SA-416 File No. 3-1593

AIRCRAFT ACCIDENT REPORT

AIR SOUTH, INC.

BEECHCRAFT B-99, N844NS

NEAR MONROE, GEORGIA

JULY 6,1969

Adopted: AUGUST 26, 1970

NATIONAL TRANSPORTATION SAFETY BOARD
Bureau of Aviation Safety
Washington, D. C. 20581

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SYNOPSIS

An Air South, Inc., Beechcraft B-99, N844NS, crashed near Monroe, Georgia,—at approximately 2122 eastern daylight time, July 6, 1969. The aircraft, operating as Air South Flight 168, was en route from Atlanta, Georgia, to Greer, South Carolina. The 12 passengers and two crewmembers received fatal injuries in the accident and the aircraft was destroyed.

An eyewitness to the accident stated that the aircraft descended in a near-vertical dive, with no change in attitude after it had nosed down. The weather in the accident area was reported to be clear and the wind was calm.

The Board determines that the probable cause of this accident was an unwanted change in longitudinal trim which resulted in a nosedown high-speed flight condition that was beyond the physical capability of the pilots to overcome. The initiating element in the accident sequence could not be specifically determined. However, the design of the aircraft flight control system was conducive to malfunctions which, if undetected by the crew, could lead to a loss of control.

On August 1, 1969, the Board recommended that the Administrator, Federal Aviation Administration, establish emergency recovery procedures from unwanted or adverse longitudinal trim conditions and publish them in the FAA-approved Flight Manual. The Board also recommended that a horizontal stabilizer' "In-Transit" warning system be installed in E99 aircraft and that the horizontal stabilizer trim range be restricted to prevent excessive aircraft nosedown trim while in flight.

The Administrator replied on August 6, 1969, that he had taken action to carry out the Board's recommendations.

In addition, the Administrator took a number of other corrective actions relating to the longitudinal control system of the E-99.

Finally, the Board recommends that direct FAA participation in the certification of new items be mandatory; that information gained from the investigation of large aircraft accidents be used by the FAA in the certification of mall aircraft; and that the FAA review the existing fault analysis system and require the completion of hazard analyses of the type required by par. 5.8.2, Military Standard 882, dated July 15, 1969.

1. INVESTIGATION

1.1 History of the Flight

Air South Flight 168, a Beechcraft B-99, N844NS, departed from Atlanta International Airport, Atlanta, Georgia, at 2107 e.d.t., 1/ July 6, 1969, destined for Greenville/Spartanburg Airport, Greer, South Carolina. The flight was cleared and handled in a routine manner, and reported level at its assigned cruising altitude of 7,000 feet m.s.l. 2/ at 2113:05. This was the last recorded transmission from the flight.

At approximately 2125:25, the radar controller in the Atlanta Air Route Traffic Control Center (ARTCC) noted that the aircraft's radar target had disappeared from the radarscope. He was unable to establish radar contact or radio contact with the flight after that time.

Eyewitness account

The only eyewitness known to have seen the accident reported that at approximately 2120, he saw an airplane coming and, "... I heard the motor cut off just after passing. I then looked up to see it and the motor cut on and (sic) off again. After it cut off the second time it started down. I could see it was a twin engine plane a little larger than those in town. As it went down it pitched down real fast and went into a straight dive. ... As it came down it seemed to pick up speed making a humming sound getting louder and louder. Before it went out of sight behind the trees the lights went out. I could see it until it went out of sight behind the trees. After it disappeared I heard a loud thump and then a boom almost immediately.

"The weather was fair. There was no lightning and no wind.

"... I might add that when the engines came on each time they backfired and after the backfire the second time all was quiet. It was completely quiet after each backfire."

The witness also stated that when he first saw the aircraft it was flying level. "After the second time the engine cut off, it backfired and didn't catch backup." The aircraft went "just a little piece and then nosed down toward the ground." king the dive, the aircraft did not nose up at all. The witness did not see anything separate from the aircraft.

The accident occurred at approximately 2122 in twilight and the wreckage was located at latitude 33" 53' 20" N. and longitude 83° 46' 10" W. at an elevation of 880 feet.

1/ All times herein are eastern daylight based on the 24-hour clock.

2/ All altitudes are mean sea level unless otherwise indicated.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	2	12	0
Nonfatal	0	0	0
None	0	0	

1.3 Damage to Aircraft

The aircraft was destroyed.

1.4 Other Damage

None.

1.5 Crew Information

Both the assigned pilots were properly certificated and qualified for the performance of this flight in accordance with the current FA4 and company regulations. (For details see Appendix B.)

1.6 Aircraft Information

The aircraft records indicate that the aircraft was properly certificated and airworthy at the time of takeoff from Atlanta. The weight and center of gravity were computed to have been within limits at the time of takeoff and at the time of the accident. The c.g. limits were 179 inches forward and 195 inches aft of the datum. The takeoff c.g. was 187 inches aft of the datum. The maximum takeoff weight was 10,400 pounds and the takeoff weight was computed to be 9,710 pounds. The aircraft was fueled with aviation kerosene. (For details see Appendix C.)

1.7 Meteorological Information

Witnesses in the area of the accident reported that the weather was. clear or there were scattered clouds, and the winds were light. While some of the witnesses did see lightning in the distance, no thunderstorms or rain were reported in the accident area. There were no reported thunderstorms or other severe weather phenomenon between the point of departure and the location of the accident site.

The weather along the route of flight was forecast and reported to be generally scattered to broken clouds at 4,000 to 5,000 feet, with visibility generally in excess of 3 miles. The winds aloft were forecast and reported to be southwest-to-west at less than 10 knots. The freezing level was above 10,000 feet.

Radar weather observations made before, during, and after the accident showed no areas of severe weather on the route of flight the aircraft was reported to have flown. There were thunderstorms to the north, east, and south of the accident site. The radar weather observer testified that according to his radarscope, the accident site was 258° at 24 miles from the center of his radarscope depiction. He also stated that in photographs taken of the weather radarscope at 2115, 2120, and 2125, the accident site was just south of a weak precipitation echo. According to his testimony, light to moderate turbulence might be expected near a moderate shower or thunderstorm. The degree of turbulence would depend on the aircraft's proximity to the center of the cell.

The flightcrew was reported to have telephoned the Fulton County Airport Flight Service Station (FSS) for a weather briefing before departing from Atlanta.

The accident occurred in twilight.

1.8 Aids to Navigation

Not applicable to this accident.

1.9 Communications

Radio communications between the ground stations and the aircraft were reported to be normal and routine until the aircraft disappeared from the radarscope and radio communications were lost.

1.10 Aerodrome and Ground Facilities

Not applicable to this accident.

1.11 Flight Recorders

No flight recorders were installed or required by regulations

1.12 Wreckage

The aircraft crashed approximately 6 miles northwest of Monroe, Georgia, on relatively flat terrain.

The impact; of the aircraft left three craters in the hard clay soil. These craters were joined by narrow scars, and a line through the centers of the craters was oriented along a magnetic bearing of 180" to 360°.

The right engine was found in the southernmost crater, the left engine was in the northern crater, and the fuselage components were in and around the center crater. The nose wheel was on the western edge of the center crater, the center wing section fragments were west and north of the center crater, and most of the empennage was northeast of the center crater.

The outer wing panels and associated control surfaces were in the general area of the craters. The left-hand outer wing panel was 150 feet south of the craters, and the right-hand outer wing panel was 180 feet south of the craters. With the exception of the right aileron, located 465 feet east of the craters, most of the parts which separated from the outer wings were found southwest of the craters.

The breakup of the aircraft was extensive in all areas of the structure. Much of the forward section of the aircraft could not be identified. The pieces of the structure were generally crushed longitudinally. In the empennage, the front spars and leading edges of the control surfaces were crushed flat against the rear spars of the surfaces to which they were attached. The rear spars showed evidence of having been crushed into the front spars and leading edges of the surfaces.

Parts of all major components of the structure and of all the control surfaces were found in the primary wreckage area. No evidence of fire, fatigue, or prior damage was observed.

The fuselage was fragmented; however, pieces from all the major areas of the fuselage were recovered and identified. Sections from the forward part of the fuselage were generally smaller and more fragmented than those from farther aft.

Only small fragments of instrument panels, control mechanisms, crew seats, etc., were identified from the cockpit area.

The lower aft end of the aft fuselage structure was recovered in one flattened section. Portions of both horizontal stabilizer pivot support brackets were in place on this piece, as were the elevator torque shaft support brackets. These brackets were spread outward by the flattening of their attach structure and the torque shaft was separated from the brackets.

All or portions of the cabin door, cargo door, aft baggage door, right-hand nose baggage door, and a section of one of the two emergency doors were identified.

The left-hand outboard wing panel was recovered upright and nearly intact. The outer panel was crushed chordwise and was deflected downward so that the tip assembly was 3-1/2 inches lower than the root section, measured along the front spar. The wingtip chord was 46" leading-edge-down with respect to the chord at the wing root.

There were spanwise buckles in the upper wing skin just aft of the front spar from WS 106 to WS 155. The skin just forward of the rear spar was buckled diagonally for almost the entire span, the buckles extending forward and outboard from the spar. The leading edge and upper skin were crushed chordwise from WS 150 to the extreme outboard edge of the tip assembly, and the forward portion of the tip assembly was mangled.

The lower wing skin exhibited a number of diagonal and spanwise buckles just aft of the front spar. Most of these buckles progressed aft and inboard from the front spar, except that those nearer the wingtip extended outboard as they progressed aft.

The front spar lower cap exhibited scoring on the exposed area between the leading edge and box section skin from near WS 200 to the tip assembly. The scoring was wider as it neared the tip section.

The right-hand outboard wing panel was relatively intact to WS 239 where the upper surface had a ragged chordwise fracture, the torn edges of which were bent or crushed aft and outboard. The outboard extremity of the lower surface fracture was at WS 250. The deformation of the wing panel was similar to that of the left-hand panel, and the entire panel exhibited a general downward bend. Scoring on the leading edge of the lower front spar cap on the right wing panel extended from the root fitting to the tip assembly, across the fracture area.

The upper surface skin was buckled spanwise between the spars from WS 172 outboard. The lower skin had sharp spanwise buckles just aft of the front spar. A number of chordwise tears were noted along the rib fastener lines in the area of these buckles.

Both of the outer wing panels separated at their attach points to the center wing section, and the failures on each side were very similar. The remaining tangs and structural members that were attached to the center wing section were bent downward and some portions had an aftward bend as well. Some of the tangs and structural fragments were also twisted toward the leading edge of the wing. This damage was typical of that which would have been caused by a downward wing failure, with the wing leading edge rotating down and aft at the same time.

