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

December 31, 2010

Airworthiness Group Chairman's Factual Report

Addendum 1

WPR09MA159

ACCIDENT

Operator:	Eagle Cap Leasing Inc.
Location:	Butte, Montana
Date:	March 22, 2009
Time:	1430 Mountain Daylight Time (MDT)
Aircraft:	Pilatus PC12/45, N128CM

SUMMARY

On March 22, 2009, at 1430 mountain daylight time, a Pilatus PC-12/45, N128CM, descended to ground impact near the approach end of runway 33 at the Bert Mooney Airport, Butte, Montana. The airplane was owned and operated by Eagle Cap Leasing Inc., 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 and there were no reported ground injuries. The flight departed Oroville, California, at 1210 mountain 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.

ADDITIONS

D.11 Fuel System Placards

According to the PC-12 Approved Airplane Flight Manual (AFM), Section 2, Limitations, a placard (see **Figure 1**) specifying the type of fuel to be used, as well as the total and usable fuel capacity of each fuel tank, was placed near each fuel filler port.

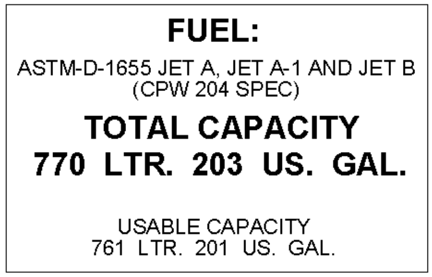

Figure 1 – PC-12 Fuel filler placard

Fuel filler placards from several other single-engine turbo-propeller powered airplanes are presented in Attachment 1: Exemplar Single Engine Turbo-Propeller Powered Airplane Fuel Filler Placards.

D.12 Fuel System Certification and Testing

D.12.1 Crane (Lear-Romec) Testing

The PC-12 fuel system configuration and principle of operation were based on the established design and components utilized by the Pilatus PC-7 and PC-9 military trainer airplanes. Some fuel system test conditions were demonstrated by Lear Siegler (Romec Division) in support of the Pilatus PC-7 fuel system development and qualification. The system was originally outfitted with electric fuel boost pumps designated model number RR53710B. The performance of the system was documented in engineering report number TR-2198 (dated October 31, 1977). Testing was conducted with JP-4 fuel heated to a temperature of 110 degrees F, and to simulate JP-5 fuel at -49 degrees Fahrenheit (F), number 2 fuel oil cooled to a nominal temperature of 10 degrees F was used to simulate the cold fuel's kinematic viscosity (about 20 to 22 centistokes). No low-temperature testing was conducted with any 'Jet' fuels.

The accident airplane was equipped with a Lear-Romec (a division of Crane Co.) electric fuel boost pump model RR53710K. The qualification of the boost pump was detailed in Lear-Romec Qualification Test Report TR-3556 (dated January 2, 1995). According to the test report, the RR53710K was a simplified version of the existing model RR53710B fuel boost pump that incorporated product improvements that reduced weight, increased motor life, reduced manufacturing complexity, and maintained pump interchangeability in the field. The objective of the tests documented by TR-3556 was to verify that the 'K' model fuel boost pump met the requirements of the 'B' model boost pump design. Several specific tests were conducted by Lear-Romec on the pump in support of the qualification including: acceptance test, break-in run, calibration, altitude performance, pressure loss (idle pump), explosion proof, endurance, and dry run. All of the tests were performed with fuel temperatures greater than 32 degrees F.

D.12.2 Pilatus Testing

During the initial development of the PC-12 fuel system, testing was conducted by Pilatus to ensure operation throughout the range of temperatures that the airplane was expected to encounter during its service life. This testing included demonstration of compliance with Title 14 Code of Federal Regulations (CFR) Part 23.951(c), which stated, "Each fuel system for a turbine engine must be capable of sustained operation throughout its flow and pressure range with fuel initially saturated with water at 80 degrees F and having 0.75cc of free water per gallon added and cooled to the most critical condition for icing likely to be encountered in operation." The testing utilized a representative simulation of the aircraft fuel system housed in a cooling chamber.

According to the Pilatus Engineering Report ER-12-28-01-001 (dated July 1993), "At the start of the first 30 minute, 700 [pph] run, fuel temperature was -5 degrees Celsius (C). As the fuel temperature dropped through -8/-10 degrees C, pressure fluctuations were observed at the air separator tank and [engine driven fuel pump] output pressure indicators. On reaching -20 degrees C it was not possible to maintain the then selected 100 [pph] flow at adequate pressure required for normal engine operation. Boost pumps were selected on, but were unable to restore normal pressure and flow. The test was prematurely terminated and the fuel filter element removed for examination." Examination of the fuel filter element revealed that it was "blocked with ice particle buildup." It should be noted that the system utilized for this test did not incorporate a fuel bypass valve, as is present on the aircraft fuel system.

A second test was then performed utilizing the same test system as was used in the previous test, but with 0.08 percent by volume of fuel system icing inhibitor (FSII) added. Fuel flows and pressure remained consistent down to an ambient temperature of -53 degrees C (fuel temperature -48 degrees C). Upon completion of the test, the fuel filter element was partially obscured on the topmost surface of the filter disc stack by a thin film of water/ice slush, which was insufficient to restrict the passage of fuel through the filter.

D.12.3 Pilatus Fuel System Hazard Analysis

Pilatus conducted a fuel system hazard assessment in order to comply with the requirements of Title 14 CFR Part 23.1309. The regulation, as effective beginning November 26, 1990, stated in-part:

…Equipment, systems, and installations.

