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CIVIL AERONAUTICS BOARD

AIRCRAFT ACCIDENT REPORT

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BRANIFF AIRWAYS, INC., LOCKHEED ELECTRA, N 9705C,
BUFFALO, TEXAS, SEPTEMBER 29, 1959

SYNOPSIS

A Braniff Airways Lockheed Electra, Model L-188A, N 9705C, broke up in flight and was further destroyed by ground impact and fire 3.19 miles east-southeast of Buffalo, Texas, on September 29, 1959, about 2309 c. s. t.. All occupants, 27 passengers, six crew members and one company employee, were killed.

Flight No. 542 of September 29, scheduled between Houston, Texas, and New York International Airport, with stops at Dallas, Texas, and Washington, D. C., departed the Houston Airport at 2237. The flight reported to San Antonio Center over the Leona VOR at 2305 at an altitude of 15,000 feet. It then made its final radio contact with company radio at 2307.

Structural failure of the aircraft occurred at approximately 2309 while on course to the next fix, Trinidad intersection. Weather at the time and flight altitude was good with scattered clouds above 20,000 feet and with visibility of 10-15 miles. A review of all records and crew reports indicated a routine operation from Houston, except that upon departure a terminal strip on No. 3 propeller was not properly bonded and the No. 3 fuel tank sump pump became inoperative shortly after takeoff.

The probable cause of this accident was structural failure of the left wing resulting from forces generated by undampened propeller whirl mode.

Investigation

The Flight

Flight 542 departed the ramp at Houston at 2237 [1], 22 minutes behind schedule with a total of 34 persons including a crew of six consisting of Captain Wilson Elza Stone, First Officer Dan Hollowell, Second Officer Roland Longhill, and Stewardesses Alvilyn Harrison, Betty Rusch and Leona Winkler, none of whom survived. The delayed departure was due to a mechanical discrepancy involving No. 3 generator. This generator was inoperative on arrival of N 9705C at Houston. Prior to departure from Houston the Nos. 3 and 4 voltage regulators were interchanged.

Actual gross weight upon departure was calculated at 83,252 pounds, including 17,000 pounds of fuel, and was 16,548 pounds less than the authorized gross weight

[1] All times herein are central standard time based on the 24-hour clock.

of 99,800 pounds. The estimated time en route to Dallas was 41 minutes.

The flight was given an instrument-flight-rules clearance which was to the Leona omni, via Victor Airway 13 west to the Gulf Coast intersection, direct to Leona, to maintain 2,300 feet altitude to Gulf Coast, then to climb to and maintain 9,000. At approximately 2240 the flight was cleared for takeoff and at 2242 it reported ready for takeoff and was airborne at 2244.

After takeoff Houston departure control advised that it had the flight in radar contact and requested it to report when established outbound on the 345-degree radial of the Houston omni. Flight 542 complied and subsequently was cleared to 9,000 feet and advised to contact San Antonio Center on 121.1 mcs. upon passing the Gulf Coast intersection.

Flight 542 reported to company radio at 2251 as blocking out of Houston at 37, taking off at 42, to cruise at 15,000 feet when so cleared, estimating Dallas at 2325, and that the Center had this information. At approximately 2252 Flight 542 reported to San Antonio Center as being over Gulf Coast intersection at 9,000 feet. The flight was then issued its destination clearance to the Dallas Airport via direct to Leona, direct to Trinidad, direct to Forney, direct to Dallas, to maintain 15,000 feet. The flight was cleared to climb to its cruising altitude.

The next transmission from Flight 542 was to the San Antonio Center, giving the time over Leona as 05 at 15,000. San Antonio Center acknowledged, and requested Flight 542 to change over and monitor the Fort Worth frequency of 120.8 mcs. at this time. The flight acknowledged.

Shortly thereafter Flight 542 contacted company radio with a message for maintenance, advising that the generators were then OK but that there had been insufficient time for maintenance to insulate the terminal strip on No. 3 propeller at Houston and it would like to have it done in Dallas. At this time the flight also said it would give the communication center a Dallas estimate of 25. This was then followed by one other item for maintenance, which was that No. 3 sump pump was inoperative. This was the final transmission from the flight and was logged as completed at 2307.

Structural failure of the aircraft occurred at approximately 2309 on course to the next fix, Trinidad intersection. The radial from Leona omni to Trinidad intersection is 344 degrees. The main wreckage was located 19.7 miles [2] north of Leona omni, 3.19 miles east-southeast of Buffalo, Texas. The time, 2309, correlates closely with the information obtained from witnesses to the accident as well as the time indicated on impact-stopped watches recovered at the scene.

Investigation disclosed that there were no radar or radio contacts established with Flight 542 nor were any emergency calls received on guard frequencies or en route frequencies after 2307.

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[2] All mileages herein are in nautical miles.

Investigation also disclosed that there was no known traffic which could account for a violent evasive maneuver in the immediate vicinity of N 9705C at the time of the accident, nor were there any missiles or unmanned aircraft in the region, according to the Department of Defense.

Weather

Surface weather charts for the late evening of September 29, 1959, and early morning hours of September 30, 1959, show a very weak pressure gradient from southwestern Texas east-northeastward to western Alabama. A diffuse quasi-stationery front along the Appalachians reached into central Mississippi and extended along a line running from near Shreveport, Louisiana, to Fort Worth, Texas, thence southwestward to Junction, Texas, and west-southwestward to the Mexican border. The leading edge of this front was some 125 miles to the north of the crash site at the time of the crash.

The route from Houston to Dallas was 60 miles or more east of an area in which U. S. Weather Bureau forecasts called for severe thunderstorm activity. For this route the aviation area forecast, issued at 1852 by the U. S. Weather Bureau at San Antonio, indicated scattered clouds at 4,000 to 5,000 feet and a broken ceiling at 10,000 feet in the vicinity of a few isolated dissipating cumulonimbus, mostly over extreme southern Texas, until 2100 and broken to scattered clouds above 10,000 feet elsewhere on the route. Additionally, the forecast indicated that low stratus, scattered to broken at 1,500 feet with its tops at 5,000 feet along the coast, was expected to spread inland and lower to broken to overcast 1,000 to 2,000 feet by 0200 locally 800 to 1,000 feet, overcast; visibility five miles in fog over interior sections after 0200 September 30, 1959.

During the early evening of September 29, 1959, weather reports show that over southeastern Texas there was generally 1/10 to 5/10 of altocumulus clouds at 12,000 feet; 6/10 to 9/10 of cirrus clouds above 20,000 feet; and a few isolated dissipating cumulonimbus with bases around 4,000 feet. A small area of locally heavy thunderstorms, which developed near San Angelo at 1500 and moved east-southeastward, had reached the vicinity of Kerrville and Fredericksburg, Texas, by 2200 decreasing to 10 to 15 miles in diameter and was dissipating in the vicinity of Blanco, Texas, around 2200. At 2200 significant, but isolated, radar echoes were reported southeast of Shreveport, Louisiana, and thunderstorms were visible from Lufkin, Texas. At the same time, lightning from the thunderstorm area near Blanco was visible from Waco and Austin. College Station, Tyler and Gregg County were clear with visibility 15 miles, while Houston reported only high thin cirrus clouds above 20,000 feet. Dallas had scattered clouds at 12,000 feet in addition to the high thin cirrus deck.

By 2300, surface observations and radar reports showed almost all thunderstorms to be dissipating except for an area extending from 25 miles north-northeast to 25 miles north-northwest of Waco. A second area showing on the radar scope at Waco was five miles wide and extended from 30 miles west-southwest of Palestine, Texas, to 30 miles east-southeast of Waco. The latter report places a radar echo approximately eight miles northwest of Buffalo, Texas, and is interpreted as being a rainfall echo, most likely the intermediate and higher level remnants of earlier thunderstorms moving from the west-southwest. A Grumman Mallard pilot en route from Dallas to Houston later reported encountering intermittent light rain and moderate turbulence at 7,000 feet northwest of Buffalo, Texas, and observed shallow buildups, estimating the tops to be about 10,000 to 12,000 feet. He also reported distant lightning to the west of course. This pilot reported that no weather was encountered south of the Leona intersection, which is about 23 miles south and slightly west of the accident site. The pilot of a military C-47 flying from Shreveport to Houston some 80 or more miles east of the accident site reported in smooth air at 6,500 feet and that the weather was clear between the C-47 and what was later determined to be the burning airliner.

