NATIONAL TRANSPORTATION SAFETY BOARD
Office of Aviation Safety
Washington, D.C. 20594

September 3, 2021

Airworthiness Group Chairman's Factual Report

CEN20FA022

A. ACCIDENT

Location:	Near the Chamberlain Municipal Airport, (9V9), Chamberlain, South Dakota
Date:	November 30, 2019
Time:	1235 central daylight time
Airplane:	Pilatus PC-12/47E airplane, N56KJ

B. SYSTEMS GROUP

Chairman: Scott Warren
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C. SUMMARY

On November 30, 2019, about 1230 central standard time, a Pilatus PC-12/47E airplane, N56KJ, was destroyed during an impact with terrain near the Chamberlain Municipal Airport, (9V9), Chamberlain, South Dakota. The pilot and 8 passengers were fatally injured. Three passengers sustained serious injuries. The airplane was registered to Conrad & Bischoff, Inc. and operated by the pilot as a Title 14 *Code of Federal Regulations* Part 91 personal flight. Instrument meteorological conditions prevailed, and the flight was operated on an instrument flight rules flight plan. The flight originated from 9V9 shortly before the accident and was destined for Idaho Regional Airport (IDA), Idaho Falls, Idaho.

The Airworthiness Group convened on December 2-3, 2019, in Chamberlain, South Dakota to examine and document selected portions of wreckage.

D. DETAILS OF THE INVESTIGATION

1.0 Cockpit Area

The cockpit area wreckage was extensively damaged during the impact. The control wheels, throttle quadrant, rudder pedals, and instrument panel all sustained significant impact damage. The left hand display was still attached to the instrument panel. The center displays were broken from their mounting, and the upper display was cracked. The right hand display was loose from its mounting and was cracked.

Figure 1
Cockpit Area

The cockpit pedestal was displaced from its normal position. The manual override (MOR) lever and power lever were both at a position of approximately 75% of maximum. The flap lever was at the 15 deg position.

Figure 2
Cockpit Pedestal – side view

Figure 3
Cockpit Pedestal – top view

The landing gear selector level was found in the up position.

The circuit breaker panels were examined, and several breakers were in the extended (tripped) position. It could not be determined if the tripping of these circuit breakers was due to a fault prior to the accident or due to the impact and subsequent breakup of the aircraft.

The overhead control panel was located, and the following switch positions were determined:

The left and right fuel pump switches were in the AUTO position;
The engine start ignition switch was in the ON position;
The AV1 BUS switch was in the ON position;
The STBY BUS switch was in the ON position;
The GEN 1 switch was in the ON position;

The BUS TIE circuit breaker was out;
The BAT 1 switch was ON;
The EXT PWR switch was OFF;
The WING, NAV, BEACON, STROBE, TAXI, and LANDING switches were all in the OFF position;
The PULSE RECOG switch was in the center position;
The CABIN BUS switch was in the ON position;
The AV2 BUS switch was in the ON position;
The GEN 2 switch was in the OFF position;
The BAT2 switch was in the ON position;
The MASTER POWER switch was in the ON position;
The EPS switch was in the ARMED position;
The NO SMOKING switch was in the ON position;
The SEAT BELT switch was in the OFF position.

Many of the switch and lever positions in the pedestal and overhead panel were determined to be susceptible to being altered during rescue efforts.

The cockpit panel containing the ICE PROTECTION switches was not located.

1.1 Avionics Units

The lightweight data recorder (LDR) was recovered on December 2 and sent to NTSB headquarters for further review. The unit was manufactured by L3 Aviation Recorders in Sarasota, FL. The part number was 1000-1000-00, and the serial number was 000891678.

Figure 4
LDR mounted in the aft fuselage

Figure 5
LDR data plate

The emergency locator transmitter (ELT) unit was recovered. The switch on the unit was in the ARMED position.

The modular avionics unit (MAU) was recovered, and it was removed from the general wreckage for possible further evaluation. The unit was manufactured by Honeywell Intl, Inc. The part number was 7029782-1912, and the serial number was 13066773.

The KMH-920 multi-hazard awareness unit (with combined EGPWS and TAS traffic awareness) was recovered, and it was removed from the general wreckage for possible further evaluation. The unit was manufactured by Honeywell. The part number was 066-01178-2102, and the serial number was KMH920-B1616.

1.2 Flight Control System

The flight control system is conventional using push-pull rods and carbon steel cables. Electric trim systems are provided for the aileron, rudder, and elevator. All trim systems can be disconnected in the event of a runaway condition.

