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

April 24, 2000

# AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT OF INVESTIGATION

A. ACCIDENT: NYC99MA178

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| LOCATION: | Vineyard Haven, Massachusetts |
| DATE: | July 16, 1999 |
| TIME: | 2141 eastern daylight time (EDT) |
| AIRCRAFT | Piper PA-32R-301; N9253N |

B. GROUP MEMBERS

| | |
|---|---|
| Chairman | Jeffrey B. Guzzetti<br>National Transportation Safety Board<br>Washington, D.C. |
| Member | Robert Hancock<br>National Transportation Safety Board<br>Parsippany, New Jersey |
| Member | William O. Herderich<br>Federal Aviation Administration<br>Aircraft Certification Office; Atlanta Georgia |
| Member | Eric West<br>Federal Aviation Administration<br>Flight Standards District Office; Boston, Massachusetts |
| Member | Paul Lehman<br>The New Piper Aircraft, Inc.<br>Vero Beach, Florida |
| Member | Phil Goettel<br>AlliedSignal Avionics<br>Olathe, Kansas |

## C. SUMMARY

On July 16, 1999, about 2141 eastern daylight time, a Piper PA-32-R301, Saratoga II, N9253N, was destroyed during a collision with water about 7 miles southwest of Gay Head, Martha's Vineyard, Massachusetts. The certificated private pilot and two passengers were fatally injured. Night visual meteorological conditions prevailed and no flight plan had been filed for the personal flight conducted under 14 CFR Part 91. The flight originated from Essex County Airport (CDW), Caldwell, New Jersey, and was destined for the Martha's Vineyard Airport, (MVY), Vineyard Haven, Massachusetts.

The Airworthiness Group was formed to document the accident site, airplane structure, airplane systems, and maintenance records. The group began examining the wreckage at Coast Guard Air Station Cape Cod at Otis Air Force Base, Massachusetts, on July 24, 1999, and concluded on July 26, 1999. Follow-up activities by the Group were as follows:

> Examination of avionics equipment at AlliedSignal Aerospace[1] in Olathe, Kansas, on July 29 - 30, 1999.
>
> Re-examination of airframe wreckage at Otis Air Force Base on August 1 - 2, 1999.
>
> Disassembly and examination of cockpit instruments, flight control components, and engine accessories at the NTSB Materials Laboratory in Washington, DC, on August 11 - 12, 1999.
>
> Examination of navigation/communication transceivers and autopilot servos at AlliedSignal Aerospace in Olathe, Kansas, on October 13 - 14, 1999.
>
> Ground and flight examinations and demonstrations related to the PA-32R-301 pitch trim and autopilot systems at The New Piper Aircraft, Inc., in Vero Beach, Florida, on December 3, 1999.

This factual report summarizes the Airworthiness Group's findings for all on-site and follow-up activities. Supporting documentation include photographs (appendix A), sketches of structural damage (appendix B), technical information regarding the automatic flight control system and global positioning system (appendix C), and pertinent copies of airframe and engine maintenance records and statements (appendix D).

[1] Formerly Bendix-King Avionics.

## D. DETAILS OF THE INVESTIGATION

### 1.0 Accident Site Documentation and Wreckage Distribution.

The airplane was found by U.S. Navy divers on July 20, 1999, at a depth of about 120 feet in the Atlantic Ocean. According to U.S. Navy personnel aboard the recovery ship *USS Grasp*, all of the recovered wreckage was distributed along an area that was about 120 feet in length and oriented along a magnetic bearing of about 010 / 190 degrees. The main cabin area was found in the middle of the wreckage distribution area at the following coordinates: North 41 degrees, 17 minutes, 37.2 seconds; West 070 degrees, 58 minutes, 39.2 seconds.

On July 21, 1999, the main cabin area was raised and placed on the *USS Grasp*. Five additional dives were made on July 22, 1999, and additional wreckage retrieved from these dives was also placed aboard the *USS Grasp*. On July 23, 1999, about 2100, the wreckage was transferred from the *USS Grasp* to trucks and brought to a hangar at Coast Guard Air Station Cape Cod located inside Otis Air Force Base, Massachusetts.

### 2.0 Structures

#### 2.1 Fuselage

About 75 percent of the fuselage structure was recovered (sketches of fuselage damage are included in Appendix B). A section of aft cabin roof from the aft and right side of the airplane (about 5 feet by 3.5 feet) had separated from the fuselage and was found in the main debris field. This section included the airframe-mounted hinge of the left side cargo door, a partial frame of the left side cabin door, and a piece of the right side cabin window. The left side of this section exhibited "accordion" crush damage in the aft direction and contained multiple folds that were about 5 inches in depth. Very little fuselage structure from the left or right side of the cabin area was recovered, except for a small piece of skin (about 2 feet by 2 feet) located beneath the left-side passenger window frame. The belly skin and floor structure of the fuselage was intact aft of the main spar box carry-through section. The recovered floor structure forward of this section was fragmented. Portions of five of the six seats were found inside the fuselage. The majority of the fuselage structure aft of the cabin area was recovered and found loosely attached to the fuselage.

