National Transportation Safety Board

Office of Aviation Safety

Washington, DC 20594

WPR23FA034

STRUCTURES

Group Chair's Factual Report

August 22, 2024

A. ACCIDENT

Location: Snohomish, Washington
Date: November 18, 2022
Time: 1019 Pacific Standard Time (PST)
1819 Coordinated Universal Time (UTC)
Airplane: Textron Aviation Inc. 208B EX, N2069B

B. STRUCTURES GROUP

Group Chair	Clinton R. Crookshanks National Transportation Safety Board Denver, Colorado
Group Member	John Baldessari Textron Aviation, Inc. Wichita, Kansas

C. DETAILS OF THE INVESTIGATION

The airplane wreckage was recovered to the AvTech facilities in Auburn, WA, after the accident. The group examined the airplane December 20-21, 2022.

The right wing, most of the horizontal stabilizer and both elevators separated from the airplane during the accident and were found separate from the main wreckage that included the remainder of the airplane. The main wreckage sustained fire damage. The right wing with the strut attached was recovered about 580 feet north of the main wreckage. The horizontal stabilizer, both elevators, and several pieces of right flap were scattered up to about 1830 feet west of the main wreckage.

D. FACTUAL INFORMATION

1.0 Airplane Overview

The Textron Aviation Inc. (formerly Cessna) 208B EX Caravan is a single engine, propeller driven, single pilot, high wing airplane originally designed as the 208 in the early 1980's (Figure 1). The 208B incorporates a fuselage extended by 4 feet and was certified as an 11-seat passenger airplane in 1989. The 208B EX has a more powerful engine and was certified in 2012. The high wing airplane is equipped with wing struts, a conventional tail, fixed tricycle landing gear, and an underbelly cargo pod. The airplane is 41 feet, 7 inches long, 14 feet, 10 inches high at the tail and has a wingspan of 52 feet, 1 inch. The airplane is powered by a single Pratt & Whitney Canada PT6A-140 turboprop engine driving a McCauley 4-blade constant speed, full feathering, reversible pitch propeller. The accident airplane was serial number 208B5657 and was built in 2021. The airplane station diagram is shown in Figure 2 for reference.

On November 1, 2022, AeroAcoustics Aircraft Systems Supplemental Type Certificate (STC) SA01213SE, Aircraft Payload Extender III, was installed on the airplane. The STC adds wing stall fences to both wings outboard of the landing lights, installs new main landing gear axles, and installs 29” main landing gear tires to increase the maximum takeoff weight to 9,062 lb, increase the maximum landing weight to 9,000 lb, and increases the payload and range. On November 11, 2022, AeroAcoustics Aircraft Systems STC SA01805SE, Aircraft Payload Extender STOL, was installed on the airplane. The STC installs a scalloped Gurney flap on the trailing edge of each flap to improve the low-speed aerodynamics of the wing.

On November 14, 2022, the FAA issued a Special Airworthiness Certificate in the Experimental category for the purpose of Research and Development for the airplane. The certificate was requested by Raisbeck Engineering to perform company flight testing for the development of an STC for the airplane. The airplane was undergoing baseline flight testing at the time of the accident.

