NATIONAL TRANSPORTATION SAFETY BOARD
Office of Aviation Safety
Ashburn, Virginia 20147

June 25, 2010

Airworthiness Group Chairman's Factual Report

WPR09MA159

A. ACCIDENT

Operator:	Eagle Cap Leasing Inc.
Location:	Butte, Montana
Date:	March 22, 2007
Time:	1430 Mountain Daylight Time (MDT)
Aircraft:	Pilatus PC12/45, N128CM

B. AIRWORTHINESS GROUP

Chairman:	Dennis J. Diaz National Transportation Safety Board Ashburn, Virginia
Member:	Rick E. Koffman Federal Aviation Administration Helena, Montana
Member:	Bryan Hanson Federal Aviation Administration Helena, Montana
Member:	Robert Renshaw Pilatus Business Aircraft, Ltd. Broomfield, Colorado
Member:	Steven T. Kurt Pilatus Business Aircraft, Ltd. Broomfield, Colorado
Member:	Konrad Oetiker Pilatus Aircraft, Ltd. Stans, Switzerland
Member:	Thomas Berthe Pratt and Whitney Canada South Burlington, Vermont

Member: Tom McCreary
Hartzell Propeller Inc.
Piqua, Ohio

C. SUMMARY

On March 22, 2009, at 1430 mountain daylight time, a Pilatus PC-12/45, N128CM, descended to ground impact in a cemetery near the approach end of runway 33 at the Bert Mooney Airport, Butte, Montana. The airplane was owned and operated by Eagle Cap Leasing, of Enterprise, Oregon, as a personal transportation flight under the provisions of 14 Code of Federal Regulations Part 91. The airplane was destroyed in the collision sequence and post crash fire. All 14 persons onboard the airplane were killed in the accident. There were no reported ground injuries. The flight departed Oroville, California, at 1210 Pacific daylight time on an instrument flight rules (IFR) flight plan and clearance destined for Gallatin Field, Bozeman, Montana. The airplane was diverting to Butte at the time of the accident. Visual meteorological conditions prevailed at both the Bozeman and Butte airports.

On March 23, 2009, the NTSB convened an Airworthiness Group at the accident site, which was located within the Holy Cross Cemetery in Butte, Montana. During the period of March 23rd through 24th the Airworthiness Group conducted an assessment of the accident site, and the wreckage. The wreckage was subsequently recovered to a facility on March 25th, where further examination of the wreckage and maintenance records took place through March 27th. Several follow-up examinations (including component data download/extraction) took place during the following months, involving various members of the Airworthiness Group, along with other participants (generally, component manufacturers).

Details of the on-scene activities, component examinations, and component data extractions are documented in this factual report.

D. DETAILS OF INVESTIGATION

D.1 Site Examination

The accident site was located about 3,900 feet from the runway 33 threshold at BTM, and about 2,100 feet west of the runway centerline. The entire debris field was contained within the bounds of the Holy Cross Cemetery. The initial impact point (IIP) was located at 45 degrees 57 minutes 01.64 seconds north latitude, 112 degrees 30 minutes 15.16 seconds west longitude. The IIP was a crater that measured 23 feet wide, 9 feet long, and 16 inches deep. The wreckage was deposited along a path that radiated about 245 degrees magnetic from the IIP. A tree and numerous tombstones, several of which were disturbed by the impact, were located along the wreckage path.

Figure 1 - Photo Showing Initial Impact Point and Approximate Energy Path

Components of the airplane found within the IIP included: the left main landing gear (which was oriented with the wing attachment point into the ground), portions of the left wing main spar, the hydraulic power pack (normally located near the left wing root-fuselage attachment point), the hydraulic selector valve, the inboard portion of the left wing flap, and the main aircraft battery (normally located in the tail, aft of the pressure bulkhead).

The right wing flap was separated from the right wing, and both were located about 16 feet forward of the IIP, and to the right side of the energy path. The left rear wing spar and the left aileron were located about 25 feet from the IIP, and on the left side of the energy path.

The fuselage was located along the energy path, with the tail being closest to the IIP at a distance of 20 feet. The remainder of the fuselage was located forward of the tail, oriented roughly 290 degrees magnetic. The fuselage was lying on its right side, and exhibited three main fracture separations. The engine remained partially attached to the firewall by the engine mounts.

The remaining portions of the left wing were distributed further along the energy path, up to 150 feet beyond the IIP.

Figure 2 - Photo of Fuselage (red bar highlights area from tail to engine)

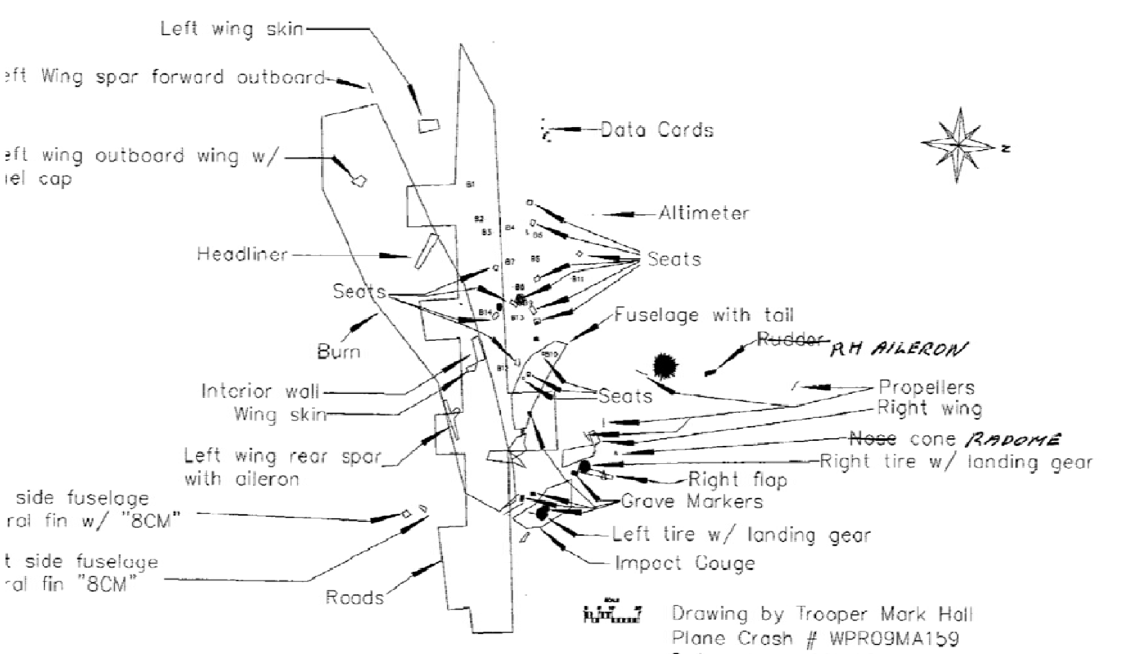

Figure 3 - Accident Site Diagram (prepared by the Montana Highway Patrol)

D.2 Airframe Inspection

D.2.1 Fuselage

The fuselage came to rest in three main pieces. The first piece consisted of the engine/engine mount and the forward fuselage. The cockpit was partially destroyed by fire. The second piece contained the fuselage from the approximate area of the emergency exit, and the third piece contained the empennage from the rear pressure bulkhead to the aft-most portion of the tail which came to rest on its left side, shearing the horizontal stabilizer off at its hinge points.

D.2.2 Left Wing

The left wing sustained heavy damage at wing root. The left main landing gear was fracture-separated from its attachment points, along with hydraulic power pack and the inboard portion of the rear spar. The left inboard flap actuator remained attached to the inboard portion of the rear spar, and was in the fully retracted position. The main battery was found immediately to the left of the main landing gear leg. The wheel rim was fractured into several pieces and the tire came to rest immediately to the right of the main landing gear leg. The left flap was separated from the wing, and was found to the left of the main landing gear leg and immediately to the left of the airplane's main battery. The rear spar was separated into three pieces. The outboard portion retained the outboard flap actuator, which was in the fully retracted position. The front lower spar fractured approximately 6 inches outboard of its main attach points. It carried approximately 3 feet of the carry-through spar with it. The carry-through spar piece came to rest a few feet forward and to the right of the initial impact point. The AOA plate and mast were found still attached to the wing. The vane was missing.

D.2.3 Right Wing

The right wing was sheared at its wing root. The inboard aileron control rod was separated immediately outboard of its threaded end. The outboard section came to rest ahead and to the right of the inboard wing section. The flap was separated from the main wing portion.

Figure 4 - Photo of Right Wing (from rear)

D.2.4 Landing Gear

The nose landing gear sheared approximately 10” above the wheel fork. The left main landing gear was torn from the left wing, was found at the IIP. The left wheel was shattered, the tire was torn off, and the landing gear actuator ram was sheared just outboard of the jam nut. The landing gear leg came to rest inverted about 1-2 feet below the surface of the ground. The right main gear trailing link was sheared. The upper attach point of shock absorber was attached to gear leg. The lower portion of shock absorber was torn away.

D.3 Flight Controls and Aerodynamic Surfaces

D.3.1 System Description

The flight control system utilized push-pull rods and carbon steel cables. An aileron/rudder interconnect system was installed to improve lateral stability and turn coordination. Electric trim systems were provided for the aileron, rudder, and stabilizer.

Aileron and horizontal stabilizer trim operation were controlled by switches on the outboard horn of each control wheel, while rudder trim operation was controlled by a switch located on the Engine Power Control Lever. Prior to engaging pitch or aileron trim, the pitch trim engage switch located on the forward side of each outboard control wheel horn had to be depressed. An electric actuator drove the movable horizontal stabilizer to adjust airplane pitch trim. The secondary trim motor, installed in the same actuator, was controlled by the autopilot, and could also be used as a backup pitch trim system. Alternate pitch trim was activated by pressing the ALTERNATE STAB TRIM switch in the desired direction.

Uncommanded trim operations ("runaway trim") could be stopped by pressing the TRIM INTR switch located forward of the Engine Control Quadrant on the center console.

Each wing trailing edge was equipped with a single piece Fowler-type flap supported by three flap arms. The flaps were controlled by a selector handle located to the right of the engine power controls on the center console. The flaps could be set to one of the four preset positions: 0°, 15°, 30° and 40°, by moving the handle to the appropriate position. If the flap lever was not at one of the four preset positions, the Flap Control and Warning Unit (FCWU) would drive the flaps to the nearest preset position. A flap position indicator was located near the top of the left instrument panel.

The flaps were electrically actuated by a single flap Power Drive Unit (PDU). The PDU drove four screw actuators (two for each flap) through flexible shafts. The screw actuators were connected to the flap actuating arms.

The flap control system incorporated a failure detection system. The system was designed to detect asymmetric flap extension, failure of the flaps to extend or retract to the selected flap position, and failure of an actuator. If the FCWU detected a failure, it would disconnect the power to the PDU and the illuminate the CAWS FLAPS caution annunciation light.

The airplane was equipped with a stick shaker-pusher system that was intended to prevent the airplane from inadvertently entering a stall condition. The stick shaker-pusher system contained two Angle-of-Attack (AOA) sensors, a single dual channel computer, a single stick shaker, a single aural warning device, and a single stick pusher. Either computer channel could independently provide stall warning (stick shaker and aural warning), but both computer channels were required to actuate the stick pusher.

D.3.2 Flight Control System Examination

All flight control surfaces were accounted for during the examination.