The vertical stabilizer was fragmented but the leading edge, front spar, and rear spar were continuous, although separated from each other. The stabilizer spars had separated from the fuselage at the attach fittings.

The horizontal stabilizer was fragmented and crushed. The stabilizer had been fractured approximately 3 feet from the center section on the right side and 8 feet from the center section on the left side. In the center section, the spars were crushed to a chordwise separation of approximately 3 inches rather than the normal dimension of approximately 2 feet.

The horizontal stabilizer trim actuator support fittings and the stabilizer pivot support fittings were intact and in place on the stabilizer structure. The trim actuator rod ends were separated from the actuator and remained in place on the stabilizer support fitting. Both rod ends were free to move by hand.

Al the flight control surfaces were recovered in the primary wreckage area. Because of the extensive fragmentation, the integrity of the flight control system prior to impact could not be determined.

The ailerons were separated from the wings but were recovered in two major pieces each. The hinges on both sides showed evidence of having overtravelled in a downward direction.

The rudder was basically intact except for the balance weight which was separated and found nearby. The leading edge was crushed against the spar and contained numerous tears. The skin aft of the spar was crushed forward and ballooned out around the spar. The ballooning appeared to be symmetrical about the spanwise centerline of the spar.

The elevators had separated and were recovered in pieces. The left-hand elevator was found in several pieces, while the right-hand one was in two main pieces. Both elevators were similarly damaged with the leading edge structure crushed aft or flattened. The elevator hinges had separated and were bent approximately 90° outboard. The elevator control tubes had failed. Because of reported service cracks in these tubes, they were examined by the Board's metallurgist. No evidence of prior damage was found in the fractures on these parts.

The elevator control horns remained attached to the stabilizer structure. There was no evidence of repeated impacts on either the up or down stops on the right control horn. The up stop on the left horn showed no evidence of pounding, but no determination of damage could be made regarding the down stop.

The elevator torque shaft at FS 442 and other mechanisms in the rear of the aircraft were separated from their attach structure. The cable, which normally controlled upward elevator movement, was connected to a small piece of the aft elevator bellcrank, and the cable was traced to its attachment to the forward bellcrank. The down elevator cable was traced from the forward bellcrank to a point near the aft fuselage turnbuckle, where it was separated in what appeared to be a tension failure.

The horizontal stabilizer trim actuator was found in a number of pieces. The case was fractured, one jackscrew and the primary motor were found separately, and the secondary motor was held on to the case by its electrical wiring. The fuselage mounting brackets had separated from the fuselage and remained in place on the actuator. The jackscrew extension shafts were fractured, with sharp 45° edges, approximately flush with the rod end inserts. Both jackscrew rod ends were in place on the horizontal stabilizer trim actuator support fittings. With the broken rod ends held in place on their respective shafts, the dimensions from the flanges of the housings to the centerlines of the stabilizer attach bolts measured 8-3/8 inches on the right and 8-1/2 inches on the left side. According to the manufacturer, the corresponding dimensions with the stabilizer leading edge in the aircraft full nosedown trim position was 8-1/2 inches.

The horizontal stabilizer trim unit was recovered, separate from its mounts and broken. The setting of the actuator was measured and calculated to be equal to a full aircraft nosedown trim position. The manufacturer reported that the jackscrew had not reached the mechanical stop but was in a position corresponding to the electrical retract stop position. The trim position actuator potentiometer was in a position appropriate to the jackscrew position. The manufacturer also reported that it was not possible to determine whether the unit had been functioning on the primary or auxiliary mode, since the associated limit switch positions had apparently been altered by impact.

The inboard landing flaps were recovered attached to the wings. The flaps were crushed against the rear spars and were generally distorted. The left inboard flap was approximately in the retracted position with respect to the wing spar, with the flap actuator detached. The right flap was so distorted that the position of the flap could not be accurately estimated.

One of the two inboard flap jackscrews was found, detached from both wing and flap. With the pieces of the jackscrew assembly held in their respective positions, the extension of the jackscrew was 10.125 inches between the centerlines of the attach bolts. According to the manufacturers, the jackscrew extension was 9.88 inches when the flaps were retracted, and when the flaps were in the approach setting, the jackscrew measured 11.82 inches.

The left outboard flap was recovered in two pieces, separated from the wing. The right outboard flap was attached to the wing and was approximately in the retracted position. The jackscrew was extended 10.750 inches. According to the manufacturer, the retracted measurement should have been 9.78 inches, and the approach setting measurement should have been 11.65 inches.

The landing gear was recovered in a number of pieces which were found in the area of the craters. Both main gear struts were crushed into their respective wheel well and wing center section structure, with the unbroken drag leg assemblies in the jackknifed position. These legs are jackknifed when the landing gear is in the retracted position.

The powerplants were found in the crater area in their normal positions relative to the aircraft. They were in a crater approximately 6 feet deep, and the engines were compressed from their original length of 5 feet 2 inches to 2 feet for the No. 1 engine, and 16 inches for the No. 2 engine.

The propeller reduction gearboxes were examined and the gears showed no evidence of distress. The axial compressor disc of each engine was found with compressor blades broken off at the platform area opposite to the direction of rotation. There was no evidence of penetration of the engine cases by compressor or turbine disc components. There was no evidence of in-flight or ground fire on either engine.

The propellers were imbedded in the craters with the engines. The manufacturer computed the average blade angle at impact to be 27.5°.

The electrically operated fuel shutoff valves were found in the open position, and the cross feed valve was found in the closed (normal) position.

Most of the aircraft system components and instruments were crushed or damaged so that no useful information could be determined from them. The electrical system was reduced to fragmented wire bundles and pieces of relays and contactors too mall to identify. Examination of many wire bundles and heavier current-carrying cables showed no evidence of arcing or short-circuiting. There was no evidence of arcing or shorting within the remains or the voltage regulator.

The left engine generator was recovered with no evidence of arcing. There were scoring marks on the generator end plate.

Only a few instruments and switches were recovered from the cockpit area. A propeller tachometer was dismantled and the indicating needle was found embedded into the face of the instrument at the 1,900 r.p.m. position. Both intermediate turbine temperature (ITT) instruments were recovered. One instrument face had a mark at 520" C. which was matched to an instrument pointer.

Important

Longitudinal trim on the Beech 99 was accomplished through an electric pitch trim system which moved the horizontal stabilizer to the selected position. The aircraft was equipped with a dual horizontal stabilizer trim system. The main trim system was armed by a switch mounted on the pedestal between the pilots, and a thumb-actuated dual element switch on the control wheel permitted control of the main trim actuator by either the pilot or the copilot. The main trim system could be momentarily interrupted by pressing a switch on either control wheel. The standby trim system switches were located on the pedestal and were armed by a separate switch on the pedestal as was the primary system. The position of the horizontal stabilizer was displayed to the pilot by a pedestal-mounted indicator.

During the before-takeoff checklist, the pitch trim indicator was to be compared with the stabilizer position noted during the preflight inspection of the tail. The secondary pitch trim system was to be checked and then turned off. The primary pitch trim system was to be checked for operation and emergency trim release, and the stabilizer set at the predetermined takeoff position. The check of each system was to operate the trim in both the up and down positions and to see that the indicator showed a movement in the correct direction and, for the primary system, to insure that the trim release interrupted trim movement when it was actuated. After landing, the trim was to be set at the zero position.

The emergency procedures outlined in the Pilots Operating Manual, revised November 8, 1968, discussed the procedures for an inoperative pitch trim system. When the primary trim system was inoperative, the pilot was to turn the primary pitch trim master switch off, turn the secondary pitch trim master switch on, and trim the aircraft with the secondary pitch trim switch as required.

The manual advised the pilot to maintain airspeed for low control forces if both the primary and secondary pitch trim systems became inoperative. For landings in this condition, the pilot was advised to use landing flaps as required to reduce pull forces as speed decreased. He was also advised to avoid stick push forces by using only enough flaps to give desired control forces. A note in this section of the manual stated that with the stabilizer inoperative in the cruise position, extending full flaps would give zero elevator force at 100 to 125 knots.

1.13 Fire

No fire occurred in this accident.

1.14 Survival Aspects

This was a nonsurvivable accident.

1.15 Tests and Research

Because of the nature of this accident, part of the investigative activities involved the search for evidence of possible interference with the flightcrew.

A ground search was conducted in the wreckage scatter area and the area in and immediately around the impact craters. During these searches, no evidence of a weapon or any other dangerous article was found.

Two days following the accident, a small pocket knife, about 3 inches long with black handles, was found. "The knife had only one remaining blade and this blade was open and broken half off." This knife was found about 12 feet from the impact craters by a policewoman and an Air South captain. The policewoman stated she placed the knife on a fencepost near the entrance to the accident site.

On the second or third day following the accident, a county deputy sheriff found a pocket knife which was described as ". . . a small pocket knife, which some refer to as a pen knife, with one blade open. About one quarter inch of the blade was broken off. The other blade was shut into the knife. The 'knife had blood on the blade and also three-quarters of an inch on to the knife handle. The color of the handle was brown." This knife was found between the impact craters and an abandoned house near the crater. The deputy stated that he stuck this knife in a crack on top of one of the fenceposts near the entrance to the accident site.

On the third day after the accident, the attention of one of the Board's investigators was directed to a knife stuck in a fencepost near the entrance to the accident site. This knife was two-bladed, with the mall blade stuck in the fencepost. The knife had a brown stag handle and was approximately 3 inches long, 5/16 inch thick, and 3/8 inch deep. The small blade was about 1-1/4 inches long and appeared to be clean. The larger blade was broken off approximately 1 inch from its hinge point. The outer portion was missing. The knife handle did not appear to be distorted or broken, and both blades were open. The investigator was advised that the knife had been used at the accident site to cut rope.

Efforts to establish the identity of the owner or owners of the knife or knives referred to above have been unsuccessful. None of the persons involved in the search and rescue operations at the accident site, or the investigators working on the accident, has provided any information regarding the ownership of these items.

Inquiries were made regarding the purchase of flight (trip) insurance by the passengers on this flight. There was no record of such a purchase by any passenger.

The crewmember remains were identified by fingerprints. It was impossible to make any type of pathological examination or to analyze their physical condition prior to and during the accident sequence.

The FAA medical records of the flightcrew were reviewed and the only abnormality noted was a "soft systolic heart murmur" recorded during the examination of the copilot in June 1968. The Aviation Medical Examiner indicated that he felt that this "very small systolic murmur was functional in nature and that it had no clinical significance."

The service history of the aircraft and the B-99 fleet was reviewed. (For history of N844NS, see Appendix C.) Witnesses who represented other carriers using the B-99, the FAA, and the manufacturer were called to testify at the public hearing.