(a)Each item of equipment, each system, and each installation:

> *(1) When performing its intended function, may not adversely affect the response, operation, or accuracy of any—*
>
> > *(i) Equipment essential to safe operation; or*
> >
> > *(ii) Other equipment unless there is a means to inform the pilot of the effect.*
>
> *(2) In a single-engine airplane, must be designed to minimize hazards to the airplane in the event of a probable malfunction or failure...*

and

…(b) The design of each item of equipment, each system, and each installation must be examined separately and in relationship to other airplane systems and installations

to determine if the airplane is dependent upon its function for continued safe flight and landing and, for airplanes not limited to VFR conditions, if failure of a system would significantly reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions. Each item of equipment, each system, and each installation identified by this examination as one upon which the airplane is dependent for proper functioning to ensure continued safe flight and landing, or whose failure would significantly reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions, must be designed to comply with the following additional requirements:

> *(1) It must perform its intended function under any foreseeable operating condition.*
>
> *(2) When systems and associated components are considered separately and in relation to other systems—*
>
> > *(i) The occurrence of any failure condition that would prevent the continued safe flight and landing of the airplane must be extremely improbable; and*
> >
> > *(ii) The occurrence of any other failure condition that would significantly reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions must be improbable.*
>
> *(3) Warning information must be provided to alert the crew to unsafe system operating conditions and to enable them to take appropriate corrective action. Systems, controls, and associated monitoring and warning means must be designed to minimize crew errors that could create additional hazards.*
>
> *(4) Compliance with the requirements of paragraph (b)(2) of this section may be shown by analysis and, where necessary, by appropriate ground, flight, or simulator test. The analysis must consider—*
>
> > *(i) Possible modes of failure, including malfunctions and damage from external sources;*
> >
> > *(ii) The probability of multiple failures, and the probability or undetected faults;*
> >
> > *(iii) The resulting effects of the airplane and occupants, considering the stage of flight and operating conditions; and*
> >
> > *(iv) The crew warning cues, corrective action required, and the crew's capability of determining faults...*

The assessment (Pilatus Engineering Report ER12-28-00-001, dated September 30, 1993) categorized failure conditions into four categories, which included minor, major, severe major, and catastrophic. The failures were further analyzed to determine their probability, that showed that any failures deemed major or severe major were improbable (defined as having a probability less than 10^{-5} or 10^{-7} respectively). Catastrophic failures were defined as failures which would make continued safe flight and landing of the airplane impossible, and must have shown a probability less than 10^{-9}. Operational experience and reliability data from other Pilatus aircraft utilizing the same fuel system as the PC-12 were used to support the calculations made in the assessment. An installation appraisal also considered the effect that faulty operation, incorrect servicing, or incorrect operating procedures would have on the operation of the fuel system.

Several system fault outputs and aircraft failure conditions were considered including a significant fuel imbalance between both wing fuel tanks. According to the assessment, "The difference in fuel weight will produce a rolling moment on the aircraft. This moment may be counteracted by changing the trim setting and the failure condition may be removed by differential operation of the booster pumps…" Additionally, "In the event of a major fuel imbalance which cannot be corrected by operating the booster pumps, the rolling moment may become too large to be counteracted by trimming and it may be necessary to amend the planned mission." The failure condition was subsequently considered major.

An excessive difference in the wing fuel tank levels was considered by the assessment to be the result of a combination of faults that must occur in sequence. Those faults were a significant differential in fuel flow from the left and right wing tank, and the inability to select the electric fuel boost pump in the heavier wing to restore balance. The sequence of events depicted in **Figure 2** below determined the failure state to be improbable.

An addendum to the report further considered the failure of the automatic fuel balancing system, and its effects on the overall fuel system. According to the addendum, it was considered a major failure if a fuel imbalance between both wing fuel tanks exceeded 25-percent of the full fuel tank load. At that point, "…the resulting rolling moment cannot be corrected by trimming alone and the control column must be used. This increases pilot workload and decreases the aircraft safety margin in the event of a manoeuvre requiring higher than usual levels of piloting skill (i.e. approach and landing in turbulent weather conditions) being called for."

Pilatus' analysis further describes that the failure condition would arise in the event of a loss of fuel flow from one of the wing tanks followed by a failure of the automatic fuel balancing system to recognize or correct the resulting difference in wing tank fuel levels. Following this, the failure may be prevented if the fuel imbalance is detected and corrected by the flight crew manually activating the required fuel boost pump. This would not occur if the flight crew were inattentive, deceived by an incorrect fuel quantity indication, or if there was a malfunction of the required fuel boost pump. The failure condition could also be created by a malfunction of the automatic fuel balancing system resulting in the continuous operation of one fuel boost pump. Once the situation was observed the flight crew, the failure condition may simply be prevented by manually deselecting the automatic fuel balancing system or deactivating the circuit breaker of the 'runaway' fuel boost pump. The analysis discussed above is depicted in **Figure 2** below.