The main spar box carry-through section was intact and slightly twisted. The upper main spar cap on the right side of the carry-through section was twisted aft relative to the lower main spar cap, and the left side of the carry-through section exhibited twisting in the opposite direction. The forward-facing web of the main spar box carry-through section exhibited a spanwise buckle that produced a bulge in the aft direction.

The engine was separated from the engine mount truss. The structural tubing on the right side of the engine mount truss was missing. The engine mount truss was deformed to the right and fractured in numerous locations. The upper left engine mount ear and both lower mount ears were fractured. The upper right engine mount ear was bent. The engine and propeller were retained by the Poweplants Group for additional examination.

## 2.2 Wings

### 2.2.1 Right Wing

About 60 percent of the right wing structure was recovered, including the entire span of the main spar. The right wing was separated into multiple pieces (sketches of wing damage are included in appendix B), and was more damaged than the left wing. The right wing main spar had separated into three pieces. The spar had fractured at its attachment to the main carry-through section. The fracture occurred at the outermost row of bolts splicing the wing spar to the carry-through section. The upper and lower spar cap fractures exhibited tension on their forward edges and compression on their aft edges. The spar web exhibited aft bending and tearing in this area. Another upper spar cap fracture was found about 26 inches outboard of this fracture. The remaining span of the right outboard main spar was bent about 90 degrees beginning at this point, producing an "L" shape. Compression was noted on the forward side of the upper spar at this bend, and tension was noted on the aft side of it.

Immediately outboard of the 90-degree bend, black rubber transfer marks were found on the aft face of the spar web, and a fracture of the upper spar cap was found. The fracture surface exhibited evidence of forward bending. There was also a puncture mark in the spar web adjacent to this area that protruded forward. The main landing gear attach fittings were found fractured in this area, and the spar web was bowed forward in the vicinity of where the main landing gear wheel assembly would normally reside in its retracted position.

The next complete spar separation fracture (continuing outboard) was found 133 inches from the main spar carry-through spar fracture. The upper and lower spar caps at this fracture exhibited tension on their forward edges and compression on their aft edges. A portion of the web in this area was folded aft.

No evidence of upward spar bending damage was found, nor was any evidence of metal fatigue found in any of the fracture surfaces.

The outboard portion of the wing leading edge exhibited rearward accordion crush damage and was separated from the remainder of the wing. The majority of the upper and lower wing skin located along the inboard 83 inches of wing span between the main spar and trailing edge was recovered.

The entire span of the right flap was recovered; it had separated into two sections (chordwise fracture), and both sections had separated from the right wing. Both flap sections did not exhibit any bowing, bulging, or planar deformation. The underside of the inboard flap section exhibited wrinkling and blue/maroon paint transfer.[2] About 33 inches of the right aileron was recovered, and the leading edge of this section exhibited rearward crush deformation.

### 2.2.2 Left Wing

About 80 percent of the left wing structure was recovered, including the entire span of the main spar. The left wing exhibited less deformation than the right wing. The left wing main spar had separated into several pieces. The wing spar was fractured near the left edge of the main spar carry-through section. The upper and lower spar cap fractures in this area exhibited tension on their forward edges, and compression on their aft edges. The spar web also exhibited aft bending and tearing in this area.

Outboard of this fracture, black rubber transfer marks were found on the aft web of the main spar as well as the fractures of the upper and lower spar cap. The fracture surfaces exhibited evidence of forward bending. There was also a puncture mark in the spar web that protruded forward. The main landing gear attach fittings were fractured in this area, and the spar web was bowed up and down in the vicinity of where the main landing gear wheel assembly resides in its retracted position. The wing section outboard of these fractures exhibited uniform aft crush deformation. The main spar was bent aft and converged (beginning inboard and converging outboard) with the trailing edge cove along the outboard half of the wing.

No evidence of upward main spar bending damage was found, nor was any evidence of metal fatigue found in any of the fracture surfaces.

About 90 percent of the upper and lower wing skin between the main spar and trailing edge cove was recovered. The upper skin in the vicinity of the left wing tip was flattened out. The leading edge skin in the vicinity of the inboard portion of the left wing, near the stall warning port, exhibited uniform hydrodynamic deformation in the aft direction.