The Chief of Maintenance, Air South, testified that there had been some problems with the pitch trim actuator. Most of these problems were directly concerned with the primary actuator motor, but had never involved both trim systems at the same time. When the primary system malfunctioned, the secondary system had operated properly. He also stated that N844NS had two writeups reporting slow operation of the primary trim system, but that both writeups related to the same malfunction because the system checked out during a ground test. When the aircraft was released for flight, the malfunction was repeated. The actuator was changed after the second writeup, but the cause of the malfunction was not determined. The company had never experienced a runaway or unwanted trim condition in flight.

The witness also reported that he had not had a primary trim switch failure since Air South had been using the E-99. Prior to the accident, he had received no reports of problems with the secondary trim switches but since the accident, and prior to the hearing, he had changed two secondary trim switches. Significant

The only difficulty he reported with the empennage was some cracking of the skin on the trailing edge of the elevators. He reported no difficulties with distortion, hinge cracking, torque tube cracking, or other malfunction.

A witness from a second carrier that operated B-99's testified that during 15,992 hours of aircraft time, they had experienced three longitudinal control problems.

These problems involved two pitchdowns and one pitchup. In each case, the degree of pitch was not reported but the crews did report heavy control forces resulting from these maneuvers. Investigation had not disclosed the cause of two of these incidents. The third incident was found to have occurred when the two pilots attempted to activate the primary trim system in opposite directions. This resulted in the opening of the primary trim system circuit breaker which deactivated the trim system. The crew experienced heavy nosedown control forces, but were able to recover when the circuit breaker was closed and the trim system reactivated. Attempts to duplicate the experience of the other two crews were unsatisfactory and the aircraft reactions could not be reproduced.

The Director of Maintenance for this carrier testified that, with regard to the longitudinal control system, they had changed "many" trim actuators. Some of these changes were made for the purpose of upgrading the system but others were for cause. In those cases where actuators were changed for cause, the problem generally manifested itself as "an inoperative position" or a burned out motor. There had been no cases of a reported runaway trim. The company had experienced no difficulty with the primary trim switch but had experienced some problems with sticking of the secondary switch. None of the sticking secondary switches caused the stabilizer to move to either a full nosedown or full noseup position. His investigations indicated that the problems with the secondary trim switches were the result of contamination of the switches by coffee and cigarette ashes.

The only difficulty this carrier reported with the empennage section was some cracking of the stabilizer skin, which was repaired in accordance with a service bulletin from Beech Aircraft.

The Vice President for Operations and Maintenance for a third carrier testified that his organization operated five B-99's and had approximately 10,200 hours of experience with them.

This carrier had experienced some difficulties with the trim motors similar to that reported above. In addition, the carrier reported cracking of the elevator skin around the trailing edge, one or two cracked elevator torque tubes, and elevator hinges cracking at the attach point. In addition, pilots had reported porpoising of the aircraft in flight. In one case, the aircraft was returned to Beech who reported that the aircraft had a warped elevator. The elevator was replaced, the installation of the horizontal stabilizer was checked, and the stabilizer index system was verified.

The porpoising reported by the pilots of this carrier generally occurred in turbulence and/or icing conditions and there were approximately six or seven reports of this problem.

Pilots for this carrier also reported three incidents of a trim system malfunction, all of which resulted in an unwanted pitchup. Investigations of these incidents have not revealed any discrepancy in the trim system or the flight control system that might have caused the unwanted pitchup. The investigation of one incident indicated that there was a possibility that an inadvertent partial extension of the landing flaps may have caused the crew to believe that they were experiencing an unwanted noseup trim condition. This could not be definitely established, however.

In general, the experience of this carrier was the same as that reported by the other witnesses.

The Chief, Engineering and Manufacturing Branch of the FAA regional office responsible for the certification of the Beech 99, appeared as a witness at the public hearing and testified regarding the certification of the aircraft.

This witness testified that the Beech 99 was certificated under a program designated as the Delegated Option Authority. Under this procedure, a manufacturer was authorized by the FAA to certificate the product he manufactured, and this certification was accepted by the FAA and used as a basis to issue the appropriate airworthiness certificate for the product. In the exercise of this program, the FAA retained the right to impose special conditions on certifiable products and to examine portions of the certification program prior to issuing a certificate of airworthiness.

Basically, special conditions were applied where the existing rules did not cover the safety aspect of a new design. In the case of the B-99, special conditions were established for the propulsion system, ~~pressurization system~~, and the electrical system. Compliance with these special conditions was established by an FAA review of the fault analyses of the systems, a review of circuit diagrams, and flight checks. None of the special conditions applied to this aircraft related to the longitudinal control or trim systems.

The expressed intent of fault analysis was to ensure that no single fault in a system could lead to a failure of that system. The fault analysis performed by the manufacturer of the horizontal stabilizer actuator installed in E99 aircraft dealt with the trim system actuator and related components which made up the dual mode of operation. The analysis appeared to be cumplete and correct for the actuator system, but did not relate to the interfacing of the system as it affected the performance of the aircraft. Consideration of human response rates, recovery factors in time, and variables such as c.g., airspeed, etc., were not contained in the report.

The ramifications of possible faults of the trim system could only have been evaluated in the performance of the aircraft and the pilot. Then corrective action could be placed in effect as necessary. This type of analysis was not performed on the E99 nor was it required by regulation.

Beech Aircraft applied for a type certificate for the Beech 99 July 8, 1966. The regulations that were applicable to the certification were FAR 23, dated February 1, 1965, including Amendments 23-1, 23-2, 23-3, and the special conditions referred to above.

Under the delegated option authority, the FAA did not review all the certification data but only such areas as they felt were necessary. Other certification data were prepared by the manufacturer and forwarded to the FAA for approval. The FAA did not witness any of the certification of the stabilizer system or the longitudinal trim system of the B-99.

Flight-test work under the delegated option authority was performed primarily by the manufacturer who then documented this work to the FAA. Normally, the FAA participated in flight tests only when a new regulation, with which the manufacturer had no previous experience, was being applied or when a previously uncertificated design feature was introduced. The trimmable stabilizer in the B-99 was a new design feature that had not been previously certificated by Beech.

A review of the E99 accident/incident history, as formally reported to the Board, indicated that the aircraft had been involved in 14 accidents or incidents as of June 1970. Of these occurrences, two accidents and one incident directly involved the longitudinal control of the aircraf't. The first pertinent accident occurred when the aircraft entered a very nose-high attitude after takeoff and, following an apparent loss of control, crashed on the airfield. Investigation of this-accident revealed that the horizontal stabilizer was in approximately the full aircraft nose-up position. The second pertinent accident was the Air South accident covered by this report.

The pertinent incident was an occurrence where the flightcrew initiated a descent and noted an unwanted "pitch over." The crew reported that both the primary and secondary pitch trim systems were ineffective during their attempt to recover. The crew found the primary trim circuit breaker tripped, and when it was reset, the aircraft responded to normal trim commands, and a recovery was completed after an altitude loss of about 900 feet. After trim control was recovered, the flight was completed without further incident.

The investigation following this incident revealed that the secondary trim actuator switches of the aircraft had a tendency to stick in other than the off position. This condition, if not corrected, caused an [illegible]ted trim input. The secondary trim actuator switches from two other B-99 aircraft were examined and found to be sticking in a similar manner, With the exception of these sticking switches, no other malfunctions were found.

As a result of various of these occurrences, the FAA has conducted flight test programs and investigated the handling characteristics of the B-99. The Board also conducted a simulator study of the handling characteristics of the B-99 in an effort to reproduce the flightpath of N844NS, as reported by the eyewitness, and in an effort to provide a rational explanation for the flightpath of the aircraft from its cruising altitude to the impact point on the ground.

The first flight test program conducted July 1, 1969, by an FAA team, reported that the airplane was unsafe when the longitudinal trim in an extreme position, and that the aircraft was in noncompliance with the applicable FAR's when that condition existed. The investigators recommended that a warning system be installed in the B-99 to alert the pilot should the stabilizer be positioned outside a Dredetermined satisfactory area for takeoff. The team also recommended-that action be taken to reduce the longitudinal control forces.

As a result of this report, a second team was formed to make an overall review of the problem with the intent of exploring means to provide an acceptable level of safety for the aircraft. This team was composed of pilots and aircraft systems and structures specialists who went to the Beech factory on July 9-10, 1969.

A series of test flights were flown by the team, with flights made at the fore and aft extremes of center of gravity and one flight made at, as near as possible, the loading and center of gravity believed to exist in the Air South accident. On the test vehicle in the accident configuration, the position of the horizontal stabilizer was at a zero angle of incidence in normal cruise. This condition left 5-1/2° of negative angle of incidence available to the pilot. When the test aircraft was trimmed to negative incidence of more than 2°, ". . . these resulted in very high pull stick forces (in excess of 75#); to attempt to place the stabilizer at its extreme negative incidence would require as much as 150# pull. If at this time the power were slowly reduced, one could not hold the nose up with both arms pulling.

"A further check showed that the limit of one hand controllability was reached in approximately four seconds. It also became apparent that, though a trim cut off was located on the wheel near the pilot's hand, he could not reach it with the fingers without letting go of the grip as the hand would have slid away from the top part of the wheel. This focused attention upon the design of the control wheel which tapers away from the trim control switches and also has a very moth porcelain finish

"It was also noted that the pilot does not have *any* ready indication that the trim control is in motion with the exception, of course, of either increasing stick forces to maintain attitude or a change in attitude."

Tests were also made to examine the problem of having the horizontal stabilizer in an extreme position for takeoff. It was found that if the trim were placed in the full noseup position, push forces of 20 to 25 pounds were sufficient to maintain normal climb speeds. With full nose-down trim, the force required to rotate the aircraft for takeoff was so high as to preclude rotation.

In the area of general controllability, the aircraft was reported to have excellent. response rates, and maneuver forces were low. However, during configuration changes without retrimming, the forces became difficult to hold with one hand. The position of the trim control on the pilot's wheel made it possible for the pilot to relieve forces immediately or to obviate the change in aircraft trim by retrimming as transient forces occurred. In this area, a qualified judgment by the team members resulted in the conclusion that the intent of the controllability requirements of the FAR's had not been met.

An in-flight examination of the automatic pilot authority in an out-of-trim condition resulted in a finding that the results were easily controlled and that the autopilot had very little authority. When the autopilot was suddenly disconnected, with the aircraft trimmed to a point where it deviated from the desired attitude, there was not a violent action on the part of the aircraft.

This team conducted a detailed evaluation of the design and operation of the longitudinal trim system. Their report stated that there was a high degree of integrity in the system and that no single system fault could be found that would induce a runaway trim. However, they did report some eccentricities in the system which they did not consider to be faulty design or hazardous in nature. It was shown that if the pilot and co-pilot opposed each other on the primary trim system, a dead short occurred that opened the circuit breaker, thus shutting down the trim system until it was restored by resetting the circuit breaker. It was also demonstrated that if a pilot were trimming with the secondary trim system, which trims at a much slower rate than the primary system, and the primary system were activated in the same direction, it would take over at a much faster rate of trim.