According to groundwitnesses in Buffalo, Texas, and the immediate area at the time of the accident, skies were partly cloudy, the visibility was good, and no lightning was observed. Shortly after the accident a few light showers were observed in the Buffalo area, but not at the accident site. By midnight the only thunderstorm in the area was located by surface and radar observation within 30 miles of the northwest of Waco.

While the flight crew of Braniff Flight No. 542 did not receive a preflight weather briefing at the Houston office of the U. S. Weather Bureau, company meteorologists provided the flight with current and forecast weather information for the route and terminals concerned.

Witnesses

All passengers aboard the aircraft when it arrived at Dallas from Chicago as trip 61/29 who could be contacted by telephone were later questioned. No unusual incidents which could be directly related to this accident were revealed.

Every known witness who either heard the aircraft at the time of difficulty or observed the fire in the sky was interrogated. Statements were obtained from all who were considered able to contribute to the investigation.

Witnesses reported hearing various noises of different intensities and of different pitches. Many of the sounds were likened to known noises such as the "clapping of two boards together," " the sound of thunder," "the roar of a jet plane breaking the sound barrier," "whoosing screaming noise," "creaking noise of a bulldozer," and "awful explosion."

The majority of witnesses observed the large fire in the sky. The geographical position of this fire was established at a location considered to be accurate within one-half mile. The elevation of the fire ball was calculated from information provided by three witnesses who were judged the best source for this information. Each had a fixed reference point for establishing the angle of elevation. This effort resulted in a minimum altitude of 17,000 feet and a maximum of 23,000 feet.

One witness stated that he observed a white light prior to hearing the unusually loud noise from the aircraft or observing the fire ball.

Following this accident, twelve known unusual noises such as jet aircraft, sonic booms, propellers at supersonic speeds, Electras cruising normally, and intentionally random noises were put on tape. This tape was played back to witnesses individually in an attempt to identify more closely the noise associated with the accident. None of the witnesses had been apprised of the source of the sounds they were about to hear. The net result of this effort was to liken the noises to those of propellers at supersonic speeds and/or those of jet aircraft.

Wreckage Distribution and General Damage

The wreckage was distributed within a long, narrow ellipse, the major axis of which was approximately coincident with the 344-degree radial of the Leona omni. The first item found at the southern edge of the wreckage pattern was a nine-inch section of hydraulic line from the left heat exchanger, and its position was fixed as 17.4 miles north of the Leona station. Proceeding northerly from this point toward the main wreckage, the major components were located in this order: No. 1 propeller and gearbox; left wing (including No. 1 engine and the No. 2 powerplant); No. 4 powerplant; left outboard stablilzer section; right outboard wing panel; followed by the main wreckage area consisting of fuselage, empennage; No. 3 powerplant, and right wing stub.

The wreckage was strewn for a total distance of 13,900 feet from the first recovered item to the nose crater, with some lateral spread of the debris, due in part to wind effect, the lighter pieces being generally east of the more dense ones. Two parts of high density, and therefore subjected to only slight trajectory deviation, were the No. 1 propeller and gearbox package and the No. 4 powerplant. The direction between these was 341 degrees, magnetic.

At the main area, 3.19 miles from the highway intersection in Buffalo and on a bearing of 92.75 degrees from that intersection, there were three basic concentrations of wreckage, one around the nose crater, one at the center section crater, and one at the tail cone. In addition, there was a wide scatter of aircraft parts and debris. Light material, such as paper, plastic, and insulation was found as far away as a half-mile to the north and northeast.

The material at and west of the nose crater was, without exception, identified as fuselage and fuselage-contained components from the nose to fuselage station 570. This debris covered an area of about 20,000 square feet of open, plowed ground. The nose crater, about four feet deep, was at the easternmost end of the area, and the fuselage material was fanned out westward for a distance of 200 feet. Approximately 90 percent of the forward fuselage was in crushed sections of two feet square or less.

The second concentration was approximately 200 feet northeast of the nose crater and in a heavy growth of scrub oak. The material in this vicinity consisted of the center section, right wing fragments, the No. 3 powerplant, rear cabin structure, and components related to these portions of the airframe. The direction of travel here at the time of impact was 320-340 degrees as indicated by tree breaks and ground furrows.

The tail section was located 250 feet northwest of the center section, with the rudder and elevator control cables lying across the tops of the intervening trees.

The trees between the three areas were undisturbed except at localized points where loose objects had passed through the branches.

Systems

The cockpit of the aircraft was almost totally demolished. Those portions which were recovered were found at the bottom of the crater made by the nose of the aircraft. These were damaged so extensively that few of them were even recognizable. Nevertheless, great effort was put into studying this debris, including the remnants of flight and powerplant instruments, but this study yielded no information significant in establishing the cause of the accident.

The recovered flight engineer's log sheet for 2250 showed that the altitude at that time was 7,000 feet; indicated airspeed 210 knots; indicated outside air temperature +27 degrees, centigrade; engine and airfoil anti-icing off; and engine instruments indications were normal. The flight engineer's log sheet for 2300, which was recovered, indicated altitude 15,000 feet; indicated airspeed 275 knots; indicated outside air temperature +15 degrees, centigrade. The engine and airfoil anti-icing systems were off; engine instrument indications appeared normal. None of these readings indicate any abnormality.

Damage to the airframe had been so great that no aircraft system, as such, survived. In addition, impact and fire had destroyed or damaged individual systems components to the extent that functional checks were generally impossible. As a result, a considerable amount of time was devoted to identifying, listing and describing the damage received by systems components. It was deemed advisable to disassemble certain components thought capable of yielding useful information and, in a few instances, functional checks were possible and were performed.

The following aircraft systems were examined to the extent possible: hydraulic, electrical, radio, air conditioning, instrument and autopilot, control surface booster, air start, fire extinguishment, oxygen, fuel and anti-icing.

No indication of operational distress was found through examination of the hydraulic and electrical system components. The left inboard main landing wheel had been involved in considerable fire and it was dismantled to permit inspection of the brake assembly. No abnormal heat patterns were noted such as might be expected from excessive braking action.

No evidence of fire or overheating was noted during inspection of the recovered radio components, all of which had suffered extensive impact damage. Examination of transmitters and receivers revealed the following estimated settings:

No. 1 VHF Transmitter - 130.5 mc.
No. 2 VHF Transmitter - 120.7 <u>or</u> 120.8 mc.

No. 1 VHF Communications Receiver - 130.5 mc.
No. 2 VHF Communications Receiver - 120.7 mc.

No. 1 VHF Navigation Receiver - 110 + mc. (tenths could not be determined)
No. 2 VHF Navigation Receiver - 110.8 <u>or</u> 112.8 mc.

No. 2 Omni Bearing Indicator - 166 degrees

No. 1 ADF Receiver - 365 kc.
No. 2 ADF Receiver - 540 kc.

All recovered items of the air conditioning system received extensive impact damage with the exception of the outflow control valve. The majority of air duct had been destroyed; however, inspection of recovered duct sections and the outflow control valve disclosed no indication of smoke or fire damage.

Only three items associated with the autopilot system were recovered and these were badly crushed. All recovered instrument system components were destroyed by impact with the exception of the two fluxgate compass transmitters.

The control surface booster assemblies had suffered moderate impact damage, which prevented their being tested as complete assemblies. However, individual components capable of operation were given functional tests and those which could not be tested were dismantled and examined in detail. All discrepancies noted were attributed to crash impact damage with the exception of a failed electrical lead at the load sensor of the elevator booster assembly.

The load sensor was subsequently examined by the National Bureau of Standards whose report states, in part, "The break in the stranded wire in the sensor unit was probably caused by several cycles of reversed bending, rather than by a single tensile or bending load."

The left air compressor assembly of the air start system was recovered at the left wing impact site. The compressor had been consumed by fire; only an ash residue remained which readily broke and flaked away when the assembly was removed for shipment. The right compressor assembly was demolished by impact but showed no evidence of fire. Both of the right air bottles remained intact in the No. 4 nacelle and still retained an air charge of unknown amount, which was released as a safety measure before removal of the wreckage. Both of the left air bottles were also recovered. One was found separated from the wing structure at the impact site. It was slightly dented and the air lines had been torn off at the flanges. There was no evidence of fire. The second bottle was still in position in No. 1 nacelle.