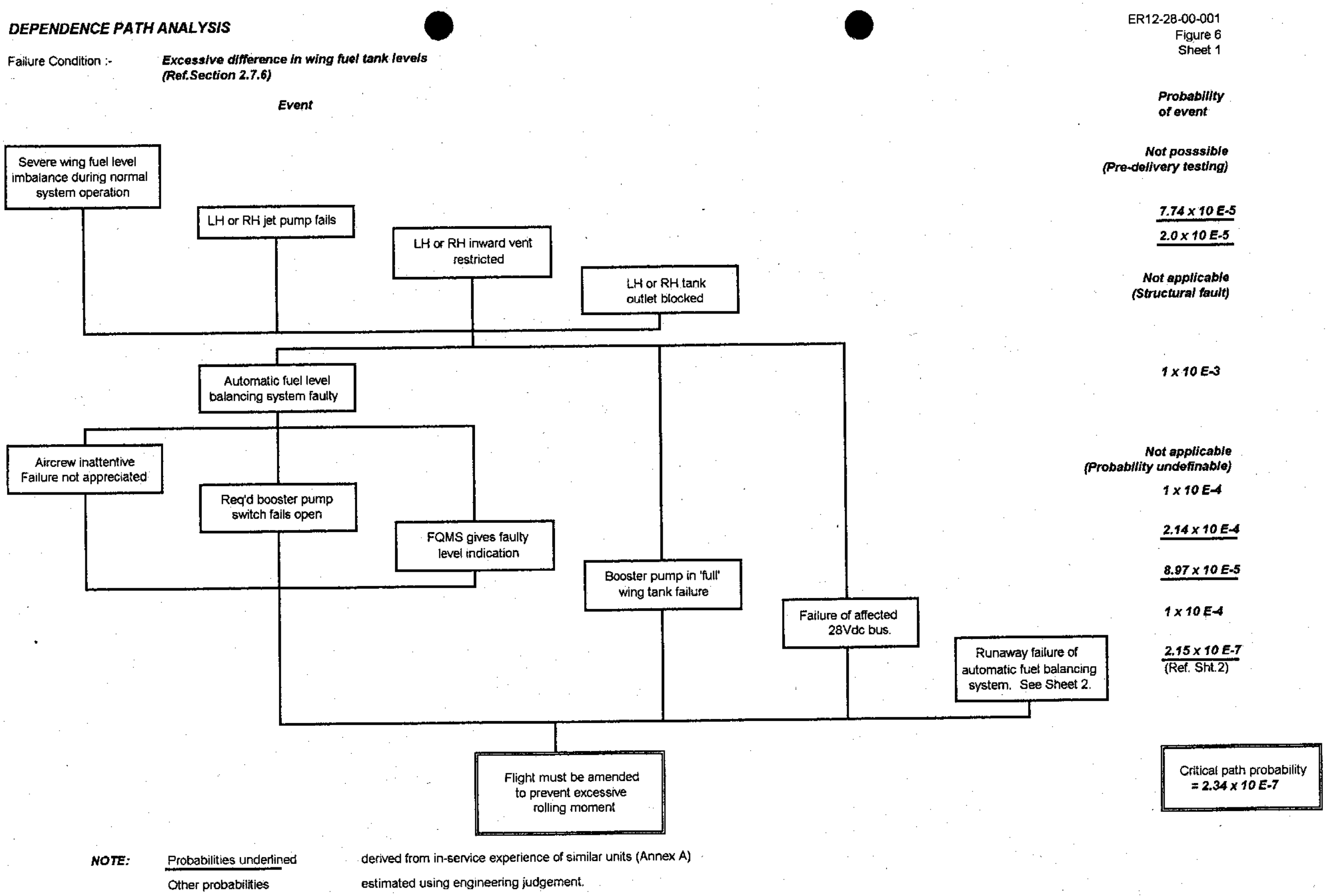

Figure 2 – Dependence Path Analysis, Excessive difference in wing fuel tank levels

Several faulty inputs to the fuel system were considered, and while a failure of the right or left fuel boost pump was not explicitly covered, a failure of the left or right fuel boost pump selector switch in a deactivated or "off" condition was. This was considered a major failure condition, though its occurrence must be in combination with other system failures.

A failure of either fuel boost pump selector switch in the off position would result in the affected boost pump not being able to operate if required (either manually or automatically). While normal operation of the fuel supply system would not be affected by this failure, a failure that coincided with another system malfunction would be considered major. The probability of failure of the electrical connector switch was considered to be 1×10^{-4}, a standard value used throughout the hazard assessment for the reliability of standard electrical components. The probability of a separate system malfunction, which would under normal circumstances require the operation of the booster pump was considered to be 1.48 x 10-5. This value was derived from the malfunctions considered in **Figure 3**.

The above probabilities resulted in an overall probability of the failure occurring of 1.48×10^{-9}, and the failure condition was considered improbable.

Maintenance/servicing errors were also addressed by the assessment. One such error was, "Failure to drain wing tanks at regular intervals. This may result in the accumulation of large quantities of water within the fuel tanks, which may cause icing problems in low ambient temperatures." Another error listed was, "Failure to ensure that fuel includes anti-icing additives. This may cause icing problems in low ambient temperatures." Each of these maintenance errors could result in a restricted or blocked fuel filter element, a failure which is considered in **Figure 4**.

The hazard assessment listed the reliability of certain fuel system components based on service experience from other aircraft that share the same fuel system with the PC-12. The fuel filter had a mean time between failure of 78,000 hours. A failure of the fuel filter could consist of either the fuel flow being restricted or completely blocked. It was assumed that 1-percent of all failures would result in the bypass valve, as well as the fuel filter being blocked. The probability of a restricted fuel filter was calculated to be 1×10^{-3}, while the probability of a blocked filter was calculated to be 1.28×10^{-5}. The probability of a blocked fuel filter bypass was 1.28×10^{-7}(note that this figure is only representative of the bypass being blocked, and not the filter and bypass).

The fuel boost pumps were listed as having a mean time between failure of 11,143 hours, and all failures were assumed to result in an inability of the pump to pass fuel flow. The probability of failure of the fuel boost pump was subsequently calculated to be 8.97×10^{-5}.

