NATIONAL TRANSPORTATION SAFETY BOARD
OFFICE OF AVIATION SAFETY
WASHINGTON, D.C. 20594

March 31, 1997

POWERPLANTS GROUP CHAIRMAN'S FACTUAL REPORT OF INVESTIGATION

NTSB ID No.: DCA97MA017

A. ACCIDENT

Location:	Monroe, Michigan
Date:	January 9, 1997
Time:	1554 eastern standard time (EST)
Aircraft:	Embraer EMB-120RT, N265CA, Comair, Flight No. 3272

B. POWERPLANTS GROUP

Gordon J. Hookey Group Chairman	National Transportation Safety Board Washington, D.C.
Karen M. Grant	Federal Aviation Administration Burlington, Massachusetts
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Martin P. Sezack	Hamilton Standard Windsor Locks, Connecticut
Dan Ford	Air Line Pilots Association Florence, Kentucky
John T. Rice	Air Line Pilots Association College Park, Georgia
Paul Picton	Greenwich Air Services-Texas, LP[1] Fort Worth, Texas
Umberto Irgang	Embraer S. José dos Campos, Brazil
Manuel Monteiro	Embraer Fort Lauderdale, Florida
Charles E. Burger	AeroControlex Richmond Heights, Ohio

C. SUMMARY

On January 9, 1997, at about 1554 eastern standard time (EST), an Embraer EMB-120RT airplane, N265CA, operated by Comair as flight No. 3272, crashed into a field in Monroe, Michigan. The airplane was powered by two Pratt & Whitney of Canada (PWC) PW118 turboprop engines, which had Hamilton Standard14RF-9 propellers installed. The airplane departed controlled flight after descending and leveling off at 4,000 feet, reducing its speed to 150 knots, and banking to the left. The airplane was operating on an instrument flight rules (IFR) flight plan under the provisions of Title 14 Code of Federal Regulations (CFR) Part 135 as a regularly scheduled flight from the Greater Cincinnati Airport to Detroit, Michigan. The airplane was destroyed by impact forces and fire. The 25 passengers, 1 infant, and 3 flight crewmembers were fatally injured.

[1] Greenwich Air Services-Texas, LP, had acquired Aviall, which had accomplished the last overhaul on the No. 1 engine.

The Powerplants Group commenced the on-site examination of the engines and propellers on January 10, 1997. The left and right engines and propeller assemblies were found in the craters on the left and right side, respectively, of the crater made by the fuselage. All eight propeller blades were recovered at the crash site. The right propeller hub was embedded into the ground deeper than the left propeller hub. The left and right engines did not have any indication of in-flight fire, case rupture, or uncontainment. The examination, excavation, and removal of the engine and propellers from the crash site was completed on January 14, 1997. The Powerplants Group completed its on-site activities on January 16, 1997.

The disassembly of the propeller assemblies commenced on January 21, 1997, at Hamilton Standard, Windsor Locks, Connecticut, in the presence of the Powerplants Group. The examination of the propeller blades showed all of the blade spars were bent away from the direction of propeller rotation. There was no indication that any of the eight propeller blades had any preexisting damage. Measurements taken during the disassembly showed that the propeller blade pitch angles were within the governing range. The servo piston and sleeve assemblies and pitch change actuators did not show any indications of any operational distress. The disassembly of the propellers was completed on January 23, 1997. The Powerplants Group reconvened at Hamilton Standard's facility on March 6, 1997, to examine the left and right propeller deicing boots and propeller hub bulkhead slip rings. The examination, to determine if the deicing boots were operational at the time of impact, was inconclusive.

The disassembly of the engines commenced on January 28, 1997, at the PWC facility at Lonqueuil, Quebec, Canada, in the presence of the Powerplants Group. The damage that was noted to both of the engines was consistent with engine rotation at the time of impact. There was no indication on either engine of any in-flight fires, case ruptures, uncontainments, or pre-impact distress or damage. The disassembly of the engines was completed on January 31, 1997.

The left and right auxiliary electric pumps (propeller feather pumps) were disassembled at the AeroControlex facility, South Euclid, Ohio, on February 4, 1997, in the presence of the Powerplants Group. The examination of both pumps did not show any indication of rotation on either of the pumps or electrical motors at the time of impact.

D. DETAILS OF INVESTIGATION

1.0 Powerplant Information

1.1 Powerplant History

Comair records indicated the following powerplant-related components were installed on EMB120RT airplane, N265CA. The part numbers and serial numbers of the powerplant components were verified during the on-site examination or the disassembly of the respective components, except as noted.

	LEFT ENGINE	RIGHT ENGINE
Serial number	115483	115576
Total time (hours)	12,621.4	11,776.0
Total cycles	12,836	11,748
Time since overhaul	1,496.0	30.5
Cycles since overhaul	1,560	29
Last overhaul agency	Aviall	Lufthansa

	LEFT PROPELLER			RIGHT PROPELLER		
	Part No.	Serial No.		Part No.	Serial No.	
Propeller assembly	790100-8	890323		790100-8	910528	
Hub	782501-6	1735		782501-6	1184	
Actuator	790201-3	901258		790201-3	891102	
Propeller control unit	782490-24	910364		782490-46	850730	
Propeller blade No. 1	RFC11AA1-6A	887792	L4[2]	RFC11N1-6A	872581	R1
Propeller blade No. 2	RFC11AA1-6A	887782	L3	RFC11U1-6A	868008	R2
Propeller blade No. 3	RFC11AA1-6A	887779	L2	RFC11N1-6A	860455	R3
Propeller blade No. 4	RFC11AA1-6A	887784	L1	RFC11N1-6A	872093	R4

	LEFT AUXILIARY ELECTRIC PUMP	RIGHT AUXILIARY ELECTRIC PUMP
Part No. (Ham Std)	802248-3	802248-3
Part No.(AeroControlex)	4122-009006	4122-009006
Serial Number	195	086
Total time (hours)	13,599.1	14,688.5
Total cycles	13,816	15,310
Delivered to Comair	6/30/91 (a/c SN 245)	12/11/90 (a/c SN 217)

The left and right propeller control unit (PCU) and left and right auxiliary electric pump part numbers and serial numbers could not be confirmed because the respective data plates were not recovered at the crash site.

For more detailed information about the history of the powerplant components, refer to the Maintenance Records Group Chairman's Factual Report of Investigation.

[2] Numbers that were marked on the blades during the on-site examination.

1.2 Powerplant description

The PW118 engine is a twin-spool turboprop engine with a free turbine that is made by PWC. The engine features a reduction gearbox module and a turbomachinery module, which has a 1-stage centrifugal flow low pressure compressor (LPC), 1-stage centrifugal flow high pressure compressor (HPC), reverse flow annular combustor with 14 fuel nozzles, 1-stage axial flow high pressure turbine (HPT), 1-stage axial flow low pressure turbine (LPT), and a 2-stage axial flow stage free turbine. The HPT, LPT, and free turbine rotors are interconnected to the HPC, LPC, and the reduction gearbox (RGB), respectively, with three concentric drive shafts. The free turbine drive shaft is within the LPT drive shaft, and both shafts are within the HPT drive shaft. The PW 118 turboprop engine has a takeoff shaft horsepower (shp) rating of 1,800 shp that is flat-rated to 91°F (33°C).[3]

The Hamilton Standard 14RF-9 propeller is a four-bladed, full feathering, constant speed, fully reversible propeller.[4] The propeller blades have a solid aluminum spar that is covered with a fiberglass exterior shell and a polyurethane foam filling. The four propeller blades are secured to the hub, which contains the pitch control actuator mechanism. The normal propeller blade pitch angle is controlled by engine oil through the oil transfer tube that is metered through the acme screw to control the fore and aft position of the propeller pitch change assembly, thereby changing the propeller blade angle. In the event that engine oil pressure is unavailable, the propeller blades can also be feathered with the electric feather switch or the engine fire handle that activates the auxiliary electric pump, which uses a reserve supply of oil to move the pitch control actuator, and consequently the propeller blades to the feather position. The propeller blades have deicers, which are electric heater elements that are enclosed in a rubber boot that is a molded part that is bonded to the leading edge at the base of each of the propeller blades. The electrical power for the heater elements is supplied through three slip rings on each bulkhead. The deicers can be off or activated by the flightcrew through an airframe-supplied timer that has two operating cycles: 10-seconds ON, 60-seconds OFF, or 20-seconds ON, 60-seconds OFF. Power is directed to the deicing boots so that two opposing propeller blades are deiced and then the other tow opposing propeller blades are deiced. The deicing operation and choice of cycle can be selected by the pilot depending upon the ambient weather conditions.

The auxiliary electric pump is a 28 volt direct current (dc) electric motor connected to a gear pump. The pump can be activated by the electric feather switch, which is located on the cockpit overhead panel, or by the engine fire handle. The pump, when activated, temporarily restores system pressure, permitting oil to be directed to the propeller hub dome to force the propeller blades into the feathered position. The pump does not turn during normal operations and is only activated when tested, before each flight during the required auto-feathering system, before the first flight of the day, after an engine and/or propeller change, and during an emergency propeller feathering procedure. The manufacturer, AeroControlex, stated

[3] Flat-rated to a specific temperature indicates the engine will be capable of reaching the rated shaft horsepower up to the specified temperature.

[4] Feathering a propeller refers to rotating the propeller blades so the leading edge is into the wind to reduce the frontal area to an absolute minimum and to minimize or stop rotation of propeller blade, to reduce the drag caused by the propeller.

that when activated, the pump will accelerate to 8,000 revolutions per minute (RPM) in 250 milliseconds.

2.0 *On-site examination*

The following people participated in the on-site examination and recovery of the engines and propellers from January 10 to January 15, 1997: Gordon Hookey-NTSB, Karen Grant-Federal Aviation Administration (FAA), Ted Orr-FAA, Michael Dages-Comair, Thomas Berthe-PWC, Martin Sezack-Hamilton Standard, Dan Ford-Air Line Pilots Association (ALPA), John Rice-ALPA, and Paul Picton-Greenwich Air Services-Texas,LP.

2.1 Wreckage

The left and right engines and propellers were found on the left and right side, respectively, of the crater made by the fuselage. The engines were laying upright in the wreckage with the left engine oriented on a heading of about 290° and right engine was oriented on a heading of about 240°.[5] The Structures Group Chairman had reported the fuselage was oriented on a heading of about 245°. Both of the engines were sooted on the exterior surfaces from the post-impact ground fire, but there was no indication that either engine had experienced an in-flight fire. The left and right engines did not have any indications of any case ruptures or uncontainments.

Both of the propeller and hub assemblies were separated from the engines. The left and right propellers were embedded into the ground about 23-feet apart.[6] The left and right propeller hub were 2 feet 8 inches and 4 feet deep into the ground, respectively. The angle of both of the propellers into the ground, as measured from the rear face of the gearbox bearings in line with the yoke relative to the horizon was 50°.

2.2 Left propeller

The left propeller assembly, with all four propeller blades, was located in a crater to the left side of the crater made by the fuselage. There were three propeller blades that had separated from the left hub and were removed separately during the excavation of the propeller assembly. Three of the four blades of the left propeller assembly were full length. The remaining blade was missing a 28-inch long x 6-inch wide section of the fiberglass shell and foam core at the tip, which was recovered about 75 feet due east of the aircraft and engine wreckage. All of the propeller blades leading edges were to the clockwise side and all of the leading edge nickel sheaths were recovered at the site.

The overspeed governor and pump, auxiliary electric pump (feather pump), brushblock assembly, and the propeller speed (Np) sensor, which was mounted on the RGB mating bracket, were recovered from the RGB that was crushed. The forward section of the

[5] All headings are magnetic and are referenced as viewed from the rear of the airplane or engine and looking forward, unless otherwise specified.
[6] The distance between the two engines and propellers on the EMB-120RT airplane is 21.59 feet.

transfer tube was extended out through the dome and bent over at an angle of about 90°. The PCU was broken up. The ball screw was contained within the servo piston and sleeve along with the governor drive gear. A measurement of the ball screw extension was not possible.

2.3 Right propeller

The right propeller assembly, with all four propeller blades, was located in a crater to the right side of the crater made by the fuselage. Three of the propeller blades remained attached to the right hub. The one propeller blade that was separated from the right hub was removed separately during the excavation of the propeller assembly. All of the propeller blades were full length. All of the propeller blades leading edges were towards the clockwise side and all of the leading edge nickel sheaths were recovered at the site.