The fire bottles of the No. 3 nacelle had not been discharged electrically but they had been broken by impact forces and contained no extinguishing agent. The bottles from the No. 2 nacelle had been involved in fire and both discharge heads had been fired. These were examined by the manufacturer who concluded that the outboard unit probably was discharged as a result of thermal discharge of the actuating cartridge and that the inboard unit was discharged through the safety disc by excessive pressure resulting from the fire, the actuating cartridge being subsequently discharged by action of the fire. Both rotary selector valves of the extinguishing system were recovered and their internal porting was determined to have been normal.

Two oxygen bottles (1800 p.s.i.) were recovered minus their regulator caps, which had been broken off. One crew bottle (39.4 cu. ft.) was recovered only slightly damaged and with its valve in the open position. The flight engineer's oxygen panel was found badly crushed. The oxygen mask was still attached to the regulator. No body tissue was found in or around this mask. Five additional masks were recovered in a torn condition with face glasses missing; none of these had evidence of human tissue on their inner surfaces.

The four fueling valves were functionally checked and then dismantled and inspected. Each valve required replacement of its impact damaged solenoid, after which its mechanical functioning was found to be within operating limits. Inspection disclosed no defects or abnormal wear. No significant contaminants were found within the valves.

The pilot valves from tanks 1, 2, and 4 were recovered. Only the No. 4 valve could be tested and it functioned normally. No signs of pre-existing abnormalitie were noted during examination of these valves.

The fuel dump valves were impact damaged but were in the closed position. One tank shutoff was found in the closed position and one emergency shutoff valve was found in a partially closed position but was free to move. Three electrically operated shutoff valves were found minus their motors. One was closed and two were open.

The fuel gages of the fueling panel were examined by the manufacturer who determined their final indications by two separate techniques, one of which involv gear measurements. The results of this method are considered to have an estimated accuracy of plus or minus 62 pounds and were as follows:

No. 1 tank - 3,960 pounds, No. 2 tank - 3,610 pounds

No. 3 tank - 4,080 pounds, No. 4 tank - 4,080 pounds

Insufficient recovery was made of anti-icing system components to provide any useful information.

Powerplants

A great amount of the powerplant investigative effort was directed toward determining if a failure or malfunction of any of the engines, propellers, or their associated systems had contributed to or caused the accident. This activity covered the following areas:

1. Oil systems for significant contamination.

2. Propeller reduction gear and accessory drive systems for gear and/or bearing failures.

3. Torquemeters for rotational interference.

4. Power section rotors for over-temperature indications, bearing failures or rotor failures.

5. Fuel pumps and fuel controls for failures.

6. Propeller pitch change mechanisms and controls for failure.

Detailed examinations in these respects did not reveal any evidence of failure or malfunction of the powerplants prior to the start of the separation of the No. 1 engine power section at the air inlet housing to the compressor split line.

The investigation of the powerplants as well as other investigations revealed specific items which warrant discussion.

Some witnesses reported hearing noises (this subject will be detailed later in this report), which from their various locations suggested possible engine overspeeding. Examinations of the engines and propellers were made in detail for overspeed evidence of the kinds that were noted during development tests. The first evidence of overspeed from tests, perceptible turbine and compressor tip diameter growth and resultant compressor tip rub, occurs at 20 percent overspeed (16,600 engine r.p.m.). At increasingly higher overspeeds, compressor tip rub is more pronounced, turbine blade tip rub and some bearing distress becomes evident. No measurable growth of turbine or compressor diameters or bearing distress of the kind associated with overspeed was noted. Based on propeller development work, the first evidence would be brinelling of the blade bearing races and it would occur at about 53 percent (21,120 engine r.p.m.) overspeed. Forty-one percent (19,500 engine r.p.m.) overspeed tests showed no brinelling. No brinelling of the kind that would result from overspeeding was noted on any of the propeller bearing races.

Attention was directed to the No. 3 powerplant by unusual markings on the safety coupling, the 50 percent closed position of the electrically operated oil shutoff valve and the totally closed position of the actuator of the electrically operated fuel shutoff valve located within the fuel control. These shutoff valves are operated by the cockpit powerplant emergency control which among other functions feathers the propeller. Operating times from "open" to "closed" of these valves are fuel, .3 to .4 seconds and oil, .5 to .97 seconds.

The safety coupling functions to disconnect the propeller from the engine in the event other protective devices have failed to function and the propeller is furnishing energy (negative torque) by windmilling action to drive the engine. This action by the safety coupling is generally termed "decoupling" and occurs when negative torque reaches approximately 1,700 shaft horsepower. Comparison of the marks on the inner and intermediate members of the No. 3 coupling with like marks on couplings known to have operationally decoupled and ratcheted revealed a dissimilar pattern. Metallographic and visual study revealed that high negative torque loads were applied while the intermediate member was out of alignment with the outer member. Impact loads between the inner and intermediate members were applied in both the positive torque and axial direction.

Separation of the No. 1 engine at the air inlet to compressor case split line occurred early in the sequence of events as evidenced by the parts forward of the separation line being the first major component along the flight path. Except for a section of the air inlet casting flange between 5:00 and 7:30 o'clock location which broke away and remained with the compressor flange, the 1/4-28 cap screws separated by tension failures and the 5/16-24 cap screws pulled the insert from the air inlet castings. Cap screw inserts which pulled out at the 10:00 to 11:0 o'clock location wiped metal from the face flange. The direction of this wiping action indicates the inlet housing rotated with respect to the compressor about a point measured radially outward at 11:00 o'clock and five to six inches outside of the bolt circle. Direction of rotation was clockwise relative to the compressor case and looking forward. The air inlet housing flange showed compressive loading between the 10:00 to 11:00 o'clock location. A visual and metallographic examination indicated that most of the scrape marks at the holes where the bushings pulled out had been made by the external threads of the bushings and there was no evidence found of a reversal of the scraping direction or repetitive movement of the bushings across the scraped areas. Hardness tests of the stud and case materials were satisfactory.

Marks were made by contact of the leading edge of the first stage compressor blades with the surface of the shelf just rearward of the inlet guide vanes .

Rubs were confined primarily to the areas between 3 degrees and 90 degrees and between 176 degrees and 230 degrees, starting from the top and progressing clockwise. The rub marks were not truly circumferential in that those between 3 degrees to 90 degrees angled forward about 6 degrees and those between 176 degrees and 230 degrees angled rearward about 3 degrees. The directions are referenced to the counterclockwise rotation of the compressor rotor.

The internal spline on the compressor stub shaft and the male spline of the compressor extension shaft showed contact marks on their normally loaded sides. The contact marks were made during the final 1/8-inch mesh of the splines as separation occurred.

No. 1 propeller blade angle when recovered was in the order of 51 degrees to 56 degrees. The remaining propellers were at or near feathered.

Structures

A major portion of the aircraft structure was shipped to a Dallas warehouse for further study. All structure was examined for break patterns, fire damage, stress patterns, explosive damage, and mechanical defects, with many of the individual pieces and/or sections being subjected to laboratory examination and evaluation. Certain sections of the structure were assembled in mockup form to help define failure, breakup and fire patterns. All of the structural damage was classed as from one or more of the following: airborne disintegration, ground impact, airborne fire, and/or ground fire. After a basic study of wreckage distribution, it became evident that the aircraft had experienced airborne disintegration which broke the aircraft up into a number of major sections as outlined under Wreckage Distribution and General Damage.

The left wing struck the ground butt-end first, right side up, after passing through trees approximately 50 to 70 feet high. Included with the left wing were the left landing gear, No. 2 QEC [3] unit and the No. 1 engine (minus propeller, gear box, air inlet housing and QEC structure). The wing was subjected to intense ground fire as a result of the ignition of fuel from the No. 1 fuel tank. The ground fire area extended 150 to 200 feet ahead of and approximately 100 feet behind the wing, but laterally only a few feet beyond the tip and the root. Some portions of the left wing in the trees showed no evidence of fire, whereas others directly over the principal wing wreckage showed light deposits of smoke. The starter compressor (magnesium), located normally in the rear of the No. 2 nacelle area, was completely consumed by fire and its louvered cover panel lying under it showed signs of heavy black smoke exiting through the louvers; however, the adjacent cover panel was found outside of the ground fire area and showed no evidence of ever having been subjected to fire or heat. The initial left wing separation occurred between the No. 2 nacelle and the center section. During the mockup of this area, approximately 80 percent of the lower planking in the No. 2 fuel tank area was accounted for and fitted into place. The upper planking of this area, in contrast to the lower planking, had been shattered into many small fragments. This made it difficult and in many cases impossible to fix the exact location for each piece; some pieces could only be fitted into a general wing station area.