A third design feature considered in this portion of the investigation was the trim cutoff switch on the pilot's control wheel. This switch opened the primary trim circuit when it was depressed but when it was released, the circuit was restored and the trim would function again. If the pilot wanted to stop a trim action with this switch, as in the case of a runaway trim, he would have to hold the switch down until the system could be deactivated by some other means.

A "minor" design deficiency reported during this investigation was the location of the trim control relays which were mounted in an exposed position under the cockpit floor.

The review of the other aircraft systems established the need for additional protection in the wing flap drive system. It was possible for a single fault to cause the flaps to be driven up or down, and the team considered this to be an unsatisfactory condition.

Among other things, the team concluded that the trim changes with configuration change resulted in higher than desirable forces, but could be readily alleviated by the pilot because the trim control was on the control wheel. "Though not meeting the intent of FAR 23.145(b) in the estimation of the evaluators, the airplane can be safely controlled and has compensating features."

The team also concluded that: ". . . the pilot needed to be warned when the longitudinal trim system was not positioned within a safe band for takeoff and also informed by some positive unmistakable means when the trim was in motion; . . . the flap drive system needed redesign to preclude a single fault driving the flaps to an unwanted position; . . . the control wheel needed to be redesigned to provide a better grip and position for the trim cutoff switch; . . . the airplane flight manual should be amended to have specific limitations regarding dispatching with any malfunction in the primary or secondary trim system; . . . the flight manual should include procedures for checking the trim system operation prior to flight and appropriate turbulent air penetration procedures and; . . . that all training programs should be reviewed and assure that they objectively cover normal and emergency procedures to assure proper and safe operation of the airplane."

Recommendations were made to implement the above conclusions, and to consider adding a trim range restricting system to prevent excessive trim when flaps were up.

Because the Beech 99 longitudinal control forces were considered by the FAA to be excessive, the FAA wanted to limit the movement of the horizontal stabilizer from positions not normally required during cruise, climb, and let down. To delineate these limits, the FAA conducted additional flight tests August 27-29, 1969, in an effort to evaluate the pitch forces and pitch stability characteristics of the stabilizer-elevator combination.

Four' flights were conducted to determine the longitudinal control forces on the E-99. Longitudinal control force data were obtained at the forward, mid, and aft c.g. locations for out-of-trim conditions. The limiting factors for obtaining these data were either full travel of the stabilizer jackscrew or 150 pounds stick force. Data were also obtained on the effects of flap position and power control force changes.

An oscillograph was used to record aircraft speed, elevator control force, stabilizer position, and elevator deflection. Engine power (torque), propeller r.p.m., outside air temperature, pressure altitude, and indicated airspeed were hand-recorded and numbered for correlation with the oscillograph data.

As a result of these tests, the FAA concluded that the stick forces generated by changes in power, by themselves, were acceptable. However, power changes made in conjunction with upsets due to turbulence, inadvertent flap operation, or improper stabilizer position could induce additive control forces which could become uncontrollable.

Inadvertent operation of the landing flaps could result in stick forces which were high and could easily be confused with unscheduled stabilizer trim motion. The flap system was subject to single fault activation, did not have an automatic overload retraction device, and did not have adequate flap in motion warning.

Finally, they reported that excessive trim for the aircraft nosedown condition was available in the cruise configuration.

The nosedown stick forces could become "extremely large" by a combination of an out-of-trim effect combined with power reductions and the unstable slope of the longitudinal stability curve (the more the speed increased, the more the stick force increased).

Based on these findings, the FAA team recommended that: the airplane flight manual be revised to contain procedures outlining the effect that power changes and flap operations have on the aircraft and to provide an understanding of how these items could be used to help reduce control forces if an out-of-trim condition should occur; the leading edge up (aircraft nosedown) travel of the stabilizer be limited to approximately 3.5"; the longitudinal control system be redesigned to reduce elevator unbalance and stabilizer power, along with an investigation of the effects of downwash on the existing installation.

To implement these recommendations, the FAA has issued five Airworthiness Directives and Beech has issued Service Instructions, changes to the aircraft flight manual. and undertaken other corrective actions to comply with the recommendations of the Administrator. (See Appendix D.)

Since the trimmable stabilizer was found in the full airplane nose-@ position, and had been driven electrically to that position prior to impact, tests were required to determine the consequence of superimposing this trim setting on trimmed cruise flight conditions.

In an effort to evaluate the resultant induced motion of a B-99 as modified by variable pilot response and reaction times, a series of flights were simulated February 2-6, 1970, using the Beech Aircraft Corporation engineering flight simulator. "he simulator was programmed to represent a E99 in a 3° of freedom longitudinal simulation. In each of the simulated flights, a runaway of the primary trim system was Introduced after the aircraft had been trimmed for a Riven cruise condition.

Seven series of flights were simulated to attempt to determine the effect and significance of a number of variables affecting aircraft recovery subsequent to the introduction of a runaway of the primary trim system to the aircraft nosedown position. The variables introduced included pilot reaction time, type and sequence of pilot response, and the magnitude of pilot force inputs. In each flight series, the initial simulated cruise altitude and airspeed was 7,000 feet and 180 knots, except for the last two series where the airspeed was increased to 200 knots.

Table I beiow summarizes the nature of each flight series:

TABLE NO. I

Flight Series No.	Condition Imposed	Pilot Reaction/ Recovery Sequence	Approximate "Full Yoke" Reaction Time Range	Comments
1	Primary trim runaway	Reduce power, pull yoke	4-13 sec.	Test pilot A/S 180K
2	"	Full yoke, reduce power	0-2 sec.	Test pilot A/S 180K
3	"	Full yoke, stop trim runaway	1-2 sec.	Test pilot A/S 180K
4	"	pull yoke	1-10 sec.	Programmed response, stick force limit = 200 lb. A/S 180K
5	"	pull yoke	1-14 sec.	Programmed response, stick force limit = 400 lb. A/S 180K

6	Primary trim runaway	Pull yoke	1-8 sec.	Programmed respons stick force limit 200 lb. A/S 200K
7	"	Pull yoke	1-12 sec.	Programmed respons stick force limit 400 lb. A/S 200K

In the first series of tests, the pilot, at fixed time intervals after initiation of a runaway trim, responded by first reducing the power to flight idle and immediately thereafter pulling back on the control wheel. The horizontal stabilizer was driven from the trim position for cruising flight to the full nosedown position in about 10 seconds. Recoveries occurred in each of these simulations except the last one. In this case, the pilot waited 11 seconds to reduce the power and 1 or 2 seconds more before pulling the control wheel back. The recovery stick force was initiated at a time when the load factor imposed on the aircraft had reached a value approximating the negative static limit load of -1.32g. During the next few seconds, the stick force increased to 320 pounds and the load factor increased to a peak value of + 1.8g and then decreased asymptotically toward a limiting value of + 1.2g. In connection with this event, it was noted that, although the control wheel was held on the aft stop throughout most of the maneuver, with a maximum stick force of about 340 pounds, the maximum up-elevator recorded was about 6'. This was found to result from the fact that at pull stick forces in excess of approximately 130 pounds, the control cables stretch and preclude attainment of the elevator maximum static updeflection of 12°, + 1°, -0°.

The second series of tests attempted to isolate *any* significant effect on the recovery maneuver resulting from increasing the delay between the initiation of the trim runaway and the power reduction. These delays varied from 4 to 10 seconds. A recovery was safely accomplished in each case.

The third series of "flights" tested the effect of stopping the runaway trim by deactivating the primary trim system. The power was left at the cruise setting and recovery was made by turning off the trim system and applying back pressure on the control wheel. The trim was allowed to run away for periods of time that varied from 2 to 10 seconds, and recovery was safely effected in each case.

The fourth series of flights were conducted to better define the effectiveness of a given pull force input applied at various delay times after initiation of the runaway trim. The computer was programmed to impose a stick force that would increase from 0 to 200 pounds in a period of 2 seconds at prescribed intervals following the time of runaway and remain at this value throughout the recovery maneuver.

In those flights where the pull force was delayed 8 to 10 seconds, the flights terminated with impact. In both of these cases, the pull force was initiated with a negative load factor on the aircraft and the airspeeds exceeded the maximum allowable for the aircraft.

The fifth series was similar to the fourth series except that the programmed stick force response was increased to a maximum value of 400 pounds. In four of the eight tests conducted in this series, the trim actuator lower clutch limit value of 3,200 pounds was reached in less than 2 seconds following application of the programmed recover stick force.

When the time delays reached 12 and 14 seconds, the flights terminated in impact.

The maximum stick forces actually recorded during this series varied from 220 to 350 pounds. This difference between the programmed force and the recorded force was, as previously explained, a result of the elastic relationship between the stick force and the elevator deflection commanded.

The sixth and seventh series of tests were similar to series Nos. 4 and 5 except that the initial airspeed was increased to 200 knots rather than the 180 knots previously examined.

In series No. 6, with 200 pounds stick force programmed, delayed reaction in power reduction of 5, 6 and 8 seconds resulted in impact.

The maximum stick force was again limited to 200 pounds and was attained in each flight. This series demonstrated that the increase in initial airspeed decreased the critical reaction time. The maximum delay time available in series No. 4 was 6 to 8 seconds, but in series No. 6 it decreased to 4 to 5 seconds.

In series No. 7, with a maximum of 400 pounds of stick force programmed, delays of 10 and 12 seconds were terminated by impact. Again, the increased initial airspeed appeared to reduce the available reaction time.

2. ANALYSIS AND CONCLUSIONS

2.1 Analysis

Both pilots were properly certificated and had been trained for the performance of this flight. The aircraft was properly certificated and, according to the aircraft records, was in compliance with the existing airworthiness directives. The aircraft had been maintained in accordance with existing FAA and Air South requirements. The weight and center of gravity were within limits at takeoff and at the time of the accident.

There is no evidence that weather was a causative factor in this accident. The weather in the accident area was clear, visibility was in excess of 5 miles, and the winds on the surface were calm. The investigation of the weather indicates that, at the time of the accident, the aircraft was clear of clouds several miles south of some cumulus buildups, which were depicted on the weather radar as weak precipitation echoes. If turbulence existed in the area, it would have been light or possibly moderate.

The possibility that horizontal vortices extended outward from the cumulus buildups was examined but no evidence was found to support the existence of such a phenomenon. Other pilots who flew through the area reported that they had no difficulty in circumnavigating the buildups and encountered only light turbulence.

The flight of N844NS was apparently routine and without reported difficulty until approximately 2122. At this time, the aircraft had been cruising at 7,000 feet for approximately 11 minutes.