D.3.2.1 Elevator System

Continuity of the elevator control system was verified from both cockpit controls to the elevator. The forward elevator bellcrank assembly was inspected. The right cable and attach hardware were found intact, and both turnbuckle locking clips were found attached. The left elevator cable end was found with all attach hardware intact. The forward pair of elevator pulleys was free, with no obstruction to cable movement. The left elevator cable turnbuckle was located and both turnbuckle clips noted in correct placement and locked. The first elevator cable (still clamped to the stick pusher cable) was broken 100 inches aft of the aft stick pusher cable clamp, with the break displaying signatures consistent with overload separation ("broomstrawed"). The second elevator cable was found broken with "broomstrawed" cable strands 163 inches aft of its turnbuckle end. The matching ends of the elevator cable breaks were noted “broom-strawed” just forward of the elevator servo in the tail. One cable was broken at 7 feet 5 inches forward of the pitch servo cable clamp and the other was broken 68 inches forward of its respective pitch servo cable clamp.

Continuity of the elevator cables aft of their respective breaks was confirmed to their respective terminations at the elevator control lever in the vertical stabilizer. Elevator control rod continuity was verified intact up to the bellcrank, where it had sheared at its attach point. Cable cuts in the pass-through hole of the vertical stabilizer were noted.

The elevator servo was found intact on its mount, and cable clamps and hardware were still in place on their respective elevator cables. The left horizontal stabilizer hinge was still attached, while the right stabilizer hinge was found torn from stabilizer. The left tip of the horizontal stabilizer was compacted at a 45-degree angle, beginning at the forward tip.

D.3.2.2 Rudder System

Continuity of the rudder system was verified from cockpit controls to the rudder. Both rudder cable turnbuckles were found secured to the rudder bellcrank, and the turn-buckle locking clips were all secure. The first rudder cable was found with a “broom-straw” break 136 inches aft of the forward attach point. The second rudder cable was found with a “broomstraw” break 96 inches aft of the forward attach point. Rudder cable continuity from the breaks of both cables was confirmed to the tail. The left cable was out of its pulley and the right cable had partially exited its pulley. Cable connections at the rudder horn were secure.

The rudder servo clamps were found still secured to the cables, and the servo cable was still attached to servo. One rudder cable was broken 219 inches from its servo clamp forward end.

Figure 6 - Photo of Horizontal and Vertical Stabilizers (from rear)

D.3.2.3 Aileron System

Continuity of the aileron system was verified from cockpit controls to the ailerons. The pilot's aileron sprocket wheel and associated chain were found with the stick shaker actuator. The left half of the chain connection, its chain segment and cable attachment were found with hardware intact. A "broomstrawed" cable break was found on the cable 105 inches from its forward attach point. The remaining 50percent of the tie rod was found attached to the co-pilot's control yoke. The co-pilot's aileron sprocket wheel attached to a portion of its associated control yoke were found, along with the tie rod and approximately 50percent of its chain connection with 11 inches of aileron cable attached. The cable end displayed "broomstraw" breakage.

The left aileron cable, with a portion of its associated quadrant and cable, forward to the turnbuckle and 1 1/4 inches of cable forward of the cable swage to a "broomstraw" break, was found in the forward fuselage. A second part of the aileron cable was found with breaks at both ends, and was 107 inches long. The right aileron cable with approximately 75 percent of the aileron quadrant, and 40 inches of cable from the swage at the quadrant, was also located in the forward fuselage. A partial break of the cable and a complete "broomstraw" break was visible. The forward ends of the two fuselage aileron rods were attached to the aileron quadrant, but both rods were crushed and broken aft of the quadrant. The aft portion of the fuselage left aileron rod had ½-inch of the left aileron arm attached to it. The pivot shaft and outer arm were intact, and the attach hardware was secure. The inboard wing aileron rod was attached to the outer arm, but was broken at the base of the rod threads. The right aileron arm (inside the fuselage) was broken from its pivot shaft, with only the threaded portion of the right fuselage aileron rod attached to it. The pivot shaft and outer aileron arm were attached, but the end of the arm was broken off.

The aileron servo was found near the aileron quadrant with one cable still attached to the quadrant. Cable routing was good on the servo capstan. The other end of the cable was pulled from the quadrant with a portion of it broke away.

The aileron-rudder interconnect arm was found still clamped to its respective cable, which was broken in a "broomstraw" fashion 20 inches forward of the clamp.

D.3.2.3.1 Left Aileron

The center aileron rod was sheared at mid-span, just inboard of the outer flap actuator and remained continuous until the left aileron bellcrank. This area sustained considerable damage and the aileron was severed from the wing. The end of the outer aileron rod was sheared off, consistent with overstress.

D.3.2.3.2 Right Aileron

The wing inboard aileron rod was broken at the end of its threads. The clevis end was missing. Aileron rod continuity was present to the outer end of the wing outboard aileron rod, where the rod had sheared at the ends of the threads. The clevis end was still attached to the aileron bellcrank.

D.3.2.4 Trim System

All three control surface trim actuators were of a jack-screw-type design, and thus were not susceptible to movement by impact forces. The aileron and rudder trim actuators were manufactured by Electromech Technologies (part number EM4069-3). The horizontal stabilizer pitch trim actuator was manufactured by Elektro-metall (part number 129-1-1100-02).

The extension of each trim actuator's jack screw was measured, and then correlated to the respective control surface trim component position. The aileron trim actuator jack screw was extended 1.10 inches, from a possible range between 0.28 inches to 1.15 inches. This jack screw extension correlated to a 95-percent extension, with 0-percent being full left wing down trim and 100-percent being full right wing down trim. The measured position correlated to a nearly-full, right-wing-down aileron trim. The rudder trim actuator had the same extension range as the aileron trim actuator. The rudder trim actuator extension measurement was 0.28 inches, which correlated to a full nose-left rudder trim position.

The pitch trim actuator had an extension range between 3.69 inches and 4.95 inches. The actuator extension measurement was 4.78 inches, which was within the normal takeoff range, towards an airplane-nose-down position.

Figure 7 - Photo Visible Extension of the Aileron Trim Actuator

D.3.2.4.1 Aileron Trim Actuator Follow-Up Examination

A follow-up examination of the aileron trim actuator was conducted at the Electromech Technologies facility located in Wichita, Kansas, on April 9, 2009. In attendance were the Airworthiness Group Chairman, a representative of Pilatus Aircraft, and representatives of Electromech Technologies. The component was subjected to inspection and test according to a group-approved plan. The plan consisted of an external condition inspection, an external electrical continuity test, an internal component inspection, and an internal component electrical continuity test.

The aileron trim actuator was damaged during the crash sequence. The frame was bent along the actuator's axis, toward the center of the unit. The unit was externally fire-damaged. The external case screws remained attached, and the actuator rod was resistant to lateral or longitudinal movement. Measurement from the fixed end side of the gear housing to the movable end of the actuator ram tube was 5.089 inches (minus an airframe hardware thickness of .129 inches, plus .37 inches to reach the nominal distance to the bearing measurement point), which totaled 5.33 inches. A factory unit extended to its electrical limit stroke would be expected to extend to 5.32 inches.

A multi-meter was connected to the frayed and burned wires emanating from the unit. No reliable readings of electrical continuity could be obtained.

The external cover was removed and electrical continuity of the extended limit switch and position sensing potentiometer were recorded (see results below). Further disassembly revealed that the motor field magnets were impact-damaged and broken. The remaining mechanical components (including armature, gears, lead screw threads, etc) were intact and exhibited no anomalies.

The movement of the actuator was mechanically limited by an acme nut (or lead screw nut) and the stop tube. The as-found distance between the two components was roughly .010 inches, where a factory component would generally be expected to stop with .030 inches clearance (it should be noted that the deformation of the external case may account for this difference). The extend limit switch actuating spring was resting on the limit switch plunger.

Wires were internally connected to the extend limit switch. Electrical continuity testing of the extend limit switch revealed that based on the as-found condition, the switch was not contacting (and thus would not stop extension of the actuator). Given the physical condition/deformation of the unit and the observations made during the internal examination, the continuity test may not have been valid indication of the pre-impact condition. The internal potentiometer, which when connected to the airplane's instrumentation for aileron trim position, sensed the difference in resistance between two leads. The resistance measured between these two points was 1 k-ohm, which was also the total resistance provided by the potentiometer, and equated to full extension of the actuator.

D.3.2.5 Flap System

The flaps were single piece fowler-type flaps supported by three stations per wing. The flap actuators were of a jack-screw-type design. On each of the four actuators, the eye bolt connecting the actuator to the flap linkage was broken off immediately after the lock-nut of the ball-screw. All four flap actuators were set to a position consisted with the flaps retracted at impact.

Figure 8 - Photo Showing One of Four Fully Retracted Flap Actuators (left inboard shown)

D.3.2.6 Other Flight Control Information

The stick pusher servo, cable, and clamps were inspected. The length of cable extrusion from clamps was measured, and revealed that no clamp slippage had occurred. All clamp hardware was intact.

D.3.2.6.1 Other Flight Control Follow-Up Examination

The Airworthiness Group met to examine the Shaker/Pusher Computer at the manufacturer, EMCA Electronic, in Horw, Switzerland on May 12, 2009. After a brief meeting, it was confirmed that the Shaker/Pusher Computer did not contain any non-volatile memory, with the exception of "zero settings" for the angle of attack resolvers. Additionally, the unit was significantly impact-damaged and was deemed non-functional and no further functional testing could be performed.

D.4 Cockpit Systems

D.4.1 On-Scene Examination

D.4.1.1 Overhead Panel

The overhead panel was found with its faceplate missing. The External Power, Avionics 1, Avionics 2 and Non-Essential switches were missing. The Generator 2 switch was in the ON position. The Inverter switch was in the BAT position; the Standby Power was in the OFF position; Battery 1 was ON, Generator 1 was in the RESET position. Aside from the Generator 2 switch, all others listed above were broken internally, so visible switch positions are not reliable.

D.4.1.2 Instrument Panel

Pilot's Altimeter:
Thousands feet needle missing, hundreds needle indicated 780, which agreed with the altitude window. Altimeter setting indicated 29.57.

Co-pilot's Altimeter:
Only the face of the instrument was found with an altimeter setting of 30.14.

Airspeed Indicators:
The first indicator (installed position unknown) had a broken glass face with no needles attached. No witness marks were visible. The second indicator had a broken glass face with only the VMo needle attached.

Cabin Differential Pressure Indicator:
Differential pressure indicated 0.4 psi.

Vertical Speed Indicator:
One VSI face was found. Possible witness marks were observed on the face at the 300 to 400 fpm climb marks.

Flap Position Indicator:
Needle pointed to zero degrees flaps.

Cabin Vertical Speed Indicator:
Needle pointed to 2,200 fpm climb.

D.4.1.3 Engine/Cockpit Controls

The power quadrant was recovered, but all linkages and cables had been severed. The emergency fuel shutoff lever was in the stowed position. The emergency landing gear hand pump was broken from its mount position, but in the stowed position. The landing gear selector had substantial damage, and the landing gear handle was missing.

Figure 9 - Photo Showing Power Quadrant

D.4.1.4 Avionics

The airplane was not equipped with a cockpit voice recorder (CVR) or flight data recorder (FDR) and was not required by Federal regulations to be so equipped.

During the on-scene examination of the avionics, units were divided into two categories that determined the subsequent follow-up action. Units having non-volatile memory (NVM) were separated and retained for possible memory extraction. All other avionics were examined on-scene only.