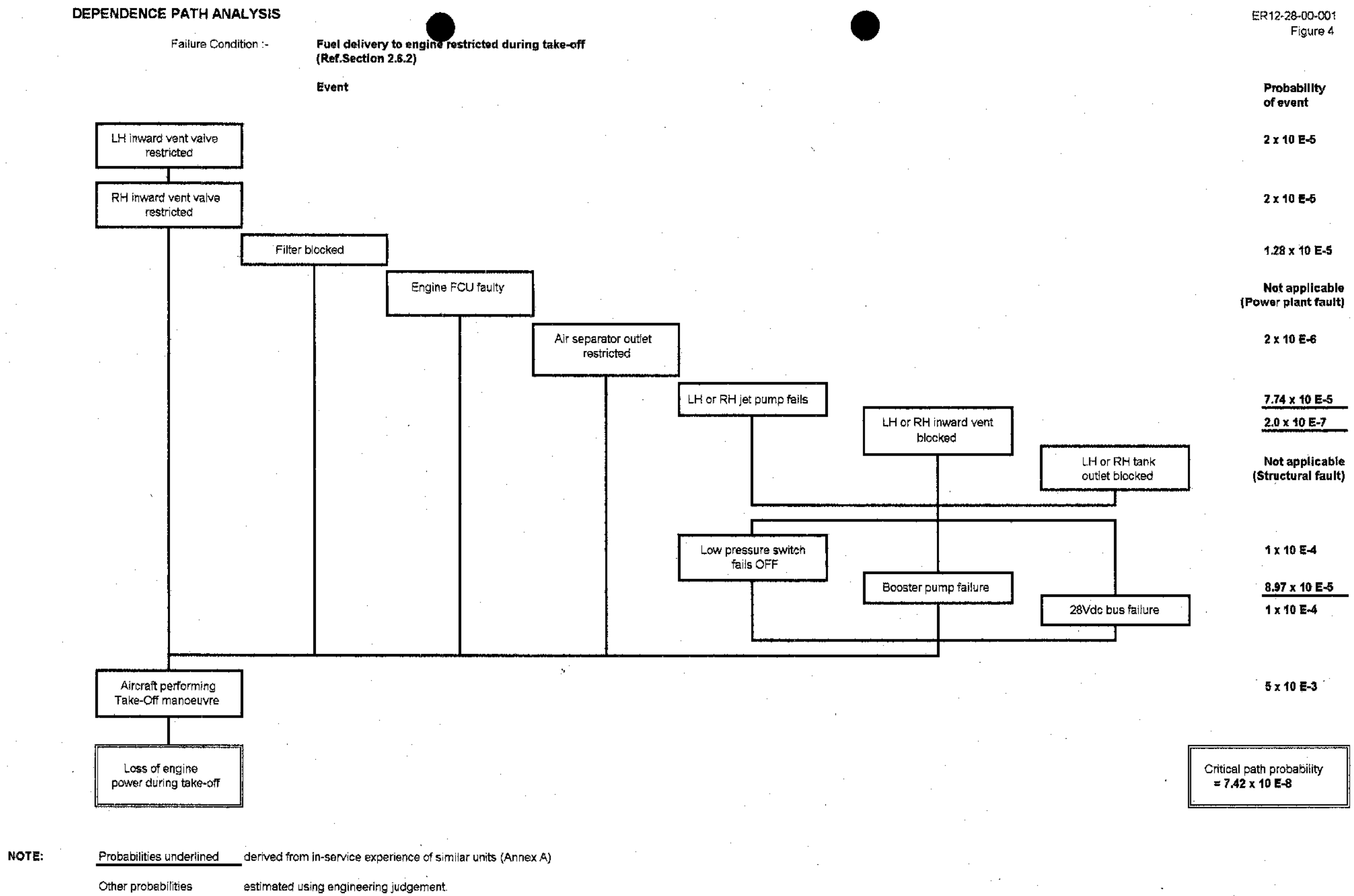

Figure 3 – Dependence Path Analysis, Fuel delivery to engine restricted during take-off