The overspeed governor and pump and auxiliary electric pump (feather pump) were recovered from the area of the RGB, which was crushed. The PCU was found to have been broken up. The ball screw assembly was still attached to the transfer tube. The extended ball screw, as measured from the retaining nut, was 1 $^{7}/_{16}$-inches long. The PCU transfer tube was bent about 90° from the axis of propeller rotation, just forward of the quill-ball screw interface. The section to the rear of the bend contained the ball screw/servo piston sleeve assembly.

2.4 Left engine

The RGB housing was shattered into many small pieces. The accessory drive section was intact. The RGB input shaft, layshafts, and the pinion gears had been ejected from the RGB and were recovered separately from the impact crater. The bull gear and propeller shaft were recovered with the propeller hub. The airframe hydraulic pump was intact and in place on the accessory drive cover. The airframe auxiliary generator was found laying adjacent to the airframe hydraulic pump and had severe impact damage.

The front inlet case housing was shattered, exposing the front face of the power turbine (PT) shaft torque coupling. The rear inlet case and accessory gearbox were shattered, exposing the LPC inlet. The low pressure (LP) diffuser case was intact, but the diffuser tubes were deformed. The intercompressor case was intact. The gas generator case had a 360°-circumferential fracture just forward of the bleed valve, exposing the combustion chamber liner and the No. 5 bearing housing. The combustion chamber was intact and did not have any metallization on the dome. The remainder of the gas generator case was intact.

The torque signal conditioning unit (TSCU) had impact damage, and had been broken away from the front inlet case housing. The TSCU remained attached to the front inlet case only by the wiring harness. The oil-to-fuel heater and fuel filter housings were in place and had severe impact damage. The hydromechanical fuel control and fuel pump were structurally separated from the accessory gearbox, and were held in place only linkages and external fuel lines. The starter generator was structurally separated from the rear inlet case and was held in place by the wiring harness. The fuel manifold was intact and in place on the gas generator case. The fuel manifold and adjacent areas did not have any apparent indications of an in-flight fire. The turbine support case was intact. The interturbine temperature (ITT or T6) harness was in place, but had been deformed. The turbomachine oil filter housing was fractured and the filter element was deformed. The filter element did not have any apparent debris entrapped. The chip detectors were not recovered.

The LPC impeller vane was rubbed and gouged circumferentially on the leading edges. The vane tips were rubbed and bent away from the direction of engine rotation. The shroud housing had circumferential rubs and gouging from contact with the impeller vane tips. The gas generator forward diaphragm and diffuser tubes were intact, although the tubes had impact damage. The PT, LPT, and HPT shafts were all fractured adjacent to the rear face of the No. 5 bearing housing. The PT and LPT shaft fractures were at an approximate angle of 60° to the shaft center lines. The HPT shaft was fractured at an approximate angle of 90° to the shaft center line. The fracture surfaces on the three shafts had a grainy, flat-gray appearance and were at a 45° angle to the cross section of the shaft. The outer diameter (OD) surfaces on each of the shafts did not show any discoloration. The second stage PT disk and blades were intact. The second stage PT blade trailing edges were rubbed and bent over away from the direction of engine rotation. The second stage PT guide vane ring, the second stage PT vane airfoils, and the first stage PT blades were intact. There was no metallization on the PT airfoil surfaces.

There were no apparent indications of any uncontainments, case ruptures, internal or external in-flight fires, or operational distress.

2.5 Right engine

The RGB housing was shattered into many small pieces. The RGB input shaft, layshafts, and pinion gears had been ejected from the RGB, and were recovered separately from the impact crater. The bull gear and propeller shaft were recovered with the propeller hub.

The rear inlet case and accessory gearbox were completely shattered, exposing the LPC inlet. The LP diffuser case forward flange was deformed. The diffuser tubes were deformed, and were separated from the case around the lower half of the case. The intercompressor case was intact. The intercompressor case diffuser tubes between 10:00 and 2:00[7] location were punctured on the front side, and the edges of the punctures were deformed radially outboard. The upper section of the gas generator case had a circumferential fracture adjacent to the forward flange exposing the LP diffuser tubes and the combustion chamber liner. The turbine support case was deformed from impact around the lower section of the case. The ITT harness was in place and intact, but was deformed from impact. One of the two chip detectors was recovered. The filters were not recovered from the crash site.

The LPC vane airfoils were rubbed and gouged circumferentially, and the vane tips were bent away from the direction of the rotation. The LPC impeller shroud was rubbed

[7] All references to position, or direction, as referenced to the clock are as viewed from the rear looking forward, unless otherwise specified.

circumferentially 360° around and was deformed radially outboard on the right side from contact with the impeller vane tips. The PT, LPT, and HPT shafts were fractured adjacent to the No. 5 bearing housing. The fracture surfaces on the three concentric shafts were on the three shafts were in a spiral pattern and had a grainy, flat-gray color that were at a 45° angle to the cross section of the shaft. The combustion chamber liner was intact and did not have any metallization on the dome. The fuel manifolds were in place and intact. The tubes were deformed from impact around the lower section of the engine. Over half of the second stage PT blades were fractured transversely across the airfoil adjacent to the blade root platform. The remaining blades in a continuous arc from about 3:00 to 9:00, were fractured transversely across the airfoil at about ¾-span. The PT shroud around the second stage PT blades was crushed inward to the fractured ends of the ¾-span second stage PT blades. The second stage PT guide vane trailing edges were gouged and deformed from the broken second stage PT blades. The first stage PT blades appeared to be intact. The right engine exhaust duct did not show any indication of metallization or metallic impacts.

There were no apparent indications of any uncontainments, case ruptures, internal or external in-flight fires, or operational distress.

3.0 Propeller disassembly

The following people participated in the disassembly of the propellers at Hamilton Standard, Windsor Locks, Connecticut, on January 21 to January 23, 1997: Gordon Hookey-NTSB, Martin Buckman-FAA, Michael Dages-Comair, Thomas Berthe-PWC, Martin Sezack-Hamilton Standard, Dan Ford-ALPA, John Rice-ALPA, Paul Picton-Greenwich Air Services-Texas,LP, and Umberto Irgang-Embraer.

The Powerplant Group reconvened at Hamilton Standard, Windsor Locks, Connecticut on March 6, 1997, to examine the left and right propeller assembly deicing boots. The following people participated in the examination of the deicing boots: Gordon Hookey-NTSB, Michael Dages-Comair, Martin Sezack-Hamilton Standard, Dan Ford-ALPA, and Manuel Monteiro-Embraer.

3.1 Visual inspection

The shipping containers containing the left and right propeller assemblies were opened in the presence of the Powerplants Group. The examination of the containers and propellers did not show any apparent damage or tampering during the shipment from Michigan to Hamilton Standard.

All eight of the propeller blade shanks and spars were intact and did not have any apparent indication of any preexisting damage before the impact. All of the blades were bent at varying degrees away from the direction of propeller rotation.

3.2 Blade butt pin plateau impact mark angle measurements

It was possible to measure the propeller blade pitch angle at the time of impact by measuring the angle of the imprint made by the pitch change actuator yoke assembly on the blade pin plateau on the propeller blade butt. The assembly of the propeller results in the angle between the propeller yoke and the blade pin plateau being a fixed relationship. At the moment of impact, the blade pin plateau is driven against the propeller yoke that causes a witness mark on the plateau to the right of the blade pin.[8] The angle of the witness mark in relation to the inboard edge of the plateau was measured by placing the T-end of a protractor against the bottom of the plateau and rotating the movable edge of the protractor until it was aligned with the visual estimate of the centerline of the most prominent witness marks. The angle of the witness mark on the blade pin plateau minus 53°[9] defines the angle of the blade at blade station 42.00.

	LEFT		RIGHT	
Propeller blade No.	Measured	Calculated	Measured	Calculated
1	92°	39°	91°	38°
2	91°	38°	90°	37°
3	91°	38°	91°	38°
4	97°	44°	91°	38°

The propeller blade butt pin plateau inspection sheets for each blade are attached.

3.3 Servo piston and sleeve assemblies

Both of the PCUs were destroyed by the impact forces and were not recovered from the crash site. The left and right servo piston and sleeve assemblies, including the ball screws, were recovered and examined at Hamilton Standard.

Another method to determine the propeller blade angle at the time of impact is to measure the height of the ball screw extension from the base of the ball screw servo. The ball screw, and the oil transfer tube that is splined into the inner diameter (ID) of the ball screw, rotate counter-clockwise and clockwise, as well as translate out and in, to increase or decrease the propeller blade pitch angle, respectively. The left and right propeller ball screw extensions were measured from the base of the ball screw servo to the extended ball screw minus the damper and quill. The measurement was then cross referenced to a Hamilton Standard-provided conversion table and graph to determine the propeller blade angles. (A copy of the

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[8] As viewing the end of the propeller blade butt with the blade pin oriented towards the top.

[9] The 53° angle is the rotation of the blade axis from the blade butt to blade station 42.00, which is the airfoil reference station that is 42.00 inches from the centerline of the propeller assembly.

Hamilton Standard-provided 14RF-9 ball screw extension to propeller blade angle conversion table and graph is attached.)

	LEFT	RIGHT
Ball screw extension height	1.203 inches	1.205 inches
Propeller blade angle (approximate)	41°	41°

The left servo piston and sleeve assembly ball screw internal splines, the internal and external splines on the quill, and the transfer tube external splines did not show any apparent operational distress. The quill was properly secured to the ball screw with the retainer wire. The examination of the ball screw showed the damper was installed and shimmed.

The right ball screw was bent and distorted. As a result of the impact damage to the ball screw, the quill could not be removed. The quill was properly secured to the ball screw with the retainer wire. To facilitate the examination of the ball screw internal splines, it was necessary to cut the quill off at the damper. It was also necessary to cut off about half of the circumference of the ball screw to facilitate the examination of the splines. The right servo piston and sleeve assembly ball screw internal splines, the internal and external splines on the quill, and the transfer tube external splines did not show any apparent operational distress. The damper was not recovered. The shims for the damper were still in place. The records for the right propeller assembly indicated that a secondary drive quill (SDQ) had been installed in accordance with Hamilton Standard Service Bulletin 14RF-9-61-76, but the SDQ was not recovered from the crash site.

3.4 Hubs

The left hub rear mounting flange was crushed. The arm bores at propeller locations L2, L3, and L4 were broken out towards the rear and in the direction of propeller rotation. The blade at location L1 was retained in the hub. The pitch change actuator was not removed from the hub because of the impact-related compression damage.

The right hub rear mounting flange was intact. The arm bore at propeller blade location R4 was broken out towards the rear and in the direction of propeller rotation. The propeller blades at locations R1, R2, and R3 were retained in the hub. The pitch change actuator was removed from the hub intact. The hub was damaged and distorted from the impact, which the removal of the actuator difficult. The actuator feather stop was bent about 90° from its original position. The oil transfer tube could be rotated 360° within the pitch lock screw, without any movement of the screw. The transfer tube connection pin was fractured, which permitted the transfer tube to be positioned forward of the required location.

It was possible to determine the pitch angle of the propeller blades at the time of impact by measuring the displacement of the pitch lock screw on the pitch change actuator in the hub. The pitchlock screw is rotated counter-clockwise and clockwise by the oil transfer tube and also translates out and in with increasing and decreasing propeller blade pitch angles, respectively. On the left and right hubs, the displacement of the pitch lock screws from the end of the screw to the top point of the pitch lock nut were measured and found to be 1.861 and 1.698 inches, respectively. Cross referencing these dimensions on the Hamilton Standard-provided pitch lock screw dimension to propeller blade angle table and graph provided a propeller blade pitch angle of 40 and 45° for the left and right propellers, respectively. (A copy of the Hamilton Standard-provided 14RF-9 pitch lock screw dimension to propeller blade angle table and graph is attached.)

The propeller blade pitch angle at the time of impact was determined by measuring axial position of the yoke from the mounting pad flange. The dimension was cross referenced to the propeller blade angle on the Hamilton Standard-provided 14RF-9 yoke-to-mounting pad dimension to propeller blade angle table and graph. (A copy of the Hamilton Standard-provided 14RF-9 yoke position to propeller blade angle conversion table is attached.)

