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

August 24, 2021

System's Group Chairman's Factual Report

NTSB No: CEN20MA044

A. ACCIDENT:

Operator:	Cheyenne Partners LLC
Aircraft:	Piper PA-31T, registration number N42CV
Location:	Lafayette, Louisiana
Date:	December 28, 2019
Time:	0921 Central Daylight Time

B. GROUP MEMBERS:

Chairman:	Mike Hauf National Transportation Safety Board
Chairman:	Tom Jacky National Transportation Safety Board
Member:	Kathryn Whitaker Piper Aircraft Inc.

C. SUMMARY:

On December 28, 2019, about 0921 central standard time, a Piper PA-31T airplane, N42CV, impacted terrain shortly after takeoff from the Lafayette Regional Airport/Paul Fournet Field (LFT), Lafayette, Louisiana. The commercial pilot and four passengers were fatally injured; one passenger sustained serious injuries. Two individuals inside a nearby building sustained minor injuries and one individual in a car sustained serious injuries. The airplane was destroyed by impact forces and a postimpact fire. The airplane was owned by Cheyenne Partners LLC and was piloted by an employee of the LLC. The personal flight was conducted under the provisions of Title 14 *Code of Federal Regulations* Part 91. Instrument meteorological conditions prevailed and a Federal Aviation Administration (FAA) instrument flight rules (IFR) flight plan was filed for the flight. The flight was originating at the time of the accident and was en route to the Dekalb-Peachtree Airport (PDK), Atlanta, Georgia.

The airplane wreckage, including the right and left engines, both propeller assemblies, were recovered from the accident site and relocated to Southern Aircraft Recovery in Baton Rouge Louisiana for storage and future examinations.

Note: The systems group was formed after the airplane wreckage had been relocated to Southern Aircraft Recovery in Baton Rouge Louisiana. Any information contained in this report referring to the accident site was provided to the Systems group by the IIC and Piper Aircraft.

D. DETAILS OF THE INVESTIGATION:

D.1 Airplane System Components Recovered during the Wreckage Examination:

On February 10th, 2020, two members from the System's group (National Transportation Safety Board (NTSB) & Piper Aircraft) convened at the Southern Aircraft Recovery facility in Baton Rouge, Louisiana and recovered the system components identified in Table 1 for further examination. The System's group completed this activity on February 13th, 2020.

Table 1 Components recovered

COMPONENT	PART NUMBER	SERIAL NUMBER
Autopilot Pitch Servo	065-0024-02	1572
Autopilot Roll Servo	065-0024-02	1428
Autopilot Yaw Servo	065-0014-11	2738
Pitch Trim Servo	065-0039-03	1468
Angle-of-Attack (AoA) Sensor	861CD2	1536
Stability Augmentation System Computer	-------	8700
Garmin GRS 77 (AHRS)	011-00868-10	42033548

D.2 Flight Control System:

D.2.1 Description:

The Cheyenne II (PA-31T) is equipped with a mechanically operated flight control system for pitch, roll, and yaw. The primary elements of the flight control system are the ailerons and elevators, which are controlled by dual control yokes, and the rudder which is controlled by the rudder pedals. The ailerons and rudder are interconnected through a cable spring system to provide coordination in normal turns.

Secondary flight control is provided by the aileron, elevator, and rudder trim tabs; their position is shown on an indicator mounted next to each trim control on the control pedestal.

The position of the aileron and rudder trim tabs are controlled by trim wheels located on the center pedestal in the cockpit. The rudder trim wheel, for nose left or nose right correction is located below the power controls and the aileron trim wheel, for right wing down or left wing down correction is located on the left side of the pedestal. The trim wheels are mechanically connected to cable drums, which when rotated, drive cables that are connected to another cable drum/screw which moves forward and aft to position the trim tabs.

D.2.2 Vertical Stabilizer:

Note: The vertical stabilizer was documented at the accident site.

The vertical stabilizer remained mounted to the empennage and the rudder had separated from the vertical stabilizer and rudder control sector; the separation of the rudder appeared to be consistent with impact damage. The cables remained attached to the rudder control sector.

D.2.3 Elevator/Stabilizer:

Note: The elevator and the stabilizer were documented at the accident site.

The empennage remained attached to the aft fuselage and came to rest at the end of the debris field adjacent to the inboard left wing and the left engine. The empennage was impact and fire damaged.

The left and right horizontal stabilizer and their respective elevators were fragmented and distributed through the debris field. The right horizontal stabilizer exhibited leading edge damage about mid span consistent with tree impacts.

One elevator control cable was continuous from the elevator control bellcrank and terminated in a ball swage. This swage was found separated from the elevator control quadrant attached to the control yokes; the separation was consistent with impact damage. The other elevator control cable displayed a broomstraw cable separation near the control quadrant.

D.2.4 Elevator Trim:

On the PA-31T, the pilot can control the position of the elevator trim tab by manually rotating the elevator trim wheel located on the left side of the pedestal. The elevator trim tab can also by trimmed electrically by the electric pitch trim control system. This system has two modes of operation: a manual mode and an automatic mode.

The manual mode is operational only when the vertical navigation mode of the autopilot is OFF (Disengaged). Manual electric pitch trim is activated by a dual action pitch trim switch that requires both halves of the switch to be moved simultaneously for actuating up or down trim commands.

When the vertical navigation mode of the autopilot is ON (engaged), the autopilot pitch servo will detect any differential elevator cable tension caused by an out-of-trim condition. To correct the detected differential cable tension, the automatic pitch trim control system will provide a command to the pitch trim servo to move the elevator trim tab to re-trim the elevator. Any attempt to overpower the autopilot pitch axis, such as pulling on the control yoke, will cause the automatic pitch trim system to oppose the applied force, resulting in an out-of-trim condition and high yoke forces.

Additionally, when the vertical navigation mode of the autopilot is ON (engaged), the pilot's pitch trim switch on the control wheel becomes inoperative and any movement of the switch will cause the autopilot to disconnect.

The left and right elevator trim tabs and their respective actuators (drum/screw) were located and documented at the accident site. To determine the position of the trim setting, measurements were taken of the exposed screw on each actuator. It was found that both actuators were extended about 1.25 inches (about 11 threads). Figure 1 shows the right elevator and tab assembly, and Figure 2 shows a measurement being taken on the right elevator trim tab drum/screw. Figure 3 shows the left elevator and tab assembly, and Figure 4 shows a measurement being taken on the left elevator trim tab drum/screw. According to Piper Aircraft, a dimension of 1.25 inches is consistent with the elevator tab positioned 20° trailing edge down (at its stop – or a full aircraft nose up trim setting).

Figure 1 Right elevator and tab

Figure 2 Right elevator tab drum/screw assembly

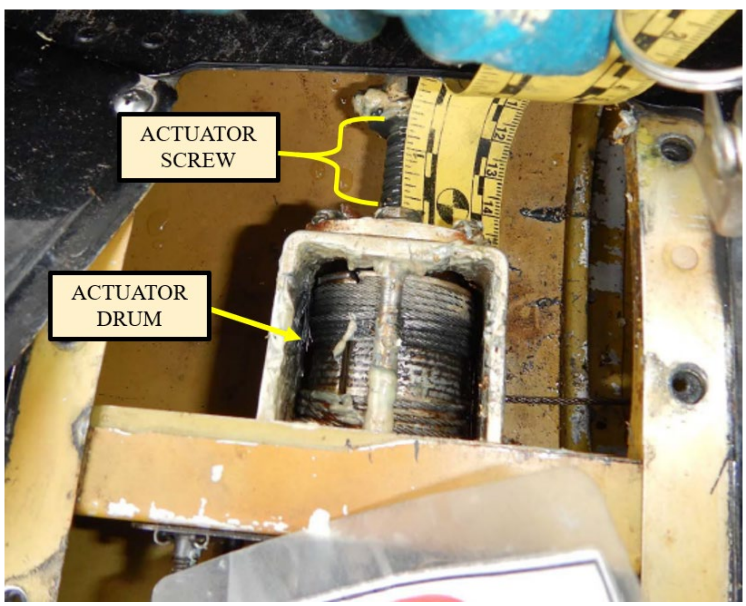


Figure 3 Portion of the Left elevator and tab

Figure 4 Left elevator tab drum/screw assembly

D.2.4.1 Pitch Trim Servo:

The KSA 373 pitch trim servo, part number 065-0039-03, serial number 1468, was located and recovered during the wreckage examination (Figure 5). The unit was placed in a shipping container and sent to the NTSB for further examination. The unit was then shipped to the Duncan Aviation facility in Lincoln, Nebraska, for examination. The servo was held in secured storage while at Duncan Aviation.

Figure 5 Pitch trim servo after removal from airplane

On June 2, 2021, the Systems group met virtually at the Duncan Aviation facility in Lincoln, Nebraska, to examine the KSA 373 pitch trim servo. A local FAA inspector provided in-person oversight at Duncan Aviation and, when the examination began, he witnessed the pitch trim servo being removed from its shipping container. The servo was then examined by Duncan Aviation's technicians under group instruction.

The servo's cover was removed, and a visual inspection was completed. The servo appeared to be in good condition, except for some small surface scratches and dents. No internal or external physical damage was noted that would prevent testing.