A 27-inch inboard section of the wing flap section was recovered and the leading edge of this section exhibited aft accordion crush damage. The flap section did not exhibit any bowing, bulging, or planar deformation. The entire span of the left aileron was recovered. It had separated into two pieces. The outboard section of the aileron was curled downward. The top of the aileron exhibited paint chipping and wrinkling, and the bottom side did not exhibit as much wrinkling and had no paint chipping.

[2] The aircraft identification lettering on the sides of the fuselage consisted of blue paint. The blue hues were identical to the blue marks found on this section of flap.

### 2.3 Empennage

The vertical stabilizer and rudder was separated from the aft fuselage. The horizontal stabilator was separated from the aft fuselage attach points and had fractured into five pieces. Two of the pieces consisted of left and right outboard sections (about 22 inches in length) and exhibited symmetrical aft crush marks that were semi-circular with diameters of about 5 inches. The fracture surfaces of the left side outboard section of the stabilator exhibited tearing in the aft direction. The fracture surfaces of the right outboard surfaces of the stabilator exhibited forward and upward tearing. The left inboard section of stabilator was more intact than the right inboard section. The leading edge of the right stabilator section exhibited rearward uniform crush damage along its entire leading edge.

The lower portion of the rudder was separated from the vertical stabilizer (fin) structure and remained attached to the torque tube bellcrank assembly and fin aft spar. The rudder was folded over toward the right side of the airplane. The vertical stabilizer was also twisted, bent and curled around toward the right. The structure surrounding the dorsal fin area was deformed symmetrically upward.

### 2.4 Landing Gear

All three landing gear assemblies had separated from the airframe. The retraction/extension actuating cylinders associated with the nose gear and the left main gear were found in the fully retracted position. The retraction/extension actuating cylinder for the right main gear was not recovered.

## 3.0 <u>Systems</u>

### 3.1 Flight Controls

#### 3.1.1 Ailerons

An examination of the aileron control cable circuit and associated hardware did not reveal any evidence of a preexisting jam or failure. Flight control cable continuity for the entire right aileron control circuit, including the entire balance cable that links the right aileron to the left aileron, was verified. The control cable continuity for the left aileron could not be verified due to impact damage and fragmentation. All of the ends of the separations of the aileron control cable circuit exhibited evidence of tensile overload. The stops for the ailerons were also examined and no evidence of severe repetitive strike marks or deformations were noted.

#### 3.1.2 Stabilator.

An examination of the stabilator control cable circuit and associated hardware did not reveal any evidence of a preexisting jam or failure. Flight control cable continuity for the stabilator was established from the control surfaces to the cockpit controls. The stabilator balance weight had separated from the stabilator and the fractures associated with the separation were consistent with tensile overload. The stops for the stabilator were examined; no evidence of severe repetitive strike marks or deformations were noted.

#### 3.1.3 Stabilator Trim

An examination of the stabilator trim control cable circuit and associated hardware did not reveal any evidence of a preexisting jam or failure. The control cable circuit for the stabilator trim was documented from the control surfaces to the cockpit area. An examination of the stabilator trim barrel jackscrew revealed that one full thread was protruding out of the upper portion of the trim barrel assembly housing.[3] The barrel assembly was free to rotate and had the trim control cable wrapped around it. The cable ends were separated about 41 inches and 37 inches respectively from the barrel assembly winding. Examination of the separations revealed evidence that was consistent with tensile overload.

According to technical specifications provided by The New Piper Aircraft, Inc., the following parameters apply to the stabilator trim tab positions:

0 exposed threads (at the stop pin) = maximum airplane nose down trim
5 exposed threads = neutral trim
16 exposed threads = maximum airplane nose up trim

Range of deflection from neutral position to maximum nose-down (stabilator trim tab up) position = 2 ½ degrees

Range of deflection from neutral position to maximum nose-up (stabilator trim tab down) position = 10 degrees

Subsequent ground and flight examinations and demonstration of similar PA-32R-301 airplanes revealed that the nominal (trimmed) stabilator trim barrel jackscrew position for the following conditions is about one exposed thread: 160-knot descent at 700 feet per minute, gross weight 3,500 pounds, 89.5 inches aft of datum, and pressure altitude of 5,500 feet at top of descent.

#### 3.1.4 Rudder

An examination of the rudder control cable circuit and associated hardware did not reveal any evidence of a preexisting jam or failure. Flight control cable continuity for the rudder was established from the control surfaces to the cockpit controls. The stops for

[3] The jackscrew must translate an additional ¼-inch of travel beyond its final thread into a non-threaded portion until it reaches the full nose-down trim stop.

the rudder were examined; no evidence of severe repetitive strike marks or deformations were noted.