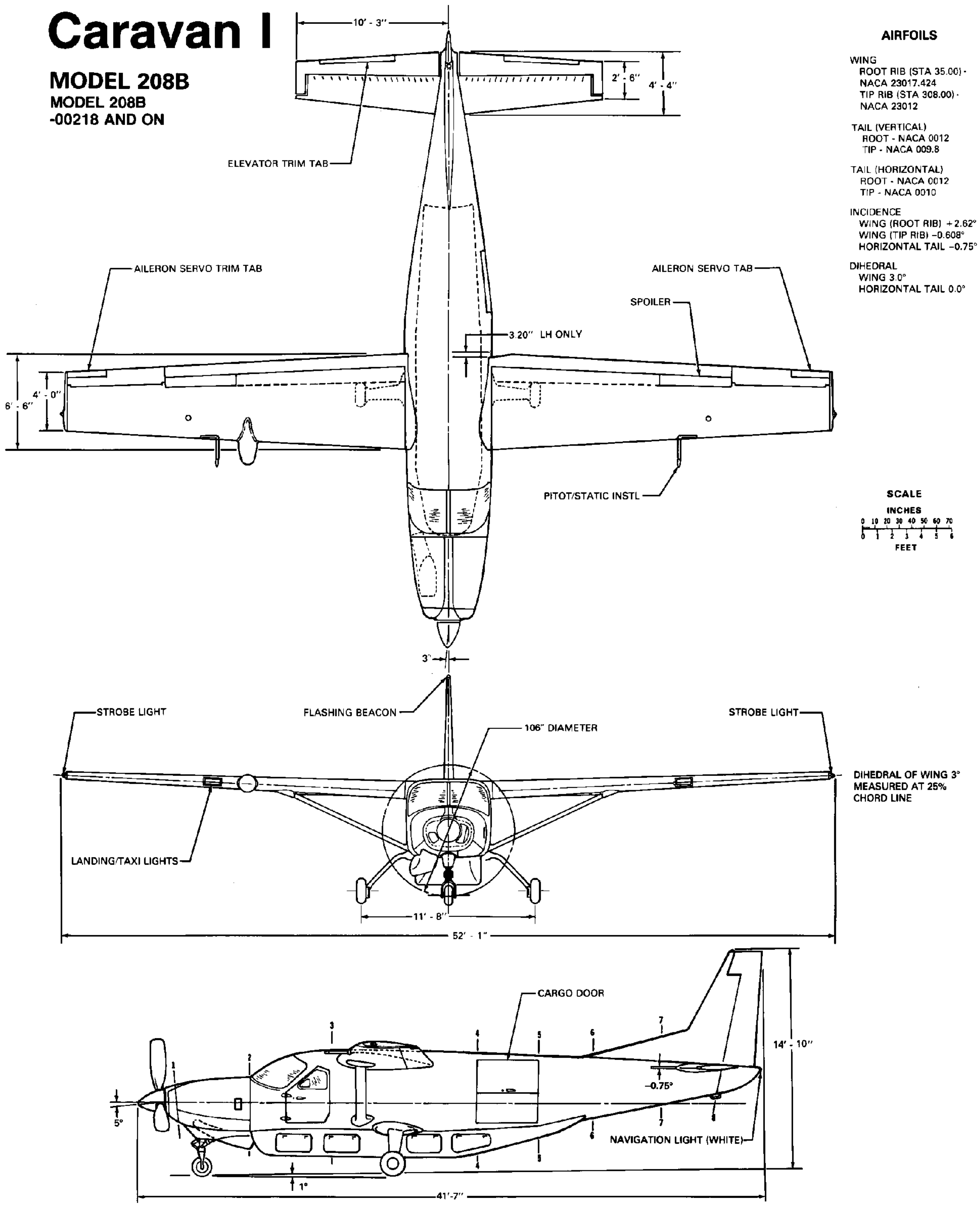

Figure 1. Textron Aviation, Inc. 208B 3-view drawing (courtesy of Textron Aviation)