The servo was connected to the Duncan Aviation KTS 147 Flight Control System Component Bench Tester and the KSA 737 mounting fixture. The Bendix/King KSA 373 Acceptance Test Protocol (Revision 7, June 2005) was performed on the servo. During the test, the following points were noted:

1. Solenoid Engage Test (Section 5.2.3.3) – The solenoid did not pull the plunger down far enough to fully engage and mesh with the gears.
2. The solenoid plunger binds during part of its travel. It appeared that there was friction in the mechanical movement of the plunger.
3. The solenoid disengages (smoothly) when commanded.
4. Because the solenoid plunger did not fully engage, there was play between the capstan and the servo gear teeth; the "degree turn" play is not acceptable.
5. Steps 6 through 8 – With the Test Set Voltage set to 20.5 Volts, the mounting fixture was turned upside down and the clutch engaged. When engaged, the solenoid appeared to be in "about the same position" as the solenoid engage test per a Duncan Aviation technician.

6. Manual Trim Test (Section 5.2.3.4) – The actuator passed each section of this test. The actuator turned 3 complete clockwise (CW) revolutions in 20 seconds, and 3 complete revolutions counterclockwise (CCW) in 20.5 seconds.
7. Auto Trim Test (Section 5.2.3.5) – The actuator passed each section of this test. The capstan rotated in the proper direction, depending on the set voltage on either the CW (+27.60 Volts) or CCW (-27.61 Volts) direction.
8. KSA 373 Torque Tests (Section 5.2.3.6) – The actuator was tested for proper torque output and disengagement while under 60 inch-pounds load from a slip clutch tool attachment. For Steps 3 (CW) and 6 (CCW), the actuator rotates but skips from teeth not fully engaging. Need to push the solenoid to get the plunger to fully engage the gear teeth.
9. The solenoids disengaged in each direction under load.
10. Visual examination of the actuator gear teeth found no visible evidence of gear teeth wear due to the teeth not engaging fully.

    Additional voltage checks were conducted at the servo's solenoid. A measurement of 28 Volts was noted across the solenoid when applied from the test stand. A measurement of 15.54 Volts was measured across one of the solenoid's coils, but since the solenoid had two coils, about one-half of the applied 28 V was expected.

    The servo's spindle/cable holder was examined. The spindle could be rotated by hand, with no defects noted.

D.2.5 Flaps:

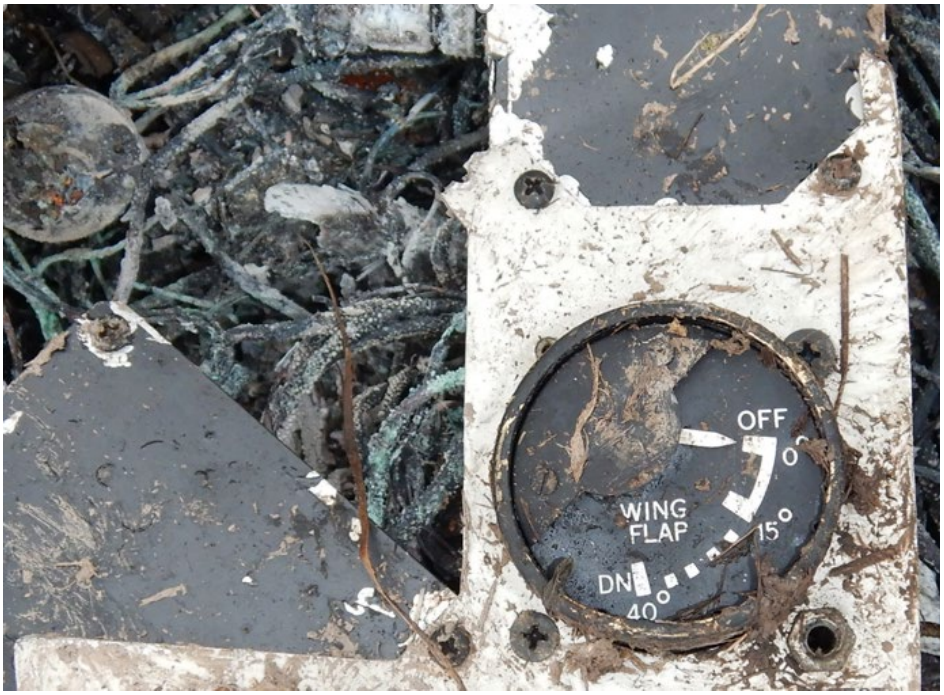
Note: The following information was documented at the accident site.

The wing flap control system provides continuous control and monitoring of flap position and condition over its full range. Wing flap position is controlled by a selector switch mounted on the instrument panel immediately to the right of the control pedestal. The flap position indicator is located to the left and above the selector switch. The position indicator 40° arc is segmented at approach (15°) flap range and in increments of 5° between the 15° and 40° arcs. An OFF position is also provided to indicate zero (0) voltage to the system. Flaps are deployed mechanically by a single motor driving through two flexible shafts connected to individual ball screw actuators.

A photograph of the flap position indicator (Figure 6) was taken at the accident site showing that the flaps were positioned to "OFF" (0°).

The right flap assembly came to rest in the field; its flap jackscrew was found separated from the flap assembly. Visual inspection of the jackscrew found that no threads were visible. According to Piper Aircraft, this is consistent with the flaps being in their fully retracted position. The inboard portion of the left wing including a portion of the left flap and the left nacelle, came to rest adjacent the fuselage and empennage. A visual inspection of the left flap jackscrew found that no threads were visible consistent with the flaps being fully retracted.

Figure 6 Flap position indicator



D.3 Wings:

A visual inspection of the left and right wings was conducted at the accident site.

D.3.1 Right Wing:

The right wing had separated from the fuselage and was found impact damaged, fire damaged and fragmented; its inboard section was in the parking lot near an inverted vehicle.

The right aileron was impact damaged, separated and fragmented into two large sections. The right aileron control cables were impact damaged in several locations and exhibited "broomstraw" signatures and therefore control cable continuity could not be confirmed[1].

The right wing tip tank had separated from the wing. Witness marks on the inboard side of tank and impact marks on trailing top edge of the tank were consistent with impact with a light pole.

D.3.2 Left Wing:

The left wing had separated from the fuselage and was found impact damaged, fire damaged and fragmented. The inboard portion of the left wing, to include a portion of the left main gear, left flap and the left nacelle, came to rest adjacent the fuselage and empennage.

[1] Right aileron control cable continuity could not be confirmed because the right aileron control cable end that mounts to the yoke chain was not observed on or off-scene.

The left aileron cable remained with the wing and exhibited "broomstraw" signatures consistent with impact and overload and therefore control cable continuity could not be confirmed[2].

The left wing tip tank had separated and was located on the left side of the debris field on the side of the road adjacent to a tree. The left tip tank contained fuel. Witness marks on top of the tank were consistent with an impact with transmission lines.

D.4 Landing Gear:

The position of the landing gear was documented at the accident site. Both left and right main landing gear remained attached to their respective wing within their wheel well. The nose landing gear was impact separated from the surrounding structure and was observed in the parking lot near the beginning of the wreckage path.

D.5 Throttle Quadrant:

The throttle quadrant was located to the right side of the debris field near the inverted car. The throttle quadrant was impact and fire damaged and did not provide any reliable positions or indications.

D.6 Pitot /Static System:

D.6.1 Pitot Probe:

A visual inspection of the pitot probe during the wreckage examination revealed that the probe along with its mounting bracket remained loosely contained within the nose structure; it was impact damaged and folded aft (Figure 7). Normally, a flexible hose and two electrical "heating element" wires are connected to the probe assembly. Inspection revealed that neither the flexible hose nor the two heating element wires remained connected to the probe assembly.

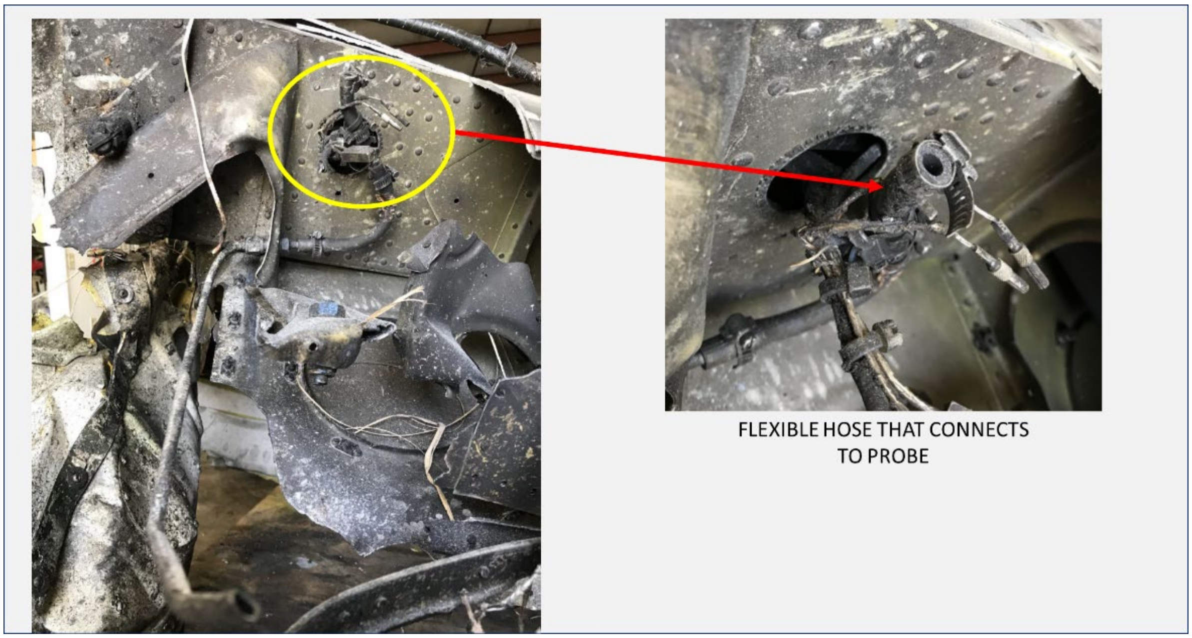
Figure 7 Pitot probe at found at wreckage examination

[2] The left wing cables, cable to the yoke, and balance cable continuity could be established through broomstraw cable separations.