#### 3.1.5 Wing Flaps

The electrically-driven wing flap jackscrew actuator was not recovered. The flap switch in the cockpit was destroyed. The leading edge of the left inboard flap section exhibited aft accordion crush damage and did not exhibit any bowing, bulging, or planar deformation. Both recovered sections of the right flap did not exhibit any bowing, bulging, or planar deformation.

#### 3.1.6 Cockpit Flight Controls

The left-side control yoke grips were both broken near the center of the yoke where it connected to the column. The autopilot trim disconnect switch, control wheel steering switch, and microphone switch were intact on the left grip of the left-side control yoke. The pitch trim switch, also mounted on this grip, was broken and missing both buttons. The right control yoke grips were intact and attached to the column.

The left and right rudder pedals of the left seat position were separated from each other, but each pedal was intact. The left and right rudder pedals of the right seat position were disintegrated.

### 3.2 Automatic Flight Control System

#### 3.2.1 System Description and Operation

The airplane was equipped with a Bendix/King 150 Series Automatic Flight Control System (AFCS), which was approved for use in Piper PA-32R-301 model airplanes by the FAA on November 1, 1982. The AFCS provided two axis control for pitch and roll. It also had an electric pitch trim system which provided autotrim during autopilot operation, and manual electric trim for the pilot. (Additional technical specifications and operator information is attached as appendix C.)

The AFCS consisted of a computer/controller/annuniciator unit, roll servo, pitch servo, pitch trim servo, slaved directional gyro, flux detector, pictorial navigation indicator[4], and flight command indicator[5]. According to AlliedSignal, the flight computer

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[4] The pictorial navigation indicator is a primary cockpit instrument that provides a pictorial presentation of aircraft displacement relative to VOR radials, localizers, and glide slope beams. It also gives heading reference with respect to magnetic north. It is also commonly referred to as an horizontal situation indicator (HSI).

[5] The flight command indicator is a primary cockpit instrument that displays aircraft attitude and computed pitch and roll steering commands. It is also commonly referred to as an attitude indicator.

is designed to compute pitch and roll steering commands, which are routed through the flight command indicator where they are displayed on the flight command indicator as visual guidance commands. The steering commands are fed to the autopilot computation circuits contained in the flight computer which generates the commands for the individual servos to manipulate the ailerons, elevator, and elevator trim. Using the same pitch and roll commands for flight director and autopilot provides flight director steering command and autopilot control. The autopilot converts the pitch and roll steering commands from the flight computer displayed on the flight command indicator to the required elevator and aileron position commands via the servos.

The AFCS installed on the accident airplane was capable of allowing the pilot to select altitude hold, which could allow the airplane to maintain the altitude that it was at during the moment that the altitude hold was selected. However, the AFCS did not have the option of allowing the pilot to "preselect" an altitude so that the autopilot could fly to and maintain the preselected altitude as it climbed or descended from another altitude.

The AFCS also had a vertical trim rocker switch installed so that the pilot could change the airplane's pitch up or down without disconnecting the autopilot. The rocker switch could allow the pilot to make small corrections in the selected altitude while in the altitude hold mode, or allow the pitch attitude to be adjusted at a rate of about 0.9 degrees per second when not in altitude hold.[6]

The AFCS also incorporated a flight director, which must be activated before the autopilot will engage. Once activated, the flight director could provide commands in the flight command indicator to maintain wings level and the pitch attitude existing at the time of engagement. To satisfy the command, the pilot could have manually flown the airplane by reference to the guidance received in the flight command indicator, or the pilot could have engaged the autopilot and let it satisfy the commands by maneuvering the aircraft in a similar manner via the autopilot servos.

The AFCS also incorporated a "navigation" mode that could provide guidance to the pilot, or the autopilot, in intercepting and tracking VOR or global positioning system (GPS)[7] courses. While engaged in this mode, the AFSC could receive input signals from either the selected VOR frequency and course, or from GPS course data selected for presentation on the pictorial navigation indicator. The flight command indicator could then command the required bank to maintain the selected VOR or GPS course with automatic crosswind compensation, and the autopilot, if engaged, would have been able to satisfy those commands.

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[6] According to AlliedSignal, an airplane will typically overshoot a desired altitude by about 15 percent of the vertical descent rate if a pilot selects the altitude hold at the desired altitude as the airplane descends through the desired altitude. For example, if an airplane is descending at 500 feet per minute, and the pilot selects altitude hold as the airplane passes through 2,500 feet, the airplane will continue to about 2,425 feet before the autopilot commands the airplane back up to 2,500 feet.

[7] A complete description of the GPS is contained in Section 3.9 of this report.