BLADE	DIMENSION	ANGLE
R1	4.308 inches	43°
R2	4.332	44°
R3	4.388	46°
R4	4.365	45°

The compression damage to the left hub assembly precluded determining the propeller blade angle by measuring the distance between the mounting pad and yoke. In addition, the damage to the right hub pitch change actuator also precluded determining the propeller blade angle by measuring the displacement of the pitch lock screw.

3.5 Deicing boots

Each of the four left propeller blades had different sized pieces of the deicing boots attached to the blade. Propeller blade L1 had the deicing boot remain attached to fiberglass shell, which had separated from the foam core and aluminum spar. The connectors were attached to the camber side of the blade. The camber side of the boot had a 3-inch long radial outboard tear from the leading edge that started from the rear edge of the boot 7-inches from the inboard edge of the boot that was over a break in the fiberglass shell. The full length of the rear section of the face side of the boot was missing from a radial break in the fiberglass shell at the leading edge radius. A continuity check of the connectors leading to the bottom of the deicing boot showed that there was no continuity.[10] Propeller blade L2 had the full length of the deicing boot attached to the fiberglass shell, which was separated from the foam and spar. The

[10] A John Fluke Manufacturing Fluke 8012A digital multimeter, Hamilton Standard SN 6101-0010, with the next calibration due on 11/21/97, was used for all of the continuity checks.

connectors were missing from the camber side of the blade. The boot had an 11-inch long radial split from the inboard end on the face side just aft of the leading edge radius. The continuity check of the broken ends of the wires within the deicing boot showed there was no continuity. Propeller blade L3 had the deicing boot connectors attached to the camber side of the blade. The remainder of the deicing boot and the fiberglass shell were missing. The continuity check of the deicing boot connectors from the lead end to the broken end showed there was continuity. Propeller blade L4 had the deicing boot attached to the fiberglass shell, which had separated from the foam and spar. The connectors were broken off from the camber side. The boot and fiberglass shell were torn radially from the camber side outer end rear corner to the leading edge about 12-inches from the inner end and then axially from the leading edge across the face side to the rear edge except for a ¾-inch wide section at the rear edge. There was a 7-inch long radial x 1-inch wide axial section of the boot missing at the inner end. It was not possible to check the continuity of the connectors or the deicing boot because there were no exposed wires.

The left propeller bulkhead was missing an approximate 90° segment. The front face of the bulkhead and the three out of the four leads that were attached and did not have any apparent fire or arcing damage. The inner and outer slip rings were broken and the center slip ring was intact. The inner ring did not have any missing material and the outer ring was missing an arc segment of about 135°. The slip ring surfaces did not exhibit any arcing, pitting, or unusual wear. The check of the physically severed connectors to L2 and L4 and associated bulkhead leads showed there was continuity. The leads and connectors to L1 and L3 were missing from the bulkhead.

On the right propeller assembly, all four of the propeller blades were missing the leading edges, thus the deicing boots were also missing. All four of the propeller blades had the connectors still attached to the camber side of the blade. The continuity check of the leads showed that there was continuity on both leads for blades R2, R3, and R4. On blade R1, one lead was found to have continuity and the other lead could not be checked because the broken end of the wire could not found in the deicing boot.

The right propeller bulkhead had sooting and heat damage on the forward face. Two of the leads, which were adjacent to propeller blade R3, had the insulation melted. Propeller blade R3 had heat damage to the trailing edge of the airfoil adjacent to the inboard end of the blade.

The three slip rings were all intact, and had no indication of any arcing, pitting, or unusual wear.

The continuity check of the leads on the bulkhead to the four propeller blades showed there was continuity on each lead.

4.0 Engines

The following people participated in the disassembly of the engines at PWC: Gordon Hookey-NTSB, Glorianne Messemer-FAA, Robin Lau-Transport Canada, Michael

Dages-Comair, Thomas Berthe-PWC, Giancarlo Masciotra-PWC, Frank Brazauckas-PWC, Martin Sezack-Hamilton Standard, Dan Ford-ALPA, John Rice-ALPA, and Manuel Monteiro-Embraer.

4.1 Left engine

4.1.1 Visual inspection

The shipping container containing the left engine was opened in the presence of the Powerplants Group. The examination of the container and the engine did not show any indication of damage or tampering during the shipment of the engine to PWC.

The engine displayed severe impact damage, which included the complete structural separation of the RGB and the accessory gearbox housings. The gas generator case housing was structurally separated adjacent to the forward flange and was held in place by external lines and linkages. There was some fire damage and sooting on the external surfaces of the engine due to the postimpact ground fire.

4.1.2 Reduction gear box

The RGB housing was extensively fractured. The housing fragments and internal gears of the RGB were recovered from impact crater separately from the main part of the engine. The bullgear, input driveshaft, helical gears, pinion gears, second stage reduction gear, propeller shaft, and their associated bearing were recovered and all displayed damage and deformation that was typical of impact. None of the recovered pieces of the RGB components displayed any indications of any distress that existed before impact. A fragment of the rear of the RGB housing had circumferential scoring with concurrent static imprint marks of the second stage reduction gear teeth on the inner surface.

The RGB shaft was deformed and compressed, which displaced the shaft at a right angle to the shaft centerline. The forward and aft ends of the RGB shaft had tearing that was consistent with a clockwise torsional overload.

4.1.3 Inlet/accessory gearbox

The front and rear inlet cases and accessory gearbox were extensively fractured with only the inner plenum of the inlet case remaining attached to the main part of the engine. The fragments of the inlet case and accessory gearbox were recovered separately.

4.1.4 Low pressure compressor

The LP diffuser case housing was essentially intact. The diffuser tubes were deformed around the upper half of the engine. The tubes around the lower right side of the engine were completely separated from the case.

The LP impeller was intact. The LP impeller vane leading edges were severely gouged and torn, and tip leading edges were deformed and bent away from the direction of engine rotation, consistent with the ingestion of debris. The vane tips were circumferentially rubbed and had discoloration with frictional heating and material smearing. There was corresponding evidence of axial and radial contact on the No. 2 bearing housing. The ID of the shroud was also circumferentially rubbed and scored, and had areas of frictional heat discoloration and material smearing. There was corresponding evidence of axial and radial contact to the LP impeller.

The LPT shaft was fractured completely about 5-inches forward of the of the LP turbine coupling. The fractured ends of the LPT shaft were typical of a counter-clockwise torsional overload. The rear section of the LPT shaft had a 2 ½-inch wide area of severe circumferential rubbing and scoring, with frictional heat discoloration and material smearing, corresponding with rub marks on the HPT shaft.

The No. 3 ball bearing did not show any rotational distress. The No. 3 bearing airseal stators were scored circumferentially that corresponded to the rotor knife edge seals.

4.1.5 High pressure compressor

The intercompressor case housing was essentially intact. During the disassembly of the intercompressor case, dirt and metal debris was found in the bottom of the LPC discharge air (P2.5) plenum.

The gas generator case housing had a 360°-circumferential fracture immediately aft of "F" flange.[11] The rear portion of the case was essentially intact.

The HP impeller was intact. The HP impeller vane leading edges had been circumferentially gouged and deformed from ingested debris. The impeller vane tips were rubbed circumferentially, with frictional heat discoloration and material smearing, corresponding to damage to the impeller shroud. The shroud face had severe, heavy circumferential rubbing and scoring, with frictional heat discoloration and material smearing and transfer. There was corresponding evidence of axial and radial contact on the impeller.

The HPT shaft was fractured immediately aft of the No. 5 bearing inner race. The fracture surfaces were typical of a tensile overload type fracture. The shaft ID was rubbed and scored circumferentially, with frictional heat discoloration and material smearing that corresponded to rubs on the OD of the LPT shaft.

The No. 4 ball bearing could be rotated freely by hand. The bearing was not disassembled.

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[11] Gas turbine engine industry convention is to identify the engine case flanges alphabetically from front to rear. (A copy of a cross section of the PW100 engine with the flange locations identified is attached.)

4.1.6 Combustor

The combustion chamber liner had impact-related deformation around the lower half. The liner did not have any apparent indications of any operational distress. The inside of the combustion chamber liner was sooted.

The combustion chamber small exit duct did not have any apparent distress.

The No. 5 bearing inner race and cage were intact, and the bearing rollers were secure in the cage pockets. The cage and rollers could be rotated by hand on the inner race. The outer race was fractured. The bearing rollers and the outer race had discoloration from frictional heat and material smearing. The bearing did not have any indications of any operational distress.

4.1.7 High pressure turbine

The HPT vane airfoil leading edges were fractured and deformed. There was corresponding damage to the vane ring inner drum. The downstream side of the inner drum was rubbed and scored circumferentially that corresponded to damage to the HPT rotor. The HPT shroud had circumferential scoring that corresponded to circumferential damage noted on the HPT vane ring.

The interstage airseal rotors and stators were circumferentially rubbed.

The HPT disk was intact. The upstream side of the HPT disk was scored circumferentially and the disk balancing ring was deformed that corresponded to circumferential damage on the rear face of the HPT vane ring. All of the HPT blade airfoils were intact. The airfoil leading edges and blade platforms were scored circumferentially that corresponded to damage to damage on the rear face of the HPT vane ring. The blade tips were rubbed circumferentially, with minor fractures at the trailing edges. There was corresponding rub damage on the ID of the HPT shroud.

4.1.8 Low pressure turbine

The LPT disk was intact. The downstream side of the disk at the outer rim had circumferential scoring that corresponded to circumferential damage on the front of the first stage PT vane ring inner drum. All of the LPT blade airfoils were intact. The blade tips were rubbed circumferentially and had frictional heat discoloration and material smearing. There was corresponding damage to the ID of the LPT shroud. The front face of the blade roots were scored circumferentially corresponding to damage on the rear of the LPT vane ring inner drum.

The LPT vane airfoils were fractured at the outer ends in line with the inner drum and the interstage airseal. The upstream side of the inner drum and interstage airseal was circumferentially scored that corresponded to circumferential damage on the rear face of the HPT disk. The downstream side of the inner drum and interstage airseal was scored circumferentially that corresponded to damage on the front of the LPT disk. The shroud had severe circumferential scoring, with frictional heat discoloration and material smearing. The LPT blades had corresponding damage to the tips. The interstage airseal had severe circumferential scoring that corresponded to the airseal rotor knife edges.

4.1.9 Power turbine

The turbine support case was deformed radially inward around the lower half, just aft of "K" flange and at about the 8:00 location adjacent to the aft rim. The deformation of the turbine support case necessitated that the case be sectioned with a carbide cutting disk at the 8:00 location to permit removal of the exhaust duct. The case was then further sectioned using a plasma arc cutting torch around the circumference of the case just forward of the second stage PT vane ring to facilitate removal of the PT balancing assembly.

The PT shaft was fractured approximately 21 ½-inches aft of the shaft forward face. The fracture surfaces had deformation and tearing at an approximate 45° angle to the shaft centerline, consistent with a clockwise torsional overload.

The first stage PT disk was intact. The downstream side of the PT disk outer rim and blade platform ID was scored circumferentially that corresponded to circumferential damage on the front of the second stage vane ring inner drum. The first stage PT blades were all intact. The blades were scored circumferentially on the front face of the roots and on the outer section of the leading edges corresponding to damage on the rear of the first stage PT vane ring. The blade trailing edge tip shrouds and vane airfoil outer shrouds were scored circumferentially and deformed. There was corresponding damage to the front of the second stage PT vane ring.

The second stage PT disk was intact. The downstream side of the disk on the outer rim and blade platform IDs were rubbed circumferentially, with frictional heat discoloration and material smearing. There was corresponding damage noted on the front of the exhaust duct inner cone. The second stage PT blades were intact. The front face of the blade roots had circumferential scoring corresponding due damage on the second stage PT vane ring inner drum. The trailing edges of the second stage PT blades were deformed and scored, approximately 1-inch wide x $^{1}/_{8}$-inch deep, adjacent to the blade tip, in line with the exhaust duct outer drum. The blade tips were rubbed circumferentially and deformed. There was corresponding damage to the ID of the shroud ring.

The first stage PT vane ring airfoils were intact. The downstream side of the vane ring inner drum had severe circumferential scoring. There was evidence of corresponding damage to the first stage PT.