Using hand pressure, the probe assembly was easily removed from its mounting position within the fuselage nose structure. To better observe the flexible hose and two electrical "heating element" wires, a small portion of the metal structure around the pitot tube mounting hole was cut and removed from the fuselage. A visual inspection was performed, and the following observations were noted and shown in the figure 8:

- A clamp remained firmly attached to the flexible hose.
- The pitot line branched into a 2 foot 8 inch section of flexible hose and a 1 foot 6 inch section of aluminum tube.
- The flexible hose terminated in a clean cut, and when tested with light air pressure, revealed no blockages. The aluminum tube was found crushed and folded in several places, and light air pressure could not flow through it.

Figure 8 Flexible hosing to pitot probe



A visual inspection of the probe was performed, and the following observations were noted and shown in figure 9:

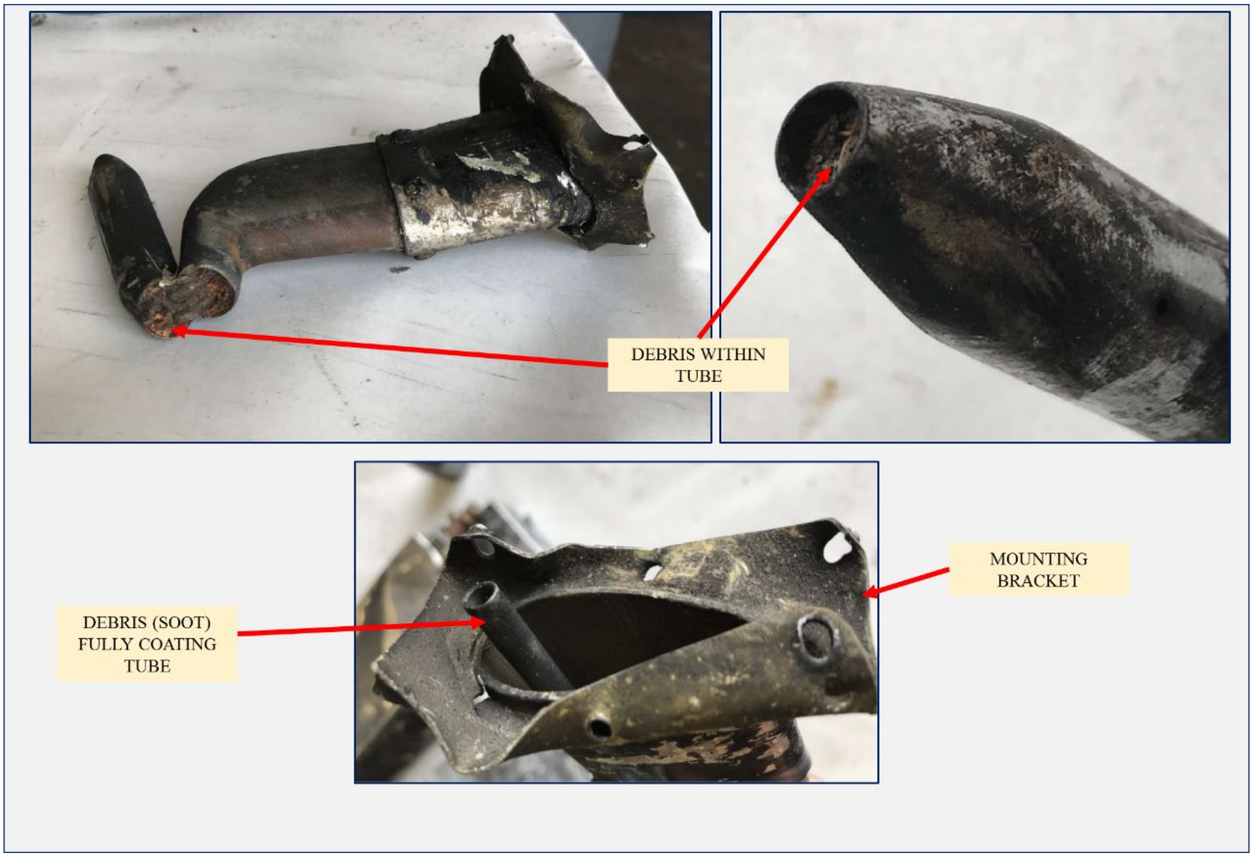
- Debris was found imbedded in the pitot tube port and the break.
- The metal "pressure" tube coming out of the probe was fully coated in black debris (soot).
- The mounting bracket was deformed and the holes for the attachment hardware were all found damaged (pull through or elongated).

A continuity test of the heating element was performed by checking the continuity between the two pins. The continuity was found open.

The probe was removed from its mounting bracket by removing four screws to determine the probe's part number and serial number. The following information was found inscribed on the tube; a serial number was not found.

- MFG part number: PH502
- Aero Instruments Co.

Figure 9 Photographs of the pitot probe



D.6.2 Static System:

The static pressure sensing system consists of a total of four static source pads (two pads are mounted on each side of the empennage near the horizontal stabilizers), an alternate static source pad; an KDC 380 air data computer (ADC); and static lines. Two static lines are routed from the empennage forward though the right side of the cabin to the instrument panel located in the cockpit. One of the static lines provides the sensed static pressure from the two aft static pads, which are crossed plumbed by flexible hosing at a T-Junction; the other static line provides the sensed static pressure from the two forward static pads (left side and right side). The air data computer is plumbed to the left side aft static source pad through flexible lines. An alternate static source pad is mounted within the unpressurized section of the nose.

A visual examination of the static sensing system revealed the following:

- Each of the four static ports were intact and remained attached to fuselage by the mounting hardware; the ports on the left side of the fuselage did not appear to be externally damaged, the ports on the right side of the fuselage appeared to have light scoring (impact damage) and scuff marks.
- All flexible static lines remained attached to their respective static ports.
- The aft flexible static line going to the air data computer was found separated about 1.5 inches from its connection at the air data computer; the line visually appeared thermally damaged (charred) at the separation.
- The flexible static lines aft of the aft pressure bulkhead showed minor thermal damage (no burn through).

- The static lines in the cabin area forward of the aft pressure bulkhead were brittle from thermal damage and sections were consumed by fire.
- The cabin area forward of Fuselage Station (FS) 200 (approximate) was fragmented and burned; the static lines in this area were not observed.
- The air data computer was soot stained. The air data computer static lines were thermally damaged. The air data computer static lines were thermally damaged and broken apart. Honeywell indicated to the NTSB that this computer does not contain any non-volatile memory.

The alternate static source pad and associated lines were impact separated from the nose section and were not located.

D.7 Stability Augmentation System:

D.7.1 Description:

The stability augmentation system (SAS), in the PA-31T airplane is required in order to satisfy certification requirements regarding static longitudinal stability.

The SAS automatically improves the static longitudinal stability of the airplane by providing variable elevator force. This is accomplished through tension changes in an elevator down spring. During normal operation above 126 kts calibrated airspeed (KCAS), the servo does not apply tension changes to the elevator down spring. If the aircraft approaches a stall, the servo applies tension to the elevator down spring, which increases the amount of force required to adjust the airplane's pitch. If the airplane enters a stall, the servo applies its maximum amount of tension to the elevator down spring.

The SAS consists of four major components plus a three position test switch. Also incorporated in the system is a power warning light, a ram warning light and a stall warning light and horn. The major components of the SAS are: a Stall Margin Indicator, Computer, an angle-of-attack (AOA) sensing vane, and a Servo Actuator. The sensing vane heat is controlled by the left pitot heat switch.

The stall margin indicator is mounted on the upper left side of the pilot's instrument panel.

The 20L1 SAS computer is designed to provide the electrical interface between the Model 0861CD2 AOAS to the PA-31T stall protection system. The Collins Aerospace Model 20L1, Mod B SAS computer was designed and qualified in 1974. This model was not designed for a specific aircraft platform and is utilized on multiple aircraft. The 20L1 SAS computer provides compensations to the AOA sensor signal as well as the following output functions:

- Stall margin related signals
- Stall warning trip point
- Stability augmentation associated functions
- Feedback signals

The AOA Vane is mounted on the right side of the nose section of the airplane; it senses the angle of attack by the airflow deflecting the vane either clockwise or counterclockwise. This information is fed to the SAS computer. The computed output derived from the AOA information is utilized as follows:

1) To drive the Stall Margin Indicator, which is mounted on the upper left side of the pilot's instrument panel, displays the airplane flight regime as follows: <u>STALL</u>, red area of indicator; <u>STALL</u>

WARNING, red and black barber pole area of indicator; 1.3 Vs[3], white area of indicator: CRUISE, green area of indicator.