1.2.1 Longitudinal control system

The elevator is a two-piece unit attached to the horizontal stabilizer at a total of five hinge points and is connected to the cockpit control wheel by carbon steel control cables. A down spring is installed in the control circuit to improve longitudinal stability.

The elevator is equipped with static wicks to dissipate static charges to the atmosphere.

Pitch trim is provided by positioning the horizontal stabilizer

Control continuity could not be established from the cockpit to the first fuselage break in the area of the wing spars due to significant wreckage deformation. Control continuity from the first fuselage break through to the elevators was established.

The complete horizontal stabilizer was found, and it was still attached to the vertical stabilizer by one attachment point. The other attachment point was broken.

The stabilizer trim actuator was recovered, and the piston extension was measured to be 4 5/8 inches. According to Pilatus, this measurement was consistent with a trim position of 0.67 deg horizontal stabilizer leading edge up. This position was within the green takeoff stabilizer trim position range for the aircraft.

The right elevator surface was bent in the TEU direction in a location consistent with impact damage.

The elevator control cables were found broken with broomstrawed ends within the vertical tail.

1.2.2 Directional Control System

The rudder is a single piece unit attached to the vertical stabilizer at two hinge points and is connected to the cockpit rudder pedals by carbon steel control cables. Both pilot and copilot rudder pedals are adjustable by use of a crank located between each set of rudder pedals. Clockwise rotation of the crank moves the pedals aft.

The rudder is equipped with static wicks to dissipate static charges to the atmosphere.

Control continuity could not be established from the cockpit to the first fuselage break in the area of the wing spars due to significant wreckage deformation. Control continuity from the first fuselage break through to the rudder was established.

The complete rudder was found, and it was still attached to the vertical tail. The rudder trim tab appeared to be faired with the rudder, and the rudder was deflected TEL which was consistent with its position resting against the fuselage.

The rudder incorporates a trim tab that is electrically operated from the cockpit. The rudder trim actuator was recovered, and the piston extension was measured to be 2 7/16 inches. According to Pilatus, this measurement was consistent with a trim tab position of 6.54 deg tab trailing edge left (rudder surface trailing edge right). This position was within the green takeoff rudder trim position range for the aircraft. The actuator was manufactured by AMETEK Advanced Industries, the part number was ACL 12A-1, and the serial number was 0817.

1.2.3 Lateral Control System

The ailerons are connected to the cockpit control wheels by control cables in the fuselage and push-pull rods in the wings. Each aileron is attached to the wing at two hinge points.

Each aileron has a moveable tab which is connected to a geared lever (Flettner) mechanism. The mechanism is installed inside the aileron and makes the tabs act as balance tabs when the ailerons are moved. They move in the opposite direction to the ailerons. The left tab is also operated electrically from the cockpit and acts as a trim tab.

Control continuity could not be established from the cockpit to the first fuselage break in the area of the wing spars due to significant wreckage deformation. Control continuity from the first fuselage break through to the ailerons could not be established due to deformation within the wings.

The left and right ailerons were found attached to their respective wings.

The aileron trim tab actuator was recovered, and the piston extension was measured to be 1 7/8 inches. According to Pilatus, this measurement was consistent with a trim tab position of 1.15 deg tab trailing edge up (left aileron trailing edge down). This position was within the green takeoff aileron trim position range for the aircraft. The actuator was manufactured by AMETEK Advanced Industries, the part number was ACL 12A-1, and the serial number was 0811.

1.3 Flap System

Each wing trailing edge has a single piece Fowler type flap supported by three flap arms. The flaps are controlled by a selector handle located to the right of the power controls on the center console. The flaps may 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 is not at one of the four preset positions, the Flap Control and Warning Unit (FCWU) will drive the flaps to the nearest preset position.

The flaps are electrically actuated. There is a single flap Power Drive Unit (PDU) installed below the cabin floor at the rear main frame. It drives screw actuators at the inboard and middle stations through flexible shafts. The screw actuators are connected to the flap actuating arms.

The left flap was found attached to the left wing, but there was significant bending in some areas of the flap.

The left inboard flap actuator was found to be broken with 5 inches of jackscrew remaining attached to the flap track. The left inboard flap actuator was found attached to wing spar structure that remained attached to the fuselage. The jackscrew section within the actuator was broken on both sides of the actuator.

The left outboard flap actuator was intact with 5 ¼ inches of jackscrew measured between the flap actuator and the center of the attachment bolt.

The left inboard and outboard flap tracks were broken and separated from the wing. The left middle flap track (containing the outboard jack screw) was intact and attached to the wing and flap.

The right inboard flap actuator was intact with 5 1/2 inches of jackscrew measured between the flap actuator and the center of the attachment bolt.