The AFCS also incorporated a "heading select" mode, allowing the pilot to select a heading by moving a "bug" on the outer ring of the pictorial navigation indicator. Once the bug was moved to the desired heading with the heading select button engaged, the autopilot could command the airplane to that heading at a bank angle of about 22 degrees.

The AFCS also had an control wheel steering (CWS) button mounted on the control yoke to allow the pilot to maneuver the aircraft in pitch and roll without disengaging the autopilot. According to AlliedSignal, after the CWS button is released, the autopilot resumes control of the aircraft at the previously selected heading and altitude that prevailed at the time the CWS button was released.

According to the FAA and Bendix/King, the trim system is designed to withstand any single inflight malfunction. Trim faults are visually and aurally annunciated in the cockpit. Through the use of monitor circuits, aircraft control is automatically returned to the pilot when a fault is detected. Internal monitors continuously check for the presence of operating pitch and roll microprocessors, adapter modules, pitch and roll reference signals, proper internal voltages, trim power, "runaway" auto trim, wrong direction of trim, abnormal pitch attitude rates, and abnormal roll attitude rates.

After the AFCS has been preflight tested, it can be engaged and disengaged by the pilot, or automatically.[8]

### 3.2.2 Autopilot Flight Computer/Controller

The autopilot flight computer/controller/annunicator (Bendix/King model KC-192; p/n 065-004202; s/n X41353) was examined. The adapter module was loose on one end and crushed on the opposite end. The face plate was intact. Corrosion and silt were noted inside the unit. Microscopic examination of each annunciator light bulb on the front face of the unit was performed at the Safety Board's Materials Laboratory Division. None of the bulbs exhibited evidence of filament stretch, including the autopilot engage, flight director, or trim failure light bulbs.[9]

### 3.2.3 Autopilot Servos

#### 3.2.3.1 Pitch Trim Servo.

The pitch trim servo (Bendix/King model no. KS-179; p/n 065-0052-06, s/n 4284) was intact and exhibited less impact damage than the other two autopilot servos. The servo remained attached to airframe structure. Its cover and cable guards were intact. The mounting stud for the electric motor was separated. Corrosion was noted throughout

[8] The following conditions will cause the autopilot to automatically disengage: 1.) power failure; 2.) internal flight control system failure; 3.) loss of a valid compass signal; 4.) roll rates in excess of 14 degrees per second; and 5.) pitch rates in excess of 8 degrees per second.
[9] These bulbs were examined by the Chief of the NTSB Materials Laboratory Division and his staff.

the unit. The engage solenoid plunger and relay both engaged and disengaged in the autopilot mode and the manual mode when electrical power was applied. A voltmeter was placed on both leads of the servo motor; the voltmeter indicated minus 8.28 volts in one direction and positive 8.2 volts in opposite direction, indicating that the servo drive circuitry was functioning. The motor initially would not rotate in the as-found condition. After the application of a greaseless lubricant, the motor was able to operate in both directions in manual mode. The motor initially would not operate under autopilot mode (which utilizes less voltage), but then moved clockwise after it worked free from binding. The manual trim voltage check indicated 12.82 volts (specifications call for greater than 11 volts and less than 14 volts). The servo clutch was then removed and placed in a fixture. The running torque was measured to be 19 in-lbs. (specifications call for 21 in-lbs. plus/minus 2 in-lbs.).

3.2.3.2 Pitch Servo.

The pitch servo (Bendix/King model no. KS-177) exhibited greater impact damage than the pitch trim servo. The servo's cover and connector plug were missing. Electrical wires emanating from the trim switches were separated. The servo's electric motor was also found separated and hanging by six wires. The engage solenoid plunger was missing. No evidence of gouging or scratching was noted in the engage solenoid plunger receptacle. The capstan could be manually rotated in either direction (after a greaseless lubricant was applied) with noticeable binding. Severe corrosion was found in the gear train of the motor. The motor could be rotated when 3 volts of electrical power was applied to it. The motor also rotated in the opposite direction under reverse polarity. Both of the trim sensing switches were able to be manually opened and closed. Continuity of one of the trim sensing switches was verified with an ohmmeter. Continuity of the other trim sensing switch could not be verified. Initial attempts to manually move the servo's clutch failed. The clutch was subsequently removed from unit and placed in a fixture. After the clutch was broken free, its breakout and running torque was measured. Breakout torque was measured to be about 55 in-lbs., and running torque was about 51 in-lbs. in both directions (specifications call for 55 in-lbs. plus or minus 5 in-lbs.).