The second stage PT vane airfoils were intact. The outer portion of the second stage PT vane leading edges had circumferential scoring, with frictional heat discoloration and material smearing, that corresponded to damage to the rear of the first stage PT. The vane airfoil trailing edges were lightly gouged corresponding to damage on the front of the second stage PT. The downstream side of the vane inner ring and outer drum had severe circumferential rubbing and scoring, with frictional heat discoloration and material smearing, that corresponded to damage noted on the front of the second stage PT. The interstage airseal stator had severe circumferential rubbing and scoring that corresponded to the rotor knife edges. There were areas of the second stage PT vane outer drum and many of the second stage PT vanes were burned and spattered with molten metal as a result of using the plasma arc cutting torch to section the PT support case at the 8:00 location.

The No. 2 bearing housing had severe circumferential rubbing and scoring corresponding to damage noted on the LP impeller inner bore. The housing could not be disassembled because of the impact damage.

The No. 6 and 7 bearings did not have any apparent operational distress. The airseal stators were grooved circumferentially that corresponded to damage on the adjacent rotors.

The exhaust duct was deformed radially inward about 2-inches around the lower half of the case that matched the deformation of the turbine support case. The inner cone OD and inner face had heavy circumferential rubbing and scoring, with frictional heat discoloration and material transfer. There was evidence of corresponding damage to the rear of the second stage PT disk.

4.1.10 Accessories

The main oil filter was fractured exposing the filter element. The filter did not have any metallic debris. The main chip detector, RGB scavenge chip detector, and the RGB scavenge oil filter were not recovered. The oil pressure transmitter did not have any apparent damage.

The low pressure fuel filter housing was fractured exposing the filter elements. There was no apparent pre-impact debris in the filter. The high pressure fuel pump had severe impact damage. The accessory gearbox input drive gear was intact. The hydromechanical fuel control unit had severe impact damage with the housing fractured exposing the internal parts and shearing the external linkages. The unit remained attached to the high pressure fuel pump. The impact damage to the fuel control precluded the disassembly of the unit. The fuel flow divider was in place on the gas generator case and had impact damage that included fracturing of the fuel inlet port and deformation of the dump housing. The fuel flow divider was not disassembled. The fuel nozzles were in place on the gas generator case. The nozzles had impact damage and were sooted. The nozzles had heat damage to the fuel transfer tube packings. The oil-to-fuel heater had severe impact damage, with only the thermal sensor body area being identifiable. The fuel pump outlet filter was not recovered.

The TSCU had severe impact damage, which included the complete separation of the top cover from the unit. The internal circuit card and microcircuitry was also damaged. The electronic control unit was not recovered. The propeller control unit hydraulic pump and overspeed governor were not recovered. The total inlet temperature (T1.8) sensor was damaged. The torque sensor was not recovered.

The igniter plugs had severe impact damage. The ignition exciter and igniter cables were not recovered.

The HP rotor speed (Nh) sensor was in place, but deformed. The LP rotor speed (Nl) sensor had impact damage to the shank. The Np sensor had no apparent damage.

The turbine interstage temperature (T6) probes had impact-related damage. The sensing tips did not have any apparent damage.

The P2.5 switching valve had impact damage with two mounting bolts fracture. The piston was free to move by hand.

4.2 Right engine

4.2.1 Visual inspection

The shipping container containing the right engine was opened in the presence of the Powerplants Group. The examination of the container and the engine did not show any indication of damage or tampering during the shipment of the engine to PWC.

The engine displayed severe impact damage, which included the complete structural separation of the RGB and disintegration of the inlet cases and accessory gearbox housings. The gas generator case housing was structurally separated adjacent to the forward flange around the upper circumference. The engine had some minor discoloration and sooting due to the postimpact fire.

4.2.2 Reduction gearbox

The RGB housing was extensively fractured. The housing fragments and gearing were recovered separately from the main part of the engine. The bullgear, input driveshaft, helical gears, pinion gears, second stage reduction gear, propeller shaft, and their associated bearings were recovered and displayed impact-related-damage and deformation. The bullgear rim was sheared off of the web. The recovered components did not show any apparent pre-impact distress. A recovered fragment of the rear housing right side airframe engine mount mounting pad displayed imprint marks that corresponded to the tooth profile of the second stage reduction gear.

4.2.3 Inlet/accessory gearbox

The front and rear inlet housings and the accessory gearbox housing were extensively fractured. The fragments of the housings of the inlet cases and accessory gearbox, as well as a limited amount of the gearing, were recovered separately.

4.2.4 Low pressure compressor

The LP diffuser case housing was essentially intact. The diffuser tubes were deformed. The diffuser tubes between 10:00 and 2:00 were ruptured adjacent to the front side of the base of the tubes. The ruptured edges of the tubes were deformed outwards.

The LP impeller was intact. The LP impeller vane leading edges were severely gouged and torn with the leading edges of the tips bent away from the direction of rotation, consistent with the ingestion of debris. The vanes tips were rubbed circumferentially and had frictional heat discoloration and material smearing that corresponded to the circumferential damage on the ID of the impeller shroud. The impeller inner bore had heavy circumferential rubbing and scoring, with a dark discoloration from frictional heat and material smearing, corresponding to damage on the No. 2 bearing housing. The shroud had heavy circumferential rubbing and scoring, with areas of frictional discoloration and material smearing. There was evidence of corresponding damage to the impeller.

The LPT shaft was fractured approximately 4-inches forward of the LPT coupling, adjacent to the aft face of the No. 5 bearing housing. The fractured ends of the shaft were typical of a fracture under a counter-clockwise torsional overload. The forward and aft sections of the shaft had severe circumferential rubbing and scoring, with frictional heat discoloration and material smearing, that matched rubs noted on the ID of the HPT shaft.

The No. 3 bearing did not show any distress. The bearing airseal stators had circumferential scoring that matched the rotor knife edges.

4.2.5 High pressure compressor

The intercompressor case housing was essentially intact. During the disassembly of the intercompressor case, dirt and metal debris was found in the bottom of the P2.5 plenum.

The gas generator case housing had a 360°-circumferential fracture immediately aft of "F" flange. The aft section of the housing was essentially intact.

The HP impeller was intact. The impeller vane leading edges were gouged and deformed circumferentially consistent with the ingestion of debris. The impeller vane tips were rubbed circumferentially, and had frictional heat discoloration and material smearing, that corresponded to circumferential damage on the ID of the impeller shroud. The shroud face had severe heavy circumferential rubbing and scoring, with frictional heat discoloration and material smearing, corresponding to damage noted on the impeller.

The HPT shaft was fractured immediately aft of the No. 5 bearing inner race. The fracture surfaces on the ends of the shaft were typical of an overload type fracture. The ID of the shaft had circumferential rubbing and scoring that matched the rubs noted on the OD of the LPT shaft.

The No. 4 bearing could be rotated smoothly by hand. The bearing was not disassembled.

4.2.6 Combustor

The lower half of the combustion chamber liner had impact-related deformation. The liner did not have any apparent indications of any operational distress. The inside of the combustion chamber liner was sooted.

The combustion chamber small exit duct did not have any apparent distress.

The No. 5 bearing inner and cage were intact, and the bearing rollers were secure in the cage pockets. The cage and rollers could be rotated by hand on the inner race.

4.2.7 High pressure turbine

The HPT disk and blades were intact. The upstream side of the disk had circumferential rubbing and scoring that corresponded to damage on the rear face of the HPT vane ring. The HPT airfoil leading edges and blade platforms were rubbed circumferentially and there was corresponding damage to the rear of the HPT vane ring. The blade tips were rubbed circumferentially that corresponded to damage on the ID of the HPT shroud.

The HPT vane airfoils were cracked and deformed that corresponded to damage to the vane ring inner drum. The downstream side of the inner drum was rubbed and scored circumferentially that corresponded to damage to the front face of the HPT. The HPT shroud had circumferential scoring and there was corresponding damage to the HPT blade tips.

The interstage airseal rotors and stators were circumferentially rubbed.

4.2.8 Low pressure turbine

The LPT disk and blades were intact. The downstream side of the LPT disk outer rim had circumferential scoring corresponding to damage noted on the first stage turbine vane ring inner drum. The LPT blade tips had light circumferential rubbing with frictional heat discoloration and material smearing, corresponding to damage on the ID of the LPT shroud. The front face of the blade roots were scored circumferentially and there was corresponding damage to the LPT vane ring inner drum.

The LPT vane airfoils had cracking at the outer ends that corresponded to damage to the inner drum and the interstage airseal. The upstream side of the inner drum and the interstage airseal had heavy circumferential scoring that matched circumferential damage noted on the HPT disk. The downstream side of the inner drum and interstage airseal were scored circumferentially that corresponded to damage noted on the LPT. The shroud had circumferential scoring, with frictional heat discoloration and material smearing, corresponding to damage to the LPT blade tips. The interstage airseal had severe circumferential scoring that corresponded to the airseal rotor knife edge seals.

4.2.9 Power turbine

The turbine support case was deformed radially inward around the lower half just aft of "K" flange and the approximate 7:00 location adjacent to the aft rim. The turbine support case had to be cut around the circumference with a carbide disk about 2 ¾-inches forward of the aft rim to facilitate removal of the exhaust duct. The case was also cut with a plasma arc cutting torch around the circumference just forward of the second stage PT vane ring to facilitate the removal of the PT balancing assembly.

The exhaust duct was deformed radially inward about 2-inches around the lower half due that corresponded to the deformation on the turbine support case. The OD of the inner cone and inner face had heavy circumferential rubbing and scoring, with frictional heat discoloration, that corresponded to damage that was noted on the rear of the second stage PT disk.

The PT rotor shaft was fractured about 11-inches aft of the rear edge of the No. 2 housing. The fractured ends of the shaft had deformation and tearing at an approximate angle of 45° to the shaft centerline that is consistent with a clockwise torsional overload. The forward face of the shaft and the No. 1 bearing housing was fractured from the shaft immediately aft of the No. 1 bearing housing. The PT shaft, adjacent to the No. 7 bearing, had a dark discoloration.

The RGB shaft was not recovered.

The first stage PT disk was intact. The downstream side of the disk outer rim and blade platform IDs were circumferentially scored corresponding to damage on the second stage vane ring inner drum. The first stage PT blades were intact. The blade airfoils had a rainbow color pattern at the mid-span area, which according to PWC mechanics assigned to the engine disassembly shop, was common on low time engines. The blade trailing edge tip shrouds and vane airfoil outer spans were circumferentially scored and deformed corresponding to damage that was noted on the second stage PT vane ring. The blade tips were circumferentially scored and there was evidence of corresponding damage to the ID of the first stage PT shroud.

The second stage PT disk was intact. The second stage PT blades had a continuous sector of about ²/₃ of the circumference that were fractured across the airfoil adjacent to the blade root platforms. The remaining blades were fractured across the airfoil at about the ²/₃ span with the fractured ends of the blades bent away from the direction of rotation. The trailing edges of the blades were deformed and scored circumferentially and there was corresponding damage noted on the second stage PT shroud. The front face of the blade platforms had heavy circumferential scoring and deformation corresponding to damage on the second stage PT vane ring inner drum. The rear face of the blade platform IDs and the disk outer rim were rubbed and scored circumferentially that corresponded to damage on the exhaust duct inner cone.

The first stage PT vane airfoils were intact. The downstream side of the vane ring inner drum had heavy circumferential scoring. There was corresponding damage noted on the first stage PT.

The second stage PT vane leading edges at the OD and the upstream side of the vane ring inner drum were circumferentially scored, with frictional heat discoloration and material smearing, with the inner drum rolled inward, corresponding to damage on the first stage PT. The vane trailing edges were deformed and circumferentially gouged. There was evidence of corresponding damage to the second stage PT rotor. The damage to both the vane trailing edges and the rotor airfoils was consistent with impacts from liberated blade debris. The downstream side of the vane ring inner and outer drums had severe circumferential rubbing and scoring, with frictional heat discoloration and material smearing, that corresponded to damage to the second stage PT. The interstage airseal stator had severe circumferential rubbing and scoring that matched the rotor knife edges.

The No. 2 bearing housing had circumferential rubbing and scoring that corresponded to damage noted on the LP impeller inner bore. The impact damage to the No. 1 and 2 bearing housings precluded access to either of the bearings.

The No. 6 and 7 bearings did not have any apparent distress. The No. 7 bearing had heavy varnishing on all surfaces. The airseal stators were circumferentially grooved.

4.2.10 Accessories

One unidentified (main or RGB) chip detector, which had been fractured from its mounting boss, was recovered. The installed position of the detector could not be confirmed. The chip detector did not show any ferrous material. The main oil filter and the other chip detector were not recovered.