2. To drive the Servo Actuator which is attached by a cable assembly to the elevator down spring. The elevator down spring is attached to the elevator horn assembly by an additional cable, which provides a variable elevator down spring tension designed to improve airplane static longitudinal stability.
3. To activate the stall warning light and horn when the airplane approaches a stall speed situation.

A SAS test panel, located on the pilot's instrument panel, provides a test switch for preflight checking of the SAS. A warning light located to the left of the annunciator panel will illuminate with any SAS malfunction. Should the SAS malfunction, the lights will illuminate continuously until the malfunction is corrected. If the malfunction is caused by a power failure, the "power" warning light will illuminate; if the malfunction is caused by a computer, a sense vane, or servo failure, the "ram" warning light will illuminate.

The SAS is equipped with a stability augmentor override system. Should the SAS fail to function satisfactorily during flight, the pilot can override the system by removing the access cover on the right side of the pilot's control pedestal and pulling the lanyard actuator handle. The override is pneumatically operated. When the lanyard actuator handle is pulled, compressed gas is released from the CO2 cartridge into a cylinder located in the aft section of the fuselage. Under normal operating conditions, the cylinder is filled with hydraulic fluid and the piston and rod assembly in the cylinder is in the down position, where is has no effect on the elevator downspring. When the lanyard actuator is pulled, the CO2 is discharged through a line and into the cylinder, which drives the piston and rod assembly upward. The rod locks into place, keeping a constant tension of about 20 additional pounds on the elevator downspring; These features are designed to provide suitable handling characteristics for the flight. At the same time, the hydraulic fluid is forced from the cylinder into the reservoir on top of the fuselage, where it is held until the override system is rearmed.

D.7.2 Stability Augmentation System - Inspection:

D.7.2.1 Stall Margin Indicator:

The stall margin indicator is mounted on the upper left side of the pilot's instrument panel. This indicator receives its signal from the AOA vane through the conditioning computer and presents a visual indication of the ratio of present speed to stall speed in the same configuration (V/Vs).

The stall margin indicator was located within the wreckage at the accident site and as shown in Figure 10, the remaining portion of the indicator needle was positioned within the red area.

[3] Vs means the stalling speed or the minimum steady flight speed at which the airplane is controllable. In this case, 1.3 Vs on this instrument indicates airspeed 1.3 times above stall speed.

Figure 10 Stall margin indicator

D.7.2.2 SAS Servo Motor:

At the accident site, the condition and position of the stability augmentation system servo was documented; it remained mounted in the aft fuselage with its servo arm positioned up. According to Piper Aircraft, its position is consistent with system activation. During the wreckage examination at the Southern Aircraft Recovery facility in Baton Rouge, Louisiana, a photograph of the servo actuator was taken Reference Figure 11.

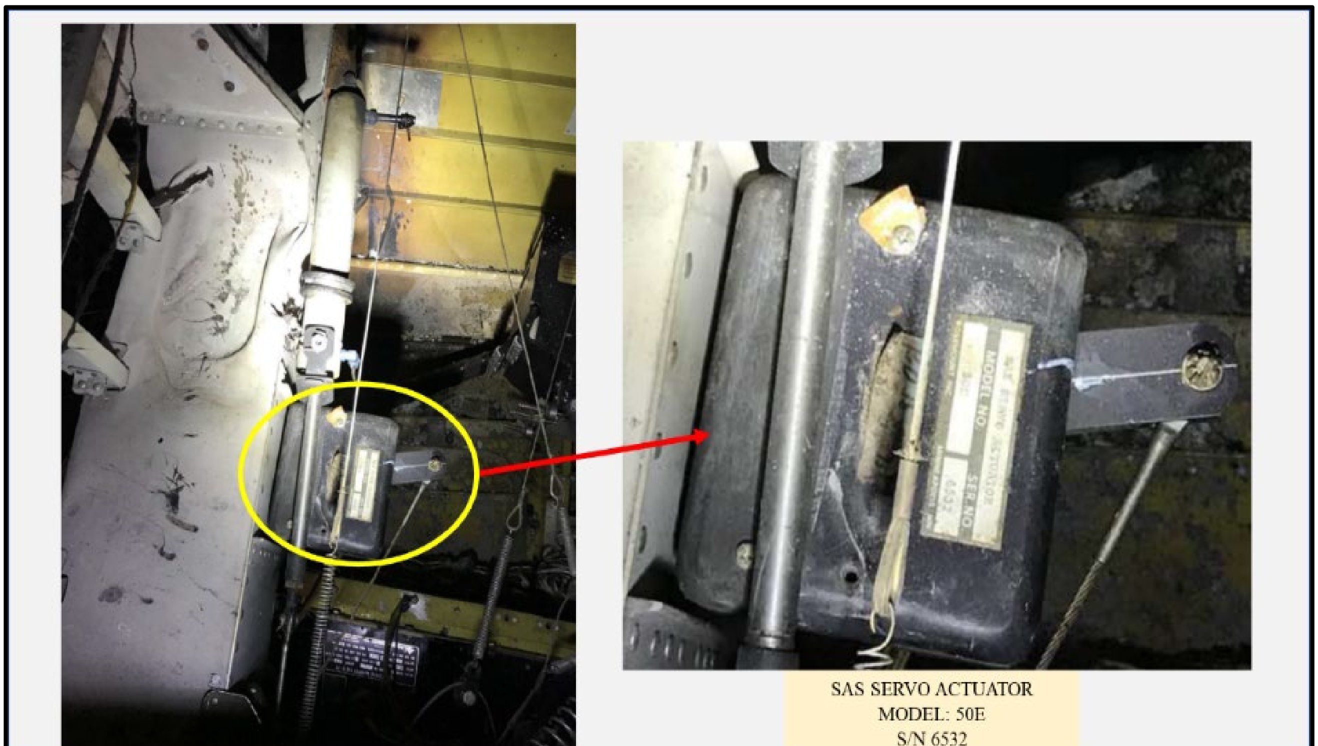
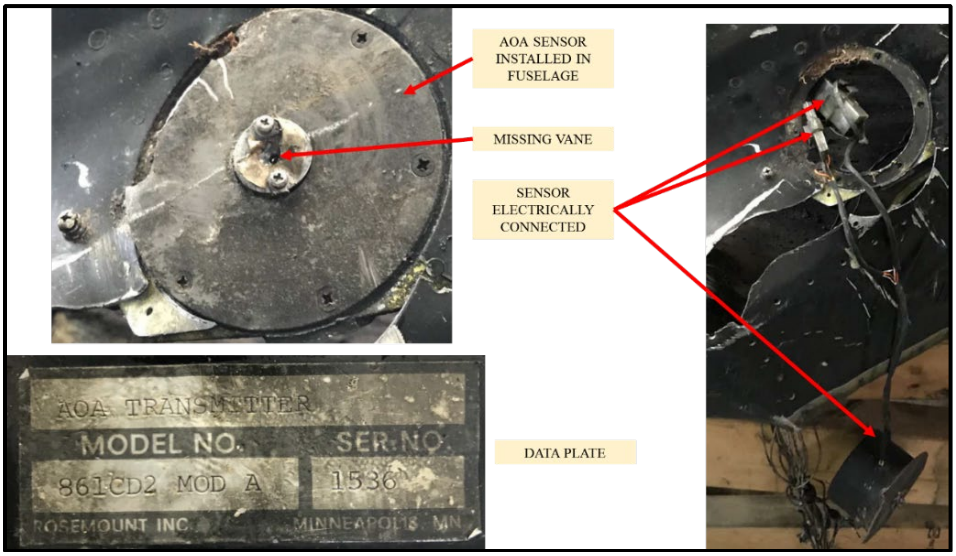


Figure 11 SAS Servo Motor

D.7.2.3 Angle of Attack Sensor (AOAS):

During the wreckage examination, the AOA Sensor (without vane) remained installed in the fuselage and electrically connected via its two electrical connectors (Figure 12). The sensor was removed from the fuselage by removing its attachment screws and cutting the electrical wiring aft (airplane side) of the connectors. The data plate indicated the following: "AOA transmitter, Model number 861CD2 (Mod A), Serial Number 1536".

Figure 12 AOA Installation



The Systems group met virtually at the Collins Aerospace facility located in in Burnsville, Minnesota on April 5, 2021, to examine the AOA sensor. Prior to the examination, the NTSB shipped the sensor to the Collins facility, where the shipping box was held in secured storage.

The examination was conducted using the investigation plan developed by Collins and approved by the group. The examination was conducted as follows.

1. <u>Production Documentation / Repair History:</u>

   Collins Aerospace reviewed production records for the Model 0861CD2, Mod A, Angle of Attack sensor and the model 20L1, Mod B, Stall Augmentation System computer to determine the manufacturing date for:
   • 0861CD2, Mod A, SN 1536
   • 20L1, Mod B, SN 8700

   Collins examined its production records systems (SAP and Oracle) and did not identify the original production records for the subject units. Further examination of Collins' archived records database (Iron Mountain) also did not reveal original production records for the subject units.

   Based on the engineering drawing documentation for the SAS computer and AOA sensor models the identified SN numbers, and the review of production data records back to 1995, Collins Aerospace estimates the date of manufacture between 1980 and 1990.

Collins Aerospace completed a facility records review of its Maintenance Repair and Overhaul (MRO) facility in Burnsville, Minnesota. The Collins MRO repairs the 0861CD2 and 20L1 products. Collins found no records of repair for the 20L1, Mod B SAS Computer, SN 8700. Collins located two repair records for the 0861CD2, Mod A, AOA:

A. Documentation reflecting the repair of a vane heater failure in March of 2000. The records indicate replacement of the vane and internal shaft, wiper CCA and potentiometers.