3.2.3.3 Roll Servo

The roll servo (Bendix-King model no. KS-178) exhibited greater impact damage than the pitch and pitch trim servos. The servo mounting was fractured, and the servo had separated from the airframe structure. The roll servo cover, connector plug, and tach generator were missing. Some transistor and power connections were separated. Examination of the circuit boards revealed numerous broken components and severe corrosion. The motor had separated from the servo and remained attached by four wires. The engage solenoid plunger was missing. No evidence of gouging or scratching was noted in the engage solenoid plunger receptacle. The capstan could be manually rotated in either direction with some binding noted following the application of a greaseless lubricant. Minor surface corrosion was noted throughout the unit. Severe corrosion was found throughout the gear train and initial attempts to manually rotate the motor failed. After the gear train was removed from the motor and more greaseless lubricant applied, the motor was able to be manually rotated with significant binding noted. The motor would not rotate under electrical power until the voltage was increased to 30 volts. Initial attempts to manually rotate the servo clutch failed. The clutch was then removed from the unit and placed it in a fixture; corrosion was evident on the exterior of the unit. After the clutch was broken free, the break-out and running torques were measured. Breakout torque was measured to be about 20 in-lbs., and running torque was about 19 in-lbs. in both directions (specifications for running torque are 25 in-lbs. plus or minus 2 in-lbs.).

### 3.3 Engine Controls and Switches

The throttle control was found in the full forward position (maximum power setting). The propeller control was found in the full forward position (maximum rpm setting). The mixture control was broken. The alternate air control was found in the CLOSED position. The key in the magneto switch was found in the BOTH position.

### 3.4 Engine Indicating

The tachometer needle was intact and fixed in place; it pointed to 2,750 rpm (red line, or maximum limit, begins at 2,700 rpm). The hour register inside the tachometer read 0663.5 hours. The manifold pressure gage needle was fixed in place and indicated 27 inches Hg. The fuel flow gage needle was slightly loose and indicated 22 gallons per hour (normal operating range is zero to 34.9 gallons per hour). The exhaust gas temperature gage needle was loose and read 1,000 degrees F. The oil temperature gauge was fixed and read 150 degree F (normal operating range is 100 degrees F to 245 degrees F). The oil pressure gage was fixed and indicated about 17 psi (normal operating range is 55 psi to 95 psi). The cylinder temperature gage needle was missing. The fuel quantity gages were obliterated.

### 3.5 Flight Instrumentation

The airspeed indicator was damaged and its glass faceplate was missing. The needle position was found off scale near the right edge of the density altitude adjustment window; it could be moved but was spring-loaded to its as-found position. Magnified examination of marks on the instrument face revealed an outline that was similar to the size and shape of the needle. This mark was located about two needle widths above the 210-knot marking, which is the maximum marking on the indicator.[10]

The vertical speed indicator needle was missing. Magnified examination of marks on the instrument face revealed an outline that was similar to the size and shape

[10] The location of the needle mark on the accident airspeed indicator is consistent with the maximum mechanical needle travel position for the airspeed indicator design. This was shown by testing a similar undamaged airspeed indicator.

of a needle. This needle mark was pointed at the maximum down limit position of 2,000 feet per minute descent.

The altimeter needle was fixed and indicated 270 feet. The altimeter setting read 30.09 inches of mercury (Hg).

The flight command indicator (Bendix/King model KI-256) was deformed and its glass faceplate was missing. The center portion of the pictorial display was partially embedded in the side of the housing in a position that indicated a right turn with a bank angle of about 125 degrees and a nose-down pitch attitude of about 30 degrees. The background portion of the pictorial display was free to move and in a position consistent with the center portion of the display. The outside bank angle ring was loose and separated. The air driven gyro housing inside of the flight command indicator was corroded and was not deformed. Disassembly and inspection of the gyro did not reveal any score marks on the spinning mass gyro and mating housing.

The turn coordinator was deformed and its glass was missing. The display was captured in a position indicating a right turn at an extreme rate. Cracks were noted in the glass covering the yaw indication groove. The cracks emanated from a common point which formed a slight depression in the front of the groove about one-third travel to the right of center. The size of the depressed area in the tube was similar to the diameter of the ball which normally rides in this groove. The electrically-driven gyro assembly inside of the instrument was removed and found free to rotate with no binding or case interference. No score marks were found on the spinning gyro mass and mating gyro housing.

The pictorial navigation indicator (Bendix/King KI-525A) was deformed and its glass was missing. The heading indicator was pointing to 339 degrees. The center navigational display needle was oriented along the 300-120 degree bearing. The heading flag was displayed. The heading bug was located at the 95-degree mark. Internal components exhibited impact deformation and corrosion. The slaved gyro assembly was partially separated from its mounting, and its case exhibited minor deformation. Disassembly and examination of the gyro revealed that its internal components were corroded. The gyro housing and internal rotor were disassembled and individually examined. The interior surface of the case and the exterior surface of the spinning mass rotor did not exhibit any deformation, impact marks, or rotational scoring.