Those units identified as possibly having viable NVM were: the Integrated Hazard Awareness System (IHAS), two Attitude Heading and Reference System (AHRS) units, the Caution and Advisory Control Unit (CACU), and the Engine Instrument System (EIS).

Other avionics found that were not able to provide pertinent information included an ADF Radio, a Transponder-KT70 (all knobs and face were broken, could not determine settings), an Air Data Computer (CIC-8800), Bendix Nav/Comm unit, avionics 1 bus, generator 2 bus circuit breaker panel, KMC321 mode controller, pilot's side lower EFIS display, emergency power supply, KNI 582 RMI Unit, King COM unit, faceplate missing, Symbol generators (2), ELT unit, KMD850 multi-function controller, EFIS controller, KN40 Navigation Receiver, triple trim indicator (faceplate was broken, indicator needles missing), and portions of the Avionics 1 and Generator 2 circuit breaker panels.

D.4.2 Integrated Hazard Awareness System (IHAS) Follow-Up Examination

The IHAS unit was shipped to the manufacturer, Honeywell, for examination under the oversight of an NTSB investigator on April 30, 2009. The unit was severely fire-damaged, and most of the electronic components were separated from the printed circuit boards (PCBs). The flash memory chips containing all stored memory for the device had also separated from their PCB. The missing flash memory chips were not located, and presumed destroyed.

Figure 10 - Exemplar PCB and memory chips (below) and accident PCB with missing memory chips (above)

D.4.3 Attitude Heading Reference System (AHRS) Follow-Up Examination and Data Extraction

Members of the Airworthiness Group and participants of the component manufacturer met at the Northrop Grumman LITEF GmbH facility in Freiburg, Germany, on May 13, 2009, to examine both of the AHRS units. The main purpose of the examination was to download recorded fault information from memory chips located within each of the units. For identification purposes, the units were arbitrarily labeled 'A' and 'B'.

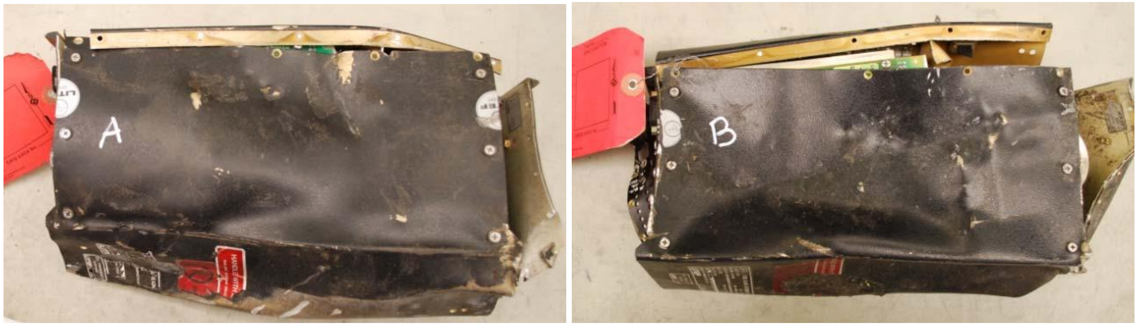

Figure 11 - Photos showings 'A' and 'B' AHRS units

Both of the AHRS units were impact-damaged. Fastening hardware was removed and/or drilled so that the component PCBs within the respective cases could be accessed. Four PCBs were contained within, with the memory located on the “processor module” board. Each of the processor modules was damaged, and data could not be downloaded with the memory chips in-situ. The decision was made to de-solder both of the memory chips (U12 and U13) from the processor modules, and re-solder them onto surrogate processor module boards.

The unit 'A' chips were removed and placed onto a surrogate processor module board without difficulty. The unit 'B' chips exhibited external damage and cracking, and the legs of the unit ‘B’ U12chip were broken, but were re-created using a “haywire” process.

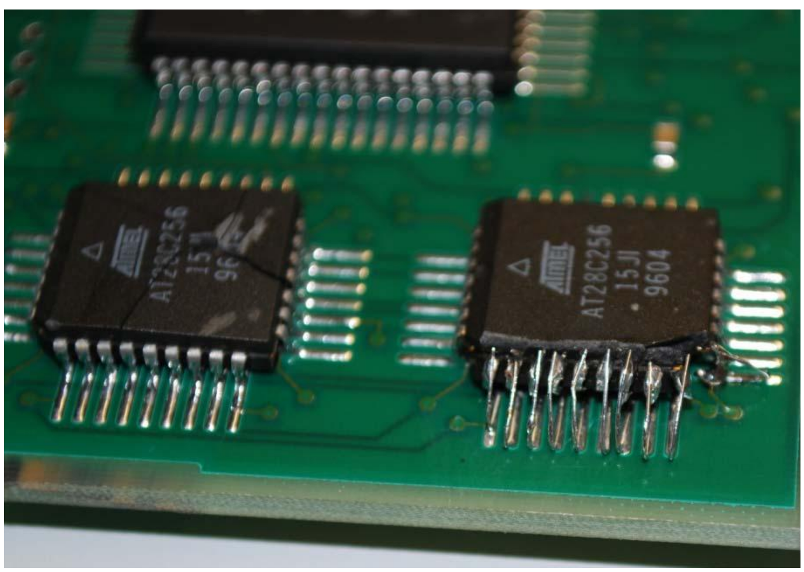

Figure 12 - Photo of unit 'B' memory chips and 'haywire' of memory chip (right)

The unit 'A' surrogate processor module was connected to the memory download device, and the contents of the memory chips were downloaded successfully. The data was then processed through an interpretation program, which produced a text output file. The raw text contained within the file can be found in <u>Attachment 1: AHRS 'A' Data – Text</u>.

The data showed that unit 'A' had accumulated 2006.7 total hours of operation, with 90 total faults logged during that time. The most recent fault was logged at 1950.7 hours, and no faults relevant to the accident flight were discovered.

The unit 'B' surrogate processor module was connected to the memory download device, and the contents of the memory chips were downloaded successfully; however, examination of the data by interpretation software revealed that the data from the U12 chip was corrupt, and therefore, invalid.

D.4.4 Central Advisory and Warning System (CAWS)

D.4.4.1 CAWS Description

The Central Advisory and Warning System (CAWS) integrated the control and display functions of aircraft systems status into a single unit. The CAWS was comprised of a Central Advisory Control Unit (CACU) and a Central Advisory Display Unit (CADU). The CACU was installed under the cabin floor between frames 20 and 21. The CADU was installed in the lower center section of the instrument panel.

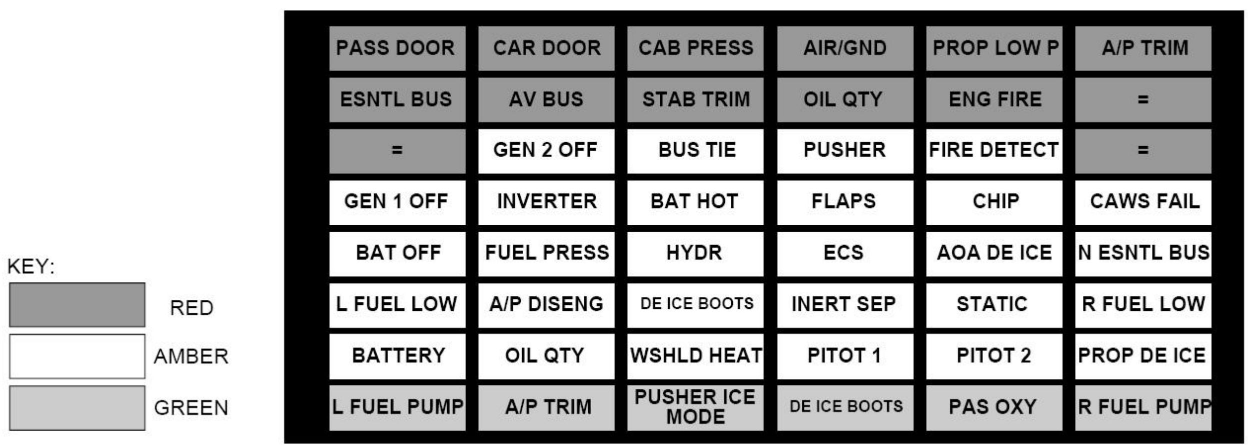

Figure 13 - CADU instrument face diagram (taken from PC-12 Approved Airplane Flight Manual)

The CADU received information regarding which annunciator to illuminate from the CACU via the serial bus. The CADU displayed 48 individual captioned annunciations. Each annunciator was comprised six colored Light Emitting Diodes (LED) connected in parallel covered with a legend panel. The CADU annunciators indicated warning, caution, and advisory conditions. As the annunciators were LEDs, and not light bulbs, no analysis could be performed to determine if any particular warnings, cautions, or advisories were illuminated at the time of impact.

A red warning light would have indicated a condition that required an immediate corrective action by the pilot. It would have been accompanied by a voice callout and illumination of the master WARNING light. An amber caution light would have indicated a condition that required a pilot's attention, but not an immediate reaction. It would have been accompanied by illumination of the master CAUTION light and an aural gong. A green advisory light would have indicated that a system was in operation.

Red master WARNING and amber master CAUTION lights were positioned on the instrument panel directly in front of the pilot and copilot. They would have alerted the pilot to changes in status of the CADU annunciators. Any condition that caused a red or amber annunciator to illuminate also would have caused the applicable master WARNING or CAUTION light to illuminate. A voice callout would have sounded through the overhead speaker and/or headset(s) any time a master WARNING light illuminated. Pushing the applicable master WARNING or CAUTION light would have extinguished the light. The CADU warning or caution annunciator that triggered the master WARNING or CAUTION light would have remained illuminated.

D.4.4.2 CACU Follow-Up Examination and Data Extraction

Members of the Airworthiness Group and a representative of the German Bundesstelle für Flugunfalluntersuchung (BFU) met at the facilities of Aircraft Electronic Engineering (AEE) GmbH in Seefeld/Droessling, Germany on May 18, 2009, to examine the CACU and attempt to extract data from the NVM. During the examination, the manufacturer explained that the components that retained NVM (contained on flash memory) were located on a sub-board, which also contained a processor and random access memory modules. The sub-board was intact, but impact damaged, and many of its input/output pins were bent. A plan was formulated to remove all of the pins that were not necessary for accessing and downloading the data stored in the flash memory. The remaining pins were straightened, and the positive (+) and negative (-) power pins were connected to a current-limited power supply. Subsequent continuity testing revealed that the pins had shorted, and that a download of the memory from the accident sub-board was not possible.

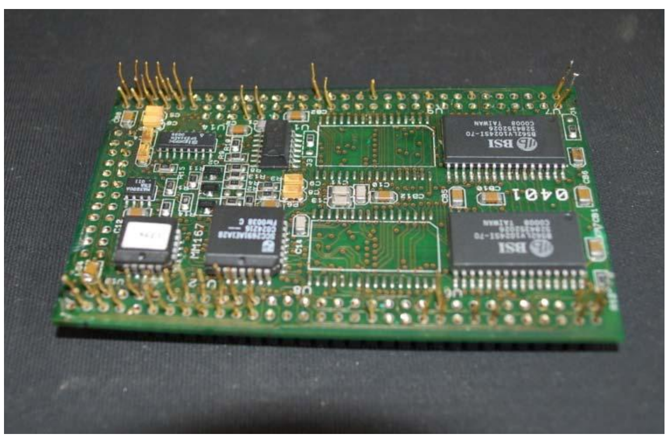

Figure 13 - Sub-board Pins Cut and Straightened

The group decided that the next best course of action would be to remove the flash memory chips, and transplant them onto a surrogate board, for subsequent download. Since AEE acquired the sub-board from a third party vendor, they did not have the equipment and experience in-house to remove the chips and re-attach them to another board. The group then enlisted the help of the German BFU to perform this task.