The high pressure fuel pump had severe impact damage. The accessory gearbox input drive gear was intact. The unit was not disassembled. The hydromechanical fuel control unit had severe impact damage, which included fracturing of the housing to expose the internal parts, and shearing of the external control arms. The impact damage precluded disassembly. The flow divider was in place on the gas generator case and had impact damage that fractured the housing. The unit was damaged by the post-impact fire that destroyed the manifold packings. The unit was not disassembled. The fuel nozzles were in place on the gas generator case. The nozzles were sooted and had heat damage to the transfer tube packings. The fuel nozzles were not disassembled. The low pressure fuel filter housing and fuel pump outlet filter were not recovered.

The TSCU had severe impact damage that included the complete separation of the top cover. The circuit card and microcircuitry were also damaged. The electronic control unit was not recovered.

The propeller control unit hydraulic pump was essentially intact, but the housing was scored. The unit was not disassembled. The overspeed governor had severe impact damage that included the complete separation of the rear housing. The unit was not disassembled. The T1.8 and torque sensor were not recovered.

The igniter plugs had severe impact damage. The ignition exciter and igniter cables were not recovered.

The Nl speed sensor was not recovered. The Nh speed sensor was in place, but was deformed. The Np speed sensor did not have any apparent damage.

The T6 probes were deformed and damaged. The sensing tips did not appear to have any distress.

The P2.5 switching valve had impact-related damage that included two mounting bolts being fractured. The piston could be moved freely by hand.

5.0 Auxiliary electric pumps

The following people participated in the disassembly of the auxiliary electric pumps at AeroControlex: Gordon Hookey-NTSB, Jim Davidson-FAA, Michael Dages-Comair, Martin Sezack-Hamilton Standard, Dan Ford-ALPA, Manuel Monteiro-Embraer, and Charles Burger-AeroControlex.

5.1 Left pump

The left pump housing bore for both gears of the pump had wear marks on about 180° of the aft side of the bores. AeroControlex reported that this wear was typical for in service parts. There was no apparent damage observed on the pump gears or the splines.

The electric motor armature laminations at the bottom were bent slightly, at the 6:00 location. The commutator also had an imprint of the brush holders at 12:00 and 6:00. There was no sliding damage on the field stators or the armature laminations. There was no indication of any arcing or overheating of the armature.

5.2 Right pump

The right pump housing bore for both gears of the pump had wear marks on the full circumference of the bores. AeroControlex reported that this wear was also typical for in service parts. There was no apparent damage observed on the pump gears or the spline.

The electric motor armature winding had an imprint of the field magnet at the 6:00 location. There was no sliding damage on the field stators or the armature laminations. The commutator also had an imprint of the brush holders at 12:00 and 6:00. There was no indication of any arcing or overheating of the armature.

[redacted]

Gordon J. Hookey
Powerplants Group Chairman

3/31/97

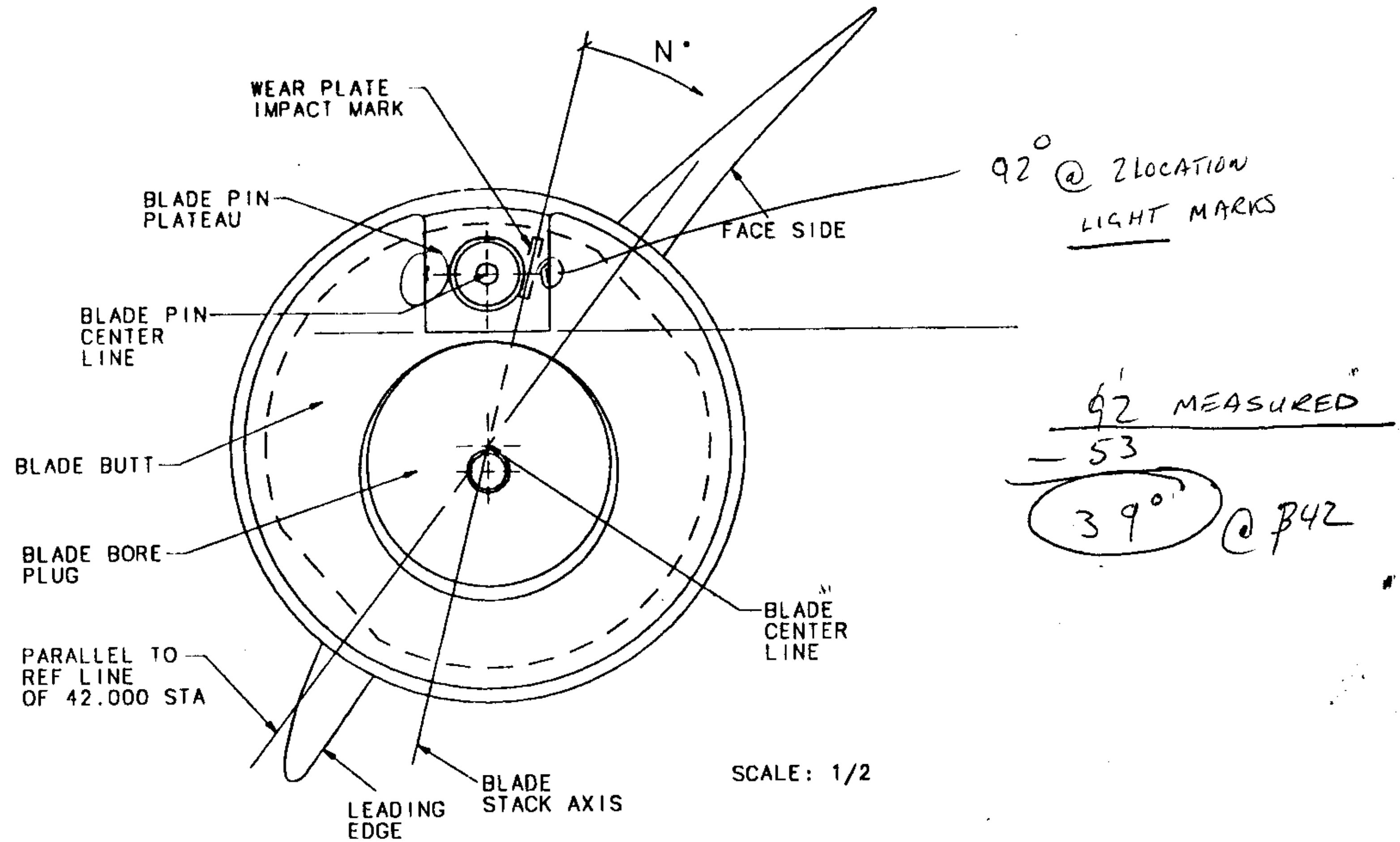
ATTACHMENTS

1. Blade butt pin inspection sheet - Blade L1
2. Blade butt pin inspection sheet - Blade L2
3. Blade butt pin inspection sheet - Blade L3
4. Blade butt pin inspection sheet - Blade L4
5. Blade butt pin inspection sheet - Blade R1
6. Blade butt pin inspection sheet - Blade R2
7. Blade butt pin inspection sheet - Blade R3
8. Blade butt pin inspection sheet - Blade R4
9. 14RF-9 ball screw extension to propeller blade angle conversion table and graph
10. 14RF-9 pitch lock screw dimension to propeller blade angle conversion table
11. 14RF-9 yoke position to propeller blade angle conversion table
12. Cross-section of PW100 turboprop engine identifying flange locations