B. Documentation reflecting the repair to a flying lead cable in June 2000. This separate repair work appears to have been performed after the March 2000 repair. The Collins MRO database stores records covering repair activity from 1995 through 2021. Table 2 summarizes all recorded MRO activities for the AOA sensor.

Table 2 0861CD2, Mod A MRO Activity

RMA Number	Model Number	Final Assembly Part Number	Serial Number	Customer Reason for Return	Repair activity	Date Unit Received	Date Repair Completed
137601	0861CD2, Mod A	00861-0850-0001	1536	Repair	Vane heater element was failed. Contamination was also found in unit. The unit was repaired with replacement of heated vane, shaft, wiper CCA assembly and potentiometers. Clean and ATP	03/16/2000	04/13/2000
140001	0861CD2, Mod A	00861-0850-0001	1536	Open Circuit	Flying lead cable insulation was broken/pinched/worn. Unit was also out of calibration Replace yellow and red/white wires in cable harness. Recalibrated. Clean and ATP	05/26/2000	06/09/2000

2. Removal from Shipping Container and Visual Inspection

The 0861CD2, Mod A AOAS was removed from the shipping packaging and visually examined. The following observations were noted:

- The sensor's part and serial numbers were confirmed.
- The sensor's vane and part of faceplate had broken off (were not present).
- The exterior was noted to be in good physical condition. Some exterior areas were noted with missing/worn paint, which Collins indicated was normal wear.
- The vane hub appeared to be positioned at the counterclockwise mechanical stop and difficult to move. The assessment by the Collins Aerospace participants was that the accident sequence impact pushed the vane to its mechanical stop. This position would be consistent with the AOA vane being moved to the stop due to a large force (impact with ground), causing vane separation, faceplate damage and internal bearing damage.
- The visual inspection revealed that the vane shaft was bent. A 3D-printed vane was placed onto vane hub to measure the approximate angle of the vane shaft. The approximate angle of the

vane was measured to be 24° counterclockwise from mechanical zero (0°) position[4].

- The vane was difficult to move. Therefore, unable to test via the acceptance test procedure (ATP).

3. Acceptance Test Protocol

The group elected to perform the available sections of the ATP (Collins Aerospace Document D8220005, Revision E), despite the condition of Angle of Attack Sensor. To facilitate the ATP, the external AOAS connectors were removed per standard procedure by cutting the wire bundle about 12 inches from the AOAS body to access the individual wires necessary for the test.

The following sections of the ATP were accomplished or designated not applicable (NA), due to the damaged condition of the AOAS, as follows:

- §3.2 Static Friction – NA (no vane)
- §3.3 Mechanical Travel - NA (no vane)
- §3.4 Electromechanical Alignment

  Potentiometer outputs were measured at current shaft position without aligning vane hub centerline with mechanical zero.
  - Potentiometer 1 = 0 V (No output)
  - Potentiometer 2 = -1.8 V (the output voltage was considered unstable)

- §3.5 Deicing Heater - NA (no vane)
- §3.6 Potentiometer Voltage Differential - N/A (no vane)
- §3.7 Case Heater Resistance

  Measurement: 45.37 ohm => Result: PA

- §3.8 Wiper Insulation Resistance

  The AOAS cover was removed for inspection of the sensor wipers. There was a dent in the AOAS cover due to accident impact damage, so the cover had to be bent for removal.

  Inspection showed that both wipers were positioned at the end of each potentiometer.

  The vane shaft was then physically moved to roughly the center position (0°) and the potentiometer outputs were recorded:
  - Potentiometer 1 = No output
  - Potentiometer 2 = 6.17 VDC

  Potentiometer outputs were measured at current shaft position without aligning vane hub centerline with mechanical zero.

  Note that a 6.17 VDC potentiometer output corresponds to about 5° from the vane center position. The potentiometer output was monitored as the vane shaft was moved, and the potentiometer output varied with the vane shaft movement as expected.

  To document the wiper contact on the potentiometers, at the conclusion of the examination,

[4] According to Collins Aerospace, the AOAS mechanical stops were located at 23±1° from mechanical zero.

Collins had the AOAS photographed for close-range photos of each wiper in contact with their respective potentiometers. Collins provided the photographs via email.

The wiper located on Potentiometer 1 did not appear to make good contact with potentiometer 1. Figure 13. Potentiometer 1 is located on the side of the AOAS with the most damage.

Figure 13 Wiper 1 contact with Potentiometer Number 1

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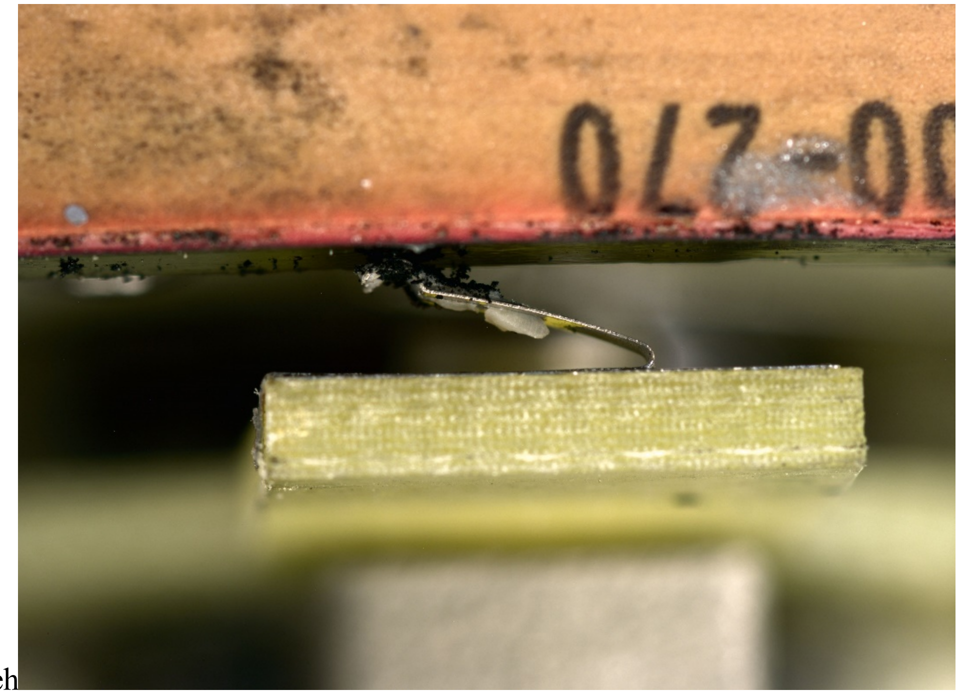


The wiper located on Potentiometer 2 appeared to make sufficient contact with Potentiometer Number 2. Figure 14.

Figure 14 Wiper 2 contact with Potentiometer Number 2

D.7.2.4 SAS Computer:

During the wreckage examination, the SAS computer remained installed in the fuselage by its mounting hardware (4 screws) and electrically connected via its two electrical connectors (Figure 15). The computer was removed from the fuselage for further examination. During the removal, the right electrical connector had to be disconnected from the computer in order to remove the unit. The data plate indicated the following:

- SAS Computer Unit
- Model No. 20 1 MOD B, Serial No. 8700
- Rosemount INC, Minneapolis, Minnesota

Figure 15 SAS Computer

The Systems group met virtually at the Collins Aerospace facility located in in Burnsville, Minnesota, on April 5, 2021, to examine the airplane's SAS Computer. Prior to the examination, the NTSB shipped the unit to the Collins facility, where the shipping box was held in secured storage.

The examination was conducted using the investigation plan developed by Collins and approved by the group. The examinations were conducted as follows.

The SAS computer was removed from its shipping container and visually examined. The computer's part and serial numbers were confirmed. The unit appeared to be in good condition, with the exception of some small surface scratches and dents on the top cover.

The cover was removed, and the internal components were visually examined. Each circuit card assembly (CCA) appeared to be in good condition.

1. Acceptance Test Protocol

The SAS Computer ATP (Collins Aerospace Document 67821, Revision B) was performed on the unit. The group decided to accomplish the following chapters:

- **§5.0 AOA Indicator Output Accuracy**
    - 0° Vane Transmitter = 0 (left of 0) units AOA Indicator - **PASS**
    - 4.2° Vane Transmitter = 0 units AOA Indicator – **FAIL**
    - 15° Vane Transmitter = 140 units AOA Indicator – **FAIL**
    - 25° Vane Transmitter = 280 units AOA Indicator – **FAIL**

- **§6.0 S/W Test**
    - Lamp turns ON – 60 units – **FAIL**
    - Lamp turns OFF – w/ in 7 units of 60 – **FAIL**

- **§7.0 AOA Functional Test**
    - PTT 1.0 – 0 units – **PASS**
    - PTT 1.3 – 165 units – **FAIL**

- **§8.0 AOA Transmitter Comparison**
    - Vane Compare HI - SAS Light turns ON – **PASS**
    - Vane Compare NORM – SAS Light turns OFF – **PASS**
    - Vane Compare LO -SAS Light turns ON – **PASS**

- **§9.0 Power Fault**
  Lights turn ON per acceptance criteria:

    - AOA – Fault Light turned ON when pressed – **PASS**
    - S/W – Fault Light turned ON when pressed – **PASS**
    - ACT – Fault Light turned ON when pressed – **PASS**

- **§10.0 Servo Alignment**
    - FB Position 1 - 150 units @ null – **PASS**
    - FB Position 2 - 200 units @ null – **FAIL**

- FB Position 3 - 250 units @ null – **FAIL**

- **§11.0 Power Supply Comparison**
  - Null meter centered – **PASS**

- **§12.0 SAS Fault**
  Lights turn ON/OFF per acceptance criteria:

  - Position 1 - Light turns ON – **PASS**
  - Position 2 - Light turns OFF – **PASS**
  - Position 3 - Light turns ON – **PASS**

Collins indicated that, overall, the results of the ATP indicated that the calibration of the SAS Computer was slightly out of tolerance, and that the results were typical of an incoming unit.