The right outboard flap actuator was intact with 5 ¼ inches of jackscrew measured between the flap actuator and the center of the attachment bolt.

According to Pilatus, these measurements are consistent with a flap position of 15 deg. The actuator measurements indicated that all of the flaps were in the same position with no indications of twist or asymmetry.

The right inboard, middle, and outboard flap tracks were damaged and separated from the wing in several locations.

1.4 Stall warning system

According to Pilatus, the stall warning system is described as:

The airplane is equipped with a stick shaker-pusher system to improve aircraft handling in the low speed flight regime by preventing the airplane from inadvertently entering a stall condition. The stick shaker-pusher system contains two Angle-of-Attack (AOA) sensors, two computers, a single stick shaker and a single stick pusher. The two computers are connected in such a way that either computer can, independently, provide stall warning (stick shaker and stall warning) but both computers are required to actuate the stick pusher.

The left and right hand Stick Pusher Computers are each provided power from the Essential and Main bus. Each computer receives inputs from its respective AOA vane and AIR/GND relay. Both computers receive inputs from the engine torque, flap position, and self test. From these various inputs, each computer independently determines the "Defined Angle of Attack" for stall warning (stall warning and stick shaker activation), stick pusher activation, and stick pusher disengagement following an actual push. A digital serial output, from the left and right computers, provide data to the Modular Avionics Unit (MAU) for the Fast/Slow pointer on the Attitude Direction Indicator (ADI) or the Dynamic Speed Bug (DSB) (Primus APEX Build 8 or higher) on the airspeed tape of the Primary Flight Display(s) (PFD's). It is also used for the display of the Low Speed Awareness Indication adjacent to the Air Speed Tape.

The stick pusher, shaker, the Flight Alerting System (FAS) visual "Stall" and aural "Stall" warnings are disabled on the ground through the AIR/GND inputs, except for the self test function. The stick pusher is inhibited for 5 seconds after lift-off. The shaker and the stall warning are operative immediately after lift-off.

The stick pusher actuator has a built-in g-switch which inhibits the stick-pusher when the airplane's normal acceleration becomes less than 0.5 g. The output torque of the stickpusher actuator is electronically-limited to have a force of 60 to 65 lbf on the control wheel. A slip-clutch on the stick-pusher capstan allows control on the elevator with a force of 85 to 90 lbf on the control wheel, in the event of stick-pusher jam. The force on the control wheel is defined when the longitudinal control is pulled to 3/4 of its travel. This allows the pilot or copilot to override the stick-pusher in the instance of an inadvertent operation.

When operated in pusher Ice Mode (to provide protection in icing conditions), all the shaker and pusher actuating points measured by the angle of attack vanes are reduced by 8°. The pusher Ice Mode is set when the propeller de-icing system is switched ON and the inertial separator is set to OPEN. When both pusher computers are set in Ice Mode, a green PUSHER ICE MODE advisory is shown in the ICE PROTECTION window of the systems MFD. If only one computer is set in Ice Mode, or if no computer is set in Ice Mode while conditions for ice mode are present, the amber PUSHER caution is activated.

After engine start on the ground, the CAS Pusher caution will illuminate until the system test has been successfully tested. The test must be done before takeoff. The engine must be operating at a minimum of 5 psi torque, the flaps set to 15°, then press and hold the STICK PUSHER switch to initiate the test. If the test switch is pressed and the test sequence does not occur and/or the CAS Pusher caution remains illuminated, the system has failed the self test and further flight before maintenance is not approved. If the test switch is pressed without the engine operating above 5 psi torque and the flaps are not set to 15°, the PUSHER annunciator will remain illuminated, the “Stall” warning and the test sequence will not occur.

Each outboard control wheel horn is equipped with a PUSHER INTR push switch providing a means to quickly disengage the stick pusher actuator in the event of an inadvertent operation.

The AOA vanes and mounting plates are electrically heated by internal heating elements. AOA vane and mounting plate heat is controlled by the PROBES switch located on the ICE PROTECTION switch panel.

The vane attached to the AOA probe aligns itself with the relative airflow. As it moves, it positions a wiper unit in the probe. This wiper unit adjusts the electrical output to its respective pusher computer. As the airplane approaches the artificial stall (5 to 10 knots before pusher actuator), the stick shaker and the “Stall” warning will activate when one of the AOA pusher computers senses the defined angle of attack for stall warning/stick shaker activation. If the “Stall” warnings are ignored and the approach to stall is continued, the stick pusher will activate when both AOA pusher computers sense the defined angle of attack for stick pusher activation. The stick shaker and “Stall” warning remain active during pusher operation.