[3] QEC is used herein for "Quick Engine Change."

Only a limited number of front spar pieces could be identified; however, the separation point was identified by fitting together two mating pieces of top cap, the inboard of which had been found at the main wreckage site and the outboard of which was dug out of the ground at the left wing site. This point was eight inches outboard of wing station No. 83. Most of the rear wing spar section from station No. 65 to station 137 was recovered except for minor fragments. That portion of the rear spar inboard of wing station No. 97 remained with the center section, while the outboard section from this wing station fell with the wing.

The lower planking inboard of wing station No. 137 showed evidence of upward bending with the bending being slight in the area of the No. 1 plank and being progressively more pronounced toward planks Nos. 8 and 9.

The lower planking stiffeners were severely column-buckled at every bay from plank No. 1 to No. 9. The rear plank was "S" shaped, being horizontal from wing station No. 65 to approximately wing station No. 75, curling sharply upward through wing station No. 85, and turning down to horizontal at approximately wing station No. 90.

The upper planking pieces in the No. 2 tank area were, in general, jaggedly rectangular with the long dimension spanwise. Only a few pieces of planking bridged a rib cap, and those which did usually contained a plank lap joint. Upper plank No. 9 from wing station No. 75 to wing station No. 101 showed the same "S" shape as did the lower plank but to a less pronounced degree. In contrast with the lower planking the stiffeners of the upper planks had separated along or very close to the radii.

The fracture faces of lower wing plank No. 3 at wing station No. 65 left, showed evidence of having recontacted each other after the fracture occurred. Microscopic examination disclosed at least three cycles of recontact. Two of these were evidenced by contact scratches; the third by a zinc chromate deposit.

The wing station No. 83 closing rib of the left leading edge contained metal-to-metal scratches at nine points. These marks were predominantly vertical, but microscopic examination showed three to four changes of direction at three different points.

Substantially the entire section of the left wing that fell separate from the aircraft showed evidence of varying fire damage. The fire pattern on the upper and lower planking of the No. 2 fuel tank was of a random configuration such that one piece or section would exhibit severe exposure to fire or heat whereas its mating piece would not. The zinc chromate in sections of the flaps aft of and inboard of the No. 2 nacelle and in sections of the fillet area between wing and fuselage had been browned by exposure to heat; however, the patterns once again were random and showed lack of continuity. This same general fire pattern was evident on pieces of wing leading edge. The left wing flap jack screws were found in the fully retracted position.

In the No. 3 fuel tank area of the right wing the lower planking was shattered into narrow spanwise strips. There was more deformation present than there was in the No. 2 tank area, particularly the stiffeners which were torn away from the planking and bent in random directions. There was some evidence of column-buckling of the stiffeners on the No. 1 to No. 4 planks, but much less severe than in the left wing.

The separation of the right wing occurred between wing station No. 329 and w station No. 346. The lower No. 3 plank inboard of wing station No. 346 showed a shear buckle pattern from wing station No. 329 to wing station No. 346. The upper surface panels were bent slightly upward in the area of the break and the broken ends of the stiffeners were pulled rearward. The right wing showed no evidence of having been exposed to heat, smoke, soot or flame impingement.

The propeller, engine gear case, air inlet housing, and the quick engine chang structure of the No. 1 powerplant separated as a unit as a result of failure of nacelle and/or QEC longerons at the QEC - nacelle fittings. The engine unit aft of the compressor front face remained in the No. 1 nacelle and descended with the left wing.

The forward attach point of the No. 1 QEC upper outboard longeron showed heavy compression loading prior to failure and further disclosed multiple directions of local bending in the several longeron members.

The forward attach area of the No. 1 QEC upper inboard longeron showed a tension failure followed by recontact of the fracture faces during a would-be compression load.

The electrical connectors and their wiring at the No. 1 nacelle firewall were failed in multiple directions of bending.

At the No. 1 nacelle firewall, the fuel line was bent up/inboard and down/outb prior to the ultimate failure which was up/outboard.

Indentations were found in the nacelle shroud which were made by the anti-swirl assembly clamp, particularly in the area of the clamp bolts. There were indications here of not less than seven contacts of the bosses with the nacelle. There were also multiple clamp marks around the shroud but less pronounced than those at the clamp split-line.

All panels and structure of the No. 2 QEC and nacelle were accounted for and included the landing gear door and starter compressor section. This entire section with the exception of the outboard starter compressor housing showed evidence of having been subjected to fire and heat exposure. A considerable amount of molten aluminum deposits was found throughout the nacelle area; however, none of the deposits showed evidence of having been blown by an airstream.

The No. 3 QEC and nacelle were both completely demolished by ground impact. All examined pieces of this area revealed no evidence of having been exposed to fire or heat nor was there any evidence of smoke or soot deposits in this area.

The No. 4 nacelle barrel outboard panel was peeled directly outboard and aft, and rivets in the lower forward corner of the inboard barrel side panel were sheared in a forward direction. All firewall fittings were bent outboard and cables notched the firewall in an outboard direction. Only those portions of the structure which were carried to the ground with the engine showed any indicatio of exposure to fire or heat.

The forward fuselage section from the nose through approximately fuselage station No. 570 was subjected to severe impact forces when it struck the ground and for the most part was reduced to crushed rubble. A thorough examination of the recovered and identifiable parts from this area showed no evidence of heat, fire, or soot.

The rear fuselage, the center section, including the right wing-to-wing station No. 329, and the rear portion of No. 3 engine struck the ground with great force. The rear half of the lower center section planking remained intact. The rest of the center section box area was reduced to hand-sized fragments. The planking and front spar of the right wing stub were shattered with only the lower planking material remaining in large sections. The fuselage side and top panels were distributed over a wide area but the individual panels were relatively free of severe impact marks and crushing effect. There was no evidence of fire or soot on the right side of the fuselage, in the center section, right wing stub, or the interior of the fuselage.

The exterior surfaces of the left fuselage panels revealed considerable evidence of inflight fire effects. The biaxially stretched plexiglass cabin windows on the left side aft of fuselage station No. 659 had been surface distorted in the form of intersecting trenches and variable size rectangular raised areas or islands. The severity of the window distortion increased progressively toward the aft end of the fuselage. This type of surface distortion is a common characteristic of this plastic when it is exposed to above normal heating effects either by direct flame impingement or by radiated heat. The exposure time, type of heat and applied temperature are all variables which will cause distortion pattern changes, i.e., depth of trenching, size of islands, and degree of edge roughness.

Lockheed Aircraft Corporation conducted tests to attempt to determine the nature of controlled variables which would cause similar type of distortion of plexiglass samples as evidenced by the fuselage windows. Reasonable correlation between laboratory tests and observed heat effects on the airplane was achieved in several instances; however, it is noted that the controlled test condition did not necessarily represent the existing airborne conditions at the time of occurrence. The following is a quoted summary of Lockheed's Report No. 14,281, dated 2/15/60:

"1. The surface heat effects observed on the cabin windows were caused by flame impingement rather than by radiated heat.

"2. Time duration of the heat exposure on window No. 18 was between 6 and 10 seconds.

"3. The flame temperature in the region of window No. 18 was approximately 2,000°F."

The blue trim paint which runs longitudinally along the left side of the fuselage was blistered in two areas: the lower half of the trim stripe on the galley door and the central part of the stripe aft of fuselage station No. 1117. The paint blistering occurred in narrow bands running parallel to the normal air stream and was most severe in the area under the stabilizer. Paint had flaked off in patches throughout the affected areas. It was noted that no paint blistering occurred in the immediate vicinity of any of the windows, even those showing the most severe heat effects.

Smoke and soot deposits existed in a number of places on the left fuselage surfaces. The white painted area above the windows from fuselage station No. 8 to fuselage station No. 1030 was heavily sooted with streaks running up and back at a 20 degree angle. The soot deposits were heavier in areas between the rings and stringers than they were directly over the ring and stringer stiffeners. A scallop pattern of soot deposit was evident for about 1.5 inches aft of each ring between stringers. This scallop pattern became progressively more evident toward the rear fuselage section. A streak of black oily substance was deposited from fuselage station No. 570 to the tail cone. This streak, unlike some of the other deposits on the aircraft, could not easily be wiped off and gave the appearance of having been partially baked to the metal. In addition to the 12 inch wide bar which started about 42 inches behind the galley door and extended all the way rear ward, numerous fine streaks existed both above and below the window line. While most of the streaks were parallel to the flight path, there were minute streaks upward and aft at an approximate angle of 30 degrees.