The BFU and AEE succeeded in downloading and decoding the data from the accident memory chips. The process and methods utilized by the BFU and AEE are detailed in Attachment 2: BFU Concise Engineering Report – CACU Data Extraction. The raw tabular data obtained can be found in Attachment 3: CACU Data – Tabular.

Interpretation of the data can be found in: Attachment 4: Non-Volatile Memory Data Study.

D.4.5 Engine Instrument System (EIS)

D.4.5.1 EIS Description

The EIS was a computer-controlled system that received input from various sensors. It displayed engine and other system information, and provided warnings when certain parameter limits were exceeded. The EIS had two independent Acquisition and Processing Units and a Display Unit, which were all combined within a single cockpit mounted housing.

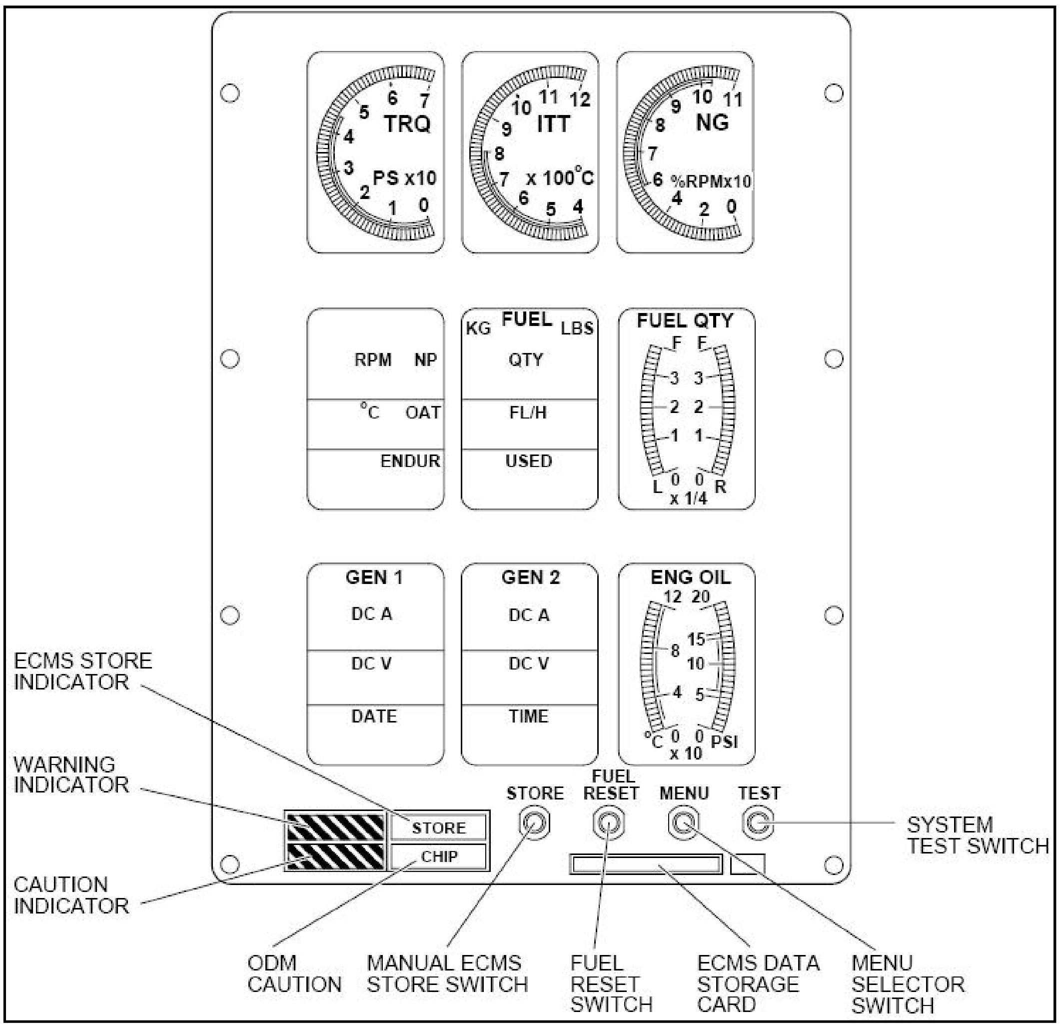

Figure 14 - EIS instrument face diagram (taken from PC-12 Approved Airplane Flight Manual)

The Engine Condition Monitoring system (ECMS) monitored the health of the engine and detected engine deterioration, thus enabling planned corrective and/or preventative maintenance actions. The ECMS periodically recorded engine data and corrected the reading to correspond to ambient conditions (air data). The corrected data was then compared with a set of typical engine characteristics where any deviation could easily be detected. Additionally, the system recorded all engine limits that were exceeded and the engine's run time. Any parameter exceedance would trigger that parameter to be recorded, along with all other parameters, from the previous 2 minutes.

In addition to exceedance information, engine trend information was recorded any time the engine was running and a pre-defined set of parameters remained stable for more than 2 minutes. Those parameters were: intermediate turbine temperature (ITT) between 600 and 800 degrees C, torque between 15 and 44.4 psi, altitude between 10,000 and 35,000 feet, no weight on wheels, engine inertial separator door closed, and engine bleed air takeoff at P2.5. Data would typically be recorded at a rate of once per second for a period of 2 minutes. Data recording would cease for a subsequent 2-minute period, then resume for another 2 minutes. If any of the previously described parameters exceeded their limits, or ceased being stable, data recording would stop immediately.

Regardless of whether or not trend or exceedance criteria were met, and data was recorded for a given flight, flight summary information was recorded at the beginning and ending of every flight. This summary information recorded would have been the maximum values reached for each parameter during the flight, along with system startup time, engine start/stop time, flight start/stop time, along with other system parameters.

When required, the ECMS processor in the EIS recorded the following parameters: ITT, torque, gas generator speed (Ng), propeller speed (Np), fuel flow, engine oil pressure, engine oil temperature, weight on wheels condition, engine P3 bleed air condition, inertial separator system mode, pressure altitude, true airspeed and total air temperature. Those parameters would have been written to a "compact flash" -type data storage card, which was inserted in the front of the EIS. The EIS display also had a manual store button and a store annunciator, which illuminated when recording was taking place. This was also an indicated to the pilot that the system was operating correctly.

A typical trend log entry can be seen in Table 1 below ("expanded" format shown).

Table 1 - EIS Trend Log Entry Example	
AutoTrending Data at [22/03/2009 11:51:36]:	
ITT	: Val = 00707.40
Engine Torque	: Val = 00027.39
Ng	: Val = 00099.42
Np	: Val = 01702.40
Fuel Flow	: Val = 00378.53
Engine Oil pressure	: Val = 00115.80
Engine Oil temp	: Val = 00053.30
Airspeed	: Val = 00162.25
Altitude	: Val = 25040.00
Total Air Temp	: Val = -0031.50
Weight on Wheels	: FLIGHT
P2.5 / P3	: P25 28VDC
Bypass	: GND
Chip Counter Cycle	= 000000
Number of flight Counter	= 001172
Number of engine run Counter	= 001116

D.4.5.2 EIS Data Card Download

The trend card was recovered from the accident EIS intact, but was slightly bent. With oversight by the Airworthiness Group Chairman, representatives of Pilatus downloaded information from the EIS compact flash card utilizing a standard laptop computer and compact flash card reader. The data was successfully downloaded using software normally provided by Pilatus to operators for analyzing engine trend information. The software provided the data in several file formats, including an expanded file format (text) that recorded both engine trends and exceedances. Raw tabular data obtained from the EIS Data Card can be found in Attachment 5: EIS Data Tabular.

Each flight was sequentially numbered by the EIS, beginning with flight 1152 on February 27, 2009, and ending with flight 1172 (the accident flight) on March 22, 2009. While the recorded date information correlated to known activity of the accident airplane, the recorded times could not be correlated to any known time base. By offsetting the stated time values by 06:45:27, a correlation with Caution and Advisory Control Unit (CACU) recorded data, and subsequently with actual time, was established (see Attachment 4: Non-Volatile Memory Data Study for further information on the CACU).

Examination of the data revealed that no engine exceedances were recorded with the exception of several propeller overspeeds. All of the recorded propeller overspeeds were recorded during an engine start. According to Pilatus, this behavior of the EIS recording system was a known software "bug," where propeller overspeed at engine start was recorded when none actually existed.

Pertinent data summarizing the events on the date of the accident flight are shown below (all times expressed in UTC). Note that since all summary information other than

EIS System Start is recorded during a normal shutdown of the system, no concluding information was recorded for the accident flight (EIS flight 1172).

Table 2 - EIS Flight No. 1170 (CACU Flt. No. 1557)	
EIS System Start	14:36:09
Engine Start	14:37:13
Flight Start	14:43:14
Flight Stop	16:34:57
Engine Stop	16:38:04

Table 3 - EIS Flight No. 1171 (CACU Flt. No. 1558)	
EIS System Start	17:09:27
Engine Start	17:10:48
Flight Start	17:16:50
Flight Stop	17:41:50
Engine Stop	17:46:38

Table 4 - EIS Flight No. 1172 (CAWS Flt. No. 1559)	
EIS System Start	18:02:36
Engine Start	N/R
Flight Start	N/R
Flight Stop	N/R
Engine Stop	N/R

Engine trend information was recorded for two periods during flight 1170 and for two periods during flight 1172. No trend information was recorded for flight 1171. Average values for selected parameters during those periods are shown below.

Table 5 - EIS Flight No. 1170 First Trend Period	
Trend Period Start	15:11:55
Trend Period Stop	15:16:52
Approximate Altitude	26,000 ft
Average OAT	-24° C
Average CAS	166 KCAS
Average Fuel Flow	352 PPH

Table 6 - EIS Flight No. 1170 Second Trend Period	
Trend Period Start	15:23:53
Trend Period Stop	16:05:05
Approximate Altitude	22,000 ft
Average OAT	-24° C
Average CAS	184 KCAS
Average Fuel Flow	352 PPH

Table 7 - EIS Flight No. 1172 First Trend Period	
Trend Period Start	18:37:03
Trend Period Stop	20:05:24
Approximate Altitude	25,000 ft
Average OAT	-32° C
Average CAS	174 KCAS
Average Fuel Flow	386 PPH

Table 8 - EIS Flight No. 1172 Second Trend Period	
Trend Period Start	20:18:33
Trend Period Stop	20:18:52
Approximate Altitude	15,000 ft
Average OAT	-10° C
Average CAS	177 KCAS
Average Fuel Flow	398 PPH

D.5 Fuel System

D. 5.1 Fuel System Description

Fuel was stored in two integral wing tanks that were each subdivided into two portions, a main tank and a collector tank. Drain valves were located in both the main and collector portions of each fuel tank. Refueling was accomplished using over-wing filler caps, located on the upper side of the main portions of both fuel thanks. Each wing tank had a usable fuel capacity of 201gallons. Fuel venting was accomplished through the use of both inward and outward vents.