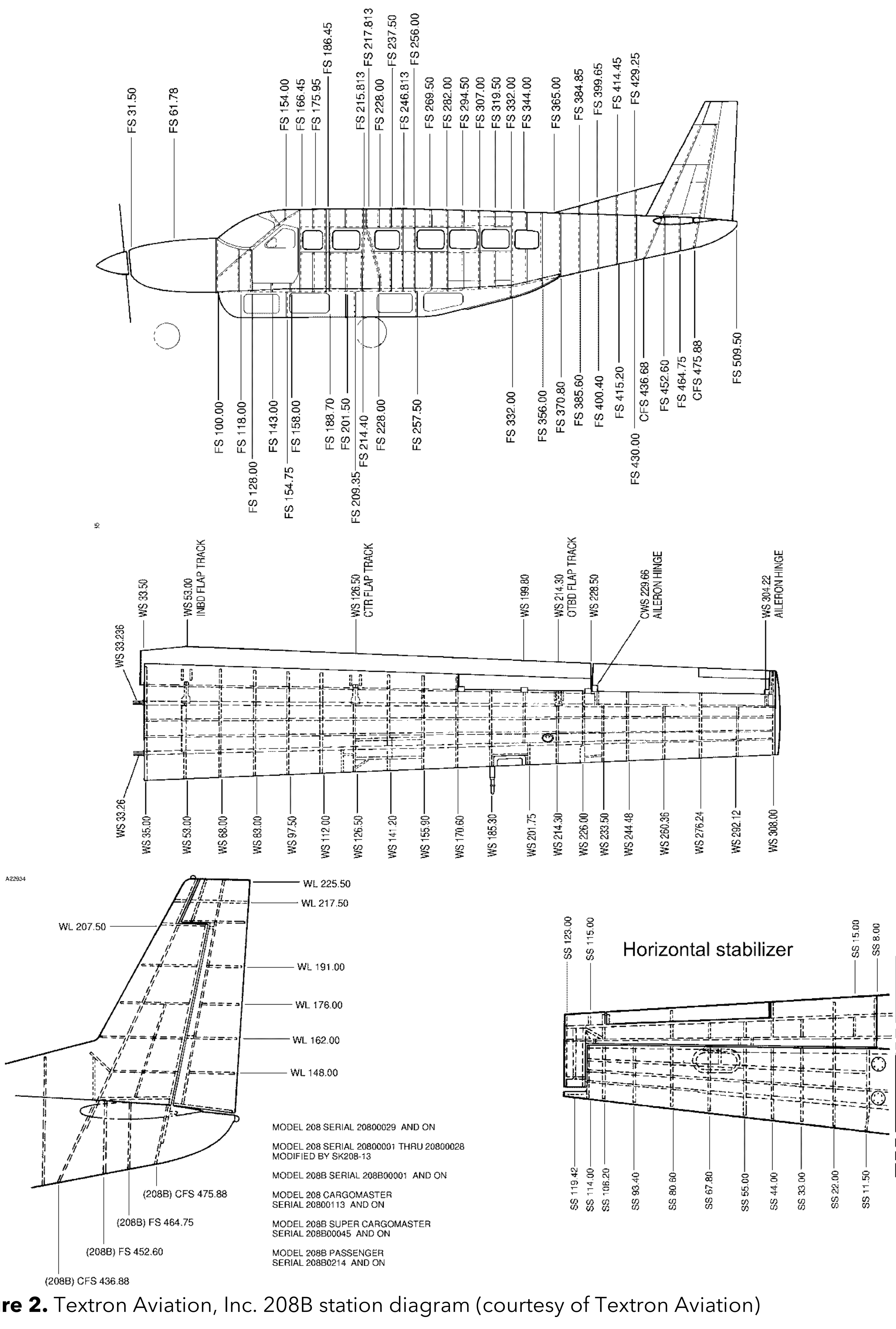

Figure 2. Textron Aviation, Inc. 208B station diagram (courtesy of Textron Aviation)

2.0 Wreckage Examination

The group focused on the right wing and empennage since these items separated prior to ground impact. All the fracture surfaces examined had a dull, grainy appearance consistent with overstress separation. There was no evidence of pre-existing cracking or corrosion noted on any of the structure examined.