The tail section consisted of vertical fin and rudder, stabilizer stubs, tail cone (fuselage station No. 1117 and aft), and the lounge floor. The direction of collapse of the right underside of the cone indicated that this section struck the ground while moving rearward, causing damage to the elevator power package, the base of the rudder, and the elevator root sections. The vertical fin was undamag except for a series of slight linear dents in the leading edge. It was found tha the wing planking stiffeners exactly fit these marks.

One other matter should be mentioned. On the ground at Houston and shortly before departure, First Officer Hollowell remarked to a representative of the eng manufacturer (Allison-General Motors), "This aircraft trims up funny." There wa no further discussion on the matter nor was it made an item of record in the air craft's logbook.

As a result of this and another accident six months later to the same model aircraft, the manufacturer instituted a searching reassessment of the aircraft's design. The work was largely analytical but also included wind tunnel testing an flight testing. The program of design reevaluation was extremely extensive.

Two questionable items in the design of the airplane came to light. One of these was that significant loads imposed on the wing intermediate ribs between th fuselage and outboard nacelles by shell distortion had not been included in the design loads. The other was that the dynamic response of the outboard nacelles i turbulence was different than that used in the original design, with the result that the torsional loading of the wing inboard thereof was increased. In additic the reevaluation program disclosed that, with the stiffness of a powerplant nacel installation reduced below normal, propeller "whirl mode" could persist undampene and couple with the wing thus exciting it to failure.

Aircraft History

N 9705C was a new aircraft. Its final assembly was started in April 1959, an the first of its three production test flights was on September 4, 1959, 25 days prior to the accident.

Braniff Airways accepted delivery of the aircraft at the factory, Burbank, California, on September 18, 1959. Acceptance had been preceded by a total of three production test flights and one acceptance flight.

The four propellers and three engines in Nos. 2, 3 and 4 positions had been installed new (zero time). The No. 1 engine had accumulated 26 hours and 25 minutes of operation at the time of installation.

Upon arrival of N 9705C at the Braniff Airways Base at Dallas, Texas, an acceptance inspection was conducted incorporating the operations of Nos. 1, 2, 3 and 4 maintenance and inspection procedures. After the acceptance inspection N 9705C operated approximately 122 hours in scheduled and training flight (total time was 132 hours and 33 minutes); therefore the first or No. 1 inspection due at 205 hours had not been performed. As a result only preflight service checks and nonroutine items were accomplished during the ten days of operation.

The only areas of chronic difficulties with N 9705C appeared to have been with the radio, navigational equipment and the generator malfunctioning during the last few flights. This latter generator malfunctioning was reported to have been corrected

Several incidents to other Electras were investigated. These consisted of: (a) possibility of excessive fuel tank pressures; (b) review of a report concerning a landing gear tire failure caused by excessive brake temperatures that resulted in an explosion of the tire approximately 30 minutes after takeoff. This caused excessive damage to the nacelle structures; (c) loss of an intermediate tail pipe cover in flight; (d) review of starter bottle compressor difficulties. (This last item has been a chronic difficulty fleetwise and the No. 2 compressor in the No. 3 nacelle had been de-activated in N 9705C at the Dallas Terminal some two hours prior to the final flight); and (e) an over-all general monitoring of L-188 difficulties for any correlation with the findings to date.

All areas investigated resulted in negative findings. All squawk items during factory flight tests were signed off as corrected. All maintenance items on this aircraft, including all checks and inspections as well as correction of all items pertaining to airworthiness appearing in the flight log (squawks) had, according to company records, been complied with by Braniff personnel in full accordance with prescribed and approved methods.

Braniff Airways maintains a special technical group to monitor the Lockheed L-188 operation. A folder is kept for each aircraft as a means for keeping individual aircraft chronological records. No significant entries were found.

On September 22, one week before the disaster, the aircraft was used on a routine training flight. Recovery from a planned stall was made incorrectly and a secondary stall developed, attended by buffeting more severe than normally allowed. The Braniff captain in command expressed the opinion that structural integrity was not impaired and that no inspection was needed.

Crew History

Captain Wilson Elza Stone completed Lockheed Electra L-188 ground school training on April 10, 1959. The course consisted of 120 hours of instruction on aircraft systems, performance, and flight planning. His average grade for the course was 96. In addition, an L-188 refresher course and cockpit check was completed by Captain Stone on May 17, 1959, and involved a total of 12 hours attendance.

Captain Stone's L-188 flight training commenced May 18, 1959, and was completed May 27, 1959. This training covered general preflight duties; air work, en route and emergency procedures, including simulated engine failure and engine fire at altitude; simulated emergency procedures, day and night takeoffs and landings; and instrument procedures. This type rating check was given by an FAA designated ATR examiner after 8 hours and 45 minutes of flight training. His flight proficiency was above average on this check. Captain Stone then flew for 12 hours and 34 minutes with company check pilots prior to being assigned to regular line operations His total Electra time was 68 hours and 39 minutes.

First Officer Dan Hollowell completed Electra ground school training on July 3, 1959. He received an average grade of 95 for the 120-hour course. Flight training commenced July 10, 1959, and was completed July 31, 1959. First Officer Hollowell received a total of 4:30 hours of flight time in the Electra, and 8:45 hours of observation time. A review of the records indicated that he was current in all requirements. His total Electra time was 95 hours and 30 minutes.

Second Officer Roland Longhill completed the 120-hour Electra ground school course on March 20, 1959. His final examination grade was 93. He was qualified for duty as a flight engineer on Electra equipment on August 12, 1959, after completing 10:40 hours of instruction. His total Electra time was 83 hours and three minutes.

Crash Injury Research

Traumatic injuries to occupants, some of whom had fallen free of the aircraft, were severe and extensive and with much mutilation.

Examination of tissue for carbon monoxide level was made from nine bodies, one of which was that of First Officer Hollowell. It and seven others showed carboxyhemoglobin saturation of the blood and tissue at less than ten percent. Medical opinion is that this is not an incapacitating quantity. One of the nine showed a 13 percent concentration, indicating possible inhalation of smoke laden air prior to death.

Analysis

The investigation of this accident has produced such a voluminous quantity of data that this report will be confined to the discussion and analysis of only those data considered to be apropos to the consideration of probable cause. Several incidents and accidents involving Electras have occurred during the course of this investigation, all of which have also been investigated. None of these is considered to have any association with this accident except the accident to a sister aircraft at Cannelton, Indiana, on March 17, 1960. The investigational results of that accident and their relevance to the solution of this case are discussed below.

Much of the information appearing under Investigation is of a negative nature insofar as the probable cause of this accident is concerned. Such matters as atmospheric turbulence cannot be logically linked to this accident. The aircraft was operating in the clear at the time of the accident, well removed from the closest significant convective activity; and the necessary meteorological parameters for the formation of clear air turbulence were not present (i.e., vertical or horizontal wind shear, strong jet stream, sharp upper trough). The subject of pilot and flight engineer competence cannot be considered a factor, for all three were well qualified and experienced airmen despite having less than 100 hours in Electras. Also, the possibility of crew incapacitation, even in small degree, by any toxicity is without foundation and is not even suspected. The aircraft itself was virtually new and had not needed appreciable maintenance work; that which had been accomplished had been signed off in accordance with established practices. Collision or threatened collision with another aircraft or object has been ruled out and the flight was being navigated properly.

Laywitnesses are often in error, particularly in their attempts to recount time lapses, the exact sequence of events, or altitudes. It is difficult, even to a trained observer, to recall accurately the order of an unanticipated rapid succession of events. There is in this accident, however, one condition which fixes the sequence and establishes to some extent a time boundary between two important elements of observation: (1) the sound, variously described as "jet noise," "low flying aircraft," "unsynchronized motor," and (2) the observation as "a large orange ball of fire." Six witnesses were indoors when startled by a noise of sufficient intensity to get them to look or go outside.

Certainly, some of their observations cannot be reconciled such as the white light seen by one witness; nor do the various times between events check out with any high degree of accuracy. However, all of the witnesses who were indoors first heard a noise which was followed by a ball of fire.

Several witnesses gave reasonably good descriptions of objects silhouetted between them and the ball of fire. This information correlated well to fix the geographic position and an approximate altitude band for the fireball. When plotted, the altitudes of sighting varied from 17,000 feet to about 24,000 feet. While the variation here is wide, it does indicate that the fireball was at high altitude and probably no lower than the 15,000 feet reported on the radio by the crew.