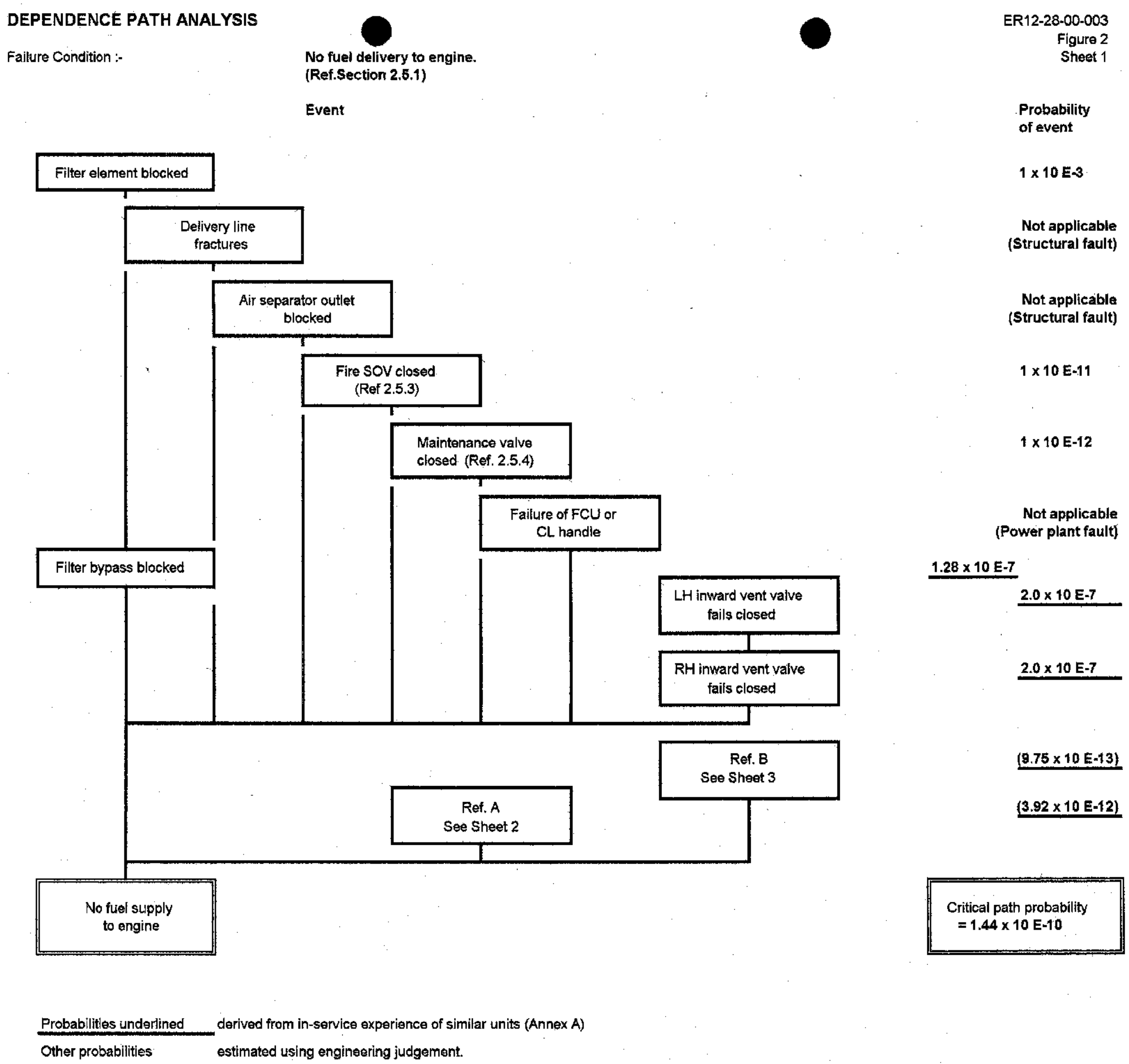

Figure 4 – Dependence Path Analysis, No fuel delivery to engine (1 of 3)

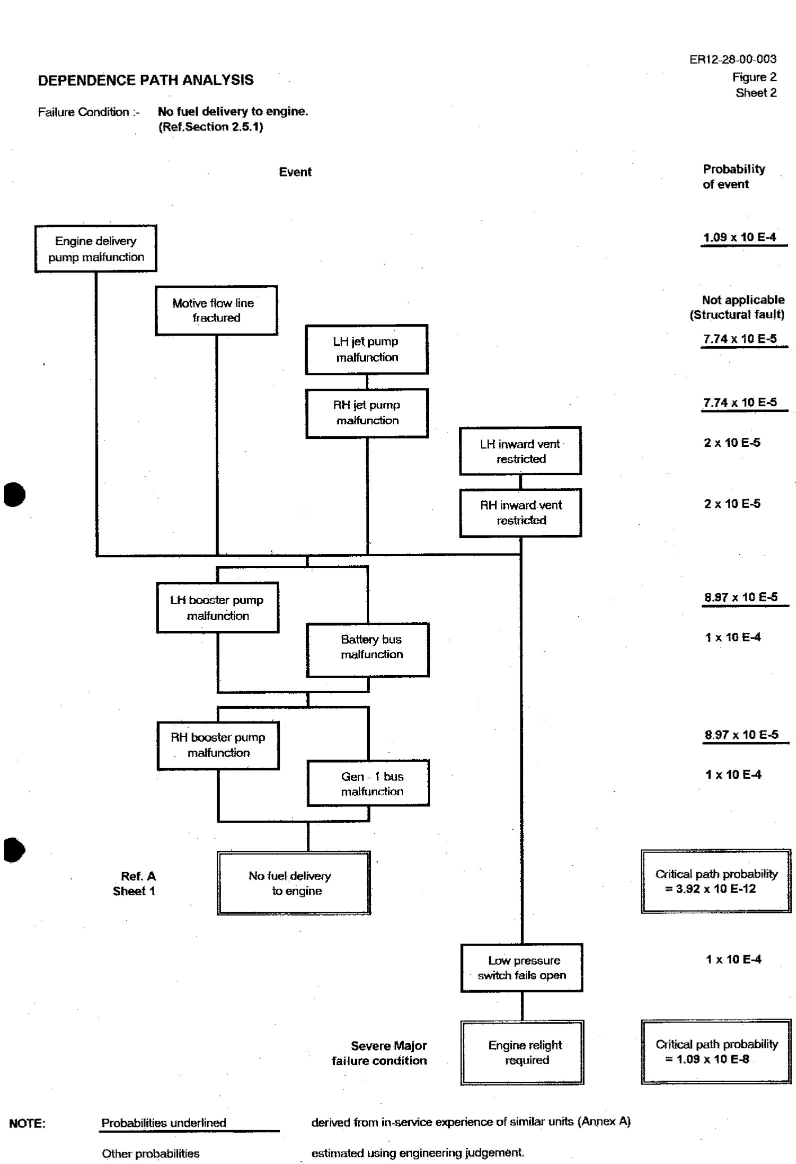

Figure 4 – Dependence Path Analysis, No fuel delivery to engine (2 of 3)