2.1 Empennage

The empennage was cut from the main wreckage for recovery near fuselage station (FS) 430. The two elevator control cables, two rudder control cables, autopilot control cables, and one of the pitch trim cables were cut during recovery. The other pitch trim cable end had a splayed, broomstraw appearance consistent with tension overload. The vertical stabilizer and rudder were mostly intact and remained attached to the empennage. The vertical stabilizer attach points on the forward and rear spars were all intact in the empennage. The rudder remained attached to the vertical stabilizer at all hinge locations and the cables were secured to the control horn. The vertical stabilizer and rudder were deformed forward and twisted leading edge right with crushing damage throughout. The aft fuselage was crushed laterally. The elevator bellcrank remained installed in the fuselage with the cables attached. The elevator control tube remained attached to the bellcrank at the forward end and to the aft control horn at the aft end but was fractured between the bellcrank and the aft control horn adjacent to some fuselage damage. The aft control horn remained attached to a portion of the horizontal stabilizer rear spar with the elevator torque tube fittings attached. The left and right elevator torque tubes were separated from the fittings with all the rivets exhibiting shear failure signatures. The vertical stabilizer spars and the rudder torque tube were cut during the exam to remove the vertical stabilizer and rudder from the empennage. The tail cone was removed from the aft fuselage.

The forward horizontal stabilizer attach points were intact with the bolts installed and sealant covering the aft ends. About 8 inches of the horizontal stabilizer forward spar remained attached at the right attach point and it was rotated down about 45° (Figure 3). The upper spar cap was partially attached, and the lower spar cap was separated. About 5 inches of the horizontal stabilizer forward spar remained attached at the left attach point and it was rotated down about 45° (Figure 4). The upper and lower spar caps were separated. The horizontal stabilizer rear spar right attach point remained attached to the aft fuselage and the spar was fractured immediately outboard of the fitting. The horizontal stabilizer rear spar left attach point was separated from the aft fuselage and remained with the left stabilizer and the rear spar fractured immediately inboard of the fitting. The center portion of the horizontal stabilizer with the trim cable and pulleys was present with significant crushing damage.

Figure 3. Horizontal stabilizer forward spar right attach point, looking forward (NTSB photo)

Figure 4. Horizontal stabilizer forward spar left attach point, looking forward (NTSB photo)

The right side of the horizontal stabilizer fractured near right horizontal stabilizer station (RHSS) 11 at the side of the fuselage. The right stabilizer was mostly intact with little damage from root to tip. All the vortex generators remained installed on the upper aft skin. A section of the forward spar lower spar cap about 17 inches long remained attached inboard of RHSS 11. There was a cable tear in the lower skin from the root out to near RHSS 44 and a portion of the trim cable was retained. The deformation of the forward and rear spars at the separation location was consistent with downward separation. The right center elevator forward hinge half was intact on the rear spar with the bolt installed. There were no remnants of the right center elevator aft hinge half retained. Both halves of the right outboard elevator hinge were intact with the bolt installed. The right outboard elevator aft hinge half brackets were pulled from the right elevator and retained in the hinge. The two right trim rods were fractured at the aft end of the rod ends. About 16 inches of the inboard portion of the right elevator was recovered. The remainder of the right elevator was not recovered. The elevator torque tube was intact but separated from the right torque tube fitting with all the rivets exhibiting shear failure signatures.

The left horizontal stabilizer fractured from the empennage near left horizontal stabilizer station (LHSS) 11 at the side of the fuselage. The left stabilizer was mostly intact from root to tip. About 13 inches of the lower skin between the spars was separated from the root outboard. There was downward bending of the spars and buckling of the skins between about LHSS 11 and LHSS 33. All the vortex generators remained installed on the upper aft skin. The left center elevator forward hinge half was separated from the rear spar. Both halves of the left outboard elevator hinge were intact with the bolt installed. The left outboard elevator aft hinge half brackets were pulled from the left elevator and retained in the hinge. The two left trim rods were fractured at the aft end of the rod ends. The left elevator was recovered intact from root to tip. The elevator torque tube was intact but separated from the left torque tube fitting with all the rivets exhibiting shear failure signatures. There was deformed and fractured upper elevator skin at the center hinge, trim rod, and outboard hinge locations consistent with elevator over travel trailing edge up. There was buckling damage to the left elevator upper and lower skins between LHSS 22 and LHSS 33 in line with the left horizontal stabilizer damage.