Pusher operation will be stopped when either AOA computer senses an angle of attack lower than the angle of attack required to active the pusher or when the airplane acceleration is less than 0.5 g.

If an inadvertent operation of the stick pusher occurs, the pilot can push the PUSHER INTR switch on the control wheel outer horn to quickly disengage the stick pusher actuator.

Activation of the stick shaker disengages the autopilot if engaged, in order to give full authority to a possible stick pusher activation. The autopilot can be manually reconnected after the angle of attack is reduced and the stick shaker has ceased operation.

The stick pusher servo actuator along with the elevator and bridal cables in the area of the stick pusher servo were removed and separated from the general wreckage for possible further examination. The servo actuator was manufactured by L3 Communications Avionics Systems in Grand Rapids, MI. The model number was SA-360D. The part number was 501-1684-04. The serial number was 80736.

Results from the accident aircraft:

According to the LDR recorded data at takeoff, the propeller de-icing system was indicted as ON, and the inertial separator was set to OPEN. These positions were consistent with the aircraft having the pusher Ice Mode set.

The left and right AOA probes were removed and separated from the general wreckage for possible further examination. The right AOA probe was found with the transmitter body separated from the probe end. In addition, the tape used to transmit the probe rotation position to the transmitter body was broken. The damage to the entire unit (including the tape) occurred during removal of the probe from the wing.

According to Pilatus:

"*The Stick Pusher Test is an essential and integral part of the BEFORE TAXIING checklist in the AFM Section 4 'Normal Procedures', to confirm availability of the stick pusher stall protection system.*

To ensure that the test is performed before each flight, the PUSHER caution is always shown when the aircraft is powered up, and the Stick Pusher Test must be performed to clear the caption. If the test is not performed and passed, the PUSHER caution on the CAS will remain, which would be recorded by the

LDR. More precisely, the LDR (and FHDB) would not record that the CAS message was cleared, i.e. would not show a transition from 'caution' to 'no caution'".

Within the LDR data for the accident flight, the master caution parameter changed from "caution" to "no caution" prior to the aircraft taxiing to the runway. This change in indication is consistent with the Stick Pusher Test being conducted and passed successfully.

During the accident flight, the LDR data recorded on the aircraft provided indications consistent with the PROBES switch being in the on position (which would have resulted in a condition where heat was provided to the AOA vanes and mounting plates), but there were also indications that the left AOA heat function was not operating continuously during the accident taxi, takeoff roll, and flight segments. During the approximately 6 minutes prior to the end of the recorded data, the left AOA heat signal was recorded transitioning from on to off and back on again multiple times. The duration of the various on and off conditions was approximately 60 seconds for each condition. During the last 38 seconds of the flight (including the takeoff itself), the left AOA heat function was indicated ON.

According to Pilatus, the following information explains the heat signal characteristics noted on the LDR:

"Each heater is equipped with a thermostat that prevents overheating of the unit. The thermostats have the following thresholds:

Open: at 450F° +/- 25F° (232.2°C +/- 13.9 °C)
Close: at 350 °F +/- 35°F (176.6°C +/- 19°C).

On the ground, with a cold soaked AOA sensor, the heater will first continuously heat until the upper threshold of the thermostat is reached, and then start to cycle on/off. The cycling frequency depends on the open/close temperature tolerance band of the particular thermostat, and is therefore normally not the same for different thermostats at a given cooling rate. The cooling rate, on the other hand, depends on ambient temperature and airflow over the unit. The system is designed so that for steady-state operating conditions at temperatures where icing conditions may be present, and with in-flight airflow over the sensors, the heaters operate continuously without reaching the temperature limit where the thermostat needs to step in to prevent overheating.

Pilatus finds that the recorded data is entirely consistent with this system functionality.
Both heaters activated when the pilot turned on the probes switch.

They continuously heated to warm up the cold soaked probes.
The heater in the left AOA probe was first to reach the thermostat threshold after about three minutes. With the aircraft standing still and some wind blowing over the sensors, the left heater started to cycle on/off regularly with constant heating and cooling periods. When the right heater reached its thermostat's upper threshold less than a minute after the left, it also started to cycle with constant (but shorter, due to different hysteresis) heating and cooling periods.

About 2 ½ minutes before lift-off, the heating periods start to become noticeably longer and the off-periods shorter. This is consistent with an increasing cooling rate (airflow) due to the aircraft taxying for take-off. When the take-off roll starts, the cooling rate due to airflow becomes such that the AOA heaters operate continuously."