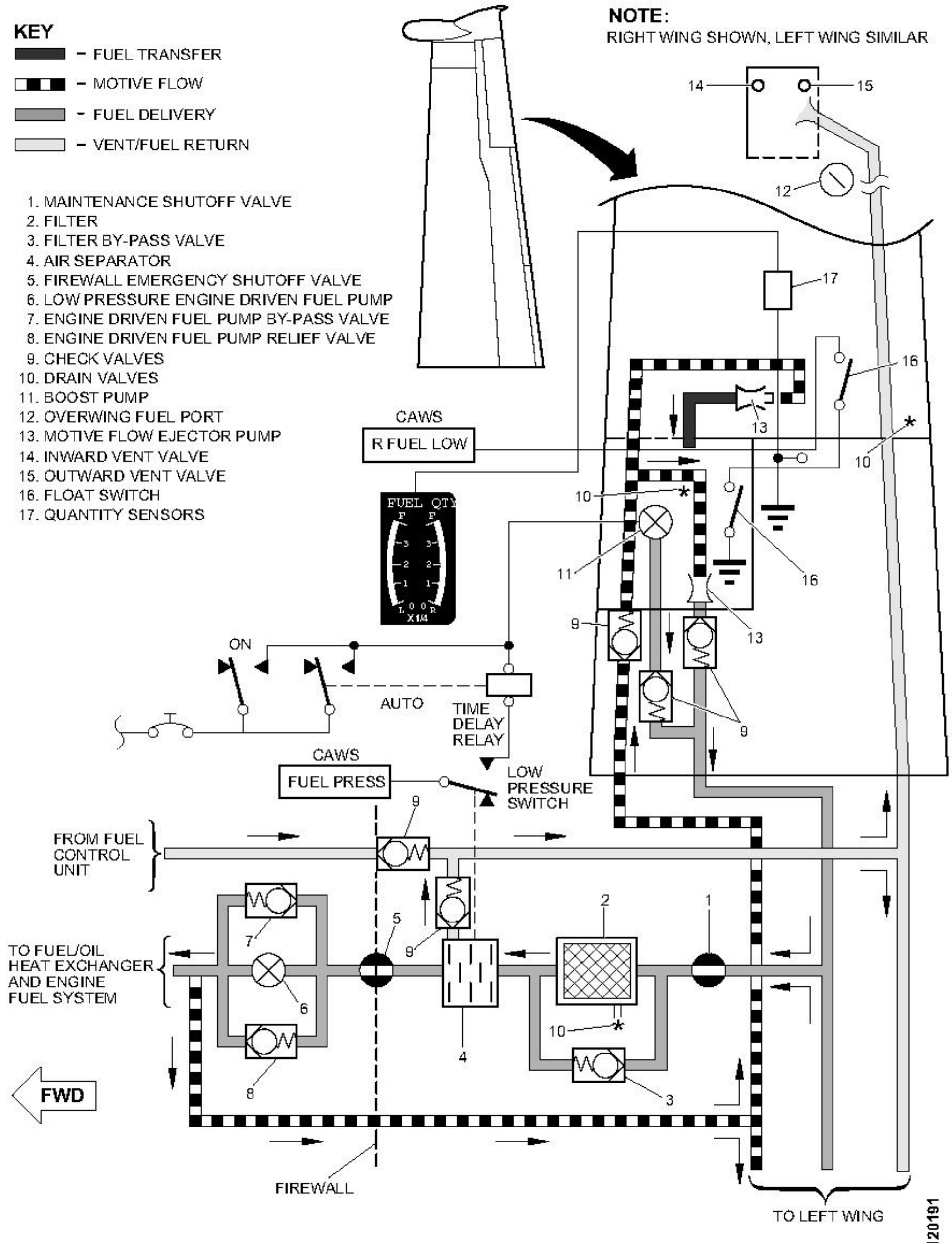

Figure 15 - Fuel System Diagram (taken from PC-12 Approved Airplane Flight Manual)

The fuel distribution system transferred fuel from the left and right main wing tanks into the respective collector tanks through one way valves located between the two fuel tanks. The transfer was facilitated by a transfer ejector pump located in each main tank. Fuel was fed from the collector tanks, through a common manifold, toward the engine primarily via delivery ejector pumps. The nominal output pressure of the delivery ejector pumps was 5 psi. Both the transfer and delivery pumps were of a 'jet pump' type, and were powered by a motive flow of fuel from the low pressure engine driven fuel pump. Electric fuel boost pumps located in each collector tank were used to provide fuel in the event that either of the delivery ejector pumps could not supply the required fuel pressure. The nominal output pressure of the boost pumps was 31 psi. The boost pumps were also used to provide fuel pressure for engine start, and to laterally balance the fuel load.

From the wing tanks, fuel flowed forward through a fuel filter, maintenance and firewall shutoff valves, an air separator, a low pressure engine driven pump, an oil/fuel heat exchanger, and a high pressure engine driven pump to the fuel control unit.

The fuel filter incorporated a bypass valve, and in the event the fuel filter became blocked, a spring loaded valve would open and allow fuel to bypass the fuel filter. The valve was calibrated to operate at a differential pressure of 8 psi (+/-1 psi). In the event of a bypass condition, a differential pressure indicator (DPI) would extend a red, resettable button that the pilot could observe during the next preflight inspection. The DPI was not visible to the pilot while in flight.

An air separator passed air in the fuel system to a vent line and incorporated a fuel low pressure switch. In the event that fuel system pressure fell below 2 psi (and the fuel boost pump switches were set to the AUTO position), both the left and right fuel boost pumps would activate and attempt to restore fuel pressure. The boost pumps would automatically turn off 10 seconds after the fuel system pressure had reached 3.5 psi. A single boost pump was capable of supplying the engine with fuel in the event of a low pressure engine driven fuel pump failure.

Fuel quantity was determined through the use of four capacitance-type fuel probes located in each fuel tank. Information on fuel quantity was computed by the EIS, and displayed to the pilot utilizing an LED display, segmented into 28 'bars' of fuel quantity for each tank. Fuel symmetry was automatically maintained by a fuel balancing device whenever the fuel pump switches were positioned to AUTO. The left and right fuel quantities were monitored to detect fuel asymmetry exceeding 5 percent of each wing's total fuel capacity (about 10.5 gallons, or 2 bars on the pilot's fuel quantity display), and would activate the fuel boost pump in the tank with the higher fuel quantity. Activation of the applicable fuel boost pump was delayed one minute to avoid pump cycling during flight in turbulence. The boost pump would continue to operate until the fuel levels equalized. The fuel balancing system will normally attempt to correct fuel imbalances up to 40 gallons (6 bars), beyond which, the system will no longer operate automatically. In the event of a system failure, fuel symmetry could be maintained by selecting the fuel boost pump to the ON position for the tank with the greater fuel quantity.

According to the PC-12 Approved Airplane Flight Manual (AFM), Section 2, Limitations, the maximum permitted fuel imbalance was 26.4 gallons (178 lbs, or 3 bars).

The CAWS captions associated with fuel system status were:

- L FUEL PUMP, R FUEL PUMP – Fuel boost pump in operation
- L FUEL LOW, R FUEL LOW – Fuel level in tank less than 20 gallons (133 lbs)
- FUEL PRESS – Fuel system pressure less than 2 psi

The L/R FUEL PUMP CAWS green advisory lights were the sole indication to the pilot that the boost pumps were operating while in flight. However, because of the way the system was configured, this light was not a positive indication of pump operation, but rather an indication that power had been applied to the relay that powered the respective pump. No positive direct indication of operation or of boost pump output was provided to the pilot. As such, per the AFM Before Starting Engine Checklist, pilots were required to manually activate each boost pump individually and audibly verify operation prior to each flight.

According to the AFM, the airplane was limited to utilizing JET-A, JET-A-1, JET-B, and JP-4 fuels (with the exception of other fuels that complied with the latest revision of Pratt and Whitney Service Bulletin 14004). All flight operations in ambient temperatures below 0 degrees C required the use of fuel ant-icing additives conforming to MIL-DTL-27686 or MIL-DTL-85470 (sometimes referred to generically as Fuel System Icing Inhibitor or FSII). The additive was to be applied in concentrations between 0.06 percent and 0.15 percent by volume. Concentrations less than 0.06 percent were considered insufficient to prohibit the formation of fuel system ice, while concentrations greater than 0.15 percent caused damage to the protective primer and sealants of the fuel tanks, as well as to the seals in the fuel system and engine components.

D.5.2 Fuel System Follow-Up Examination

Members of the Airworthiness Group met at the NTSB Eastern Regional office in Ashburn, Virginia on November 16 and 17, 2009, for the examination of the fuel system components removed from the accident airplane.

D.5.2.1 Airframe Fuel Filter

The airframe fuel filter was recovered in three pieces consisting of the fuel filter housing, the fuel filter bowl, and the fuel filter element. Each exhibited fire and varying degrees of impact damage.

Figure 16 - Airframe Fuel Filter Assembly

The fuel filter housing contained an integrated fuel filter bypass valve, which was intact (with the exception of the differential pressure indicator, which was separated). The position of the bypass valve and the intact portion of the differential pressure indicator could not be determined. The bypass valve was removed from the filter housing, and forwarded to the NTSB Materials Laboratory for further examination. Examination results can be found in NTSB Materials Laboratory Factual Report number 10-005, which resides in the public docket for this accident.

The fuel filter element was severely impact and fire damaged. No obvious evidence of blockage or deposits was observed.

Figure 17 - Airframe Fuel Filter Elements

The maintenance fuel shutoff valve was also attached to the fuel filter housing. The valve appeared partially closed, and the valve handle was not free to move. Due to the degree of damage to the valve handle, the position of the valve was considered unreliable by the group.

D.5.2.2 Fuel-Oil Heat Exchanger

Several ounces of fuel were drained from the fuel-oil heat exchanger during the initial recovery in Bozeman, Montana. The drain valve was subsequently removed during examination in Virginia. No fuel was observed at that time. The exchanger was forwarded to the NTSB Materials Laboratory for mechanical sectioning, and examination of the corrugated face of the heat transfer surface for contamination. Examination results can be found in NTSB Materials Laboratory Factual Report number 10-005, which resides in the public docket for this accident.

D.5.2.3 Engine Driven Low Pressure Fuel Pump

The pump was recovered in two major pieces, while additional, smaller remnants were not. The two recovered portions consisted of the input drive (with input driveshaft intact), and the pump rotor with vanes. The driveshaft was intact and was not sheared. The pump was forwarded to the NTSB Materials Laboratory for further examination.

Examination results can be found in NTSB Materials Laboratory Factual Report number 10-005, which resides in the public docket for this accident.

D.5.2.4 Fuel Lines

Portions of the fuel lines from the supply, motive flow, and spill return sections were identified and separated from the other non-fuel lines that were recovered, and were forwarded to the NTSB Eastern Regional Office in Ashburn, Virginia. Due to the extensive impact and fire damage, no additional information could be obtained.

D.5.2.5 Check Valves

The following check valves were recovered and identified:

- Electric boost pump check valves (left and right), with the left valve fractured into two pieces, and the right valve intact and fire-damaged. The right valve was opened by the group, and found to be absent of debris.
- Motive flow return check valves (left and right), with the left valve fire- and impact-damaged, and the right valve impact-damaged.
- Outward vent valves (left and right), which appeared relatively intact.
- Inward vent valves (left and right), which were impact-damaged, but were able to pivot about their respective hinges.
- Fuel air separator check valve, which was separated from the fuel air separator (the shell of which was torn into three pieces), but appeared intact.
- Both flapper valves from the right fuel tank, which were free to pivot. (The left fuel tank flapper valves were not recovered.)

D.5.2.6 Fuel Quantity Indicating Probes

Seven of eight fuel probes were recovered. Three were identified as being from the right wing, while the remaining four could not be correlated with either wing. Due to the impact and fire damage observed, no further information was available. One of the four fuel low level floats was found intact.