2.2 Right Wing

The right wing was recovered mostly intact from root to tip with the right aileron and right aileron trim tab intact and installed. There was diagonal buckling of the upper wing skin between about wing station (WS) 244 and 292. The right flap was separated and recovered in several pieces. The largest piece was the outboard portion about 6 feet long that was intact and mostly undamaged. The inboard end of the outboard section of right flap was mechanically cut near WS 160. The cut edges were curled upward along their length and were oriented about 25° from the longitudinal axis. The outboard flap carriage and the outboard flap track were deformed consistent with the inboard end of the flap section rotating upward. About 32 inches of damaged and deformed flap remained attached to the center flap track and the center track was deformed inboard. There was significant crushing and multiple mechanical cut areas noted with the cut ends curled upward. There was no flap structure attached to the inboard flap track. The inboard flap track was deformed in a u-shape. Two additional sections of flap were identified, one about 6 inches wide and one about 39 inches wide, each with diagonal mechanical cuts on each end. The cuts were oriented about 25° from the longitudinal axis and the cut ends were curled upward. There were 4 distinct parallel diagonal mechanical cuts noted to the wing between WS 141 and WS 170 that cut the skins and aft wing spar (Figure 5). The cuts were oriented about 25° from the longitudinal axis. The cut edges all had upward curling, and there was some black paint transfer noted.

Figure 5. Right wing lower surface diagonal cuts WS 141-170 (NTSB photo)

There were 4 distinct parallel diagonal mechanical cuts noted in the wing root area between the root (WS 35) and WS 84. The cuts were oriented about 60° from the longitudinal axis. The cut edges all had upward curling, and there was some black paint transfer noted.

Figure 6. Right wing lower surface diagonal cuts WS 35-84 (NTSB photo)

The right wing strut remained attached to the right wing at the upper attach point. The lower forward and aft attach fittings remained attached to the lower end of the right strut, but both were fractured. There was buckling of the right strut trailing edge noted along its length and the strut forward and rear spars were deformed aft (Figure 7). The upper strut attach bolt was intact and installed. There was deformation, damage, and an impact impression on the lower wing skin outboard of the strut attach point consistent with the strut rotating outboard about the attach point. The right wing was recovered laying on its lower surface with the strut oriented outboard.

Figure 7. Right wing strut lower surface, looking outboard (NTSB photo)

The lower 12 inches of the right strut was cut from the strut and the fractured forward fitting was removed before recovery. The lower aft strut fitting was fractured near the lower attach hole just above the lower edge of the strut. There was no obvious deformation of the lower aft strut fitting. The lower forward strut fitting was fractured partially through the attach bolt hole about 3 inches below the edge of the strut. The fractured end was deformed forward and twisted strut leading edge down. The strut fitting fracture surfaces had an angled, dull, grainy appearance consistent with overstress separation. The fractured ends of the lower forward and aft strut fittings were retained in the fuselage clevises (Figure 8). The retained forward piece was rotated upward relative to the aft piece. The lower strut attach bolt was intact and installed in the fuselage fittings. There was forward deformation noted to the forward fuselage clevis lug.

Figure 8. Fractured right strut lower fittings in fuselage clevis (NTSB photo)

The forward spar and aft spar wing attach fittings were fractured from the inboard end of the right wing. The spar caps and webs were fractured where they attached to the fittings. The right wing aft spar upper cap was twisted leading edge down at the fracture location and the lower spar cap exhibited s-bending deformation of the spar cap at the fracture location. The forward and aft spar wing attach fittings remained attached to the fuselage fittings. The forward and aft wing attach bolts were intact and installed. The forward wing attach fitting was jammed upward about 60° and the aft wing attach fitting was jammed upward about 10°. The forward spar wing attach fitting and the fuselage lugs were deformed forward. There was no obvious deformation of the aft spar wing attach fitting or fuselage lugs.