### 3.6 Air/Pneumatics

The engine-driven vacuum pump (Airborne s/n 12AK6853) was examined. Its drive shear shaft was intact. The drive end was removed by investigators to expose the internal rotor and vanes. The rotor showed several cracks between the bottom of the vane slots and the center of the rotor that were consistent with impact damage. All six vanes were removed intact. The rotor was removed in several pieces and the housing was examined. No evidence of scoring or rotational damage was noted. A metal straight edge was placed along the long ends of each vane and no warping or wear was noted.

The electrically-driven vacuum pump (Airborne s/n 4AK20) was examined. Its drive shear shaft was found intact. The pump was opened by investigators from the motor drive end, exposing the rotor and internal vanes. Several cracks were noted in the rotor between the vane slots and the center shaft area. Five of the six vanes were removed and found intact with no fractures or edge chipping. Approximately one-half of the rotor was removed and an examination of its housing revealed no evidence of scoring or rotational damage. A metal straight edge was placed along the long ends of the removed vanes and no warping or wear was noted.

Disassembly and examination of the vacuum system filter did not reveal any evidence of contaminants or blockages.

### 3.7 Electrical

Examination of all recovered electrical wiring and components did not reveal any evidence of arcing or fire. The circuit breaker panel was deformed and impact damaged. All of the breakers were in the tripped position except for the flap, transceiver, and DME.

### 3.8 Fuel

The fuel selector valve was recovered and examined. The bottom of the valve was missing. All three fuel line connections were broken off. The valve had separated from the fuselage attach points and was impact damaged. The selector valve linkage was deformed, and the valve itself was found in the OFF position.

A liquid that had a similar color, odor, and texture as 100 low lead aviation fuel was found in the fuel selector valve sump.

The electrically-driven fuel boost pump was removed and functionally tested with water by investigators. The test revealed that the pump was able to operate when electrical power was applied to it.

### 3.9 Navigation and Communication

Both of the airplane's communication/navigation transceivers received severe impact damage and could not be powered up. The non-volatile memory circuit chips were extracted from the transceivers, placed in a test unit, and were powered up. The following information was noted:

Transceiver No. 1 (Bendix/King KX-165; p/n 069-1025-21; s/n 105256):

Face Plate damaged and missing about 75 percent of its structure.
ON / OFF switch missing.

In-use com frequency - 132.02 (Teterboro ATIS)
Standby com frequency - 135.25 (Unknown)[11]

In-use nav frequency - 109.80 (New Haven VOR)
Standby nav frequency - 113.10 (La Guardia VOR)

Transceiver No. 2 (Bendix/King model KX-165; p/n 069-1025-21; s/n 150293):

Face Plate Intact
ON / OFF switch found in the ON position

In-use com frequency - 121.40 (Martha's Vineyard Tower)
Standby com frequency - 127.25 (Unknown) [12]

In-use nav frequency - 108.80 (Bridgeport VOR)
Standby nav frequency - 110.00 (Norwich VOR)

The top of the VOR indicator heading card was found at the 097 degree bearing.

The accident airplane was equipped with a global position system (GPS) receiver, Bendix/King model KLN-90B (p/n 066-04031-1121-R24, s/n 20209). (Technical information regarding the GPS is attached as appendix C.) The KLN-90B consisted of a panel-mounted unit containing the GPS sensor, navigation computer, cathode ray tube display, data base, and all controls required to operate the unit. The unit was also wired to four remote switches/annunicators mounted directly in front of the pilot on the top portion of the instrument panel.[13] The GPS was capable of presenting moving map displays, bearings and distances to programmable destinations such as airports and waypoints, airport information, groundspeed, and other information. It was also capable of interfacing with the AFCS and the pictorial navigation indicator.

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[11] Essex County ATIS is 135.5 MHz

[12] Martha's Vineyard ATIS is 126.25 MHz

[13] The switches/annunciators included the following (mounted left to right): 1.) NAV/GPS; 2.) GPS APR - ARM/ACT; 3.) GPS CRS - OBS/LEG; and MSG/WPT (anniciator only). A complete description of these switches/ annunciators can be found in the FAA approved flight manual supplement for the PA-32R with KLN90B GPS navigation system.

The GPS unit was crushed vertically. The display in the front face of the unit was destroyed. The on/off switch was found in the ON position. The navigation database indicated that it was effective on October 8, 1998, and expired on November 4, 1998. The wire connecting the circuitry to the lithium battery was separated and the separated end exhibited corrosion. According to AlliedSignal, the lithium battery provides electrical power to retain the volatile memory of the GPS receiver. The battery voltage was measured to be 0.2 volts and the memory was not retrieved.[14]

The automatic direction finder (ADF) receiver (Bendix/King KR-87; p/n 066-1072-04; s/n 62665) was examined. The on/off switch was found in the ON position. The non-volatile memory circuit chips were extracted from the transceivers, placed in a test unit, and were powered up. The following frequencies were displayed[15]:

Primary Frequency 400 Khz
Secondary Frequency 200 Khz

The head of the ADF needle was pointing to the 356 degrees.