D.5.2.7 Other Fuel System Components

The following additional components were recovered, but no further findings were made during their examination other than their general condition:

- Engine fuel recovery (EPA) can (which was absent of fuel).
- Firewall fuel shutoff handle.
- Firewall shutoff valve, which was free to rotate.
- Right wing fuel drain.

D.5.3 Delivery, Transfer, and Boost Fuel Pumps Examination

Members of the Airworthiness Group and participants in the examination from the component manufacturer met at the Crane Aerospace and Electronics facility in Elyria, Ohio on September 15, 2009, for the examination of the delivery, transfer, and boost fuel pumps.

D.5.3.1 Right Fuel Delivery Jet Pump

The pump exhibited fire damage, and was fractured at the conical center section. Disassembly revealed that the flapper valve was intact and free to move. No sealing ring was present on the flapper valve, but the interior displayed heat distress. The jet nozzle was absent of debris.

Figure 18 - External View of Right Fuel Delivery Jet Pump

D.5.3.2 Left Fuel Delivery Jet Pump

Only the screened portion intake was present (the flapper valve portion was not located). The jet nozzle had a small piece of debris bridging its diameter.

Figure 19 - Recovered Portion of Left Fuel Delivery Jet Pump

D.5.3.3 Right Fuel Transfer Jet Pump

The entire pump unit was recovered. The jet nozzle was absent of debris.

D.5.3.4 Left Fuel Transfer Jet Pump

The entire unit was recovered. The jet nozzle was packed with dirt.

D.5.3.5 Right Fuel Boost Pump

The entire pump was recovered, but exhibited extensive fire and heat damage. The impeller housing was removed and the impeller (made of Torlon 4275 plastic) was melted within the housing. The impeller and housing were held for further examination in the NTSB Materials Laboratory. Examination results can be found in NTSB Materials Laboratory Factual Report number 10-005, which resides in the public docket for this accident.

The electrical motor commutator exhibited normal wear, and the brushes were in good condition and free to move. Electrical testing of the thermal fuse showed that no continuity was present (thermal fuse melts above 378 degrees Fahrenheit).

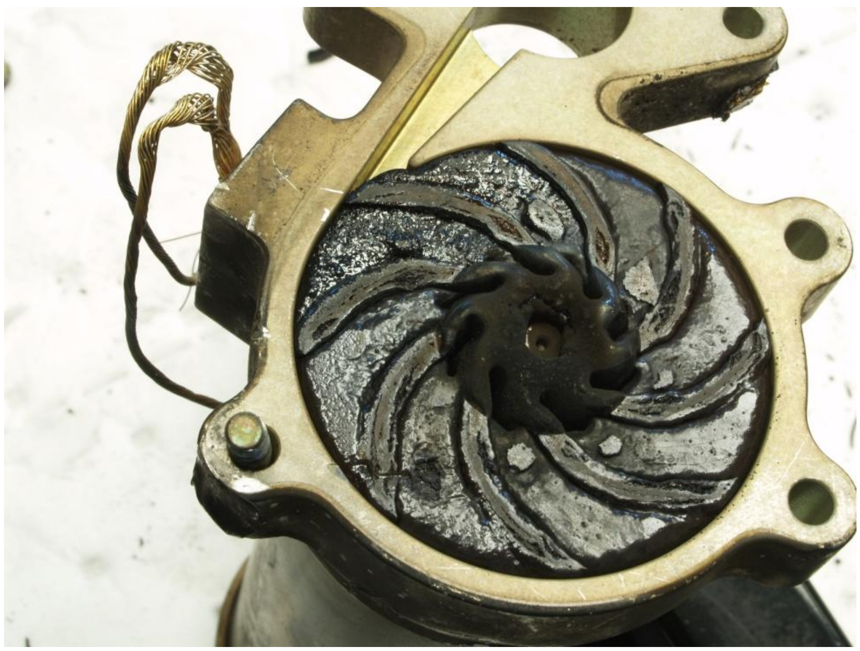

Figure 20 - Melted Right Fuel Boost Pump Impeller

D.5.3.6 Left Fuel Boost Pump

Only the impeller housing was recovered. Disassembly revealed the impeller remained intact, with the exception of two blades that were broken or cracked. The impeller and housing were held for further examination in the NTSB Materials Laboratory. Examination results can be found in NTSB Materials Laboratory Factual Report number 10-005, which resides in the public docket for this accident.

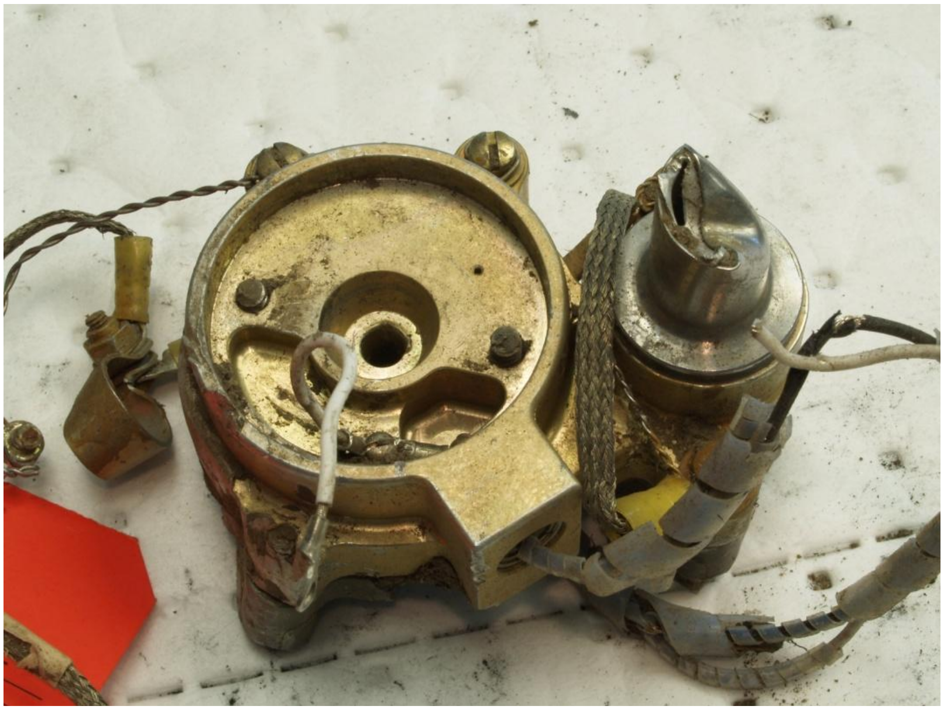

Figure 21 - Left Fuel Boost Pump Impeller Housing

D.6 Other Airframe Components and Systems

D.6.1 Anti-/De-Icing System

Inflatable neoprene boots were installed on the leading edges of the wings and horizontal tail surfaces. Their purpose was to inflate and disperse ice which may have formed on their surfaces during flight in atmospheric icing conditions. When not in use, the boots had a vacuum applied to prevent partial inflation while in flight. The inflatable pneumatic deicing boots were fixed to the leading edges of the wings (two boots per wing- inboard and outboard) and the horizontal stabilizer. Bleed air from the 3rd stage of the engine compressor section was routed to the regulator-reliever valve at a nominal 14-psi regulating pressure, then through a water separator to the ejector flow control valves. The valves, which were solenoid-operated, ported air pressure to the deicing boots in a prescribed sequence: first to the horizontal stabilizer deicer, then to the lower portion of the inboard wing deicers, the upper portion, the lower portion of the outboard wing deicers, and finally the upper portion. Progression through the sequence was controlled by an electronic timer/controller and monitored by low pressure sensing switches in each line, which were linked to annunciator lights in the CAWS.

The system was initiated by pressing the switch labeled BOOTS on the DE-ICING section of the overhead panel. When the switch was in the ON position, an ON annunciator in the switch was illuminated and a green DE ICE BOOTS annunciator on the CAWS illuminated. When activated, the deicing boot timer actuated each ejector flow control valve in the prescribed sequence, for eight seconds. The time interval to inflate and deflate all of the de-icer units was 40 seconds. There was then a dwell period of 20 seconds (if the `one minute cycle´ had been selected) or of 140 seconds (if the `three minute cycle´ had been selected) before the inflation sequence was repeated.

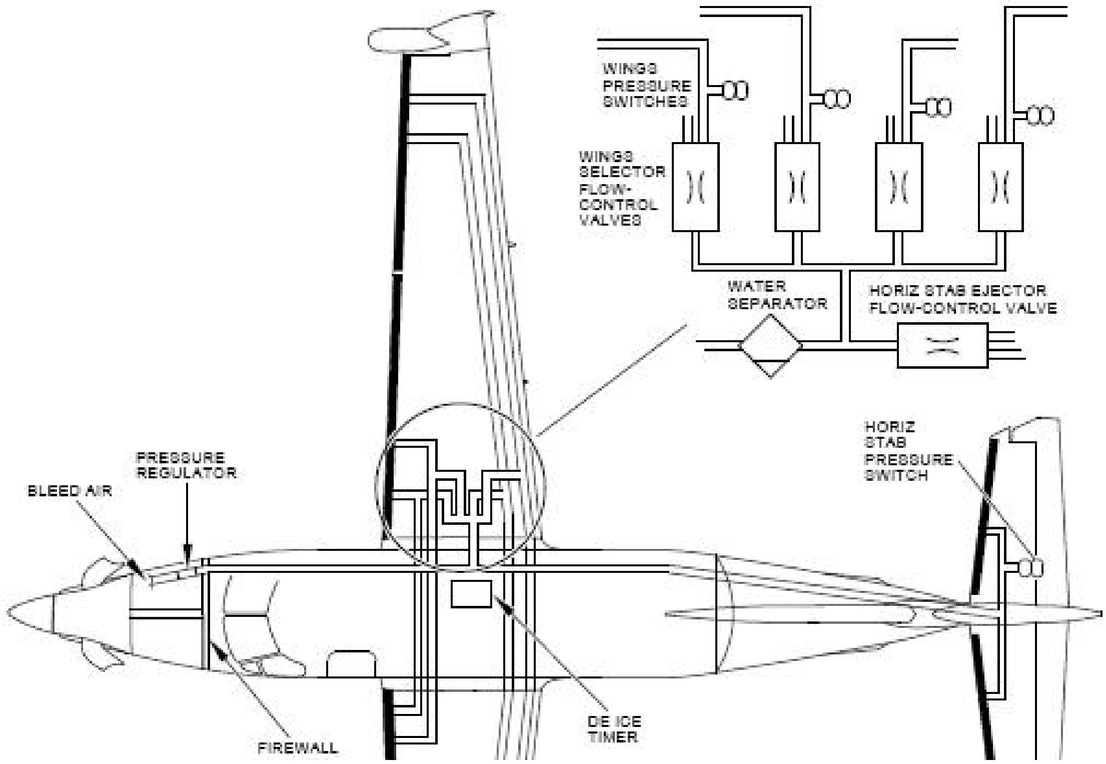

Figure 22 - Pneumatic Deice System Diagram
(taken from PC-12 Approved Airplane Flight Manual)

The inertial separator was of the `fixed geometry´ design and provided engine induction system protection when operating in icing or foreign object debris (FOD) conditions. It was comprised of a fixed mesh screen attached to the rear wall of the plenum covering a percentage of the inlet area, a moveable outlet door and electrical actuator situated directly above the oil cooler outlet exit, and a converging by-pass duct.