The right forward and aft wing attach points and the right lower strut attach point were disassembled and the fittings were separated. All 3 bolts had no obvious evidence of deformation. The right lower aft strut fitting fragment measured 0.500 inch thick and the hole measured 0.752 inch diameter vertically and 0.765 inch diameter horizontally. The right lower forward strut fitting measured 0.498 inch thick. The hole diameter could not be measured due to the fracture. The right wing forward spar attach fitting forward lug measured 0.498 inch thick and the hole measured 0.877 inch diameter vertically and 0.894 inch diameter horizontally. There was a ridge of deformed material along the outboard side of the hole. The right wing forward spar attach fitting aft lug measured 0.490 thick above the hole and 0.498 below the hole and the hole measured 0.870 inch diameter vertically and 1.151 inch diameter horizontally. The hole was visibly elongated horizontally (Figure 9). The right wing aft spar attach fitting forward and aft lugs measured 0.440 inch thick. The keyed eccentric bushings remained installed in the lugs. The holes in the bushings measured 0.623 inch diameter horizontally and vertically.

Figure 9. Right wing forward and aft spar attach fittings, aft side, after disassembly (NTSB photo)

E. TESTS AND RESEARCH

1.0 Textron Aviation Airplane Information

The Textron 208B EX Pilot's Operating Handbook and FAA Approved Airplane Flight Manual (POH/AFM) published the following limitations.

Max Takeoff Weight – 8,807 lb (without STC)
Max Landing Weight – 8,500 lb (without STC)
Max Operating Speed V_{MO} – 175 KIAS
Maneuvering Speed V_A – 148 KIAS (at 8,807 lb)

Max Flap Extended Speed V_{FE} – 125 KIAS (Land)
Max Flap Extended Speed V_{FE} – 150 KIAS (Takeoff)
Flap Operating Range – 50-125 KIAS

Center of Gravity Range – The forward CG limit is 185 inches aft of datum at 6500 pounds or less, with straight line variation to 193.37 inches aft of datum at 8000 pounds and straight line variation to 199.15 inches aft of datum at 8807 pounds. The aft CG limit is 204.35 inches aft of datum at all weights up to 8807 pounds. The datum is located 100 inches forward of the face of the firewall.

The airplane is certificated in the normal category which includes maneuvers incidental to normal flying such as stalls (except whip stalls), lazy eights, chandelles, and turns with bank angles not more than 60°. Aerobatic maneuvers, including spins, are not approved.

The limit load factors at MTOW for flaps up are +3.8g and -1.52g and for flaps extended +2.4g.

2.0 STC Limitations

The installed STCs on the airplane had further limitations beyond those published in the POH/AFM.

LIMITATIONS for SN 208B2197, 208B5000 and up (LAND PLANES ONLY)
CG Range: (Takeoff and Flight)
+185.00 inches at 6500 pounds or less
+193.37 inches at 8000 pounds
+199.15 inches at 8807 pounds
+200.00 inches at 9062 pounds (Takeoff only)
+204.35 inches at 9062 pounds (Takeoff only above 9000 pounds)
Straight line variation between points given

Limitations Pertaining to all Eligible Models

Limit Speeds (IAS):	Maneuvering Speed at 9,062 lb	143 kt
Maximum Weights	Takeoff:	9,062 lb
Limit Factor	Flaps up:	+3.36, -1.34
Limits (g's):	Flaps down (all settings):	+2.00"

3.0 Textron Aviation Loads Analysis

The NTSB requested that Textron Aviation run a loads analysis of the wing, flap, and horizontal tail loads for the full flap overspeed conditions that the airplane experienced during the accident flight using their existing airplane models. The external wing loads were calculated by a proprietary structures computer program that takes wing geometry, wing aerodynamic data, wing weight distribution, force coefficients, load factor, velocity, and center of gravity (CG) as inputs. The aerodynamic data was calculated based on standard textbook methods using known airfoil information. The external wing, horizontal tail, and flap loads were calculated for the following speeds with full flaps (30°), V_{F30}. The resulting loads were compared to the baseline loads and the design limit load (DLL) envelopes established during certification of the airplane. The ultimate load (UL) envelopes, defined as 1.5 x DLL, were included for the cases where the results exceeded DLL.