The ground witness reported that following a series of power reductions and reapplications of power, accompanied by sounds similar to backfires, the aircraft nosed over and descended nearly straight into the ground. He did not see any rolling, yawing, or other maneuvering of the aircraft as it descended, nor did he see anything separate from the aircraft. The aircraft disappeared behind trees which obstructed his view of the later portion of the descent.

The extreme destruction of the aircraft and its contents, and the depth to which the aircraft parts penetrated the hard soil are indicative of a high airspeed at impact. The symmetry of the craters and the limited throwout pattern are indicative of a very steep impact angle, approaching 90°, with little or no horizontal motion of the aircraft with respect to the ground.

Both wings failed, almost symmetrically, in a downward (negative) direction, with aft loading and leading edge down torsion. Loading of this nature is similar to that which is developed in a negative low angle of attack condition. This load condition occurs in the E99 at and above the limit dive speed of 283 knots calibrated airspeed.

In addition to the nature of the outer wing panel joint failures, the downward bow in both outer wing panels and the diagonal buckling observed on the left-hand wing panel upper surface and on the outer portion of the lower surface support the conclusions that the failure occurred under a negative low angle of attack condition.

The scoring behind the wing leading edge on the exposed surface of the front spar lower cap is consistent with a failure caused by high airspeed. This may be indicative of a failure of the leading edge caused by high aft loads and leading edge down torsional loads which occurred as a result of high airspeed. In fact, this yielding of the leading edge was probably the initial failure of the wing. Such a failure would rapidly increase the aft loading on the wing, causing total failure. In this connection, the manufacturer has estimated that an airspeed far in excess of the design diving speed would be required to cause such a failure.

All the major flight control surfaces were found in the primary wreckage area. Nothing was found which would indicate that the aircraft was not intact before the wings failed and separated from the aircraft. There was no evidence of fire in flight, or after impact, and no evidence of any significant prior structural damage. There was a possibility that control system malfunction or failures occurred but that the evidence of such a mishap was masked by the ensuing damage caused by impact. The Board believes, however, that this possibility was discounted by the terminal maneuver.

The terminal maneuver was a relatively gradual pitchover into near-vertical flight. king this maneuver, the airspeed increased beyond the limit dive speed of the aircraft, resulting in the yielding of the wing leading edges described above. Final separation of the wings occurred at low altitude, several hundred feet above the ground. This conclusion is based on the fact that the wing panels were travelling at a relatively slow speed when they struck the ground. The Board has calculated that the panels could slow from 300 knots to their terminal velocity of approximately 35 knots, in approximately 240 feet. The Board believes that the pitchover was relatively gradual since the wing failures did not occur during the initiation of the maneuver. Finally, the symmetry of the wing failures indicates that there was no appreciable aileron deflection at the time the wings failed. This is further supported by the eyewitness' statement that the aircraft did not roll or yaw during the descent, but rather just went straight down until it disappeared from his view behind trees.

Three possible causes of such a terminal maneuver have been considered by the Board. They were: flightcrew incapacitation; an upset due to some longitudinal instability; or a control system malfunction.

The condition of the pilots' bodies was such that no determination of their physical condition could be made by autopsies. Their medical records do, however, show that the only indication of a physical problem was the soft systolic heart murmur detected in the copilot during a regular physical examination. The examining physician discounted this murmur as having any clinical significance, and the pilot was certificated without a waiver being required.

It was impossible to determine whether the pilots had been incapacitated by overt action of some other person on the aircraft. In this connection, however, the Board's experience indicates that when acts of violence are committed aboard an aircraft, the investigation discloses evidence pointing toward such an act in the form of large insurance purchases, mental instability, etc., as well as evidence of the use of a weapon or explosive in the wreckage area. In this case, there were no weapons, other than one or more pocket knives, found in the wreckage area, nor was any explosive residue found. There was no evidence that anyone aboard the aircraft purchased any insurance prior to the trip. A background investigation has not revealed any evidence of emotional disturbance on the part of anyone aboard the aircraft.

Any theory concerning flightcrew incapacitation must account for the horizontal stabilizer trim position and must also take note of the lack of rolling or yawing of the aircraft during the terminal maneuver. Considering the spiral instability found in most aircraft, it is unlikely that this aircraft would have descended 6,000 to 7,000 feet without entering a spiral maneuver if its controls were left untended. Such a maneuver would have been observed by the witness and should have been apparent from the examination of the wreckage distribution.

The investigation has uncovered no evidence of longitudinal instability that would have presented a problem to the crew. Other carriers have reported pitch problems and porpoising of the aircraft in flight, but the E99 has not demonstrated any instability that could be considered dangerous.

Types of control malfunction which could produce a pitchover would be related to either the longitudinal trim system or to the primary longitudinal control system. Of the two, the trim system is the more critical, since that system is capable of producing greater pitching moments than the elevators.

The FAA flight-test programs indicated that the longitudinal control forces generated by the stabilizer were excessive and, as a result, the nosedown trim capability of the aircraft was restricted to 3.5". This program also reported that while stick forces generated by changes in engine power, by themselves, were acceptable, power changes made in conjunction with upsets due to turbulence, inadvertent flap operation, or improper stabilizer position could induce additive control forces which could become uncontrollable. Inadvertent operation of the landing flaps could result in high stick forces and could be confused with unscheduled stabilizer trim motion. User experience with the aircraft has shown that unscheduled trim actuator motion can be induced by sticking secondary trim system switches.

The fault analysis of the horizontal stabilizer actuator prepared by the manufacturer for the certification of the E99 indicated that no single fault in the actuator system could cause a failure of that system.

The fact that a sticking secondary trim actuator switch could cause the system to impose unwanted trim motion in the aircraft is considered by the Board to be a failure of the system. Beyond that, however, the fault analysis did not relate the actuator trim system to the performance of the aircraft or the crew. It was also noted that the analysis was not used to the maximum to train flightcrews in the operation of the aircraft. For example, the analysis indicated that if the pilots opposed one another on the primary trim system, the system would stop. Actually this condition would result in an opening of the trim system circuit breaker and deactivation of the trim system. This possibility was not brought to the attention of flightcrews before the Air South accident occurred.

Such an analysis was incomplete unless it was integrated with the aerodynamic characteristics of the aircraft in which the actuator was to be installed. Consideration of human response rates, recovery factors in time, and variables such as c.g., airspeed, etc., should have been included in the report. When used as a basis of oroof that the pitch system net the requirements of certification, the-analysis lacked several critical elements that should have been included.

The ramifications of possible faults of the trim system could only have been evaluated in the performance of the aircraft. Then corrective action could have been placed in effect on the aircraft, which would have eliminated some of the problems that arose when the system was evaluated in a flight test environment. Items such as the trim-in-transit Warning device might have been deemed necessary from the beginning. Other problem areas that affected the system, such as the poor placement of system switches, the inadequacies of the control grip, the possibility of contamination of switches, and others on which corrective action has already been taken, could have been anticipated and corrected before the aircraft was certificated.

The simulator study conducted under the auspices of the Board has indicated that high stick forces can be generated by out-of-trim conditions under certain circumstances. (See Appendix E.)

In comparing the simulator study results with the accident circumstances, the Board considered the following facts: a steep impact angle; a high airspeed at impact; a negative wing failure and separation at a relatively low altitude; a trimmable stabilizer at the electrical stop at the full aircraft nosedown position; a clean configuration at impact; and an idle engine power at impact.

The simulated impacts in simulator series Nos. 4 and 6 were characterized by physical factors and circumstances which appeared similar to those associated with the accident. Flights Nos. 6 and 7 of series No. 6 appeared to bear the closest correlation.

Based on an initial cruise condition of 7,000 feet and 200 knots, a set of simulation conditions, which resulted in an impact consistent with the factual accident evidence, consisted of:

a. A continuous operation of the primary trim actuator from the cruise trim position to the full airplane nosedown position.

b. A delay of about 6 seconds following initiation of the above trim change before starting recovery action at a load factor of -0.3g.

c. A yoke recovery force which increased linearly from zero to 200 pounds in 2 seconds.

With the gross weight and c.g. simulating the accident aircraft, and an initial airspeed of 180 knots, 10 seconds were required to move the stabilizer from the cruise trim to the full nosedown position. From an airspeed of 200 knots, the same amount of travel took 8.5 seconds. The testimony at the public hearing indicated that the rate of trim was nominally 2.5°/sec for the primary trim system and approximately 0.8°/sec for the secondary trim system.

The average pilot may not be able to detect rotational accelerations in any plane if the acceleration is less than $2°/sec^2$, unless some visual cue is available to assist him. The records of the simulator test flights indicated that, with a simulated primary trim system runaway, the rate of acceleration was at or less than $2°/sec^2$ for 1 to 2 seconds on each of the flights that most nearly simulated the accident flight. If a runaway secondary trim was assumed, at one-third the rate of motion, the acceleration in pitch would be about $0.8°/sec^2$. In this case, the acceleration would remain below the threshold of detection for a longer period of time.

There are some studies in existence which indicate that under rigidly controlled laboratory conditions, the detection threshold of angular accelerations may be as low as $0.4°/sec^2$, however, most literature on this subject refers to the threshold as $2°/sec^2$ 3/.

The Board has performed calculations in an effort to estimate the pull forces that an individual could apply to the control wheel of a B-99 aircraft. Note that the pull forces reported herein may not accurately reflect the pull forces that could have been exerted by the crew in this accident.

3/ AFM 51-37, April 1961; Aviation Physiology, USC 1956; W.J. Osterveld- Threshold Value for Stimulation of the Horizontal Semicircular Canals. Aerospace Medicine. 41(4) pp. 386-389, 1970.

The dimensions of the B-99 cockpit were correlated with information extracted from Table 96, The Human Body in Equipment Design, by Damon, published by Harvard University Press in 1966. Table 96, Forces Exertable on an Aircraft Control Wheel, reported the pull forces obtainable for various locations of the control wheel in relation to the Standard Seat Reference Point. The values in this table were based on the individual pull forces exerted by 33 college men subjects.

From Table 96, the mean and 95th Percentile 4/ pull force values for the right arm were obtained and these values were graphed. Interpolation of this graph provided the pull forces for the B99 control wheel, Standard Seat Reference Point distances. Similar treatment was given to the mean values of pull forces exerted by both arms used together. Table 96 did not present the 95th percentile values for simultaneous pull exertable by both arms. To obtain these values, the percent of increase from the mean to the 95th percentile values for the right arm was determined, and these percentage increases were applied to the mean values for both arms to give an estimate of the 95th percentile values for both arms. The latter value was graphed and interpolation of this graph provided the estimated pull force values for a crewman using both arms in the B-99 cockpit.