In normal operations (non icing, non FOD) the outlet door was closed which sealed the by-pass and provided the induction air with a single flow path to the plenum and engine through the porous screen. In icing or FOD conditions the actuator was retracted to open the outlet door and allow a flow path past the plenum to ambient air and increased the pressure ratio across the inlet system. The increased pressure ratio had the effect of accelerating heavy particles present in the inlet air, which then went past the plenum and into the bypass duct before exiting through the outlet door. In icing conditions, the porous screen iced to restrict the flow path of solid particles, which could not turn into the plenum

D.6.1.1 On-Scene Examination

No icing system cockpit switches were located. The inertial separator actuator extension was measured at 0.55 inches, which equated to the OPEN position of the inertial separator door. The deice boots timer was located intact, to be sent to manufacturer for testing.

Figure 23 - Photo Showing Extension of Inertial Separator Actuator

D.6.1.2 Boots De-Icing Timer Follow-Up Examination

Members of the Airworthiness Group and participants in the examination from the component manufacturer met at the Goodrich facility in Uniontown, Ohio, on April 7, 2009 for the examination of the de-icing boots timer.

The de-icing boots timer was damaged, with the top and two sides of the case were dented inward, but the unit remained intact. The unit was not fire-damaged. The safety wiring retaining the cover screws and connector receptacle was intact, as was the unit seal (although it appeared that the seal had been compromised).

A multi-meter was connected to the various electrical connector pins on the unit. The power pin (pin B) was tested against the solenoid activation pins (output pins E, F, G, H, and J). Each of the pins E, F, G, and H indicated no electrical connection (open), while pin J, indicated electrical continuity (short). According to a diagram for the unit provided by Goodrich, the results corresponded to one unit applying power to a de-ice valve solenoid at the time power was lost. Specifically, it was the final sequential sole-noid, the “wing top outboard” de-icing boot.

Figure 24 – Photo of Electrical Continuity Check of the Boots Deicing Timer Pin B to Pin J

The safety wire, retaining screws, cover seal, and cover were removed and the timer's internal components were examined. Deck-A of the rotary switch was cracked and damaged, and two small pieces were found loose inside the case. The electrical continuity that was observed on the external connector (continuity to pins B and J) was also confirmed on the corresponding internal rotary switch tab (deck C, tab 3).

Continuity was also tested for the "home" or off position between tabs 1 and 12 of the deck B rotary switch, with no continuity observed (consistent with the unit not in the off position). Continuity for the "dwell" or between cycle position, was measured between tab 12 of the deck B rotary switch and pin D on the connector. The measured value was 1.17 mega-ohms and increasing, which corresponded to the unit not being in the dwell position (it would have measured a short if it was).

D.6.2 Airplane/Cabin Environment On-Scene Examination

The airplane was equipped with two flight crew seats and eight passenger seats. The seating configuration was identified by the manufacturer as "six plus two," and was comprised of six "executive" style seats along with two "commuter" style seats. A placard noting the seating configuration was located inside the frame of the cargo door and read, "EX-6S-2".

The cabin was destroyed by impact and a post crash fire. The contents of the cabin were distributed along the wreckage path. About 300 pounds of luggage was recovered from the area along the wreckage path beyond where the fuselage came to rest. The baggage area cargo net was consumed by fire.

All eight passenger seats and both crew seats were dislodged from their mounts (the seat rail mount was separated from the airframe structure), and exhibited varying degrees of impact and fire damage. Of the eight passenger seats, only two exhibited evidence of the lap belts being buckled at impact. The remaining six seat belts exhibited too much impact/fire damage to make a reliable determination of the seatbelt state at impact.

Cabin Equipment/Furnishings:
The cargo net contained 12 tie-down points. Each had a ring on it. Of the ten rings found, seven exhibited severe elongation. One of the adjustable net straps was also recovered. One ring (lower) was found still attached to its seat rail. The ring was severely elongated.

Cargo Door:
The forward two lower two rollers were intact and still attached to the door jamb. The aft roller was sheared from its attach point. A portion of the door handle and door structure were recovered. The handle was in the closed position. The lower portion of the cargo door was recovered, and the aft cargo hook was still intact. It appeared to be in the closed position. One door shoot bolt was found in a small piece of door structure; the shoot bolt was in the extended (door closed) position. The lower portion of the cargo door frame was found with a portion of the aft outboard floor.

D.8 Engine

D.8.1 Engine On-Scene Examination

The engine was lying upright in the airframe nacelle and displayed severe impact damage. The firewall area displayed severe fire damage. The cowling was largely separated or consumed by fire. The airframe engine mounts, inlet shrouding, exhaust stubs, and pneumatic ducting displayed severe impact deformation. The propeller hub mounting flange and propeller spinner back plate remained attached to the propeller shaft, but the hub was disintegrated and the blades separated. The starter generator was fractured from its mounting boss and was located separately. The engine mount structure, oil cooler hoses, and wiring harness were cut as required to remove the engine for detailed inspection. Severe impact and fire damage precluded assessment of pre-impact continuity of the engine to firewall connections.

Figure 25 - Photo of Engine and Nacelle, Right Hand Forward View

Figure 26 - Photo of Engine and Nacelle, Left Hand View

D.8.1.1 Engine External Cases

Reduction Gearbox:
The reduction gearbox housing was largely disintegrated, exposing the propeller shaft and reduction gearing. The propeller shaft was intact. The propeller governor and torque limiter were located adjacent to the gearbox, and exhibited severe impact damage. Portions of the fractured reduction gearbox housing were located separately. The propeller overspeed governor and Np [propeller] tachometer generator were not

located. The propeller shaft and 2nd stage gearing were displaced approximately 20 degrees to the right and downwards.

Exhaust Duct:
The duct exhibited severe compressional deformation, and radial deformation to the duct underside. The duct displayed no indications of outward pockmarks or dimples.

Gas Generator Case:
The housing exhibited severe compressional and radial deformation. The compressor inlet case support struts were fractured. The compressor bleed valves were in place and intact. The fuel nozzle manifold exhibited severe deformation and multiple fractures.

Accessory Gearbox:
The rear housing was separated from the oil tank housing around the right hand circumference. The left side of the oil tank housing exhibited multiple impact fractures. The rear housing exhibited severe fire damage. The high pressure fuel pump was fractured from its mounting boss. The oil to fuel heater was in place with impact damage. The ignition exciter was not recovered. The fuel control unit exhibited severe fire and impact damage. The input lever linkage was fractured and deformed. The fuel control manual override linkage was continuous with airframe linkage. The lever was in the stowed position, and the lock wire gate was intact.

Power Control and Reversing Linkage:
The forward linkage was disintegrated. The Teleflex cable was continuous, with severe impact deformation from the flange “A” fitting to the accessory gear box fitting. The cam box and rear linkage were disintegrated.

D.8.1.2 Pneumatic Lines

Compressor Discharge Air (P3):
The line was continuous, with deformation from the gas generator case fitting to the P3 filter housing, and to the fuel control unit fitting. All connections were intact.

Power Turbine Control (Py): Continuous, with deformation from the fuel control unit fitting to fractures forward of flange “C”.

Chip Detectors and Filters:

Reduction Gearbox Chip Detector:
The mounting boss was fractured from the accessory gearbox housing and was located separately. The magnetic elements were free of ferrous debris.

Oil Filter:
Free of debris.

Fuel Filter:
Fire and heat damage to the filter housing precluded removal.

P3 Filter:
Clean. The filter could be blown through with normal lung pressure.

D.8.3 Engine Follow-Up Examination

Deformation precluded formal disassembly. The gas generator case was mechanically sectioned around flange “C” and the engine mount collar to separate the gas generator and power sections, and the exhaust duct was mechanically sectioned as practicable for inspection access.

D.8.3.1 Compressor Section

The compressor 1st stage was observed in-situ through the compressor inlet. The 1st stage blades displayed circumferential rubbing and deformation, opposite the direction of rotation, from radial contact with the 1st stage shroud. The blade leading edges displayed circumferential nicks, gouges, and tears consistent with the passage of ingested debris. The 1st stage shroud exhibited circumferential scoring abeam the 1st stage blade tips.

Figure 27 - Photo of Compressor First Stage Blades, Detail

D.8.3.2 Combustion Section

Combustion Chamber Liner:
Deformation of the outer liner precluded removal. The outer liner to inner liner mating flange was sectioned radially to facilitate removal of the power section. The inner liner was sectioned axially to facilitate separation from the power turbine housing. The inner and outer liner displayed no indications of operational distress. The flame pattern indications appeared normal.

Figure 28 - Photo of Combustion Chamber Outer Liner and Compressor Turbine

Small Exit Duct:
Observed in-situ, as damage precluded removal. The duct exhibited no indications of operational distress.

D.8.3.3 Turbine Section

Compressor Turbine Vane Ring:
Observed in-situ, as damage precluded removal. The outer drum was displaced aft, and the vane airfoils were fractured, with compressional deformation of the gas generator case. The vane airfoils displayed no indications of operational distress.

Compressor Turbine Shroud:
Displayed no indications of operational distress.

Compressor Turbine:
The vane airfoil trailing edges and tips were circumferentially machined and fractured. Under macroscopic examination, the fracture surfaces exhibited mechanical damage and coarse dendritic features characteristic of overload fracture, and no indications of fatigue or other progressive fracture mechanism. The disc upstream side exhibited circumferential rubbing. The disc downstream side displayed severe circumferential rubbing and scoring with the 1st stage power turbine vane ring and interstage baffle. The disc hub displayed heavy circumferential rubbing, with frictional heat discoloration and material smearing,.

Figure 29 - Photo of Compressor Turbine upstream Side

ITT Probes, Busbar, and Harness:
The probes were disintegrated.

Power Turbine Housing:
Displayed impact deformation. The forward flange was sectioned radially, and the housing was sectioned axially at the approximate 1:00 o'clock position to facilitate removal.

First Stage Power Turbine Vane Ring:
The vane airfoils were disintegrated The inner drum and interstage baffle were physically separated and could be removed by hand. Vane airfoil fragments were distributed within the power turbine housing. The upstream side inner drum and interstage baffle face displayed severe circumferential rubbing and scoring due to axial contact with the compressor turbine disc. The downstream side interstage baffle face displayed severe circumferential rubbing and scoring The baffle inner cup was machined from the baffle and pressed into the compressor turbine hub.

First Stage Power Turbine Shroud:
Displayed heavy circumferential rubbing.

First Stage Power Turbine:
Observed in-situ, damage precluded removal. The blade airfoils were fractured at their roots. .Four non adjacent blade platforms were displaced axially aft from their serrated fixings.. The four displaced blade platforms and fractured blade material were located within the power turbine housing. All of the blade roots were deformed away from the direction of rotation and displayed heavy mechanical damage. The recovered blade tips displayed circumferential rubbing to the blade tip shrouds. Under macroscopic examination the fracture surfaces displayed mechanical damage and coarse dendritic features characteristic of overload fracture, and no indications of fatigue or other progressive fracture mechanism. The disc face displayed heavy circumferential rubbing.

The disc hub displayed heavy circumferential rubbing, with frictional heat discoloration and material smearing,.

Figure 30 - Photo of First Stage Power Turbine

Second Stage Power Turbine Vane Ring:
Observed in-situ, damage precluded removal. The outer drum was deformed inward around the lower circumference. The vane airfoils were deformed, fractured, and circumferentially rubbed and machined. The upstream side inner and outer drums displayed heavy circumferential rubbing and scoring. The downstream side outer and inner drums displayed heavy circumferential rubbing.