14RF-9

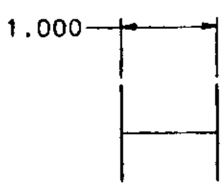
3oclk

Attachment 1



IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°



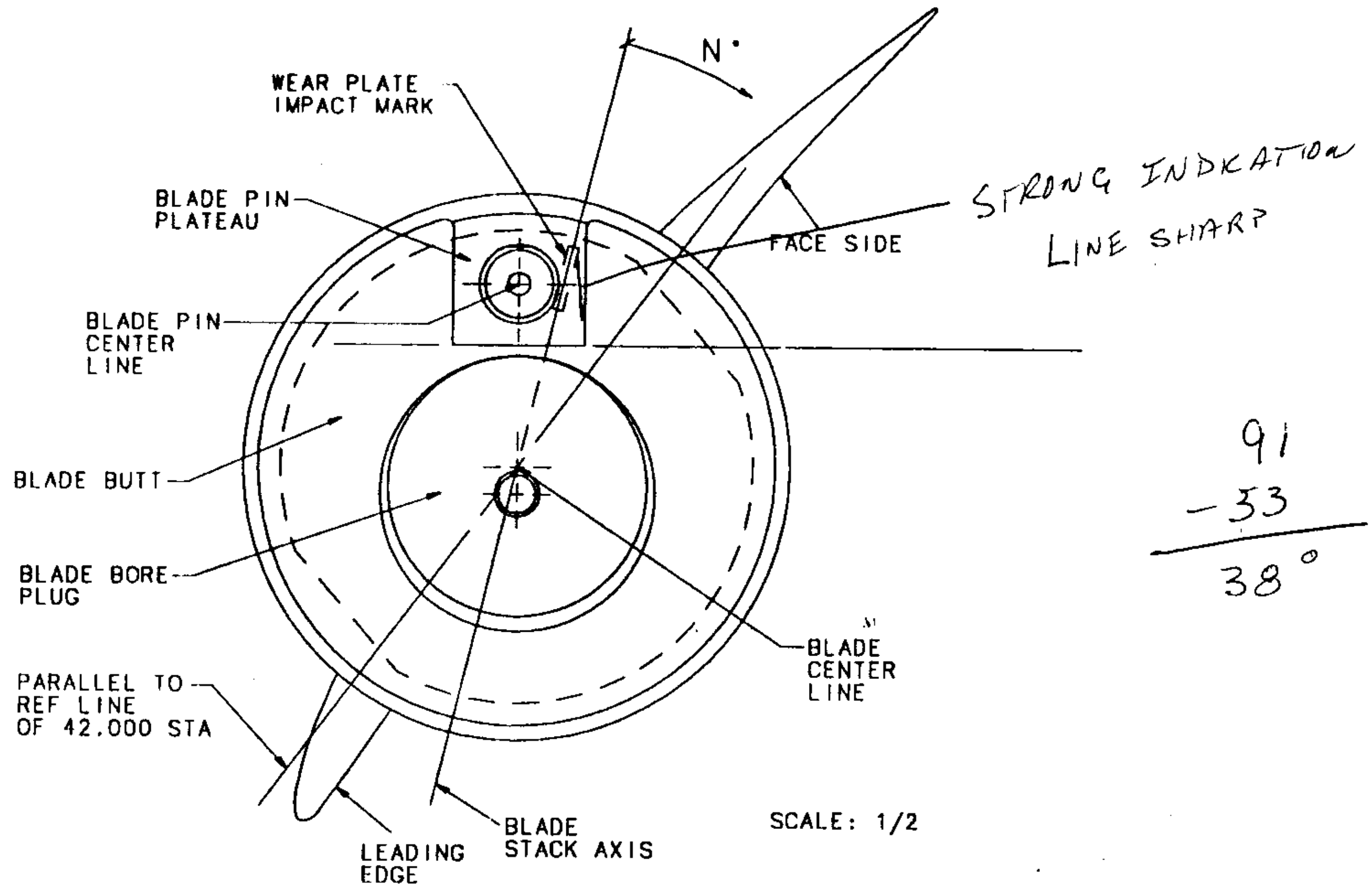
REF DWG: 802339 REV J
792232 REV AD

BLADE PN: RFC11AA1-6A
BLADE SN: 887784
INSTALLED LCTN: L4
SITE LCTN: L1

14RF-9

12 OCK

Attachment 2



91
-53
38°

IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°

1.000

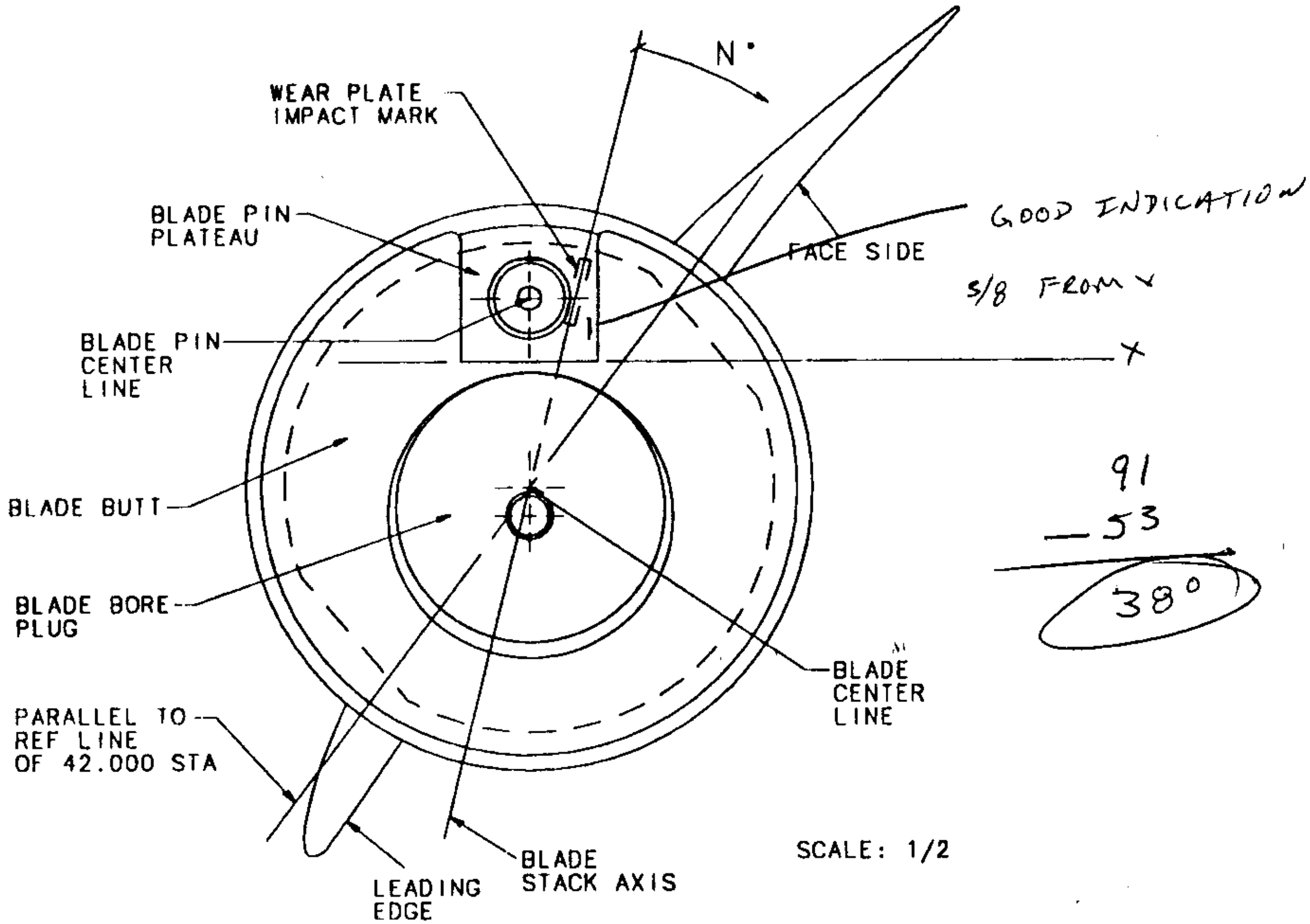
REF DWG:802339 REV J
792232 REV AD

BLADE PN: RFC11AA1-6A
BLADE SN: 087779
INSTALLED LCTN: L3
SITE LCTN: L2

14RF-9

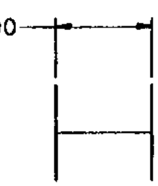
9 CLOCK

ATTACHMENT 3



IMPACT BLADE ANGLE CALCULATION

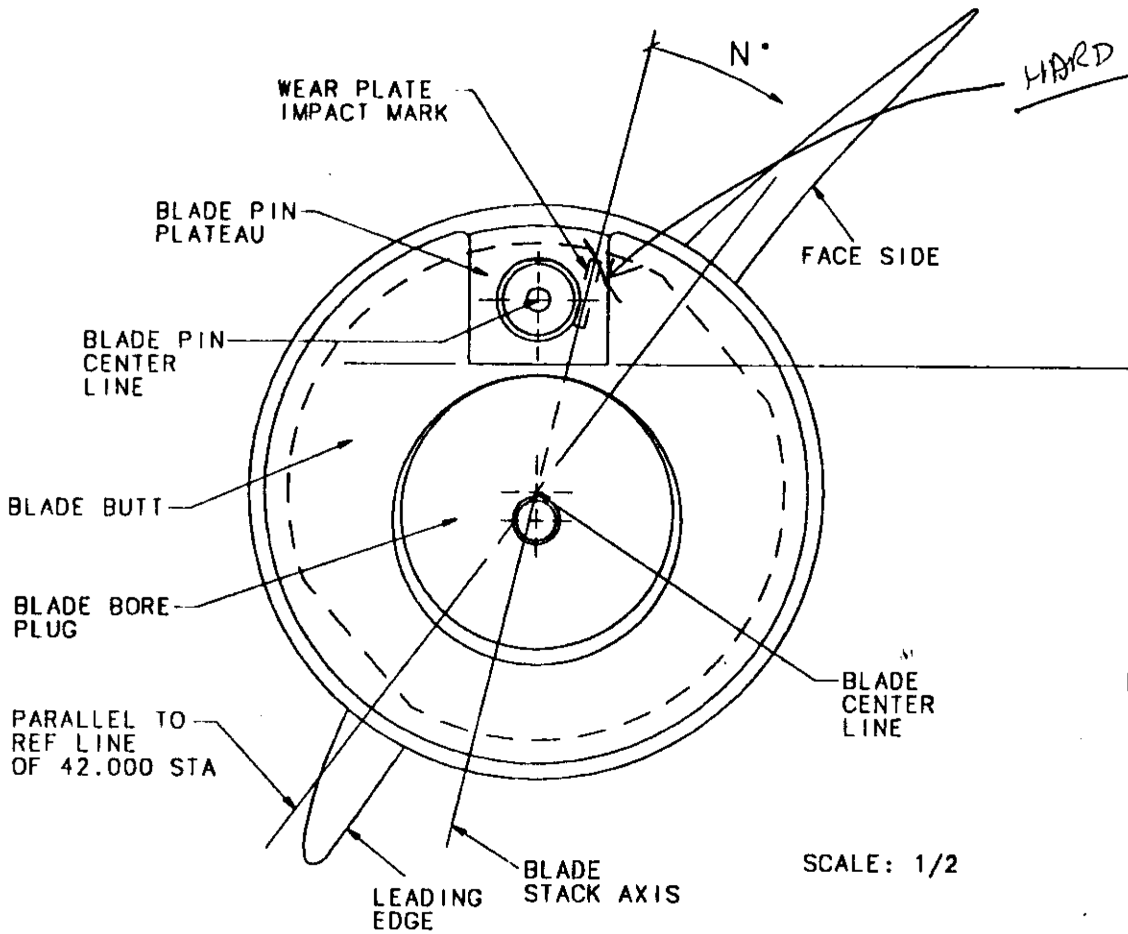
IMPACT BLADE ANGLE= N°-53°



REF DWG:802339 REV J
792232 REV AD

BLADE PN: RFC11A1-6A
BLADE SN: 887782
INSTALLED LCTN: L2
SITE LCTN: L3

6 OCLK