The following values of estimated pull by one pilot were calculated.

B-99 Wheel Position	B99 Seat Position	Distance-Wheel to Seat Ref. Point	Pounds pull Both Arms Mean	95th Percentile
Full aft	Full fwd.	12.25 inches	136	212
Full aft	Full aft	18.25 inches	182	302

These calculations represent an estimated range of pull forces exertable based on limited data. To estimate the 95th percentile values for forces exertable by simultaneous pull for both arms it was assumed that the percentage of increase from one arm mean to corresponding 95th percentile could be applied to the both arm mean and to the both arm 95th percentile values. It was also assumed that four data points would provide an accurate graph which could be interpolated. In addition, the physical capabilities of the pilots of N844NS were not known. Also unknown was the extent that their physical capability was affected physiologically by stress, or mechanically by acceleration. Finally, the seat positions used by the pilots were not known.

4/ Percentile - Value of the statistical variable that marks the boundary between any two consecutive intervals in a distribution of 100 intervals, each containing one percent of the total population.

Because of these factors and the limited data available for these calculations, the data reported above are coarse estimates of a range of pull forces a pilot might have been able to exert in this aircraft.

It appears most likely that the positioning of the stabilizer in the full nosedown position was not an action initiated by the crew.

Inadvertent flap extension, a longitudinal upset requiring full nosedown trim, or a nearly full nosedown trim condition cannot be supported by the record of this investigation. Experience has indicated that inadvertent flap extensions can be handled without undue trim application or control forces. There is nothing in the record to support a longitudinal upset, and a full nosedown trim condition for cruising flight at this accident weight and airspeed was not required, according to the testimony of the pilot who had flown the aircraft.

An unscheduled trim condition could have resulted from a malfunctioning primary or secondary trim system. In the case of the primary trim system, there is no history of such a malfunction. In the case of the secondary trim system, the record indicates that unscheduled trim conditions can occur.

6

The record indicates that it was a common practice to use the secondary trim system to make small trim changes when in cruising flight because of the slower rate of trim motion. The record also indicates that there was a history of sticking secondary trim switches. If such a condition occurred, the crew would have had a nosedown trim action at a relatively slow rate, one-third of the primary rate, with a corresponding slow change in g force. It is also possible that, with no visual cues due to the lack of a horizon in the existing weather and light conditions, or to distraction of the crew's attention, this change in pitch was below the level of perception until the g forces changed sufficiently to be sensed through the "deep muscle" source.

It was apparent from the simulator and flight tests that the time delay in initiating recovery action was critical. If recovery was started quickly, no major control difficulty was encountered by the pilots. If the initiation of control pull forces was delayed for an appreciable period, recovery became increasingly difficult and eventually impossible at the critical delay time.

A number of factors might have combined to cause a time delay in the crew response to an unwanted trim condition. These might have included the masking of the onset of unwanted trim motion by light to moderate turbulence; a lack of visual cues due to weather and light conditions; a lag in reaching the stabilized indicated airspeed corresponding to a given trim change; cockpit activities which diverted the crew's attention from the-aircraft attitude: or some malfunction of the powerplants that was attracting the crew's attention. Another possibility, demonstrated by an aircraft incident involving another B-99, was the possibility that the two pilots opposed each other on the primary trim system and caused the circuit breaker to open, thus rendering the primary trim system inoperative. The fact that this could occur was not widely known among pilots operating this type aircraft.

The crew of N844NS was apparently operating at its assigned cruising altitude at an airspeed between 180 to 200 knots, with both trim systems in the armed position. The aircraft, due to a sticking trim switch or Other condition cited above, entered into a nosedown attitude. It is most likely that based on the history of the unit, the cause of this change in attitude was a sticking secondary trim switch. Interruption of the trim motion by the trim cutout switch on the control yoke would apply only to the primary trim system. Thus, with a runaway secondary trim system, if the nose started down and the pilot attempted to interrupt the motion with the trim cutout switch, it would have no effect on the nosedown trim change. This would probably confuse the pilot and time would be required to analyze the condition, During this time, the trim would continue to run, increasing the stick forces required to counteract it. The airspeed would also begin to build up, increasing the load on the yoke. Within 5 to 8 seconds, the crew should have begun to exaerience visceral cues of a less than 1-g condition. It is likely that-up to this time, the pilot flying the aircraft, probably the captain, would have attempted to deal with the problem by himself. He was probably pulling back on the yoke and attempting to trim the aircraft with the primary trim system. If he then called for assistance from the copilot, or the copilot reacted to the nosedown condition without command, it is possible that the copilot's primary trim switch was activated in opposition to the captain's, thus opening the circuit breaker and effectively deactivating the primary trim system without their being aware of this fact. During this time, the secondary trim system would continue to operate until the electrical stop was activated and the trim action was terminated.

The total time consumed during this period would have been in excess of 10 to 15 seconds, depending on how quickly the nosedown trim condition was detected. By the time the crew could correctly assess their problem, the load on the trim actuator would have been within the clutch slippage range, and the crew would have been unable to retrim the aircraft to a noseup attitude. Their efforts to recover by back pressure on the control yoke would have been frustrated by the high loads imposed by the increase in airspeed. The aircraft would have continued to increase its velocity until the stick force's became too high for the pilots to hold, and when the back pressure was released, this imposed negative loads on the aircraft which failed both wings.

There are other conditions, as stated above, that could have initiated this sequence of events, but the one factor that must be common to all of them is a relatively long delay in the application of adequate noseup pitch command.

2.2 Conclusions

(a) Findings

1. Both pilots were properly certificated and qualified for the performance of this flight.

2. The aircraft was certificated in accordance with the existing FAA rules.

3. The aircraft records indicated that it had been maintained in accordance with existing FAA and Air South directives. There was no history of malfunctions in this aircraft that could be associated in a causal manner with this accident.

4. The weather was not a causal factor in this accident. Light to moderate turbulence might have existed in the accident area. Thunderstorms were in the area north, northeast, and east of the accident site. In the immediate accident area, there was no evidence of severe weather of any sort.

5. The flight was apparently routine and without unusual incident until approximately 2122. At that time, the aircraft had been cruising at its assigned altitude for about 11 minutes.

6. The aircraft was seen flying apparently straight and level, the power was reduced and advanced, and then reduced a second time. Following the second power reduction, the aircraft was observed entering into a descent which continued uninterrupted into the ground.

7. The aircraft was apparently being controlled to some degree because the witness did not observe any roll or yaw during the descent.

8. No evidence was recovered which would support a finding of interference with the flightcrew. There was no evidence of any medical condition that would have incapacitated the crew.

9. The wings separated from the aircraft in a negative direction, symmetrically, and simultaneously. The mode of failure indicates the application of a symmetrical negative load to both wings.

10. All the flight control surfaces were found in the primary wreckage area. There was no evidence of any structural failure prior to the wing failure. There was no evidence recovered of any malfunction or failure of the flight .control system.

11. The record of this investigation does not support a finding of longitudinal instability within the aircraft operating envelope.

12. The secondary longitudinal trim system had a history of sticking switches that could induce unwanted pitch trim changes. The primary trim system had no such history.

13. The fault analysis of the primary trim system actuator did not forecast the results of opposing primary trim applications or sticking secondary trim actuator switches.

14. The fault analysis of the trim actuator did not relate the ,actuator to its operating environment, the aircraft, or the flightcrew.

15. The horizontal stabilizer had the capability of producing greater pitching moments than could the elevator.

16. Flight tests and simulator tests have shown that excessive stick forces can be generated by out-of-trim conditions that result in higher-than-normal airspeeds. These stick forces can exceed the capability of one pilot, and in some cases two pilots, to control.

17. The simulator tests indicated that it was possible to put the the aircraft in a flight condition where the combined pull capability of the pilots would not be sufficient to effect a recovery.

18. At the time of the accident, published information was not available to flightcrews regarding certain undesirable design features of the longitudinal trim system.

19. Studies indicate that angular accelerations in any plane of less than $2°/sec^2$ may not be perceived by pilots without supplementary visual cues.

20. The angular acceleration in the vertical plane, generated by a runaway or sticking secondary trim switch condition, would be of less than $2°/sec^2$ for several seconds. Unprogrammed primary trim changes in the same plane could go undetected for 1 to 2 seconds, particularly if the initial motion was masked by light to moderate turbulence.

21. The time interval between the initiation of an unwanted trim motion and the initiation of corrective action was critical in determining whether the corrective action would be effective.

(b) Probable Cause

The Board determines that the probable cause of this accident was an unwanted change in longitudinal trim which resulted in a nosedown high-speed flight condition that was beyond the physical. capability of the pilots to overcome. The initiating element in the accident sequence could not be specifically determined. However, the design of the aircraft flight control system was conducive to malfunctions which, if undetected by the crew, could lead to a loss of control.

3. RECOMMENDATIONS AND CORRECTIVE ACTION

The testimony at the public hearing indicated that the FAA policy regarding the Delegated Option certification procedure was to accept certification data from the manufacturer and to review the data in the areas the FAA felt were necessary. The FAA also indicated that they participated in flight tests only when a new regulation was being applied to an aircraft, or when the manufacturer produced a new design feature that had not previously been certificated by them. The trimmable stabilizer in the E99 was such a new design feature, but the FAA did not participate in the flight testing of this item.

This type of stabilizer has been in use for a long period of time on various commercial and military aircraft, and the problems that were associated with it should have been well known throughout the industry. These problems have included excess stabilizer-up angle, runaway trim potential, and flight conditions where the elevator power might not be capable of overcoming the stabilizer power. Since this type of stabilizer has been in use, various devices have been incorporated in the systems to provide more information to the crew and to eliminate some of the known hazards that could evolve from its use. These devices have included audible warning of trim motion, stabilizer position indicators, restrictions to stabilizer-up angles, and published emergency procedures developed to deal with the results of various malfunctions in the system.

The Board notes that the modifications applied to the trim system of the B-99, since the accidents, are similar to those which have been previously applied to large aircraft.

The fault analysis used by the manufacturer and the FAA to certificate the longitudinal trim system of the E99 was reviewed and the Board concludes it was inadequate. As stated in this report, a fault analysis that did not consider the total operating environment was not complete. Therefore, the Board recommends that:

> The FAA review the existing fault analysis system and give consideration to requiring the completion of safety analyses in a manner similar to that required by Military Standard 882, System Safety Program for Systems and Associated Subsystems and Equipment; Requirements For.

These types of analyses should be applied to all aircraft offered for certification that can be used for the carriage of passengers for hire.

The Board recommends that the FAA take action to:

(1) require direct participation of FAA personnel in the certification of all newly designed aircraft components;

(2) review its aircraft certification system for possible procedural changes which would ensure that lessons learned in investigation of large aircraft accidents and incidents would be applied, when appropriate, to certification of small aircraft;

(3) bring recommendation (2) above, to the attention of those units within the FAA that are charged with the certification of mall aircraft.