The audio panel (Bendix/King model KMA-24; p/n 066-1055-03; s/n 100909) was examined and the following as-received switch positions were noted: The transmit selector was selected to the "Comm 1" position. The speaker audio select button for the telephone position was found in the depressed (activated) position; all other speaker audio select buttons were in the deactivated position. The headphone audio select button for the DME was found in the depressed (activated) position; all other headphone audio selector buttons were in the deactivated position.

3.10 Annunciator Lights

An examination of all recovered light bulbs from the airplane's main annunciator panel[16] and landing gear annunciator panel[17] was performed at the Safety Board's Materials Laboratory Division in Washington, DC. The examination revealed that none of the filaments exhibited stretching.

[14] According to AlliedSignal, the nominal battery voltage is 3.6 volts, and the battery must supply at least 2.5 volts to maintain the non-volatile memory.

[15] These frequencies are not similar to any ADF navigation facilities along the route of the accident flight.

[16] This panel contained an annunciator for each of the following warnings: Alternator Inoperative; Landing Gear Warning; Starter Engaged; Oil Pressure Low; Vacuum Inoperative; Air Conditioner Door; Flaps; Baggage Door Ajar; Low Bus Voltage.

[17] This panel contains an annunciator for the left main landing gear, right main landing gear, and nose gear. When the annunicator is lit, it indicates the gear is down and locked.

### 3.11 Seats and Restraint Systems

#### 3.11.1 Seat Structure

The airplane was equipped with six seats. The seats were configured in "club style" arrangement, with two forward-facing seats in row 1 (including the pilot's seat), two aft-facing seats in row 2, and two front-facing seats in row 3. Portions of five of the six seats were recovered,[18] and all of them had separated from the floor structure. An examination of the aluminum backs of both aft-facing seats revealed that they were deformed (bulged) toward the direction of flight.

#### 3.11.2 Seat Restraint System

The left and right front seats were equipped with lap belts and shoulder harnesses. None of the belts for these seats could be identified in the wreckage. The four seats in row 2 and row 3 were also equipped with lap belts and shoulder harnesses. Both sections of the lap belt for the left side aft-facing seat were found and exhibited evidence of stretching. The inboard section of lap belt for the right side aft-facing seat in row 2 had been cleanly cut about 3 inches from the male-end of the latch, and the outboard section of lap belt for this seat exhibited evidence of stretching. All of the lap belt sections for the third row of seats were identified and none exhibited evidence of stretching. The shoulder harnesses for the rear seats could not be identified in the wreckage.

### 3.12 Emergency Beacon

The airplane was equipped with an emergency locator transmitter (ELT) that was manufactured by Artex Aircraft Supplies, Inc. (model no. ELT 110-4; p/n 453-0150; s/n 54710) and conforms to the FAA Technical Standard Order (TSO) C91a type design. The ELT was found intact and attached to its mounting bracket. The ELT's antenna coaxial cable was separated between the ELT and its antenna. The ELT master switch, mounted on the ELT itself, was found in the "OFF" position, which is its normal position for installation and automatic operation. The ELT could also be controlled by a cockpit control switch which was not located in the wreckage. The ELT battery expiration date was marked as "June / 01".

The ELT failed to produce a signal when the master switch was placed in the "ON" position after the accident. A subsequent measurement of battery voltage indicated 1.6 volts.

[18] One seat from row 3 was not found, and the position (left or right side) of the other seat from row 3 could not be determined.

## 4.0 Maintenance Records

No airplane maintenance records were recovered. Records and a statement regarding recent maintenance and refueling on the airplane from fixed based operators were collected (copies of records attached as Appendix D). According to the records and statement, the last annual inspection was June 28, 1999, at a tach/hobbs of 622.8 hours. The last recent maintenance event recorded in the records was dated July 13, 1999; this entry did not include a hobbs or tach time and indicated that the compass and pictorial navigation indicator were adjusted and a microphone bracket was repaired. (A statement from the manager of the fixed base operator that last performed maintenance on the accident airplane is attached in Appendix D). Records addressing the overhaul of the engine were also collected and reviewed by investigators (pertinent copies attached in Appendix D).

[redacted] 4/24/2000
Jeffrey B. Guzzetti
Aerospace Engineer
[redacted]
4/25/00