1.5 Airfoil deice system

Inflatable neoprene boots are installed on the leading edges of the wings and horizontal tail surfaces. Their purpose is to inflate and dispense any ice which may accrete on their surface during flight in atmospheric icing conditions. When not in use, the boots have a vacuum applied to prevent partial inflation while in flight.

The airplane is equipped with inflatable pneumatic deicing boots fixed to the leading edges of the wings (two boots per wing- inboard and outboard) and the horizontal stabilizer. Air bled from the 3rd stage of the engine compressor section, is routed to the regulator-reliever valve of nominal 14 psi regulating pressure, then through a water separator to the ejector flow control valves. These valves, which are solenoid-operated, port 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 this sequence is controlled by an electronic Timer/controller and monitored by low pressure sensing switches in each line, which are linked to the Modular Avionics Unit (MAU).

In the OFF mode, the system applies a continuous vacuum to the pneumatic de-ice boots while the engine is running. The system is initiated by setting the switch labeled BOOTS on the ICE PROTECTION switch panel. The switch can be set to 3 MIN or 1 MIN and a green advisory BOOTS advisory is shown in the ICE PROTECTION window of the systems Multi Function Display (MFD).

According to the data recorded in the LDR, the wing deicing system power switch was ON, and the cycle time was set to three minutes at the time of the accident.

1.6 Landing Gear

Both main landing gears were found with the brakes and tires still attached.

The wheel hubs on both main gear wheels were found to be pressed into the wheel to the point where the bolt patterns of the wheel were visible on the wheel hubs. The normal wheel hub surface position is not in contact with the wheel surface. The deformation of the wheel hubs was consistent with the gear impacting the ground in the retracted position.

The nose gear was found in the retracted position within the nose gear well. There was significant fuselage compression extending in all directions from the nose gear location. Portions of the nose gear doors were found wedged between the nose gear and the nose gear well. In some locations, aircraft structure was deformed so that it was covering portions of the nose gear. The damage was consistent with aircraft to ground impact in this area with the nose gear in the retracted position.

1.7 Aircraft Seats

A total of eight passenger seats and two crew seats were found in the wreckage.

The seatbelts for the passenger seats were found buckled, unbuckled, and cut. According to reports from the first responders, they found some passengers buckled into their seats and some not buckled into a seat. For those passengers who were still buckled, the first responders were sometimes able to unbuckle their seatbelts to get them out of the seats, and they sometimes had to cut the seatbelts to release the occupant.

1.8 Fuel tanks

The fuel tanks within the wings both appeared to be empty, and there was a strong smell of fuel in the wreckage area.

2.0 Aircraft structure

The aircraft fuselage was found to have widespread compression damage in an area extending from the nose of the aircraft back to the fuselage lower skin below the cargo door. The compression damage in the nose area was consistent with impact related crush damage, while the compression damage aft of the nose gear well was found to be damage where the fuselage skin was pressed back into the fuselage structure.

The fuselage was broken in two main areas. One break was in the area of the wing spars, and another break resulted in the separation of the empennage from the fuselage in the area of the rear pressure bulkhead.

Fuselage structure buckling damage was noted in a location just aft of the cargo door.

The passenger and cargo doors were found to be in the fully open positions but reports from the first responders stated that they found the doors closed, and they opened the doors using the door handles to facilitate passenger rescue operations.

The left wing tip was found to be bent up, and there were several buckled areas noted in the left wing leading edge. The left wingtip light was found to be intact. The leading edge skin was separated from the wing structure along the upper rivet line. Multiple bending deformations were noted throughout the wing structure. There were several fractures noted in the lower wing skin.

The right wing tip was broken off of the right wing, and it was found near the left wing. The radome located on the right wing tip was broken loose, and the forward end of the radome was found pointed aft. The right wing was found attached to the fuselage, but the attachment points were deformed and shifted from their normal positions. The damage in the right wing was relatively minor, and the leading edge was mostly intact. There was a significant bend, and the wing structure was broken in the area just outboard of the aileron inboard edge.

3.0 Aircraft Engine

The aircraft engine was separated from the fuselage, and it was found upright with the propeller blades bent. There was evidence of fire in some areas of the engine. There was a strong fuel odor in the area of the engine.

There was deformation of some of the lower engine components in the aft direction.

According to the engine data plate:

The engine was built by Pratt & Whitney Canada;
The engine model was PT6A-67P;
The engine S/N was PCE-RYO454;
The module S/N was GG-RYG454.

Due to delays in the Pratt & Whitney representative arriving on-scene, the engine was recovered without a detailed on-scene examination.

Scott Warren
Lead Aerospace Engineer