Using a speed of sound of 1,088 feet per second, which is the standard-day average between sea level and 15,000 feet, it can be shown that from a simultaneous noise and light at 15,000 feet, an observer directly below would hear the sound about 14 seconds after seeing the light. An observer three miles away would not hear the sound for an additional six seconds. (Normal temperature variations and even strong winds will make only negligible differences in time.) The loud continuing noise, then, had to occur 14 or more seconds prior to the appearance of the fireball, plus the time interval between the witness observations of noise and light.

Analysis of the witness statements shows that the information provided by a majority of the witnesses is reasonably consistent. The average time from noise (at the source) to the appearance of the ball of fire was in the order of 33 seconds, with the largest variation from the average being about eight seconds.

The witnesses who saw the fireball from inception agree that there was no prolonged fire, but rather a small one which grew quickly into a large orange of red ball and then disappeared in a few seconds. Several witnesses observed that just prior to extinguishment, a smaller fire emerged from the large ball and fell to the northeast, dying out well before reaching ground level.

That the aircraft broke up violently is self-evident. That the breakup process was both quick and with little or no warning is also clear for two reasons. First, only one of the 37 aircraft's passenger seats recognizable as such was found with the safety belt fastened, and this probably means there was no time to order their fastening. Second, the final radio message preceded the breakup by an interval of something less than two minutes and that message gave no hint of trouble.

A definite sequence of failures and breakages appears discernible and will be mentioned because it may be considered as somewhat basic for this analysis. Separation of the left wing and the No. 1 gear box propeller and QEC structure occurred at about the same time; it is impossible to say which went first. The horizontal stabilizer then broke up under the impact of parts coming from the wing; wing planking from the right wing tip came free; the No. 4 powerplant tore loose; and the right wing outboard of No. 4 separated. All of these events happened in a short period of time. Somewhat later, at much lower altitudes, the fuselage broke in two separate portions at a point about halfway back near fuselage station No. 570.

Under Powerplants mention was made of there being no evidence of overspeeding. However, in view of the tolerance of both the engine and propeller to overspeeding before any physical evidence develops, 20 percent and 53 percent, respectively, lack of this evidence does not permit concluding an overspeed of a lesser amount did not occur. However, it is difficult to project an overspeed as such into an accident of this kind. The following devices are incorporated in the engine propeller design to protect against overspeeding and/or high drag: (1) fuel control overspeed governor, (2) negative torque signal, (3) safety coupling, (4) hydraulic and mechanical low pitch stops, (5) beta followup, and (6) pitch lock. These features, some of which function entirely independently, provide multiple protection against powerplant induced drag of a degree which would present airplane control or structural loading problems.

Also, under Powerplants there is mention of possible emergency procedures having been used on No. 3 powerplant. However, the evidence indicating that emergency action may have been taken with respect to No. 3 powerplant is not supported by the physical condition of the engine and propeller. This powerplant was the last to separate from the airplane, possibly at contact with the ground. That the oil shutoff valve was only partially closed indicates the operation was prematurely terminated, most likely by a loss of electrical power. It appears that emergency action with respect to this engine was initiated just prior to or during breakup by either the crew or by actuation of the control due to disruption by the airplane breakup. Any significance of these valves with respect to the accident is not discernible.

In reference to the statement under Powerplants that the No. 1 propeller, engine gear case, torquemeter, air inlet case, and QEC structure separated and fell as a unit, the following should be noted. This separation occurred following failures in the QEC which permitted movement of the rear of the engine. Had the engine separation occurred first the repeated markings made on the adjacent shrouding by the clamp on the rear of the engine would not have occurred. It is concluded that the normal support provided by the mounts at the reduction gear case was disrupted, thus permitting loads generated by the rotating propeller to be transmitted through the engine structure causing gyrations of the rear of the engine within the confines of the adjacent shrouding and ducting. Separation at the air inlet and compressor case junction occurred in an upward and slightly to the left direction with the forward portion also rotating clockwise about a center five to six inches outside the bolt circle positioned radially about the 11:00 o'clock postion. This separation occurred by tension failures of the 1/4 - 28 cap crews and pullout of the 5/16 - 24 inserts. A study of this separation failed to reveal any evidence of repetitive relative motion as separation occurred. The loading necessary to bring about this separation could have occurred only after the QEC structural integrity was disrupted, and propeller-generated loads that were intended to be absorbed by the Lord mounts which support the reduction gear assembly were instead transmitted rearward through the intact engine structure.

Interference of the first stage compressor blades with the air inlet housing occurred on the No. 1 engine of this aircraft and on the Nos. 1 and 4 engines of the Electra involved in the accident at Cannelton, Indiana. There was separation in flight of some portion of these three engines. These similar circumstances cannot be accepted as coincidental since like circumstances prevailed in each case. It is believed this rotational interference was caused by air inlet case deflection due to abnormal loads being applied through the engine torquemeter housing and struts. Furthermore, these abnormal loads followed disruption of the engine supporting structure such that loads normally taken out by the forward QEC Lord mounts and structure were, instead, imposed on the engine structure. It follows that the basic engine structure forward of the compressor must have been intact in order to transmit propeller generated case distorting loads. The design strength of the basic engine structure is materially greater than that required by the Civil Air Regulations for its supporting structure. This suggests that structural damage due to overloads by whatever means would be confined initially to the supporting structure. Thus, the previous conclusion that engine supporting structure disruption preceded the engine structure damage is further substantiated.

No. 1 propeller blade angle and markings on the load side of the compressor extension and stub shafts' splines indicate power was being produced when the separation occurred.

As stated under <u>Investigation</u> no indication of operational distress was found through examination of the hydraulic and electrical system components. Examination of the radio transmitters and receivers revealed no sign of malfunctioning.

Damage to the control surface boosters precluded establishment of booster selection, i.e.: "On" or "Off" or whether the autopilot had been in operation prior to the breakup of the aircraft. Although the broken lead at the elevator load sensor probably failed as the result of a few (possibly three or four) cycles of reversed bending, it is not known whether the failure occurred prior to or as a result of the accident. It may well have broken during the violent shaking which could have preceded the inflight breakup. If the failure existed in flight and the aircraft were being flown on autopilot the automatic elevator trim feature would be inoperative and any change in longitudinal trim would be accommodated by the autopilot. With the autopilot holding against an out-of-trim condition, up to the limit of its authority, sudden release of the autopilot would result in a relatively mild pitchup or pitchdown, depending upon the direction of trim imbalance.

This would not create a hazard or place the aircraft in an attitude from which recovery would be difficult.

The extremely brittle ash residue of the left air compressor of the air start system flaked away readily when handled, indicating that the compressor had burned where found on the ground at the left wing impact site. Examination of the engine fire extinguishing system showed that the selector valves were in their normal positions and that none of the fire bottles had been discharged by crew action.

The pertinent observations of the physical evidence can be summarized as follows:

1. Inflight fire was confined to the extreme inboard portion of the left wing, causing heat damage to the left windows rear of the wing trailing edge and sooting of the left rear fuselage.

2. The No. 2 fuel tank showed no evidence of internal pressure or explosion and the planking fragments were burned and sooted in a random pattern.

3. The left inboard leading edge, the lower planking and the rear spar showed that the left wing failed at the inboard one-third of the No. 2 tank in upward bending and noseup torsion. The relatively small fragments of the upper planking indicated a strong probability of failure resulting from a high positive load.

4. The wing station No. 83 closing rib of the left leading edge showed metal-to-metal scratches. Microscopic examination disclosed three to four changes of direction in these predominantly vertical marks.

5. The fracture faces of lower wing plank No. 3 at wing station No. 65, left, showed evidence of having recontacted each other after the fracture occurred. Microscopic examination revealed at least three cycles of recontact.

6. The forward attach point of the No. 1 QEC upper outboard longeron showed heavy compression loading prior to failure and further disclosed multiple directions of local bending in the several longeron members.

7. The forward attach area of the No. 1 QEC upper inboard longeron revealed a tension failure followed by a recontact of the fracture faces in a would-be compression load.

8. The electrical connectors and their wiring at the No. 1 nacelle firewall were failed in multiple directions of bending.

9. At the No. 1 firewall, the fuel line was bent up/inboard and down/outboard prior to ultimate failure which was up/outboard.