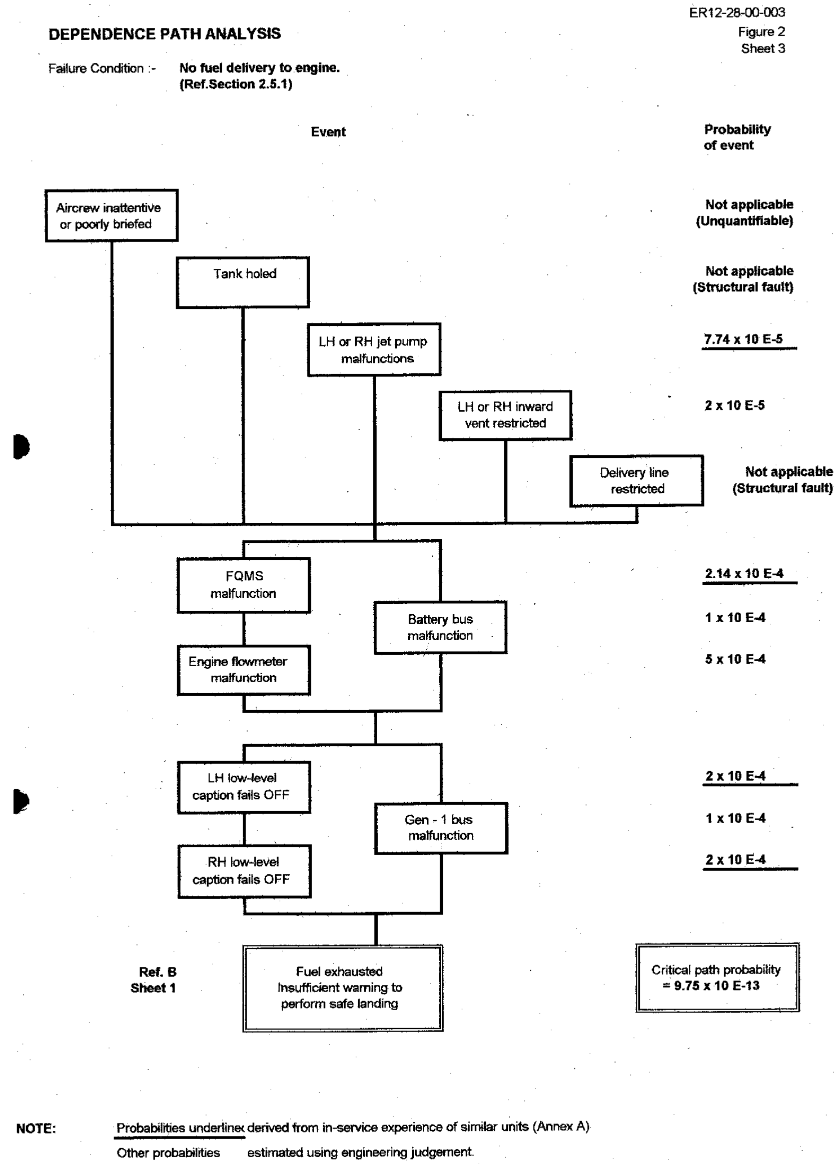

Figure 4 – Dependence Path Analysis, No fuel delivery to engine (3 of 3)

D.12.4 Pilatus Fuel System Specification

As stated in D.11.1, the PC-12 fuel system is based on the PC-7 and PC-9 fuel system. The specification for the PC-9 fuel system (Pilatus Specification PEI-9-SYS-070, dated August 19, 1983) outlined the requirements for performance, design, certification, and maintenance of the system. The specification stated that the minimum fuel inlet temperature to the engine oil-to-fuel heater was:

a) *that equivalent to 12 centistokes or*

b) *JP4, Jet B, AVGAS: -54 degrees C*
Jet A, Jet A-1: -34 degrees C
JP5: -26 degrees C

Additionally, the minimum fuel temperature to the engine inlet was "-54 degrees c or that equivalent to 12 centistokes."

Pilatus Specification ESM-12-SPEC-171 (dated November 3, 1992), defines the design and performance requirements for fuel boost pumps utilized in the PC-12. The specification outlined the operating environment temperature to include an ambient air temperature between -65 degrees C and +55 degrees C.

D.12.5 Fuel Boost Pump Service Bulletin

On August 14, 2000, a Pilatus PC-12 was on a delivery flight from the factory in Stans, Switzerland to the United States. While flying at 30,000 feet, with an outside air temperature of -40 degrees C, one of the fuel boost pump circuit breakers tripped when the fuel boost pump was activated. The airplane subsequently returned to the factory, where upon re-setting the circuit breaker, the fuel boost pump operated normally while at an outside air temperature of 5 degrees C. Two additional events prompted Pilatus to remove the subject fuel boost pumps from their respective airframes and return them to Lear-Romec for further investigation.

Lear-Romec subsequently initiated an investigation into the issues reported by Pilatus, and prepared a report of their findings. According to Lear-Romec report TR-4277 (Summary of Tests Conducted on Model RR53710K to Determine Cause of High Current at Cold Temperature, dated April 2, 2001), the manufacturing specifications provided to Lear-Romec by Pilatus for the fuel boost pump had "no clear requirements for maximum current draw and cold temperature testing requirements…" About the time that the initial issues with the fuel boost pumps began, Pilatus independently began to conduct tests to observe the current draw of operating pumps immersed in Jet-A fuel chilled to temperatures between -30 and -60 degrees C. Pumps with relatively high current draw were forwarded to Lear-Romec for further evaluation.

Two of the forwarded fuel boost pumps were initially subjected to acceptance test procedures at nominal room temperatures, and passed with no anomalies noted, despite the fact that the armature end play of both fuel boost pumps was measured to be 0.000-inches, which was below the requirement for between 0.005 and 0.015-inches. The pumps were then subjected to progressively lower temperatures, while their driveshafts were rotated. The driveshafts became increasingly difficult to turn as temperature decreased. The pumps were then disassembled, and their armature end-play were each reset to values of 0.004 and 0.005-inches before being subjected to operational cold testing, utilizing Jet-A fuel with 0.06 to 0.15 percent by volume of FSII (ethylene glycol) added. The performance of the pumps was measured at progressively lower fuel temperatures ranging from 66 degrees F to -65 degrees F. Observed current draw and pump output was observed within nominal limits.

The results of additional testing showed that cold temperature operation decreased the armature shaft end-play of the fuel boost pump. Testing of four other fuel boost pumps revealed that shaft end-play could be reduced by as much as 0.0045 inches. The testing also found that there could be as much as a 3.1 amp increase in electrical current as temperature dropped from 70 degrees to -57 degrees F. The report attributed the increase in current draw primarily to the increase in density of the cold Jet-A fuel.

Several corrective actions were implemented as a result of the testing including a Service Bulletin (Lear-Romec service bulletin number RR53710K-20-001, Pilatus service bulletin number 28-008 [dated June 19, 2001]). The service bulletin required the inspection of the shaft armature end-play, and replacement of units with end-play less than 0.005 inches. Pumps with end play measured greater than this minimum were to have their serial numbers marked with an 'A' suffix. Only a certain range of non-sequential pump serial numbers were affected, the highest serial number being B5116.

According to acceptance records, the accident airplane was equipped with RR53710K boost pumps at the time of manufacture, serial numbers B5159A and B5161A. Review of maintenance log and service invoice records revealed no evidence that either fuel boost pump had been replaced during the accident airplane's lifetime.