Second Stage Power Turbine Shroud:
Displayed heavy circumferential rubbing and scoring.

Second Stage Power Turbine:
Observed in-situ through the exhaust duct and the 2nd stage power turbine vane ring. The blade airfoils were fractured. All of the blade roots were deformed opposite the direction of rotation and displayed heavy mechanical damage. Fractured blade debris was located within the exhaust duct.


Figure 31 - Photo of Second Stage Power Turbine Vane Ring

D.8.3.4 Reduction Gearbox

The reduction gearing was observed as exposed by the fractured reduction gearbox housing. The propeller shaft and 2nd stage gear set were displaced to the right and downwards, and the 2nd stage sun gear coupling spline was partially displaced from the 1st stage planet gear carrier. All of the gearing and carriers displayed normal running wear, with no indications of any operational distress.

D.8.3.5 Accessory Gearbox

The accessory gearbox was not disassembled.

D.8.3.6 Controls and Accessories Evaluation

Due to the severe fire and damage to the engine controls and accessories, they were not removed for further evaluation, with the exception of the engine-driven low pressure fuel pump and the fuel control unit (FCU)/high pressure engine-driven fuel pump package.

D.8.3.6.1 Engine Driven Low Pressure Fuel Pump

See fuel system section D.5.2.3 for detailed information regarding engine-driven low pressure fuel pump examination.

D.8.3.6.2 Engine FCU and High Pressure Engine Driven Fuel Pump

The fuel control unit and pump were removed as a package. The exterior was blackened, and exhibited evidence of fire and impact-related damage. Upon separating the FCU from the fuel pump, it was possible to rotate the FCU driveshaft by hand, but some resistance was evident. The driveshaft was bent. The P3 bellows had separated from the bellows end piece at the FCU base side of the bellows. The solder which had a melting temperature of 430 deg Fahrenheit, had been melted. There was a puddle of solder to one side location of the bellows cusp area.

The high pressure engine driven fuel pump driveshaft was broken, with the remains of the shaft in the splines of the drive-gear. When the FCU was removed from the pump, the driveshaft could be turned by hand with normal resistance. The bypass valve was correctly oriented and was removed without incident. The packing on the bypass valve cover was damaged by the post crash fire and had a pasty texture. The drive gear and driven gear had a dark blue coloration, consistent with exposure to heat. The sealing and contact faces of the gears had a polished appearance with no leak-paths present.

Following disassembly of the pump, the bypass valve was installed in a slave pump and the function of the valve tested in accordance with test record sheet TR3360 Rev. 07 (Eaton CMM 73-10-12 Rev. 1). With the pump speed at 810 rpm, the bypass valve pressure drop was 21.2 psid (the slave pump bypass valve pressure drop at this condition was 23.6 psid). With the drive speed at 6,350 rpm, the bypass valve pressure drop was 43.2 psid (the slave pump bypass valve pressure drop at this condition was 42.1 psid). The difference in performance between the slave pump bypass valve and the bypass valve from the pump under investigation was considered to be insignificant, confirming that the bypass valve from the pump under investigation was functional.

No evidence of any pre-impact mechanical malfunctions or failures of the FCU or the high pressure fuel pump were observed.

D.9 Propeller

D.9.1 Propeller Information

General Information:
The propeller was a four-blade, single-acting, hydraulically operated, constant speed model with feathering, and reverse pitch capability. Oil pressure from the propeller governor was used to move the blades to the low pitch (blade angle) position. A feathering spring and blade counterweight forces were used to move the blades toward the high pitch/feather direction in the absence of governor oil pressure. The propeller incorporated a Beta mechanism that actuated when the blade angles were lower than the flight idle position.

The propeller utilized an aluminum hub with aluminum blades. Rotation was clockwise as viewed from the rear.

Blade angle settings (at the 30-inch station) were:

Reverse:	-17.5	degrees
Flight Idle:	19.0	degrees
Feather:	79.6	degrees

D.9.2 Propeller Examination

The propeller was severely fragmented. All four blades were found separated from the engine. The aluminum hub was fragmented. The piston/cylinder assembly had separated from the hub. The spinner dome was fragmented.

Numerous small parts were not recovered, including: three blade bearings, one blade counterweight, one blade pitch change bracket, two fork bumpers, a portion of the pitch change rod, and two blade preload plates.

Engine/Propeller Mounting:
The engine flange and propeller attachment bolts were intact. However, the propeller hub was fractured around the circumference of the mounting bolts.

Cylinder:
The cylinder was found separated from the hub. The cylinder had a deep gouge consistent with contact with a blade counterweight. The blade/counterweight was at a pitch position in the operating range (not feathered, not in reverse) when the impact mark was created.

Piston:
The piston was retained inside the cylinder and was unremarkable.

Pitch Change Rod:
The pitch change rod was bent on the aft side of the fork. The rod was fractured on the aft side of the piston. It was also fractured on the forward side of the fork. The segment between the fork and piston was missing.

Fork:
The fork had impact marks caused by inward movement of the blade pitch change knobs. Three of the four fork tines were cracked on the thin-walled (low pitch) side of the fork. Two of the four fork bumpers were fractured and missing; the other two were damaged.

Feather Spring:
The feather spring was intact and retained inside the cylinder.

Reverse Pitch Stop:
The reverse pitch stop was intact and retained in the cylinder.

Feather Stop:
The feather stop was intact. The feather stop nut position was extended 1-13/32 inch from the feather stop, which equated to an 18.5-degree blade angle. The feather spring was unable to actuate the piston all the way to the feather stop, and the piston was blocked from movement towards feather by a gouge in the side of the cylinder.
Beta Mechanism:
All components of the flight idle stop (beta feedback collar, beta rods, pick-ups, springs) were severely damaged; parts were bent, broken, and/or separated.

Hub Assembly:

The hub was fragmented; nine hub fragments were recovered. Several hub bolts were fractured and/or missing. The threaded inserts for the engine mounting bolts were intact but partially exposed due to the hub fractures. The cylinder attachment flange was separated/fractured and was retained inside the cylinder.

Blade Identification:
The blades were identified as 1-2-3-4 at the accident scene, with number 1 as the furthest from the main wreckage and number 4 as the closest. The sequence of the rotational order was unknown.

Figure 32 - Photo of the Propeller Blades

Blades:
Blades number 1 and number 2 were severely damaged. Blades number 3 and number 4 had relatively minor damage.

The number 1 blade was bent edgewise at the blade shank (leading edge bent aft). It was bent aft at 1/4 radius and was bent forward at mid-blade. There was a deep gouge in the leading edge at 33 inches from the retention shoulder. It had numerous deep gouges on the flat side of the blade. The blade retention shoulder was partially fractured. The blade pitch change knob assembly was missing; bolts stripped. The blade counterweight was separated from the blade and the bolts were stripped.

The number 2 blade was fractured at 2/3 radius. The blade was torn chordwise about 14 inches from the blade tip. There were deep chordwise gouges adjacent to both sides of the fracture. The separated portion of the blade had significant chordwise bending (flat side bent concave); the leading edge was gouged with two large chordwise tears; and a portion of the trailing edge was missing. The blade retention shoulder was partially fractured. The cam follower (bearing) from the blade pitch change knob assembly was missing and one attachment screw fractured. The blade counterweight had separated from the blade; bolts stripped.

The number 3 blade was bent forward approximately 20 degrees with a large radius bend. It had rotational scoring in the paint on the camber side. The outer 6 inches of the tip had leading edge gouges and deformation. It was bent slightly aft at the tip. The blade pitch change knob assembly was intact. The blade counterweight was separated from the blade and the bolts were stripped.

The number 4 blade was bent aft approximately 20 degrees at mid-blade and was twisted toward low pitch. It had rotational scoring in the paint at the tip. The cam follower (bearing) from the blade pitch change knob assembly was missing. The blade counterweight had separated from the blade and the bolts were stripped.

Preload Plates:
NOTE: For this propeller model, when the blade knob is aligned with the hub parting line, the blade angle at the reference station is 33.0 degrees. When a fork bumper is aligned with the hub parting line, the blade angle is 28 degrees.

The number 1 and number 2 preload plates were missing except for a small portion that was fractured and retained inside the blade bores.

The number 3 preload plate had two impact marks consistent with contact with a fork bumper about 38 degrees and 25 degrees lower than the hub parting line. The marks equated to -10.0 degrees and + 3.0 degrees blade angle respectively. It also had a light mark consistent with blade knob contact about 79 degrees, which equated to 112 degrees blade angle.

The number 4 preload plate had an impact mark consistent with contact with a fork bumper at approximately 14 degrees lower than the hub parting line. This equated to a 14.0 degrees blade angle

D.10 Maintenance Records Examination
The accident airplane was a Pilatus PC-12/45 (s/n 403), and was manufactured in 2001. The operator purchased the airplane on August 30, 2002. The airplane was domiciled at Redlands Municipal Airport (REI), Redlands, California.

The airframe, propeller, and engine logs were recovered from the wreckage, along with service bulletin (SB) and airworthiness directive (AD) compliance information, and hand-written engine trend information. The records were fire- and water-damaged, with the airframe logbook sustaining the most serious damage. Maintenance information was also provided by the airplane’s primary maintenance provider (Martin Aviation of Santa Ana, California), which dated back to the approximate time when the operator purchased the airplane. The following information was compiled during an examination of all the available records:

Data obtained from Engine Trend Monitoring Page:
An entry made on March 22, 2009, indicated airplane had accumulated 1,916.4 total hours of operation.

Airframe:

The most recent annual inspection was performed by Martin Aviation on October 9, 2008, at 1,815.3 total aircraft hours.

Propeller:
The most recent annual inspection was performed by Martin Aviation on the propeller, on October 9, 2008, and a time since new of 1,815.3 hours. The propeller was overhauled on February 26, 2008, aircraft total time 1,616.2 hours.

Engine:
The most recent annual inspection was performed by Martin Aviation, on October 9 2008, with a time since of new 1,815.3 hours, and 1,359 cycles since new.

Airworthiness Directives:
Last reviewed on October 9, 2008, during the annual inspection.
AD 2007-21-11 complied with on September 24, 2007 through SB 71-007
AD 2008-07-11 complied with on October 9, 2008 (Not Applicable)
AD 2008-06-17 complied with on December 26, 2008 through SB 27-018
AD 2008-02-03 complied with on December 26, 2008 through SB 32-020
AD 2009-05-07 Rear Stick Pusher Cable (not complied with and was not due until 150 hours time-in-service or 30 days after March 30, 2009)

Alterations:
October 3, 2005 STC SA 01376CH Parker Hannifin Brakes Conversion

Field approvals:
August 28, 2001 KGS SS50 Static Inverter
August 28, 2001 BF Goodrich Flight Systems Storm Scope WX-500
August 28, 2001 Bendix / King KMH880 Multi-Hazard TAS & GA EGPWS

DER approvals:
July 27, 2001 Tronair Tow Head Adapter

Registration History:
Eagle Cap Leasing, Enterprise Oregon submitted registration on September 13, 2002
Terry Cunningham submitted registration on September 25, 2001
Aviation Sales submitted registration on September 25, 2001
Pilatus Aircraft submitted registration on July 20, 2001

E. ATTACHMENTS

Attachment 1: AHRS 'A' Data – Text

Attachment 2: BFU Concise Engineering Report – CACU Data Extraction

Attachment 3: CACU Data – Tabular

Attachment 4: Non-Volatile Memory Data Study

Attachment 5: EIS Data - Tabular