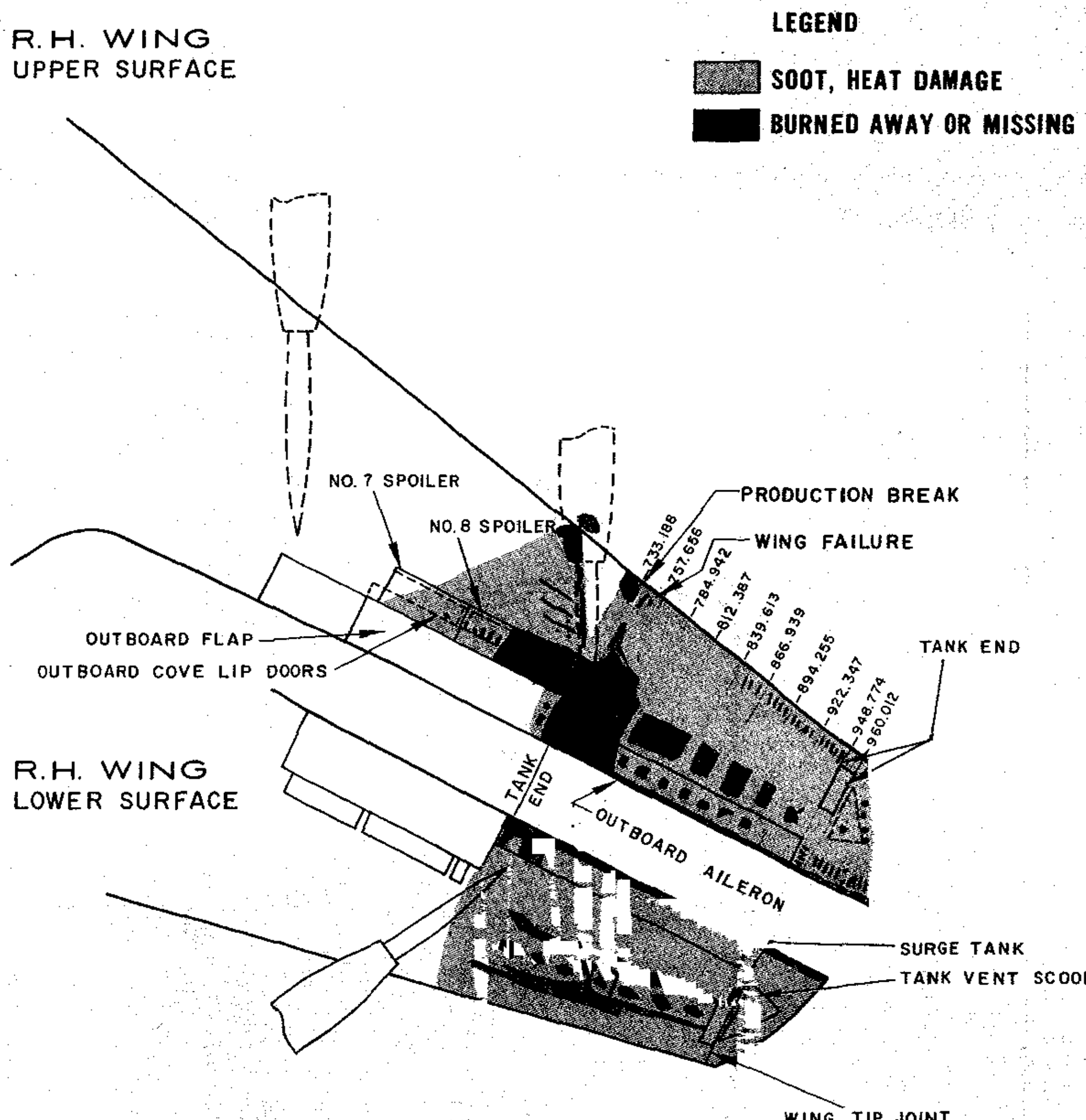
/s/ ROBERT T. MURPHY
Vice Chairman

/s/ G. JOSEPH MINETTI
Member

/s/ WHITNEY GILLILLAND
Member

/s/ JOHN G. ADAMS
Member

ATTACHMENT 1

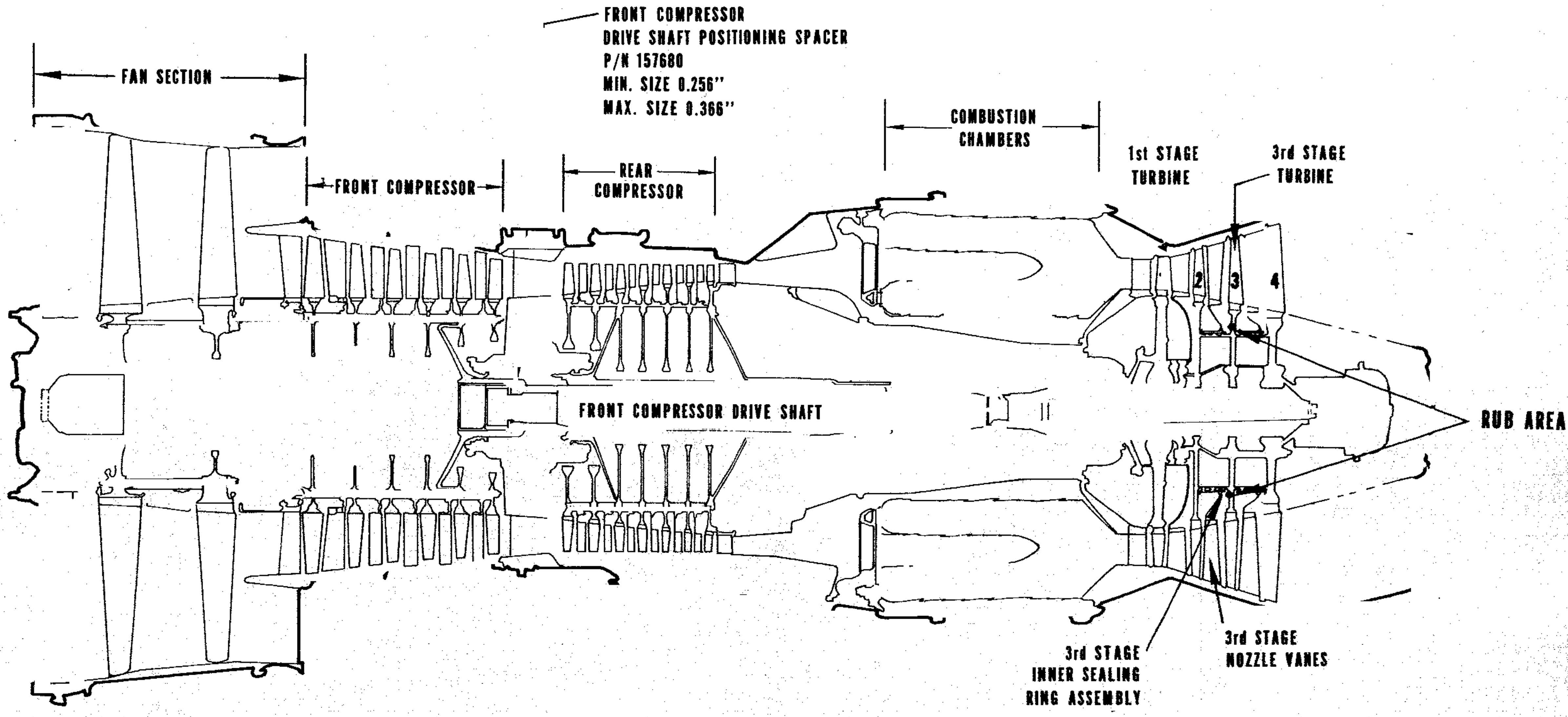


C. A. B.

BUREAU OF SAFETY

FIRE PATTERN CHART
PAN AMERICAN WORLD AIRWAYS
BOEING 707, MODEL 321 B, N761PA
SAN FRANCISCO INTERNATIONAL AIRPORT
JUNE 28, 1965

W. E. L. - 8/25/65



SIMPLIFIED ENGINE DRAWING