CORRECTIVE ACTION

On August 1, 1969, the Board recommended to the Administrator, Federal Aviation Administration, that he take certain interim actions immediately. (See Appendix F.)

On August 6, 1969, the FAA replied that they were in the process of implementing recommendations resulting from a special evaluation of the B-99 conducted July 9 and 10, 1969. This implemented recommendation included some of the actions recommended by the Board. The Administrator believed that the actions being taken by the FAA would correct the trim system deficiencies. (See Appendices D and F.)

BY THE NATIONAL TRANSPORTATION SAFETY BOARD:

/s/ JOHN H. REED
Chairman

/s/ OSCAR M. LAUREL
Member

/s/ FRANCIS H. McADAMS
Member

/s/ LOUIS M. THAYER
Member

/s/ ISABEL A. BURGESS
Member

August 26, 1970.

INVESTIGATION AND HEARING

1. Investigation

The Board received notification of the accident at approximately 2300 e.d.t., July 6, 1969, from the Federal Aviation Administration. An investigating team was immediately dispatched to the scene of the accident. Working groups were established for Operations, Air Traffic Control, Weather, Structures, Powerplants, Human Factors, and Aircraft Systems.

Interested Parties included the Federal Aviation Administration, Air South, Inc.; Beech Aircraft Corporation; Pratt & Whitney Aircraft Division; and the Federal Bureau of Investigation.

The on-scene investigation was completed July 12, 1969.

2. Public Hearing

A public hearing was held at Atlanta, Georgia, October 14-16, 1969. Parties to the Investigation were: Federal Aviation Administration; Air South, Inc.; and Beech Aircraft Corporation.

3. Preliminary Report

A summary of the testimony which was taken at the public hearing was published by the Board on November 10, 1969.

CREW INFORMATION

Captain Erwin W. Wood, 38, was employed by Air South, Inc., as a captain, December 1, 1967. On the date of the accident, he had approximately 8,753 hours' flying time according to Air South records. He had 987 hours in the Beech 99, including 87 hours of instrument time and 273 hours of night time.

Captain Wood held a valid pilot's certificate, with an airline transport rating, No. 1161500 issued August 25, 1968. He also possessed a flight instructor certificate, airplane and instruments, issued June 26, 1969. Captain Wood had a first-class medical certificate issued May 26, 1969, with no limitations attached.

On February 17, 1969, he successfully completed a 6-month proficiency check in the E99 which was given by the chief pilot of Air South and observed by an FAA inspector.

Captain Wood was off duty for 48 hours prior to reporting for this flight.

The copilot, Thomas M. Wagner, 24, was most recently employed by Air South, March 10, 1969, as a copilot. He had previously been employed by Air South for approximately 4 months in 1968, but had resigned to accept employment with a scheduled air carrier.

Mr Wagner held a current pilot certificate, with an airline transport rating issued August 25, 1968. He was also a certificated flight instructor in rotorcraft, airplanes, and instruments.

He held a first-class medical certificate issued June 2, 1969, with no limitations.

Mr Wagner had not received any proficiency checks other than efficiency reports from captains with whom he had flown. All these reports indicated his performance was satisfactory or higher.

He had a total recorded flight time of 3,898 hours, including 254 hours in the B-99. This latter time included 15 hours' instrument time and 135 hours of night time.

Mr. Wagner was off duty 48 hours before reporting for this flight.

AIRCRAFT INFORMATION

N844NS, serial No. U-16, received a certificate of airworthiness, June 20, 1968. The aircraft had accumulated approximately 2,226 hours of operating time at the time of takeoff from Atlanta.

The aircraft records indicated that the aircraft had received a postflight inspection at 1930 on July 6, 1969, which included a review of the flight logs of the day's flying. The *only* discrepancies written up by the pilot of the last flight prior to the accident related to the turn indicator, the radio panel lights, and the DME. This pilot testified that he had detected no other discrepancies in the functioning of the aircraft or its systems during his flights, which totaled 6.3 hours.

The maintenance records of the aircraft were reviewed, and they indicated that all routine aircraft inspections had been conducted in accordance with Beechcraft inspection guides and had been properly signed off. The aircraft was being maintained in accordance with a progressive inspection system with an inspection cycle of 200 hours' operating time. After each 25 hours of aircraft time, a portion of the inspection was performed. In addition, a postflight inspection was conducted following each day's operation. Any discrepancies that were found during any inspection were to be entered on a monthly work order form for corrective action. Pilot complaints were to be entered in the flight logbook and were also carried over to the monthly work form for correction.

A review of the monthly work order showed that all the uncorrected discrepancies related to the flight instruments, radios, power levers, and lights. There were no outstanding discrepancies regarding the trim system, autopilot, flight controls, or other systems relating to the control of the aircraft in flight.

All required special inspections and airworthiness directives had been complied with on the aircraft.

The only recorded complaints regarding the flight controls or flight characteristics of the aircraft were made on May 15 and 16, 1969. On May 15, a pilot reported that the main trim ran much too slow. This complaint was corrected by the installation of a new trim actuator. On *May* 16, a pilot reported that the primary pitch trim was erratic in that it ran at varying speeds from "0" through "Normal." The operation was reported as erratic and unpredictable. On May 17, another primary trim actuator was installed and the aircraft operated until the time of the accident without further reported trim problems.

The aircraft was equipped with two Pratt & Whitney PT6A-20 engines with Hartzell Model HC B3TN-3B propellers. Both engines had 2,226 hours of operation recorded, and the propellers had a reported operation of 1,226 hours.

BEECH 99 CORRECTIVE ACTION ITEMS

NO.	RECOMMENDATION	CORRECTIVE ACTION
1	Develop a pilot oral examination.	Beech printed and distributed training information - and FAA provided it to the General Aviation District Offices (GADO).
2	Develop an out of trim warning system.	AD 69-24-2 effective 21 November 1969 and due in aircraft by 15 March 1970.
3	Develop a trim in motion warning.	AD 69-24-2 effective 21 November 1969 and due in aircraft by 15 March 1970.
4	Provide protection of trim system relays.	AD 69-18-7 effective 25 August 1969.
5	Relocation of trim cut-off switch.	AD 69-24-2 effective 21 November 1969 and due in aircraft by 15 March 1970.
6	Revise trim disconnect circuit from present "momentary off" to "permanently off" type.	Due to other changes such as protection of relay terminals, preflight check of system, rework of wheel grips and relocation of disconnect switch it was determined that the "momentary" feature was acceptable.
7	Restriction of trim range when flaps are up.	AD 69-18-6 effective 25 August 1969 provided Airplane Flight Manual (AFM) information relative to reducing control forces if an out of trim condition occurs. AD 69-24-2 effective 21 November 1969 restricted the stabilizer travel (leading edge up) to 3.5 degrees. It was determined that other restrictions were not required.
8	Revise trim control system to prevent a short when pilot/copilot activate opposing trim.	AD 69-24-2 effective 21 November 1969. To be installed by 15 March 1970.

PPENDIX E

NO.	RECOMMENDATION	CORRECTION ACTION
9 -	Provide protection for flap control system (or deactivate flap circuit from "after take off until needed for landing.")	AD 69-16-3 effective 18 July 1969 required insulation of flap follow up potentiometer in trailing edge of wing. Beech issued Service Instruction 0328-158 in March 1970 which added a positive flap "up" detent such that inadvertent operation is avoided and added an improved 30% setting detent. This Service Instruction was acceptable to FAA due to the modifications to the stabilizer trim system and due to the fact that we now consider inadvertent operation of the flaps to be controllable. Beech is to advise FAA of the number of compliance cards that are returned indicating the installation has been made.
	Long range redesign of longitudinal control system to reduce control forces.	See Item B.3. and 4. below.
	Develop tools/methods to check basic elevator contours on in-service aircraft.	Beech found they were unable to provide tooling of sufficient accuracy to assure acceptable response from the elevators. Beech tightened up the manufacturing and inspection tolerances at the factory. Also, there were established flight test procedures to be used when the elevators are replaced. This information was included in AD 69-24-2 effective 5 December 1969. The fix noted in items B.3 and 4 below is expected to further alleviate this elevator problem.

NO.	RECOMMENDATION	CORRECTIVE ACTION
12	Modify control wheel for better grip.	AD 69-18-8 effective 11 September 1969 required installation of handgrips.
13	Develop an AFM change to prohibit flight when any part of the trim system is inoperative.	AD 69-16-3 effective 18 July 1969.
14	Develop AFM change to require proper operation of trim system prior to flight.	AD 69-16-3 effective 18 July 1969.
15	Develop an AFM change to include 'operating procedures in turbulence.	AD 69-16-3 effective 18 July 1969.
16	Develop a training program to assure proper operation of all aircraft systems and procedures, including turbulence penetration, thunderstorm avoidance and use of weather radar.	Beech has printed and distributed necessary information and have included this informatic in their own training program. FAA has given this information to the GADO's.
17	Develop an AFM change to include proper instructions for loading and flight of aircraft within approved limits.	AD 69-16-3 effective 18 July 1969.

B. ITEMS AS A RESULT OF 14 AUGUST 1969 LETTER RE ADDITIONAL FLIGHT CHECKS:

NO.	RECOMMENDATION	CORRECTIVE ACTION
1	Provide operators with AFM procedures for the effect that power and flap operations have on this aircraft. They should understand how these items can be used to help reduce control forces if an out-of-trim condition occurs.	AD 69-18-6 effective 25 August 1969.

NO.	RECOMMENDATION	CORRECTIVE ACTION
2	Limit the travel of the stabilizer to approximately 3.5 degrees LEU, although this does not take away excessive forces (110-pound, Forward C.G.) in some cases it does eliminate the extremely high forces (160-plus pounds). Limit speed to 174 KIAS until the stabilizer travel is restricted to 3.5 degrees LEU.	AD 69-24-2 effective 21 November 1969.
3 and 4	Redesign the longitudinal control system so as to substantially reduce the longitudinal control forces for an out-of-trim flight. Redesign the stabilizer and elevator to reduce elevator unbalance and stabilizer power. Also, investigate the downwash effects on the existing installation.	Beech is currently testing the following modifications as a part of their Beech 99 certification: a. Metal Wedges on elevators Strips are installed on the aft inboard edge of the elevators which help alleviate oscillations along the longitudinal axis. These oscillations were caused by separation of air flow along the inboard.portion of the elevator trailing edge. This separation alternated between the upper and lower surface causing the elevator to oscillate. This aerodynamic characteristic of the elevator was greatly improved by adding wedges of sufficient thickness to be affected by the active airstream (i.e., where separation has not occurred).