Following the ATP, the SAS Computer circuit card assemblies were removed for photos and then re-assembled.

D.7.3 Autopilot Servos:

The autopilot pitch, roll, and yaw servos were located and recovered during the wreckage examination (Figure 16). The units were placed in a shipping container and sent to the NTSB for further examination. The units were then shipped to the Duncan Aviation facility in Lincoln, Nebraska, for examination. The shipping box was held in secured storage while at Duncan. Visual inspection revealed the following:

1. <u>Yaw King Servo Actuator (KSA)</u>
   Part Name: KSA-370 Servo Actuator
   Part Number: 065-0014
   Serial Number: 2738

2. <u>Pitch King Servo Actuator</u>
   Part Name: KSA-371 Servo Actuator
   Part Number: 065-0024
   Serial Number: 1572

3. <u>Roll King Servo Actuator</u>
   Part Name: KSA-371 Servo Actuator
   Part Number: 065-0014
   Serial Number: 1428

Figure 16 Autopilot Servos

The group met virtually at the Duncan Aviation facility in Lincoln, Nebraska, on June 2, 2021 to examine the three King (now Honeywell) autopilot servos. Prior to the examination, the NTSB shipped the units to the Duncan Aviation facility, where the shipping box was held in secured storage.

Upon the group's arrival, the components were removed from their shipping boxes. The components were then examined by Duncan's technicians under group instruction. Each component was visually examined, bench tested if able, and then disassembled. The examinations were conducted as follows.

D.7.3.1 Examination of the Pitch Servo:

The Pitch Servo was visually examined. The part and serial numbers were confirmed. No visible defects were noted. As part of the visual examination, the wire harness connector was detached; the plug and unit connector look fine, with no defects noted. The protective tape around the servo between the back chassis and the servo body looked fine. The dust cover was removed to examine the internal components; no defects were noted, and the servo was considered testable.

The capstan was removed from the servo. One of the four connecting bolts was missing. The capstan looked okay, no defects noted.

The capstan's plastic dust cap was removed, and the capstan torque was checked – the torque was okay. No torque value was written on the slip clutch.

The servo actuator was connected to the Duncan Aviation KTS 147 Flight Control System Component Bench Tester and the KSA 371 adapter/mounting fixture. The Honeywell KSA 371 Servo Test (Revision 3, August 1983) was performed on the servo.

During the test, the following points were noted:

1. KSA 371 Servo Test (Section 6.2.3.3) - All steps conducted with no defects noted.
2. Clutch Test (Section 6.2.3.4) - The servo was attached to the Torque Testing Fixture. All steps were completed with no defects noted.
3. KSA 371 Torque Limiting Resistor and Voltage Torque Limiting Low Level (Section 6.2.3.5) – The servo was attached to the Torque Testing Fixture.
4. Step a. The CW Low Level Torque was measured at an average of about 43-inch pounds.
5. Step b. The CCW Low Level Torque was measured at an average of about 45-inch pounds.

Overall, no defects were noted, and a Duncan Aviation representative indicated the servo appeared to be in an operable state.

D.7.3.2 Examination of the Yaw Servo:

The Yaw Servo was visually examined. The part and serial numbers were confirmed. No visible defects were noted. As part of the visual examination, the wire harness connector was detached; The servo connector pins looked straight with no corrosion noted. The protective tape around the servo between the back chassis and the servo body looked fine.

The back connector, servo capstan, and all four of the mounting bolts were removed.

No binding was noted on the servo rotating spline. At least two of the metal bars around the capstan were bent; the group could not determine if the bars were bent during the accident sequence or the removal from the airplane. The bearings on the servo mount appeared normal, rotated fine. No torque value was written on the slip clutch.

80-85 one direction, 85 in the other direction.

The servo was connected to the Duncan KTS 147 Flight Control System Component Bench Tester and the KSA 370 adapter/mounting fixture. The Honeywell KSA 370 Servo Test (Revision 3, August 1983) was performed on the servo.

During the test, the following points were noted:

1. KSA 370 Servo Test (Section 6.2.3.3) – For CW direction, the measured voltage is +2.7 Volts (greater than the allowed +2.0 Volts), and CCW the measured voltage was -1.27 Volts. When the CW direction was attempted a second time, the measured voltage was +1.27 Volts. A representative from Duncan Aviation indicated that the different measurements could be from dirt in the gear train or a problem with the motor.
2. Steps 15-22: The output voltage (motor speed) was measured within tolerance in both the CW and CCW directions.
3. Clutch Test, (Section 6.2.3.4) Step 5 – The air gap is too tight with the clutch engaged. The step failed.
4. Steps 6-13 were successfully completed.
5. Step 16 – In CW direction, measured value of 10 in-lb., in the CCW direction 15 in-lb.
6. Step 24 – Clutch engaged at approximately 16.5 Volts.
7. Step 26 – Clutch disengages at approximately 5 Volts.

8. Clockwise Aileron (Section 6.2.3.5.1) – Skip step 5 because no zeiner diode.
9. Step 9 – The output torque was measured at 110 in-lb. (CW) and 115 in-lbs. (CCW) within the specifications as set in Table 6-2.
10. Clockwise Clutch Disengagement (Section 6.2.3.6.1) – Passed
11. Counterclockwise Clutch Disengagement (Section 6.2.3.6.2) – The clutch was considered sticky in the CCW direction.
12. Final Verification (Section 6.2.3.7) – Skip to step 12, noted sticky clutch again.

D.7.3.3 Examination of the Roll Servo:

The group determined that the Roll Servo was too damaged for testing. The outer case was deformed, by the accident sequence, around the actuator chassis. When the outer case was cut and pried away from the chassis, the top circuit card was found pushed down into the second circuit card.

The part number and serial numbers were confirmed.

When the wire bundle was disconnected, the rear connector was examined; all pins looked good.

The internal gear train could be moved by hand; no witness marks were noted on the gear teeth.

No evidence was noticed of burnt internal components, and no burnt traces were noted on the back side of the circuit cards.

The snap ring was missing; the group assessment was that the ring was detached during the accident sequence.

During the testing there was a group discussion regarding the initial torque settings for the servo actuators, based on the initially-installed King KFC – 300 Autopilot System. Piper Aircraft provided documentation for the STC that was installed on the airplane.

D.8 Flight Instruments:

D.8.1 New Equipment:

Review of maintenance records showed that in July 2016, the airplane was retrofitted, by Avionics Solutions LLC, with a Garmin G600 System per Supplemental Type certificate (STC) SA02153LA-D. The installation was completed in accordance with Garmin installation manual 190-00601-06 Revision N, dated October 31, 2015. The new equipment consisted of the components shown in Table 3:

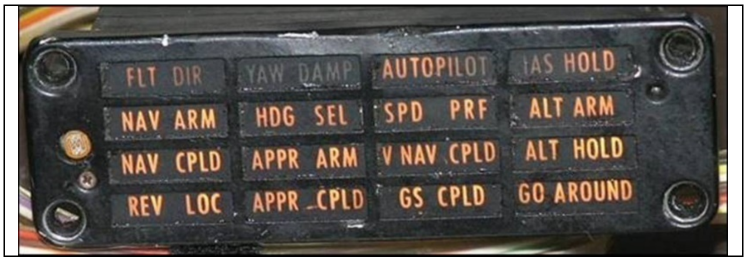
Table 3 Components Installed

Description		Part Number	Serial Number
GAD 43e Autopilot Interface		011-02349-00	1V7050752
GDU 620 Display		011-01264-00	165003702
Digital Attitude Indicator (Standby)		RCA 2600-3	916F357
GDC 74A Air Data Computer		011-00882-10	20622832
GRS 77 AHRS		011-00868-10	42033415
GMU 44 Magnetometer		011-00870-00	47525327
GA 55 Antenna		010-10600-01	87501792

D.8.2 KAP 315 Annunciator Panel Examination:

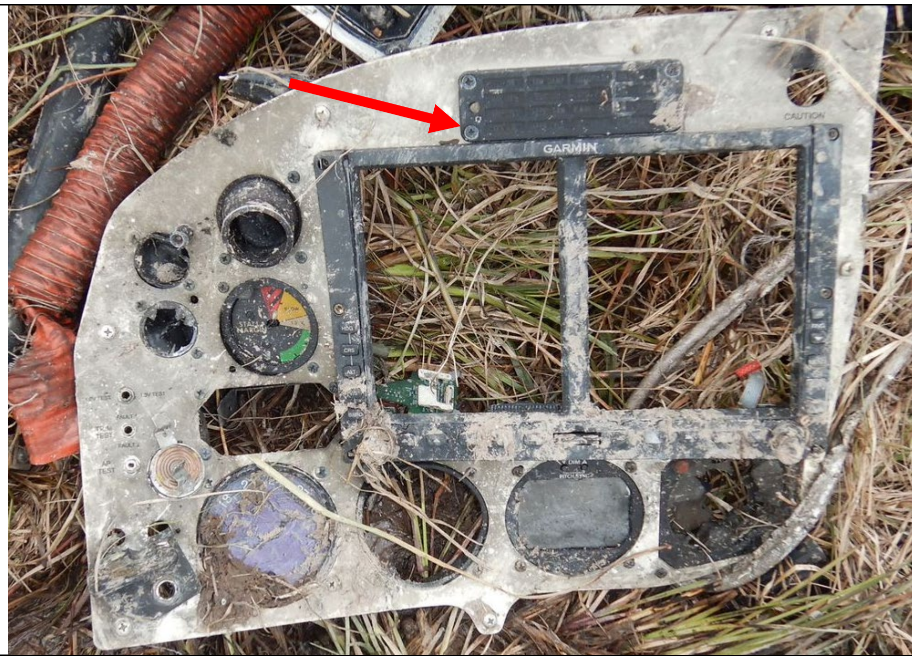
The KAP 315 annunciator panel (Figure 17) annunciates all vertical and lateral flight director / autopilot system modes, including all "armed" modes prior to capture.