10. Found in the No. 1 nacelle shroud were indentations which were made by the antiswirl assembly clamp bosses. There were also multiple clamp marks around the shroud but less pronounced than those at the clamp splitline.

11. Both No. 1 gear box Lord mounts showed evidences of repeated yaw loads and some indication of rear load. The rear mount disclosed excessive relative motion of the mount with respect to the nacelle structure.

12. The No. 1 engine's first stage compressor blades rubbed the inside of the air inlet housing.

13. Examination of the structure for fatigue produced completely negative results.

In reference to the localization of the left inboard wing fire, as mentioned, it seems proper to present the following: At no point can there be found a continuous fire or heat pattern across the rear portion of the wing, particularly along the spar, the back side of which is white, and the upper trailing edge surface, the under side of which is white. This material was clean. Two of the flap beams, flap station No. 174 and flap station No. 106, showed some sooting; however, the soot marks are not continuous across break lines. The inboard flap beam at wing station No. 72 was completely clean. This beam went into the main wreckage area with the center section. The flaps themselves had fire patterns on them; however, at any point where there was a fire pattern it could be shown that it did not exist prior to the breakup of the flap and most of this fire occurred in the area where the flap was torn through as a result of wing failure. Inboard of the station No. 72 flap beam there was evidence of inflight fire, and such would be expected since there was a ball of fire passing through this area at the time of wing failure. The only point at which fire or heat can get into the fillet area on the rear portion of the wing is through a small opening under the fillet and above the junction point of the upper cap of the rear spar to the fuselage. This area was completely clean and showed no evidence of soot, fire, or heat. This area, incidentally, is white and would show soot very readily. The only other way to get heat into the fillet area from outboard would be through the leading edge and through a similar opening from the leading edge into the fillet area; however, this did not get sooted in any way. It was noted during the mockup period that the trailing portion of the wing fillet makes a scoop or funnel capable of holding several gallons of kerosene, and ahead of this area there is a place where additional fuel could be trapped for a short period of time. This could contribute to a more prolonged fire than might normally be considered possible.

Any comprehensive analysis must consider, along with the positive evidence in the wreckage, the following negative points:

1. In the 07 radio call to the company the only maintenance items reported were an inoperative No. 3 sump pump and the bonding of a terminal strip. This was only two minutes prior to the accident.

2. There was no turbulence along the route of this flight at operating altitudes.

3. There was no record of this aircraft being subjected to a hard landing or to any appreciable turbulence during its 100-plus hours since manufacture. There could be found only one incident of any possible maltreatment of the airframe. This occurred on September 22, 1959, during a training flight wherein the pilot entered a secondary stall following an improperly executed stall recovery. Any likelihood of damage resulting from this maneuver has been evaluated and dismissed under <u>Investigation</u>.

4. According to ARTC records there was no conflicting traffic of aircraft operating on flight plan. The U. S. Navy advised that there were no aircraft

operating from the only Navy facility in the area and further that no other Na- command had aircraft operating in the vicinity of Buffalo. The Air Force repo no local flights from Barksdale Air Force Base between the hours of 2200 and 240C Connally Air Force Base had aircraft in the area, but all had landed prior to the time of the accident. Carswell Air Force Base had two KC-135's on IFR round robins at accident time. (If these two had been in the Buffalo area IFR, ARTC should have had a record of this.)

5. In all of the examination, testing, and analysis of the flight control systems, boost, and autopilot, no phenomenon could be produced which would produce or lead to a structural failure. (There was further work done in this area after the Cannelton accident.)

There is one other very important consideration. This is the Cannelton, Indiana, accident of a similar Electra, which also experienced a wing failure (right) and loss of QEC units to form a similar destruction pattern of the Buffalo accident. While a mirror image type of pattern itself is not positive proof of similarity of cause, there are indications of oscillatory motions of wing and outboard QEC structure in both the Buffalo and Cannelton wreckages.

Following the accident at Cannelton, Indiana, Lockheed undertook a reevaluation program in which the entire Electra concept and design was audited. An enormous quantity of data was produced, the majority of which was negative. It is sufficient for the purpose of this report to state that, insofar as causal factor is concerned, only one area of the program is significant. This is the phenomenon known as "whirl mode," an oscillation which under certain conditions can produce flutter.

All of the flutter tests and analyses made by Lockheed during the original certification process and during reevaluation showed the Electra to be flutter-free during and even above normal operating speeds and further disclosed that the wing has a high degree of damping. The term "damping" means that if a motion is imparted to the structure, the motion will die out when the exciting force is removed; the damping forces are those which take energy away from the oscillation. A small amount of damping is from internal energy absorption in the structure and in energy absorbers such as engine mounts. The most significant damping, however, is the result of aerodynamic forces acting in opposition, thus absorbing energy from the oscillation. Conversely, if a major change occurs that allows the aerodynamic forces to be additive to the exciting force, the oscillation grows, and the result is flutter.

Since the Electra wing is basically flutter resistant, in order to produce flutter there must be an external driving force. The possible force generators are the control surfaces and the propellers. Analyses indicated that the control surfaces would not produce wing oscillations of sufficient amplitude to produce a wing failure; consequently, further analysis was centered around the propeller.

The propellers being normally stabilizing, it was necessary to consider abnormal propeller behavior, such as overspeeding and wobbling. The studies and tests conducted during the reevaluation program proved that a wobbling outboard propeller caused by weakened nacelle and/or engine structure can induce wing oscillations.

Since a propeller has gyroscopic characteristics it will tend to stay in its plane of rotation until it is displaced by some strong external force. When such a force or moment is applied, the propeller reacts in a direction 90 degrees to the force. For example, if the propeller is displaced upward the resistance of the structure applies a nosedown pitching moment causing the propeller disc to swing to the left due to precession. The yaw stiffness resists this motion causing precession downward, resisted by pitching stiffness which produces a precessional swing to the right. This, in turn, is resisted to cause an upward precession to complete the cycle. This effect is termed "whirl mode," and its direction of rotation is counter to that of the propeller.

Normally, whirl mode can operate only within the flexibility limits of the engine mounting structure and is quickly damped. If, however, the stiffness of the supporting system is reduced through improperly installed, failed, or damaged powerplant structure, mounts, or nacelle structure, the damping of whirl mode is reduced to a degree depending on the amount of stiffness reduction.

Structural weakness or damage does not change the conditions under which whirl mode may be initiated, but in three ways it makes the phenomenon a potential danger:

1. The greater flexibility of a weakened system can allow whirl mode more freedom, hence it can become more violent. In an undamaged system the stiffness increases with increasing deflections but this is not necessarily true if the structure is damaged.

2. In a weakened installation, the increasing violence of whirl mode can further damage the supporting structure, in turn leading progressively to more violence and even further damage.

3. As the structural system is damaged, reducing the spring-constant, the amplitude of whirl mode increases and the frequency decreases from its natural value to lower values which, in the case of the Electra, approach the wing fundamental frequencies.

The natural frequency of whirl mode in an undamaged installation is approximately five cycles per second. The wing torsional frequency is about 3.5, and wing bending about two cycles per second, with some slight variation with fuel loading.

As whirl mode progresses in an overly flexible or damaged powerplant installation, its frequency can reduce from five to three c.p.s. where it will drive the wing in three c.p.s. torsional and bending oscillations. These wing oscillations will re-enforce and perpetuate the whirl mode. The three oscillations are then coupled at the same frequency of about three c.p.s., thus becoming a form of induced flutter forced by a powerful harmonic oscillation. This phenomenon can exist, as demonstrated in wind tunnel tests and in analytical methods, at an airspeed far below that at which classical flutter can develop.

The design stiffness factor for an Electra powerplant installation is 15.9×10^6 inch pounds per radian (root-mean-square). The tests indicated that at this stiffness level whirl mode cannot force wing oscillation at an airspeed lower than 120 percent of the design dive speed of the aircraft. If, however, the stiffness is reduced, forced oscillations become more likely depending on amount of stiffness reduction and on equivalent airspeed. More specifically, the data show that if the stiffness is reduced to some value less than 8×10^6 inch pounds per radian, whirl mode could become a driving force on the wing in the cruising speed range. The tests further showed that whirl mode of catastrophic proportions could develop, reduce its frequency, and couple with the wing in a period of from 20 to 40 seconds.