D.13 Fuel Boost Pump Post-Accident Testing

D.13.1 Fuel Boost Pump Cold Testing

Fuel boost pump cold testing was conducted at Crane Aerospace and Electronics' Elyria, Ohio facility on October 18 through 28, 2010 under the supervision of the Airworthiness Group Chairman. The purposes of the tests were to characterize fuel boost performance under various cold (below 0 degrees C) operating conditions. Tests were performed with jet fuel, jet fuel with water added, and jet fuel with water and FSII (Diethylene glycol monomethyl ether-type) added, to compare the performance of each fuel condition. Two test units (model RR53710K fuel boost pump), serial numbers B-4771 and B-5120, were utilized for the tests.

The test units were provided by Pilatus, and were deemed "unserviceable." The units were subsequently refurbished by Crane before acceptance tests were performed. Both units passed the acceptance test procedure prior to the commencement of testing, including measurement of armature end-play.

The test stand (**Figure 5**) consisted of a 50-gallon tank filled with Jet-A fuel, with provisions for the installation of a RR53710K fuel boost pump. The discharge port of the fuel boost pump was piped through a check valve supplied by Pilatus (identical to the check valve used in the aircraft fuel system), to a fuel flow control valve, which was installed to facilitate performing the initial calibration tests required. Pressure sensors were installed in-line with the discharge path of the fuel boost pump, both before and after the check valve, and a flow meter was installed prior to the fuel flow control valve. Fuel temperature was controlled through the use of a circulation pump that routed fuel through a heat exchanger that was chilled with methanol.

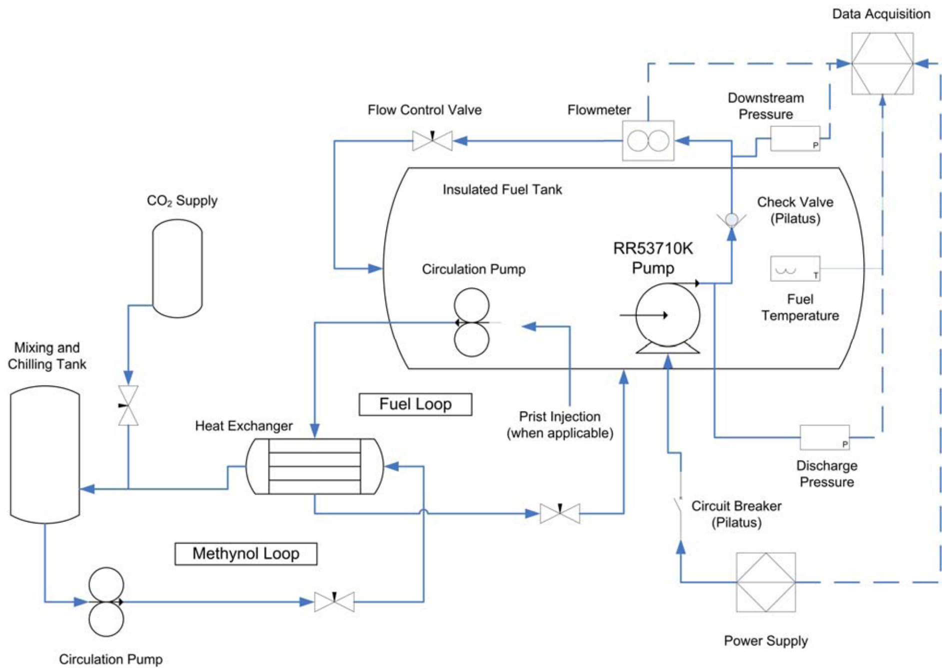

Figure 5 – Test stand

For each fuel condition, both fuel boost pumps were run through a calibration curve test and an endurance test. The fuel calibration curve was the same type of test that new fuel boost pumps were subjected to during acceptance testing. The curve was derived by measuring the fuel boost pump discharge pressure at various flow rates (0, 400, 600, 800, 1,100, 1,200, 1,400, 1,600, 2,000, 2,400 pounds per hour [pph], and maximum flow). Curves were generated for each fuel condition at fuel temperatures of 32, 5, and -22 degrees F, with the exception of the baseline fuel test, which also conducted a calibration curve at 70 degrees F. For the endurance test, the fuel temperature was stabilized at -22 degrees F, fuel flow rate was set to 900 pph, and the pump was cycled in a manner similar to that observed during the accident flight, active for 10 seconds and inactive for 1 second.

The detailed results of each test can be found in Attachment 2: Cold Test with Excess Water Report (Crane Document No. TR-4959), and are summarized in the sections below.

D.13.2.1 Baseline Cold Test

Fuel flow calibration curves were developed for each fuel pump at four fuel temperatures, and operating with Jet-A fuel (no water or FSII added). Each data point was set by adjusting the fuel flow control valve to the predetermined flow rates, ranging from 0 pph through full fuel flow (about 3,000 pph). Both fuel pumps showed nominal performance at each temperature condition, with a loss of about 4 pounds per square inch (psi) over the entire temperature spectrum.

During the endurance testing, the fuel flow rate progressively decreased over time. The decision was made to remediate the decrease in flow by adjusting the fuel flow valve to restore fuel flow with the first test unit, and tapping the fuel flow valve with a hammer during the test of the second unit. Each of these actions restored flow to nearly the 900 pph value set at the test start. It was surmised that the reduced flow was due to a buildup of ice on the fuel flow valve, and not a degradation of the pumps' performance, as action to the valve and not the pump affected the observed output.

D.13.2.2 Cold Test with Water Added

Water was added to the fuel through an atomizing nozzle. To ensure a level of water in excess of the Jet-A's saturation point, 270 cubic centimeters (cc) of water was added to the existing 50 gallons of fuel. The fuel flow calibration curves were developed at all three cold temperatures (32, 5, and -22 degrees F), and exhibited similar performance to the baseline tests. The endurance test showed similar results as the baseline test, with an accelerated degradation of fuel flow and a greater remedial interval required. Pump B-4771 also exhibited audible pitch changes at points during the test.