Figure 17 Exemplar KAP 315 annunciator panel



The KAP 315 annunciator panel was recovered from the wreckage at the accident site (Figure 18) and was submitted to the NTSB Materials Laboratory for examination of the bulb filaments in each of the annunciator lights. There are two light bulbs present for each annunciator position.

Figure 18 KAP 315 annunciator panel found on-scene



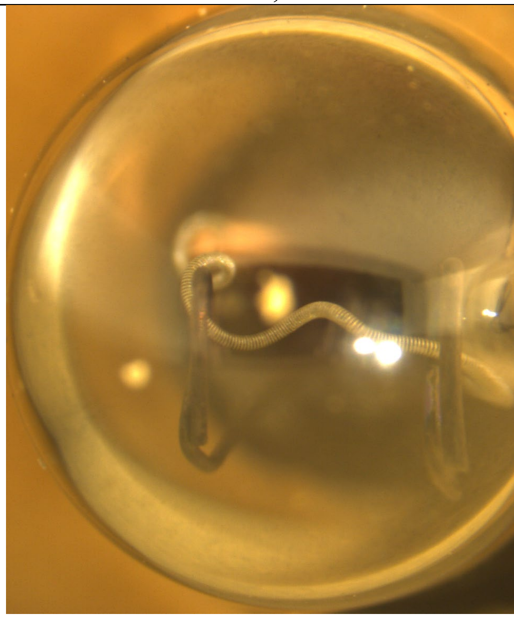
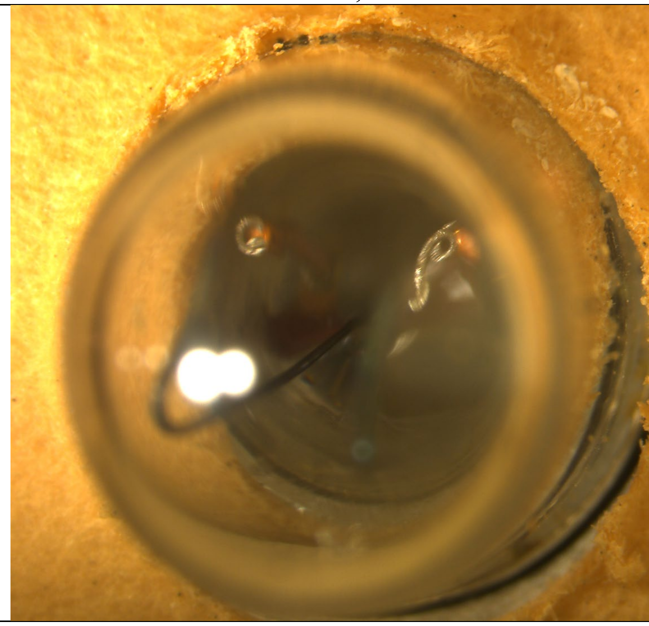
On 1/14/2020, a light bulb filament analysis was conducted on the annunciator panel at the NTSB Materials Laboratory by a Materials specialist. The front cover was removed from the autopilot panel in order to visualize the bulbs inside. The individual bulb filaments were examined using a stereomicroscope. The status of each filament is listed in Table 4. Photographs of the **AUTOPILOT** bulb filaments are provided in Figure 19

Table 4 Results of filament analysis

FLT DIR BNS INS	**YAW DAMP** BNS INS	**AUTOPILOT** BNS INS	**IAS HOLD** INS INS
NAV ARM INS INS	**HDG SEL** INS INS	**SPD PRF** INS INS	**ALT ARM** INS INS
NAV CPLD INS INS	**APPR ARM** INS INS	**V NAV CPLD** INS INS	**ALT HOLD** INS INS
REV LOC INS INS	**APPR CPLD** INS INS	**GS CPLD** BNS INS	**GO AROUND** BNS INS

LEGEND:
BNS – Broken, Not Stretched
INS – Intact, Not Stretched

Figure 19 Autopilot lightbulbs

Filament - Intact, Not Stretched	Filament - Broken, Not Stretched
	

D.9 Annunciator Display System:

The annunciator display system was recovered from the wreckage at the accident site and was submitted to the NTSB Materials Laboratory for examination of the bulb filaments in the annunciator lights. There are two light bulbs present for each annunciator position.

The long, single row annunciator panel was received by the Materials Laboratory still attached to a piece of fire damaged cabin structure. The panel also had varying degrees of fire damage. Individual lights were removed from the panel or in the case of more damaged lights, sections were separated from the surrounding structure to facilitate radiographic examination. The annunciator lights were examined using x-ray in order to visualize the individual bulb filaments. The status of each filament is listed in Figure 20.

Figure 20 Results of filament analysis

SAS	**L ENG DE ICE**	**R ENG DE ICE**	**OIL DR L ENG**	**OIL DR R ENG**	**L ENG FIRE**	**R ENG FIRE**	**L ENG FUEL PRESS**
BNS BNS	INS BNS	BNS BNS	BNS BNS	BNS BNS	BNS BNS	BNS BNS	BNS BNS

ENG FUEL PRESS	**L ENG OIL PRESS**	**R ENG OIL PRESS**	**L ENG OIL TEMP**	**R ENG OIL TEMP**	**FLAP**
BNS BNS	BNS BNS	BNS BNS	BNS BNS	BNS BNS	BNS INS

L FIRE EXTNG INOP	**R FIRE EXTNG INOP**	**TRIM**	**INVERTER POWER**	**L GEN INOP**	**R GEN INOP**

BNS BNS	BNS BNS	BNS BNS	INS BNS	BNS BNS	BNS INS

BATTERY OVER TEMP	**ANN POWER**	**CABIN PRESS**	**CABIN ALT**	**CABIN DR UNSAFE**	**NOSE/BAG DR AJAR**
No Bulbs	INS INS	BNS BNS	INS BNS	INS BNS	INS INS

D.10 Garmin G600 Avionics Display System:

The G600 system is an integrated display system that presents primary flight instrumentation, navigation, and a moving map to the pilot through large-format displays (Figure 21[5]). In normal operating mode, the Primary Flight Display (PFD) presents graphical flight instrumentation (attitude, heading, airspeed, altitude, vertical speed), replacing the traditional flight instrument cluster. The Multi-Function Display (MFD) normally displays a full-color moving map with navigation information, as well as supplemental data.

Figure 21 Garmin G600 display system

[5] The image of the Garmin display system was copied from the Garmin g500/G600 Pilot's Guide document number 190-00601-20, revision L, dated July 2019.

D.11 Garmin GRS 77H

D.11.1 Description:

The Garmin GRS 77H is an Attitude Heading and Reference System (AHRS) designed to provide attitude and heading information to Garmin Integrated Flight Decks.

D.11.2 Recovery:

The GRS 77 AHRS unit, P/N 011-00868-10, S/N 42033548, was located and recovered during the wreckage examination on February 11th (Figure 22). Visual inspection of the unit found that it remained structurally connected to its flat plate "mounting rack" and its black plastic cover was damaged; it was fractured and partially missing on one side. Although the unit's black plastic cover was damaged, the unit appeared to be structurally intact. The unit was carefully moved by hand to determine if any of the internal components had broken loose. Based on this cursory check, none of the components appeared to be loose. The electrical connector visually appeared to be structurally intact; trace amounts of fine debris (ash, dirt etc..) was observed around and within the pin holes. The unit was placed in a shipping container and sent to the NTSB laboratory for further examination.