BOTTOM BLADE

97 MEASURED
-53
44° @ β42

SCALE: 1/2

IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°

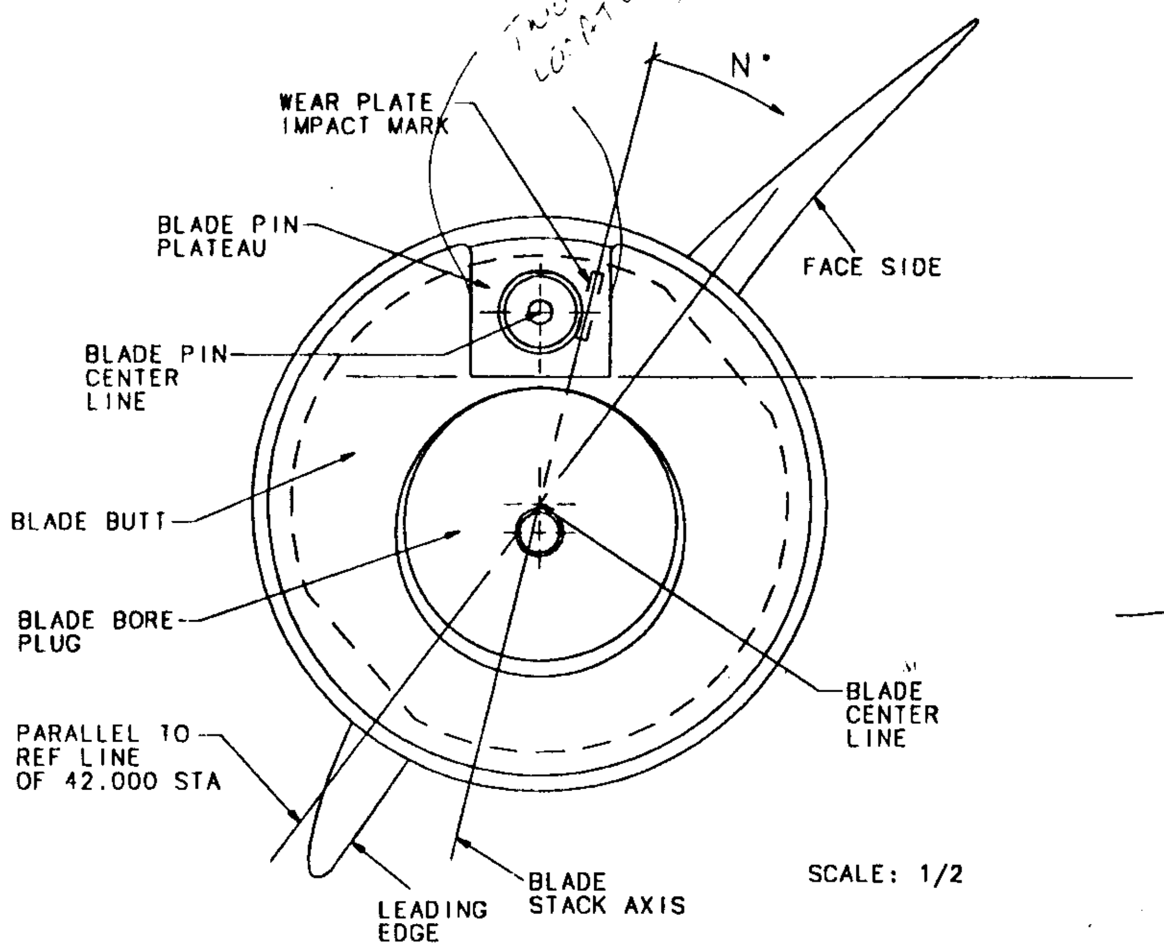
1.000

REF DWG: 802339 REV J
792232 REV AD

BLADE PN: RFC11AA1-6A
BLADE SN: 887792
INSTALLED LCTN: L1
SITE LCTN: L4

14RF-9

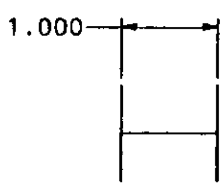
9 OCLOCK



91 EST & CHECKED
-53
38 @ β42

IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°

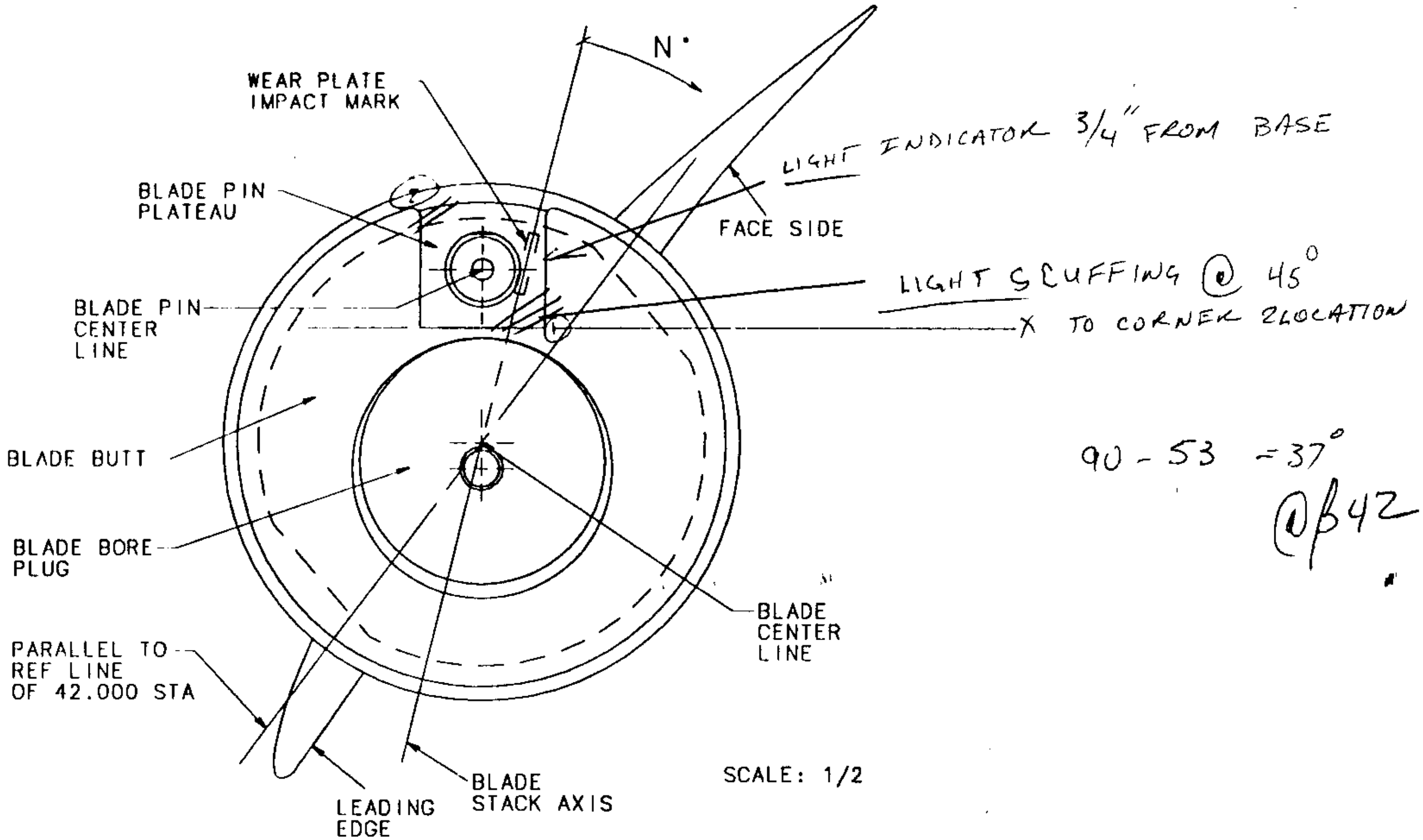


REF DWG: 802339 REV J
792232 REV AD

BLADE PN: RFC11N1-6A
BLADE SN: 872581
INSTALLED LCTN: R1
SITE LCTN: R1

12 OCLOCK

ATTACHMENT 6



IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°

1.000

REF DWG: 802339 REV J
792232 REV AD

BLADE PN: RFC11U1-6A
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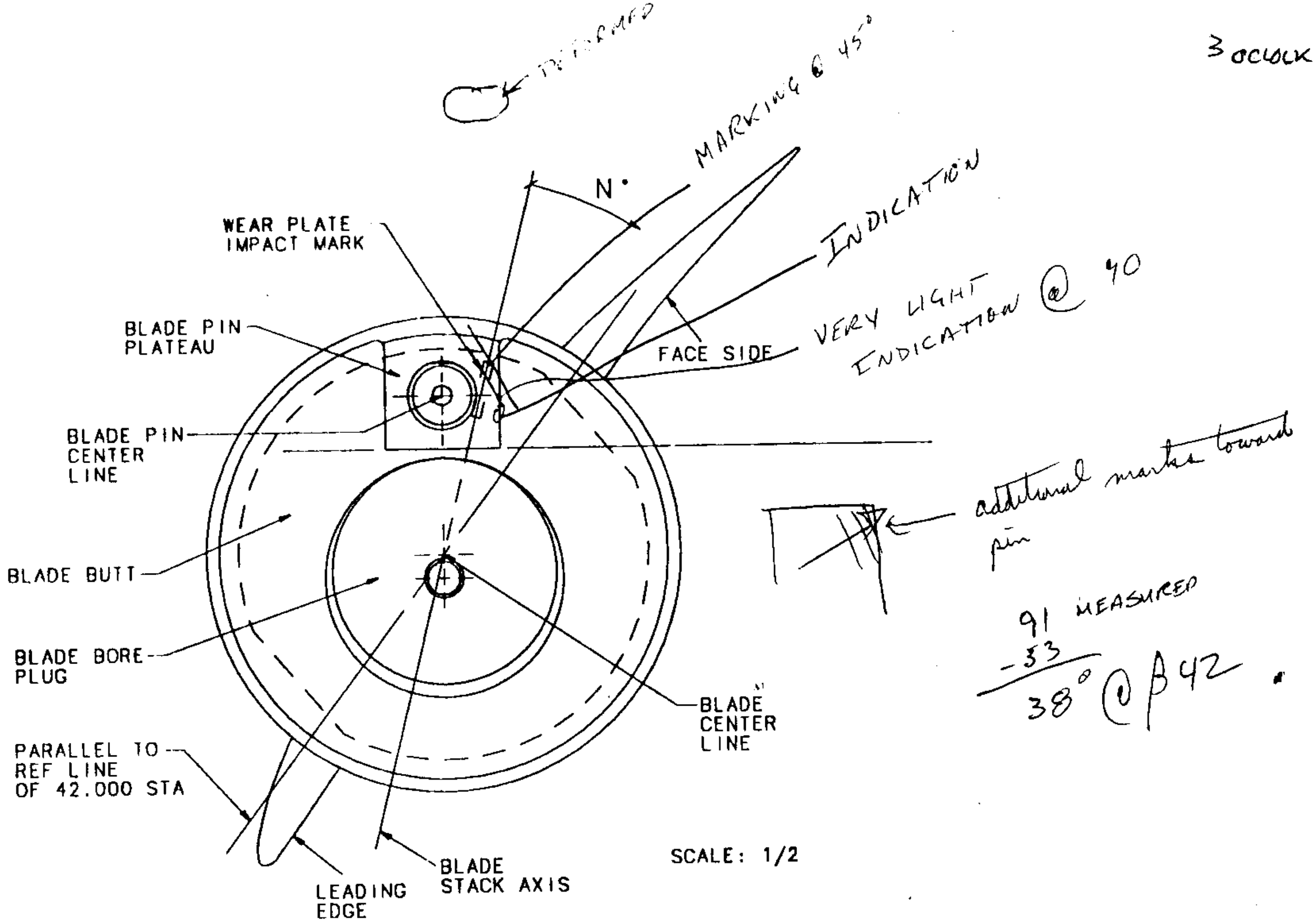
INSTALLED LCTN: R2
SITE LCTN: R2