NO.	RECOMMENDATION	CORRECTIVE ACTION

b. Mechanical Advantage in the longitudinal control system

The control wheel stroke has been increased along with internal repositioning of elevator control parts which according to Beech will reduce the longitudinal control forces.

FAA will test and evaluate as necessary upon receipt of completion of certification testing by Beech.

APPENDIX E

The following tables depict some of the pertinent data parameters recorded during the simulator tests.

TABLE NO. 1

Simulator Series No. 1	Flight No. 1	2	3	4	5
Time from trim runaway to power reduction (sec.)	3	4.7	6.8	8.6	11
Load factor at power reduction	+0.4	+0.2	-0.3	-0.6	-1.0
Time from trim runaway to "yoke pull" (sec.)	4	5.5	7.8	9.5	12.2
Minimum load factor	+0.1	-0.2	-0.6	-1.0	-1.3
Maximum load factor	+1.2	+1.5	+1.6	+1.7	+1.8
Maximum stick force - (lb.)	180	230	270	300	340
Maximum airspeed - (kt.)	180	185	213	240	290
Minimum deck angle - (deg.)	-12	-21	-39	-52	-73
Altitude loss - (ft.)	-600	-1000	-2900	-4700	-6100
Maximum compressive actuator load - (lb.)	1700	2600	3600	4400	6300+ (stylus pegged)

TABLE NO. 2

Simulator Series No. 2	Flight No. 1	2	3	4
Time from trim runaway to "yoke pull" (sec.)	1.5	1.5	0.6	0.5
Load factor at "yoke pull"	+0.8	+0.8	1.0	1.0
Time from trim runaway to power reduction (sec.)	4.0	6.0	8.0	10.0
Minimum load factor	+0.7	+0.8	+0.5	+0.8
Maximum load factor	+1.3	+1.4	+1.6	+1.4
Maximum stick force - (lb.)	180	160	160	170
Maximum airspeed - (kt.)	180	180	180	180
Minimum deck angle	-4.0	0	0	0
Altitude loss (ft.)	-150	0	0	0
Maximum compressive actuator load (lb.)	1650	1300	1350	1400

TABLE NO. 3

Simulator Series No. 3	Flight No. 1	2	3	4	5
Time from trim runaway to "yoke pull" (sec.)	1.1	1.4	1.6	1.5	1.5
Load factor at "yoke pull"	+0.8	+0.8	+1.0	+0.8	+0.8
Duration of trim runaway (sec.)	2.0	4.7	7.0	9.0	10.0
Minimum load factor	+0.8	+0.8	+0.8	+0.8	+0.8
Maximum load factor	+1.4	+1.3	+1.5	+1.4	+1.6
Maximum stick force - (lb.)	70	100	130	140	140
Maximum airspeed - (kt.)	180	180	180	180	180
Minimum deck angle	0	0	0	0	0
Altitude loss (ft.)	0	0	0	0	0
Maximum compressive actuator load (lb.)	600	800	1100	1200	1200

TABLE NO. 4

Simulator Series No. 4	Flight No. 1	2	3	4	5	6
Time from trim runaway to "yoke pull" (sec.)	1	2	4	6	8	10
Load factor at "yoke pull"	+0.9	+0.8	+0.4	+0.1	-0.2	-0.6
Minimum load factor	+0.8	+0.7	+0.4	0	-0.3	-0.6
Maximum load factor	+2.6	+ 2.5	+ 2.2	+ 1.8	+ 1.4	+ 1.2
Maximum stick force -(lb.)	200	200	200	200	200	200
Maximum airspeed -(kt.)	180	180	182	210	370	380
Minimum deck angle (deg.)	0	-1	-8	-20	-56	-68
Altitude loss (ft.)	0	0	-150	-800	-6100	-6100
Maximum compressive actuator load (lb.)	2000	2100	2200	2400	3800	3900

TABLE NO. 5

Simulator Series No. 5	Flight No. 1	2	3	4	5	6	7	8
Time from trim runaway to "yoke pull" (sec.)	1	2	4	6	8	10	12	14
Load factor at "yoke pull"	+0.9	+0.8	+0.4	+0.1	-0.2	-0.6	-0.7	-0.8
Minimum load factor	+0.8	+0.8	+0.4	0	-0.3	-0.7	-0.7	-0.8
Maximum load factor	+2.6	+2.7	+2.6	+2.4	+2.0	+2.1	+2.2	+2.2
Maximum stick force - (lb.)	220	220	220	240	265	320	350	350
Maximum air-speed - (kt.)	180	180	183	195	228	295	370	380
Minimum deck angle (deg.)	0	0	-8	-18	-32	-48	-64	-76
Altitude loss - (ft.)	0	0	-50	-400	-1200	-3600	-6100	-6100
Maximum compressive actuator load (lb.)	2400	2400	2600	3000	4100	6400+ (stylus pegged)	11,000	11,500

TABLE NO. 6

Simulator Series No. 6	Flight No. 1	2	3	4	5	6	7
Time from trim runaway to "yoke pull" (sec.)	1	2	4	4½	5	6	8
Load factor at "yoke pull"	f0.8	f0.6	f0.2	+0.1	0	-0.2	-0.8
Minimum load factor	f0.8	j0.6	40.2	0	-0.1	-0.4	-0.8
Maximum load factor	f2.5	42.2	f1.8	f1.6	+1.6	41.2	+1.0
Maximum stick force- (lb.)	200	200	200	200	200	200	200
Maximum airspeed - (kt.)	200	200	216	220	235	380	390
Minimum deck angle (deg.)	-2	-5	-13	-16	-44	-65	-73
Altitude loss (ft.)	0	-100	-500	-1600	-6100	-6100	-6100
Maximum compressive actuator load (lb.)	2000	2000	2380	2250	2200	3500	3500

TABLE NO. 7

Simulator Series No. 7	Flight No. 1	2	3	4	5	6	7
Time from trim runaway to "yoke pull" (sec.)	1	2	4	6	8	10	12
Load factor at "yoke pull"	+0.9	+0.6	+0.2	-0.2	-0.8	-1.0	-1.1
Minimum load factor	+0.8	+0.6	+0.1	-0.3	-0.8	-1.0	-1.2
Maximum load factor	+3.2	+3.0	+2.6	+2.2	+2.0	+2.0	+2.0
Maximum stick force - (lb.)	240	240	250	275	325	360	360
Maximum airspeed - (kt.)	200	200	208	232	297	380	390
Minimum deck angle (deg.)	-1	-4	-13	-25	-40	-58	-76
Altitude loss (ft.)	0	0	-250	-900	-3300	-6100	-6100
Maximum compressive actuator load (lb.)	3000	3000	3500	4250	7000	11,500	11,750

DEPARTMENT OF TRANSPORTATION
NATIONAL TRANSPORTATION SAFETY BOARD
WASHINGTON, D.C. 20591

OFFICE OF
THE CHAIRMAN

August 1, 1969

Honorable John H. Shaffer
Administrator
Federal Aviation Administration
Washington, D. C. 20590

Dear Mr Shaffer:

We are presently investigating two major accidents, as well as several incidents, involving Beech Model 99 aircraft. All of these cases involved the loss of longitudinal control during flight.

We are aware of the communications and directives issued by the Administrator regarding this subject in an effort to preclude further occurrences involving the subject model aircraft.

In view of the latest occurrence near Atlanta, Georgia, on July 29, 1969, involving again a longitudinal control problem in a Beech 99 aircraft, and the known potentially serious consequences of such occurrences, we feel that further action in this matter is indicated.

Based on the evidence indicating involvement of the pitch trim systems in the aforementioned accidents and incidents, it is recommended that the following interim action be taken immediately:

(1) Emergency recovery procedures should be established to effect timely recovery from unwanted and/or adverse longitudinal trim conditions. Such procedures should be incorporated as part of the "Emergency Section" of the FAA approved Flight Manual and include specific reference to ensuring the pitch trim circuit breaker is engaged whenever trim is attempted. Consideration should also be given to the use of flaps as an emergency method to induce nose-up pitch, should unanticipated pitch over occur and the stick forces become heavy.

(2) Provide a stabilizer "In-Transit" warning system to alert the flight crews of movement of the trim system.

(3) Consider imposition of appropriate restrictions to the stabilizer trim range in order to prevent excessive nose-down trim while the aircraft is in the cruise configuration.

General aspects of the above recommendations have been discussed with personnel of your Flight Standard Service and our Bureau of Aviation Safety Staff.

Please feel free to contact us if further information is desired.

Sincerely yours,

/s/ Louis M. Thayer
Acting Chairman

DEPARTMENT OF TRANSPORTATION.
FEDERAL AVIATION ADMINISTRATION
WASHINGTON, D.C. 20590

OFFICE OF
THE ADMINISTRATOR

6 AUG 1969

Honorable John H. Reed
Chairman, National Transportation Safety Board
Department of Transportation
Washington, D.C. 20591

Dear Mr. Chairman:

This is in reply to your letter of 1 August 1969 regarding service difficulties with Beech Model 99 longitudinal trim system and your recommendations thereto.

The Central Region and the Beech Aircraft Corporation are presently in the process of implementing the recommendations which resulted from a special evaluation of the Beech Model 99 conducted on 9 and 10 July 1969 at the Beech Aircraft Corporation plant, Wichita, Kansas. Among these recommendations were included some of those which you also have brought to our attention. Action taken as a result of these recommendations will, we believe, correct the trim system deficiencies.

The interim actions taken to preclude repetition of the service difficulties reported include an operations alert bulletin issued on 9 July 1969 which specified complete trim system preflight check procedures. This bulletin was further revised and refined by a bulletin issued 14 July 1969. Finally, as a result of the incident which occurred near Atlanta, Georgia, on 29 July 1969 a Flight Standards operations alert telegram was issued on 1 August 1969 to advise the pilots of the fact that opposing one another on the primary trim system will trip the circuit breaker and that this possibility should be checked prior to attempting to use the secondary system. It also specifies that the secondary trim system be off prior to takeoff and should not be in use when the primary system is in use. We believe this interim action will alleviate the .problem of sticking secondary switches until retrofit can be accomplished. All of the above procedures will be included in a revision to the approved flight manual.

Procedures for the use of flaps to control pitch forces at speeds below V_{FE} already are included in the flight manual. Structural considerations prohibit the use of flaps for this purpose at speeds above 175 knots where the aircraft normally cruises,

We already had informed Beech Aircraft Corporation that an in-transit trim warning system will be required, and are currently considering the feasibility of adding a trim-range restricting system to prevent excessive trim when flaps are up.

We trust that this information satisfactorily replies to the recommendations in your letter. We will appreciate any additional information you may receive on this subject.

Sincerely,

J. H. Shaffer
Administrator

NTSB
AAR
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