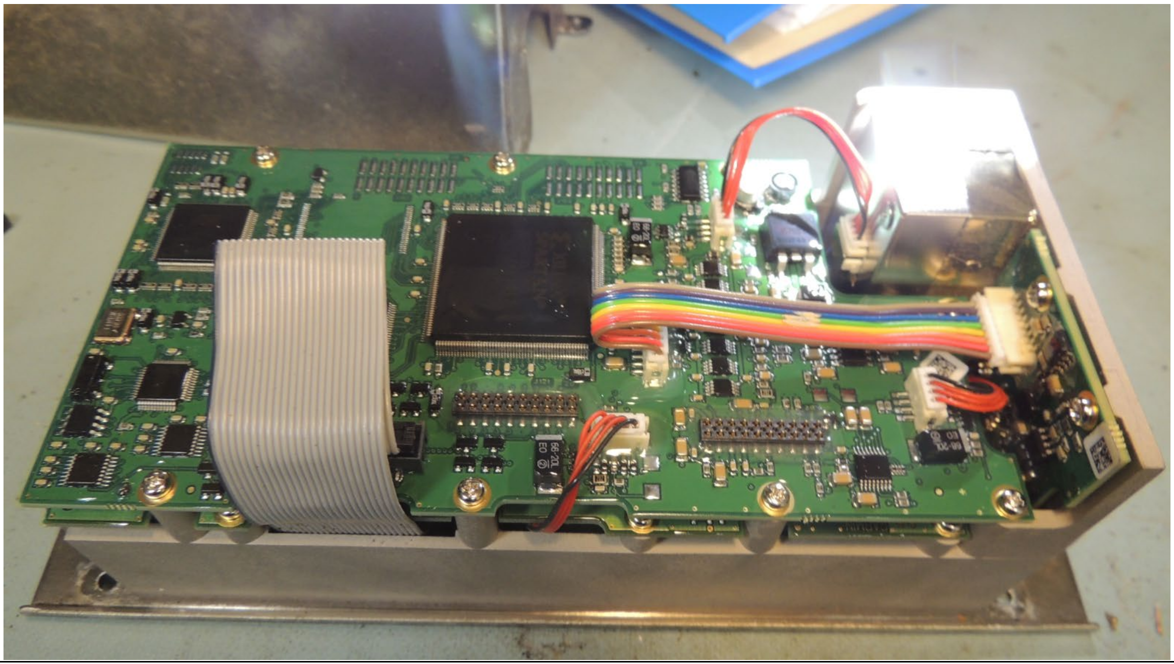
Figure 22 Photographs of the AHRS unit

View of the Data Label	View of damaged cover	Shown attached to mounting rack
		

D.11.3 Inspection:

On June 29, 2021, GRS 77 AHRS unit, P/N 011-00868-10, S/N 42033548, was submitted to the NTSB laboratory in Washington DC for an internal visual inspection by an Electronics Engineer to non-invasively check for any broken or damaged components that would prevent it from being electrically tested at its manufacturer. The unit's cover was removed, and the visual inspection did not identify any anomalies, see Figure 23. Other than the mounting plate and the cover, no additional items were removed or disconnected. After the inspection, the cover was replaced and the unit was returned to its shipping container.

Figure 23 AHRS with cover removed



D.11.4 Examination:

The GRS 77AHRS unit was shipped to Garmin, located in Olathe, Kansas, and was received by Garmin on February 15th, 2021. Prior to shipping, the AHRS unit was removed from the portion of mounting structure still attached to the unit. The structure was removed by a NTSB laboratory specialist.

The group met virtually at the Garmin facility in Olathe, Kansas on February 26, 2021 to examine the airplane's GRS 77 AHRS unit. The examination was conducted following protocols established by Garmin engineering, as follows.

D.11.5 Received Unit:

The AHRS was removed from its shipping container and was visually examined, and the following was noted:

a. The exterior was noted to be in fair physical condition (Figure 24 and 25). Some exterior areas were noted with missing areas of the plastic, protective cover, which the group concluded was a result of the accident impact sequence. The chassis case exhibited some minor damage but did not prevent the case from having been previously removed by the NTSB.

b. The unit's part number and serial number were verified.

c. The connector exhibited some debris but did not appear to be deformed.

d. The unit's base plate was bent in one corner; the unit may not be able to be installed onto the rotation table for testing.

e. The unit was rotated, and no internal loose parts were noted.

Figure 24 Photograph of the as received AHRS unit

Figure 25 Photograph of the side of the as received AHRS unit

D.11.6 Electrical Checks/Data Recovery:

The GRS 77 AHRS was then moved to a Garmin test bench. Garmin personnel then performed electrical continuity checks on the GRS 77.

The first continuity checks were accomplished at the unit level for electrical damage or any open/shorts. The connector power pins, connector pins to ground, unit and magnetometer power were checked for continuity. In each case, continuity was confirmed, the unit passed, and no internal shorts or other internal damage were determined.

Based on the electrical continuity tests, Garmin personnel indicated that the GRS 77 was fit to be connected to the Garmin bench test computer. The test bench test wire harness was connected to the GRS 77 for further electrical continuity tests. The tests were conducted without power applied to the GRS 77.

The continuity checks performed checked the ARINC 429 connection, aircraft power, configuration Module Data, Serial Busses, and the Automated Test Equipment (ATE) bus. In each case, Garmin personnel indicated the recorded values were within acceptable limits, and with no apparent damage to the internal circuitry.

Based on all the electrical continuity tests, Garmin personnel indicated that the GRS 77 was fit to have power applied from the Garmin bench test and continue the Garmin Automated Test.

Power was applied and the Garmin bench test computer was able to communicate with the GRS 77 via the ATE Bus. The GRS 77 System Software Version was recorded as 3.04. When the GRS 77's Boot Software version was requested, the unit did not respond with the information.

Garmin personnel then noted that the GRS 77 appeared to electrically reset (or reboot) every few seconds.

Garmin personnel were able to communicate with the GRS 77 and download the unit's Assert Logs, from non-volatile memory. An examination of the downloaded and decoded assert files, by Garmin personnel, indicated that no data related to the accident flight was noted in the data.

Following the examination, Garmin placed the GRS 77 back into its shipping box and returned the box to the NTSB.

Garmin, in an email message to the NTSB group chairman following the examination, indicated that the non-volatile information retrieved from the GRS 77 Assert Logs was not related to any aircraft flight. After further examination of the Assert Log information, Garmin concluded that the computer files had been recorded over during the bench testing and the records now contained non-sensical information.

D.11.7 Re-Inspection:

After receipt of the GRS 77 from Garmin, the unit was re-submitted to the NTSB Recorders Laboratory for another internal examination by the Electronics Engineer. The unit's cover was removed, and the internal components examined. After the top two circuit cards were removed for observation (Figure 26), the connector for the 24-pin ribbon cable (which ran between the top and bottom circuit cards) on the bottom circuit card was noted as, at least, partially disconnected (Figure 27). The ribbon connector was reattached and the unit re-assembled. No other anomalies were noted.

The NTSB also straightened the bent corner of the GRS 77.

Figure 26 AHRS top circuit boards removed

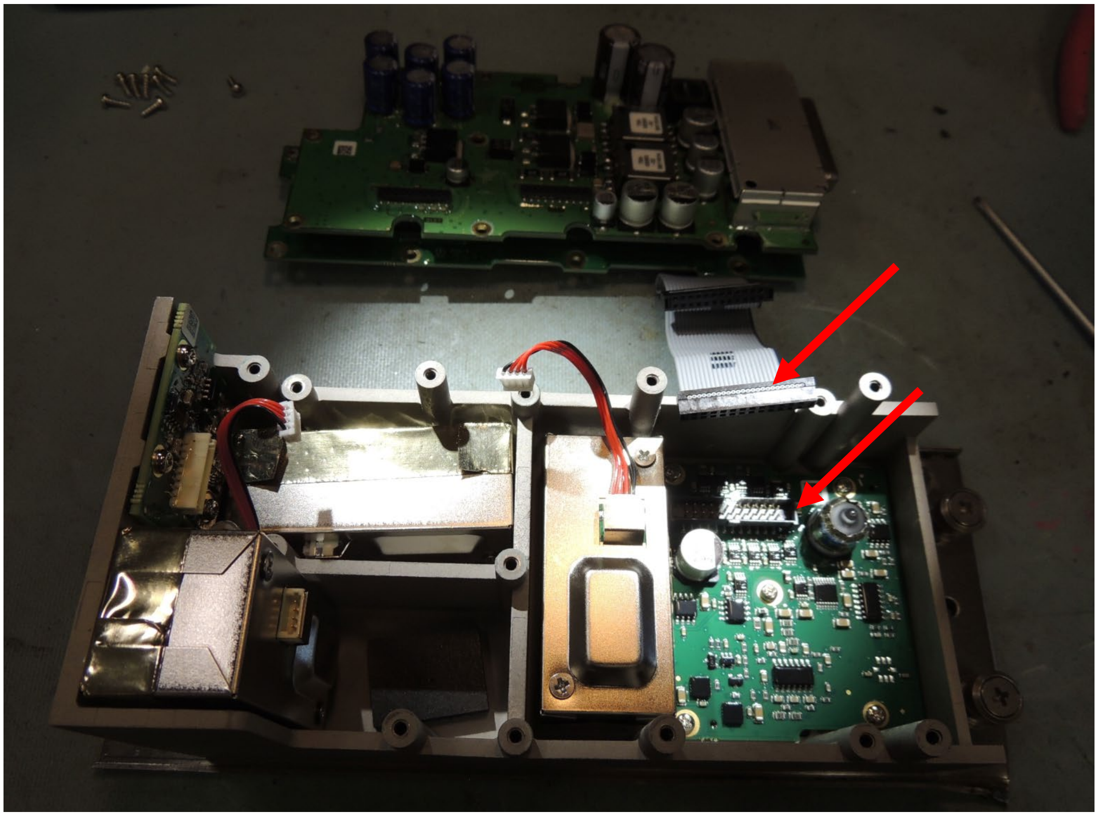


Figure 27 AHRS ribbon connector not connected