D.13.2.3 Cold Test with FSII Added

FSII was added to the fuel through a nozzle directed to the inlet of the circulating (cooling) pump while it was operating. A total of 200 cc of FSII was added to the fuel/water mixture. Calibration curves at all three cold temperatures exhibited similar performance as the previous two fuel states. Throughout the endurance test, flow remained relatively constant, and no valve adjustments or tapping were required to maintain the fuel flow rate.

D.13.2.4 Cold Test Anomaly

During the initial test run of pump B-5120, the pump performed inconsistently and exhibited a decrease in discharge pressure along with an audible change in pitch after completing the calibration curves and after one hour of continuous operation at -22 degrees F, at the beginning of the endurance test. The endurance test was discontinued after about 13 cycles. Visual examination of the pump during operation showed that the fuel venting from the top vent hole of the pump appeared dirty or contaminated, likely with brush material, exiting from the top of the unit. A subsequent repeat of the -22 degree F calibration curve showed a lower flow/pressure spread than the original curve at the same temperature. The fuel temperature was then increased to 32 degrees F, and another calibration was performed, showing normal pump performance.

As a precautionary measure, the pump was removed from the test rig, disassembled, and visually inspected. There were no visual indications in the hardware of misassembly or abnormal mechanical operation. The motor brushes were fully engaged with the commutator and moved freely. Upon further inspection, one motor brush exhibited pitting and brush material had collected on the commutator surface. The brushes were replaced with new parts and the armature/commutator replaced with a refurbished unit. Subsequent testing of the pump produced the normal results described above.

D.14 Fuel Consumption Calculations

Fuel consumption calculations for flights 1557, 1558, and 1559 are presented in Addendum 1, Attachment 3: Flights 1557, 1558, and 1559 Fuel Consumption Calculations. Figures for climb and descent were computed utilizing the charts presented in the PC-12 airplane flight manual (AFM).

The majority of flight 1557 was captured by FAA ATC radar. According to the radar data, the airplane was first observed on radar climbing through 8,700 feet at 14:47:09 Universal Coordinated Time (UTC), and the airplane reached its first cruise altitude of 26,000 feet at 15:09:05 UTC. The total radar-observed climb portion of the flight was 0:21:57. The airplane subsequently began a descent to its second cruising altitude of 22,000 feet at 15:15:43 UTC. The airplane continued at that altitude until 16:04:57 UTC, when it began to descend. The total calculated flight time for the cruise portion of the flight was 00:06:38 for the cruise portion at 26,000 feet, and 00:49:14 for the cruise portion at 22,000 feet (including the descent from 26,000 to 22,000 feet). The airplane was observed on radar in a descent until 16:26:39 UTC. The total radar observed flight time for the descent portion of the flight was 0:21:42. Given the disparity between the observed and calculated climb and descent values, only the calculated values presented by the AFM are given in **Table 1**. The radar-observed cruise flight time is used to calculate the cruise fuel consumption presented.

No definite flight time for flight 1558 could be calculated, as no radar data was available for analysis. Instead cruise flight time for flight 1558 was based on the total flight time recorded by the CAWS (takeoff to last flight entry), less the climb and descent time calculated with the AFM performance charts. The total flight time recorded by the CAWS for flight 1558 was 0:19:13, and the total calculated cruise flight time was 0:11:00. The total flight times and fuel consumed for flight 1558 are presented in **Table 2**.

The majority of flight 1559 was captured by FAA ATC radar. According to the radar data, the airplane was first observed on radar climbing through 3,900 feet at 18:11:39 UTC, and the airplane reached its cruise altitude of 25,000 feet at 18:33:36 Universal Coordinated Time (UTC). The total radar-observed climb portion of the flight was 0:21:57. The airplane continued at that altitude until 20:04:00 UTC, when it began to descend. The total calculated flight time for the cruise portion of the flight was 1:30:24. The airplane was observed on radar in a descent until 20:30:25 UTC. The total radar-observed flight time for the descent portion of the flight was 0:26:25. Given the disparity between the observed and calculated climb and descent values, only the calculated values presented by the AFM are given in **Table 3**. The radar-observed cruise flight time is used to calculate the cruise fuel consumption presented.

Table 1 - Flight 1557 Fuel Consumption by Flight Phase

Phase	Duration	Fuel Burned
	hh:mm	*lb*
Takeoff	0:00	*38
Climb	**0:25	190
Cruise 1	***0:07	41
Cruise 2	***0:49	357
Descent	**0:11	68
Total	1:32	694

Table 2 - Flight 1558 Fuel Consumption by Flight Phase

Phase	Duration	Fuel Burned
	hh:mm	*lb*
Takeoff	0:00	*38
Climb	**0:05	45
Cruise	0:11	103
Descent	**0:03	22
Total	0:19	208

Table 3 - Flight 1559 Fuel Consumption by Flight Phase

Phase	Duration	Fuel Burned
	hh:mm	*lb*
Takeoff	0:00	*38
Climb	**0:26	220
Cruise	***1:31	603
Descent	**0:09	49
Total	2:09	910

*fuel consumption prior to takeoff estimated by Pilatus
**time based on AFM computed value for particular phase of flight
***time based on radar observed value

ATTACHMENTS

Addendum 1, Attachment 1: Exemplar Single Engine Turbo-Propeller Powered Airplane Fuel Filler Placards

Addendum 1, Attachment 2: Cold Test with Excess Water Report (Crane Document No. TR-4959)

Addendum 1, Attachment 3: Flights 1557, 1558, and 1559 Fuel Consumption Calculations

Addendum 1, Attachment 4 – Pilatus Service Bulletin 28-008 and Crane Service Bulletin RR53710K-20-001