14RF-9

3 OCLOCK

Attachment 7



IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°

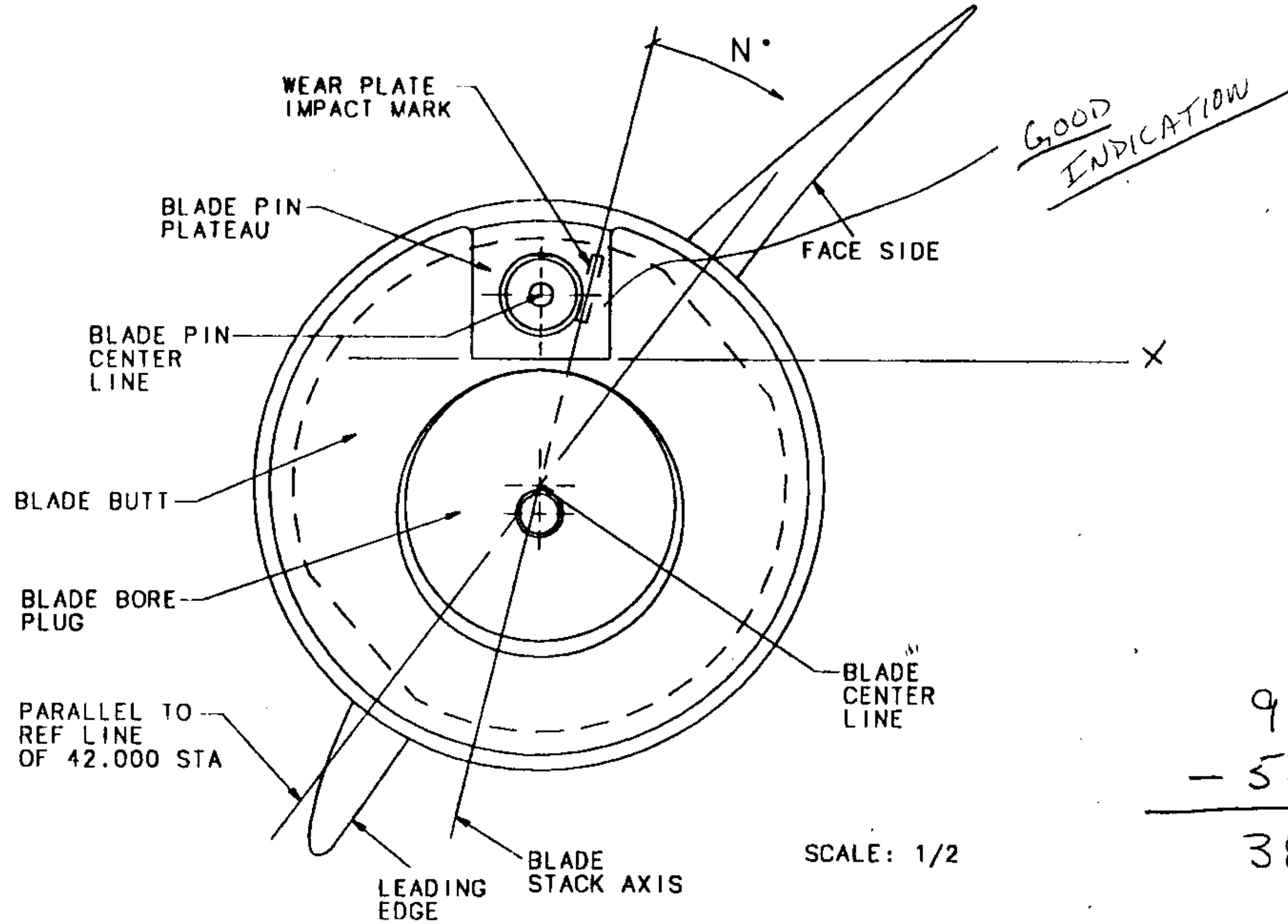
1.000

BLADE PN: RFC11N1-6A
BLADE SN: 860455
INSTALLED LCTN: R3
SITE LCTN: R3

REF DWG:802339 REV J
792232 REV AD

6 OCLOCK

LOW BLADE



Attachment 8

91°
− 53°
38° @ 42 STAT

IMPACT BLADE ANGLE CALCULATION

IMPACT BLADE ANGLE= N°-53°

1.000

REF DWG:802339 REV J
792232 REV AD

BLADE PN: RFC 11 N1-6A
BLADE SN: 872093
INSTALLED LCTN: R4
SITE LCTN: R4

Attachment 9

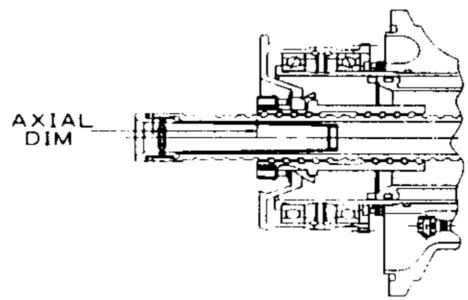
HAMILTON STANDARD

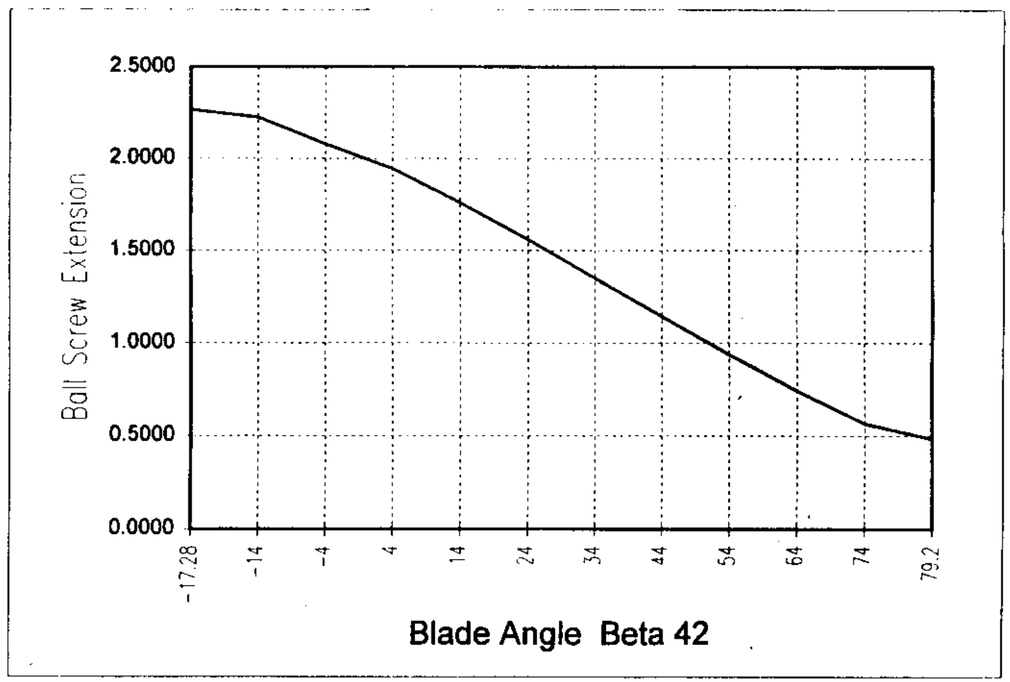
January 28,1997

(14RF PER A/S734070)

(Axial dim from face of nut to face of screw on the Ball Screw in the PCU)

BLADE ANGLE	PCU BALLSCREW EXTENSION	RADIAN EQUIVELANT OF BLADE ANGLE
-17.28	2.2660	-0.30159
-14	2.2241	-0.24435
-4	2.0780	-0.06981
4	1.9437	0.069813
14	1.7584	0.244346
24	1.5589	0.418879
34	1.3513	0.593412
35	1.3304	0.610865
36	1.3094	0.628319
37	1.2884	0.645772
38	1.2674	0.663225
39	1.2465	0.680678
40	1.2255	0.698132
41	1.2046	0.715585
42	1.1837	0.733038
43	1.1628	0.750492
44	1.1420	0.767945
54	0.9372	0.942478
64	0.7432	1.117011
74	0.5659	1.291544
79.2	0.4821	1.382301





ATTACHMENT 10

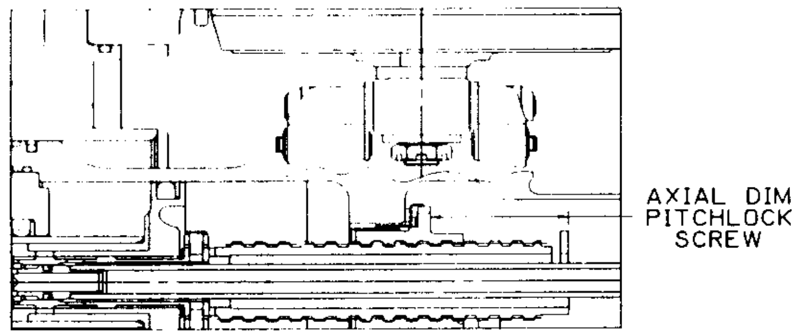
HAMILTON STANDARD

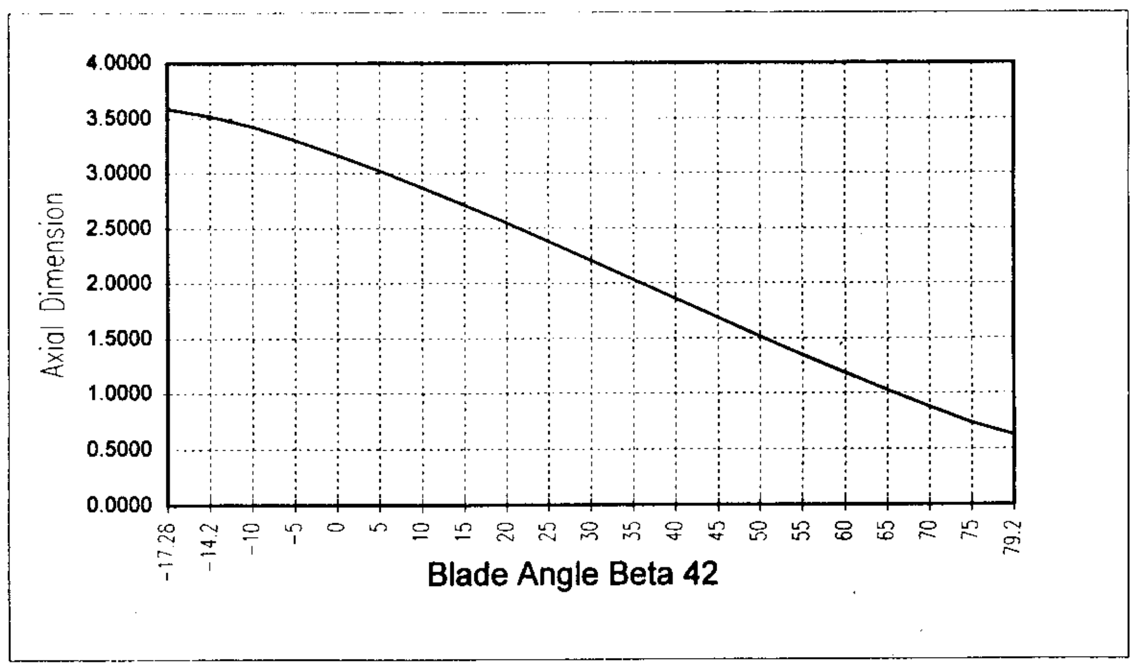
January 28,1997

(14RF per A/S734070)

Axial dimension from aft face of nut to aft face of screw on Pitch Lock Screw

	Blade Angle	Axial Dim
Reverse	-17.28	3.5883
	-14.2	3.5229
	-10	3.4265
	-5	3.3017
	0	3.1666
	5	3.0225
	10	2.8704
	15	2.7113
	20	2.5467
	25	2.3776
	30	2.2054
	35	2.0314
	40	1.8570
	45	1.6833
	50	1.5119
	55	1.3439
	60	1.1807
	65	1.0235
	70	0.8735
	75	0.7318
Feather	79.2	0.6200





HAMILTON STANDARD

January 28,1997

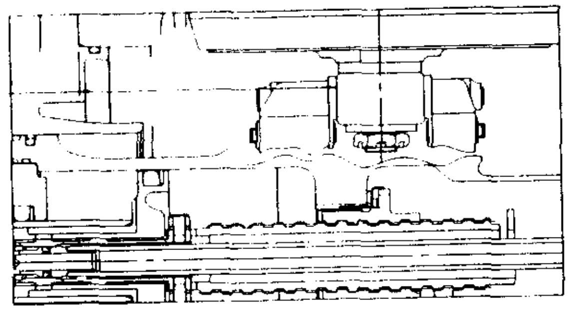
(14RF per A/S734070)

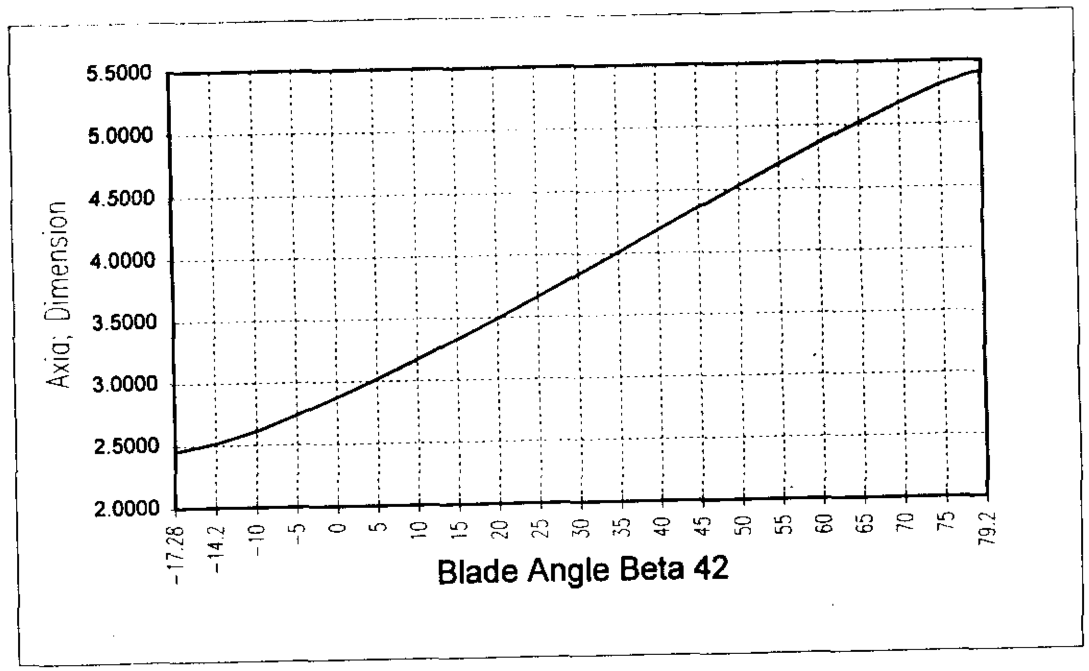
Axial dimension from fwd face of Hub to aft face of fwd wear plate on Yoke

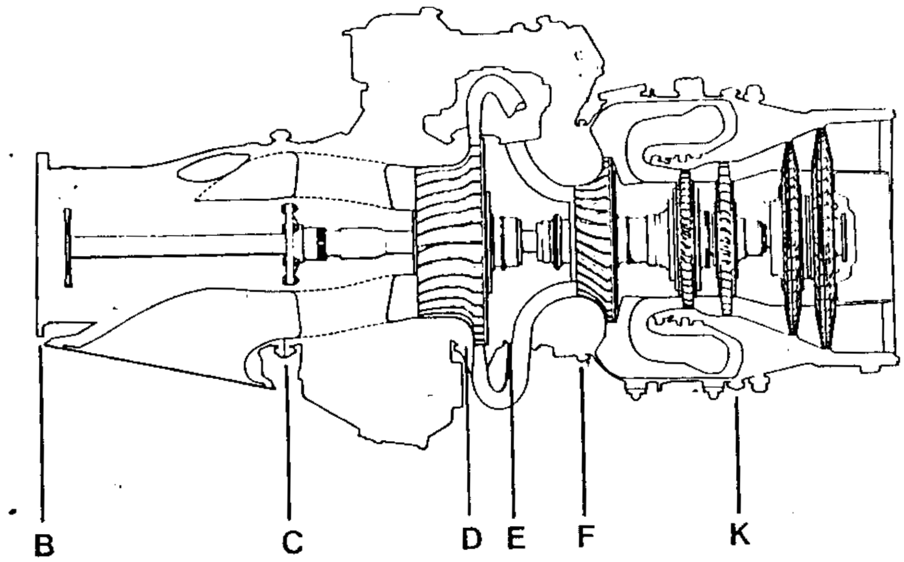
Attachment 11

	Blade Angle	Axial Dim
Reverse	-17.28	2.4528
	-14.2	2.5182
	-10	2.6146
	-5	2.7394
	0	2.8745
	5	3.0186
	10	3.1707
	15	3.3298
	20	3.4944
	25	3.6635
	30	3.8357
	35	4.0097
	40	4.1841
	45	4.3578
	50	4.5292
	55	4.6972
	60	4.8604
	65	5.0176
	70	5.1676
	75	5.3093
Feather	79.2	5.4211

AXIAL DIM
YOKE







FLANGES

B– Reduction gearbox to front air inlet case

C– Front air inlet case to rear air inlet case

D– Rear air inlet case to low pressure diffuser case

E– Low pressure diffuser case to inter compressor case

F– Inter compressor case to gas generator case

K– Gas generator case to turbine support case

ATTACHMENT 12