V_{F30} = 125 KEAS (knots equivalent airspeed), baseline condition corresponding to the maximum full flap extension speed
V_{F30} = 175 KEAS, corresponding to maximum operating speed
V_{F30} = MAS (maximum analysis speed, proprietary value)

The airplane full flap operating envelope at a maximum gross weight of 8,807 lb is limited to a minimum vertical load factor of 0 g, a maximum vertical load factor of 2.4 g, and a maximum speed of 125 KEAS. The initial calculations examined the effect of weight and CG on the wing and horizontal tail loads at the minimum weight of 5,500 lb, the maximum weight of 8,807 lb, and at the forward and aft CG limits for each weight under the baseline conditions throughout the load factor range. The results for wing beam (vertical) shear and wing chordwise shear (aft bending) showed expected variations for weight and load factor and that CG location did not have a significant effect on the loads. The balancing tail loads were calculated for the same conditions with no elevator deflection and showed expected variations for weight and load factor with less downward load on the tail required for the aft CG locations. The effect of speed on the wing and horizontal tail loads at the maximum weight of 8,807 lb and the aft CG limit of 205 in throughout the load factor range was examined at 125 KEAS and MAS. The results for wing beam (vertical) shear showed expected variations for load factor and that speed did not have a significant effect on the loads. The results for wing chordwise shear (aft bending) and balancing tail loads showed expected variation for load factor and a significant increase in the wing chordwise shear load and a significant increase in the downward force on the tail with increasing speed.

The right wing loads under the accident conditions were then examined with full flaps, an airplane weight of 7,974 lb, a CG at 203.46 in, and at load factors of 1 g and 2.4 g at 125 KEAS and 175 KEAS. The results showed that the wing beam shear and wing beam bending moment loads remain well within the DLL envelope for all conditions.

The wing chordwise shear results showed that the V_{F30} 125 KEAS 1 g and 2.4 g conditions remain within the DLL envelope along the entire wingspan. The V_{F30} 175 KEAS 1 g condition loads exceed the DLL envelope along the entire span. The loads are slightly within (less than 10%) the UL envelope from the wing root to WS 112.0, equal to the UL envelope at the strut attach point (WS 126.5), and marginally within (less than 20%) the UL envelope from WS 141.2 to the wingtip. The V_{F30} 175 KEAS 2.4

g condition loads exceed the DLL envelope along the entire span except between WS 97.5 and WS 126.5. The loads are well within the UL envelope from the root to WS 97.5 and slightly within (less than 10%) the UL envelope from WS 141.2 to the wingtip.

The DLL envelope for wing torsion is negative along the entire wingspan resulting in a leading edge down moment on the wing. The V_{F30} 125 KEAS 1 g and 2.4 g conditions remain within the DLL envelope along the entire wingspan. The results for the V_{F30} 175 KEAS 1 g and 2.4 g conditions exceed the DLL envelope from the wing root to WS 201.5 and are within the DLL envelope from WS 201.5 to the wingtip. The results for the V_{F30} 175 KEAS 1 g condition are slightly within (less than 10%) the UL envelope from the wing root to WS 126.5 and very slightly within (less than 5%) the UL envelope from WS 141.2 to WS 185.3. The results for the V_{F30} 175 KEAS 2.4 g condition are very slightly within (less than 5%) the UL envelope or exceed the UL envelope by up to 2% (at WS 112.0) from the wing root to WS 185.3.

The flap loads were examined for the V_{F30} 175 KEAS conditions and compared to the design flap loads governed by a head-on gust case at 139.8 KEAS. The flap loads were scaled by the dynamic pressure increase from the design case to the exceedance case. The flap loads were calculated to exceed the UL envelope by about 7%. The wing forward and rear spar attach point loads were examined at the V_{F30} 175 KEAS cases and the results showed that the vertical load at the forward spar attach point exceeded the UL load envelope by about 13% and about 11% at the rear spar attach point. The balancing tail loads with no elevator deflection showed that the tail loads were within the DLL envelope up to MAS.

Submitted by:

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