Certain causal possibilities can be eliminated from further discussion because of a complete lack of evidence or evidence to the contrary:

1. Collision with another aricraft
2. Structural failure due to turbulence during this flight
3. Structural failure from fatigue
4. Structural failure as a result of boost and/or autopilot malfunction
5. Sabotage

The shattered upper planking of the left inboard wing suggested a strong possibility of failure due to excessive positive loading. The horizontal tail or rear fuselage showed no such evidence; however, Lockheed testified that at 275 KIAS (last known airspeed) the wing and tail were about equally critical under positive loading. There was further testimony that above 275 knots the wing becomes the more critical of the two.

This leads to the premise that high-load wing failure (if it existed) occurred at an airspeed in the order of 275 knots (cruise) or higher. Such an overload failure, with boost, autopilot, and turbulence out of the picture, would have to develop from a pullup maneuver brought on by collision avoidance or following loss of control. Since there was no known conflicting traffic, there is nothing to substantiate a theory of collision avoidance.

Loss of control has occurred in other instances because of a pilot's inattention to duty resulting in a dive or diving spiral. An analysis of a plot of the witness sightings, however, places the ball of fire at or above 15,000 feet. If, then, the ball of fire (wing-tank fuel ignition) was at or above 15,000 feet it would require a climb, intentional or not, prior to any loss of control of a type which would create excessive airspeed. (Note: It is extremely difficult to conceive of a recovery from an "unusual position" causing structural failure without first having excessive speed, particularly at the gross weight of this aircraft at the time of the accident.) This hypotheses cannot be maintained for it first presupposes a climb for which there would be no known purpose. If it be argued that the climb is unintentional, it becomes necessary to assume an extremely lengthy inattention. It must also be remembered that a scant four minutes prior to impact, or about three minutes prior to the witnessed noise, the flight reported 15,000 feet.

All this leads to a conclusion that, even with indications of high positive loading, there is a causal factor far more insidious than excessive air loads.

It thus becomes necessary to consider "whirl mode" which has been described, a phenomenon shown by wind tunnel tests and analysis to be a potential destructor. Some evidence of oscillatory motion was found in the left wing and No. 1 QEC/nacelle. While this is not positive evidence of whirl mode, it is certainly compatible with the motions shown by tests to exist during the latter stage of excitation.

Another factor which is compatible with, but not proof of, whirl mode is the intense noise attested to by groundwitnesses. Analyses by Lockheed and Board technical personnel have shown that during whirl mode the propeller tips approach sonic velocity without increase in r.p.m. or airspeed, and probably produce a noise in the order of 120 decibels. The witnesses heard such a noise at a time which would place the noise about 33 seconds prior to the fuel ignition. Analysis has shown that whirl mode, from inception to destruction, would last about 20 to 40 seconds. No avenue of investigation has revealed any other reason for the sound described and later identified by the witnesses.

As mentioned earlier, the left wing showed indications of high positive load. This is in complete contrast to the right wing failure at Cannelton. There is no way to establish with any degree of certainty this difference in wing failure patterns, but it is possible to rationalize a possibility. The first impulse of a pilot, when subjected to either severe vibration, a runaway propeller noise, or both, is to slow the aircraft down. Normal action would be to reduce power and to climb. Of the two, climbing is the more immediately effective, particularly in the Electra, which takes several minutes to reduce speed from 275 to 200 knots by power reduction. There is, then, the possibility that in the excitement and in his desire to slow down quickly, the pilot exerted back pressure sufficient to fail the wing earlier than if failure had resulted from oscillation alone. This is not to imply that the pilot applied a stick force capable of failing a structurally sound wing, but rather that his action dictated direction and time of failure.

There remains one point, the element of "prior damage," which cannot be satisfactorily explained. According to Lockheed, the stiffness factor of the QEC must be substantially reduced to produce an undamped whirl mode, or propeller precession. This suggests damaged or failed structure, engine mounts, or engine structural components. No such evidence was found. The No. 1 QEC and powerplant were examined minutely for fatigue, with negative results. No other type of failure was discovered which could be definitely considered damage prior to whirl mode, QEC failure, and impact. There is serious doubt whether such a determination could be made with any degree of accuracy. For example, there were several pure tension and compression failures in the QEC structure which could have occurred prior to whirl mode or early in the precession. Furthermore, there is nothing in the aircraft's recent history, such as hard landings or turbulence, to indicate the possibility of prior damage, nor was there on the final flight, as far as can be determined, any incident leading to structural damage prior to the accident.

Conclusion

There was in this investigation no positive indication of the cause. For this reason, an attempt has been made in this report to eliminate certain possibilities by application of the available evidence to each of them. Once these possibilities have been disposed of, the only remaining causal factor for which there is some known basis is the condition of whirl mode. The probability that this accident was so caused is supported by the following:

1. So far as is known, the aircraft was in straight and level flight and at a normal cruise speed with no serious mechanical problems.

2. A sound identified as a supersonic or high speed propeller occurred 30 seconds prior to fuel ignition (wing failure).

3. There was structural damage evidence compatible with oscillatory motion of the No. 1 QEC and the left wing.

4. First stage compressor blades of No. 1 engine rubbed the air inlet housing supports.

5. The probable cause of a similar accident of another Electra was due to whirl mode.

If prior damage is a requirement for the necessary reduction in stiffness, it must be assumed that the evidence of such damage was either obliterated in the crash or never existed in a discernible form.

Probable Cause

The Board determines that the probable cause of this accident was structural failure of the left wing resulting from forces generated by undampened propeller whirl mode.

BY THE CIVIL AERONAUTICS BOARD:

/s/ ALAN S. BOYD
Chairman

/s/ ROBERT T. MURPHY
Vice Chairman

/s/ CHAN GURNEY
Member

/s/ G. JOSEPH MINETTI
Member

/s/ WHITNEY GILLILLAND
Member

S U P P L E M E N T A L D A T A

ıvestigation and Hearing

The Civil Aeronautics Board was notified of this accident immediately after occurrence. An investigation was started at once in accordance with the provisions of the Federal Aviation Act of 1958. A public hearing was ordered by the Board and held in Buffalo, Texas, on October 21, 1959, and in Dallas, Texas, on March 9 and 10, 1960.

The Carrier

Braniff Airways, Inc., is an Oklahoma corporation with its principal office in Dallas, Texas. The carrier holds a certificate of public convenience and necessity issued by the Civil Aeronautics Board and an air carrier operating certificate issued by the Federal Aviation Agency. These certificates authorize the carrier to engage in air transportation of persons, cargo and mail within the United States, including the route involved.

Flight Personnel

Captain Wilson Elza Stone, age 47, was employed by Braniff airways on April 22, 1939. He held a currently effective airman certificate with airline transport pilot rating number 24487. His other ratings included DC-3-4-6-7C, Convair 340-440, MEL, SEL and L-188. He had a total recorded flying time of 20,726 hours, of which 68:39 were in Lockheed Electra aircraft. He passed his last FAA physical examination September 21, 1959.

First Officer Dan Hollowell, age 39, was employed by the company on November 29, 1948. He held a currently effective airline transport rating certificate number 418671 with other ratings in DC-3 and Convair 340-440. He had a total recorded time of 11,316 hours of which 95:30 hours were in Electra aircraft. His last FAA physical was passed on June 11, 1959.

Second Officer Roland Longhill, age 29, was employed by the company July 16, 1956. He held a current airman certificate, flight engineer certificate number 1358795 and commercial pilot certificate number 1304814. He had a total recorded time of 3191:35 flying hours of which 83:03 were in Electra aircraft.

Hostess Alvilyn Harrison, age 25, was employed by the company December 29, 1953. She completed her Electra training June 4, 1959. Hostess Betty Rusch, age 24, was employed by the company on April 18, 1956, and completed her Electra training June 2, 1959. Hostess Leona Winkler, age 25, was employed by the company on March 21, 1958. She completed her Electra training June 4, 1959.

Extra crew member Wendell John Ide, age 35, was employed by the company July 9, 1951. His position was Technical Instructor to Engineer Specialists. He had mechanic's engine certificate number 1287530 issued November 30, 1955.

The aircraft

At the time of departure from Houston N 9705C, a Lockheed Model L-188A, ha[illegible] had a total time of 132 hours and 33 minutes. Thus none of the periodic inspections, the first of which was to have been at 205 hours, had become due. All of [illegible] customary preflight service checks were performed during the ten days the aircra[illegible] was in use. All pilot complaints (squawks had been signed off as corrected. Th[illegible] engines were Allison (a division of General Motors) model 501-D13 and the propellers were Aero Products (also a division of General Motors), model A6441FN-60[illegible]