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AIR SAFETY BOARD
REPORT

TO THE CIVIL AERONAUTICS AUTHORITY

AS A RESULT OF AN INVESTIGATION OF AN ACCIDENT INVOLVING AIRCRAFT

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Accident involving aircraft NX 19901

of the Boeing Aircraft Company, near

Alder, Washington, March 18, 1939.

An accident involving aircraft of United States registry NX 19901 having occurred in the vicinity of Alder, Washington, on the 18th day of March, 1939, with the resultant destruction of the aircraft and fatal injuries to all ten persons aboard, the Air Safety Board of the Civil Aeronautics Authority on the same day, directed that full and complete investigation of the accident be instigated immediately pursuant to the provisions of Section 702 (a)(2) of the Civil Aeronautics Act of 1938 (52 Stat. 973, 1013), and that the facts, conditions, and circumstances relating to the accident and the probable cause thereof be determined. It was further ordered that the investigation include such field investigation and research and such public or private hearings as might be considered necessary.

For the purpose of carrying out the above order, the Air Safety Board designated Frank Caldwell, Chief of the Investigation Section of the Air Safety Board, as Investigator in charge, and Fred M. Glass, Chief of the Examiners Section of the Air Safety Board, as legal advisor to the Investigator in charge during the field investigation and as Examiner empowered to order and conduct such public or private hearing or hearings in connection with the investigation as the Board might direct. It was further ordered that Mr. Caldwell and Mr. Glass be assisted and advised by Phil C. Salzman, Air Safety Investigator, Air Safety Board, and Osborn, Aeronautical Engineer, Air Safety Board, and pursuant to the provisions of the Civil Aeronautics Act of 1938, by the following technical personnel:

Major Carl F. Green (expert on aircraft structures),
Air Corps,
United States Army

Lt. F. R. Dent (expert on flutter and vibration),
Air Corps,
United States Army

Alfred S. Niles (expert on aircraft structure),
Leland-Stanford University
Palo Alto, California

R. D. Bedinger, Regional Supervisor,
Civil Aeronautics Authority

J. N. Boudwin, Senior Engineering Inspector,
Civil Aeronautics Authority

G. W. Haldeman, Engineering Inspector,
Civil Aeronautics Authority

H. C. Sine, Associate Aircraft Inspector,
Civil Aeronautics Authority

A. D. Niemeyer, Air Carrier Inspector,
Civil Aeronautics Authority

O. A. Rosto, Air Carrier Inspector (Maintenance),
Civil Aeronautics Authority

F. Hammerberg, Associate Aeronauticl Engineer,
Civil Aeronautics Authority

M. P. Crews, Aeronautical Engineer,
Civil Aeronautics Authority.

The air Safety Board further ordered that all phases of the investigation, research and hearing or hearings be carried out under the direct supervision of Thomas O. Hardin, Vice Chairman, and C. B. Allen, Member, Air Safety Board.

Investigation of the accident was begun on the 19th day of March, 1939, by the above-named personnel and the public hearing in connection therewith was ordered and held in the City of Seattle, State of Washington, on the 30th and 31st days of March, 1939, and the 3d, 4th, 5th, 6th and 7th days of April, 1939. In addition to the personnel already named, the assistance of the following agencies and organizations was solicited and obtained:

Civil Aeronautics Authority

Air Corps, United States Army

Bureau of Aeronautics, United States Navy

Bureau of Standards, United States Department of Commerce

Federal Bureau of Investigation, Department of Justice

National Advisory Committee for Aeronautics

Having considered the evidence adduced during the investigation, the following facts, conditions, and circumstances relating to the accident and conclusion as to the probable cause thereof are hereby reported, and recommendations, which, in the opinion of the Air Safety Board, will tend to prevent similar accidents in the future, are hereby made, to the Civil Aeronautics Authority:

FACTS, CONDITIONS AND CIRCUMSTANCES

Aircraft NX 19901, manufactured by the Boeing Aircraft Company of Seattle, Washington, a corporation organized and existing under and by virtue of the laws of the State of Washington, was a four-engine all-metal low-wing land monoplane designed for commercial operation and known as Boeing Model S-307. It had a wing span of 107' 3" and an overall length of 74' 4". Provision had been made for optional installations requiring crews of 3 or 6, during commercial cperation, and for a maximum of 33 passengers. The aircraft, which weighed 28,900 pounds empty and bore the manufacturer's serial number 1994, was issued a temporary experimental certificate by the Civil Aeronautics Authority on December 30, 1938, authorizing flight with a standard gross weight of 41,000 pounds. This authorization was amended in January, 1939, to permit a standard gross weight of 41,000 pounds and a provisional gross weight of 45,000. Both authorizations prohibited the carriage of persons other than bona fide members of the crew.

GENERAL DESCRIPTION OF AIRCRAFT

Power plants

This aircraft was powered with four Wright Cyclone GR-18200-102 engines manufactured by the Wright Aeronautical Corporation. Each of these four engines had aporoved power ratings as follows:

(a) Maximum, except take-off, at 6000 ft. pressure altitude - 900 horsepower with 35 inches of mercury manifold pressure and 2200 r.p.m.

(b) Maxium, except take-off, at sea level pressure altitude - 900 horsepower with 36.7 inches of mercury manifold pressure and 2200 r.p.m.

(c) Take-off (one minute) 1100 horsepower with 43 inches of mercury manifold pressure and 2200 or 2350 r.p.m.

The propellors installed on the aircraft were of the hydromatic quick-feathering type with constant speed control, and were manufactured by Hamilton Standard Propellers, Division of United Aircraft Corporation, East Hartford, Connecticut. Three of the propeller hubs were Model No. 23E50-31 and the fourth hub, which was installed on No. 2 engine, was Model No. 23E50-85. All of the propeller blades were Hamilton Standard Model No. 6153A-18. All hubs and blades installed on the aircraft had approved power and speed ratings which were satisfacory for the approved power and speed ratings of the engines.

Provision had been made in the aircraft for a total fuel capacity of 1700 gallons, to be carried in six tanks. A main tank with a total capacity of 425 gallons, and two auxiliary tanks, each with a capacity of 212½ gallons, were located in the inboard panel of each wing. A twenty-five gallon oil tank was installed in each engine nacelle, giving a total oil capacity of 100 gallons.

Aerodynamic Design and Performance

This aircraft was of exceptionally clean aerodynamic design, all wing and tail surfaces being full cantilever internally braced and any longitudinal section through the body being symmetrical.

The airfoil sections of the wing were the NACA 0018 symmetrical section at the root, tapered into the NACA 0010 symmetrical section at the tip. The wing was tapered in plan form, the root chord being approximately two and one-half times the tip chord. The wing had $4\frac{1}{2}°$ dihedral and was set at $3\frac{1}{2}°$ incidence with respect to the body center line.

Symmetrical airfoil sections were also used for the empennage surfaces, which consisted of a single fin and rudder and right and left horizontal stabilizers and elevators.

Test flights of the aircraft indicated that it would have a maximum speed in level flight at sea level of about 230 m.p.h., and a maximum level flight speed of approximately 250 m.p.h. at 9000 feet density altitude. The maximum L over D appeared to be about 15.

STRUCTURE

Design Loads

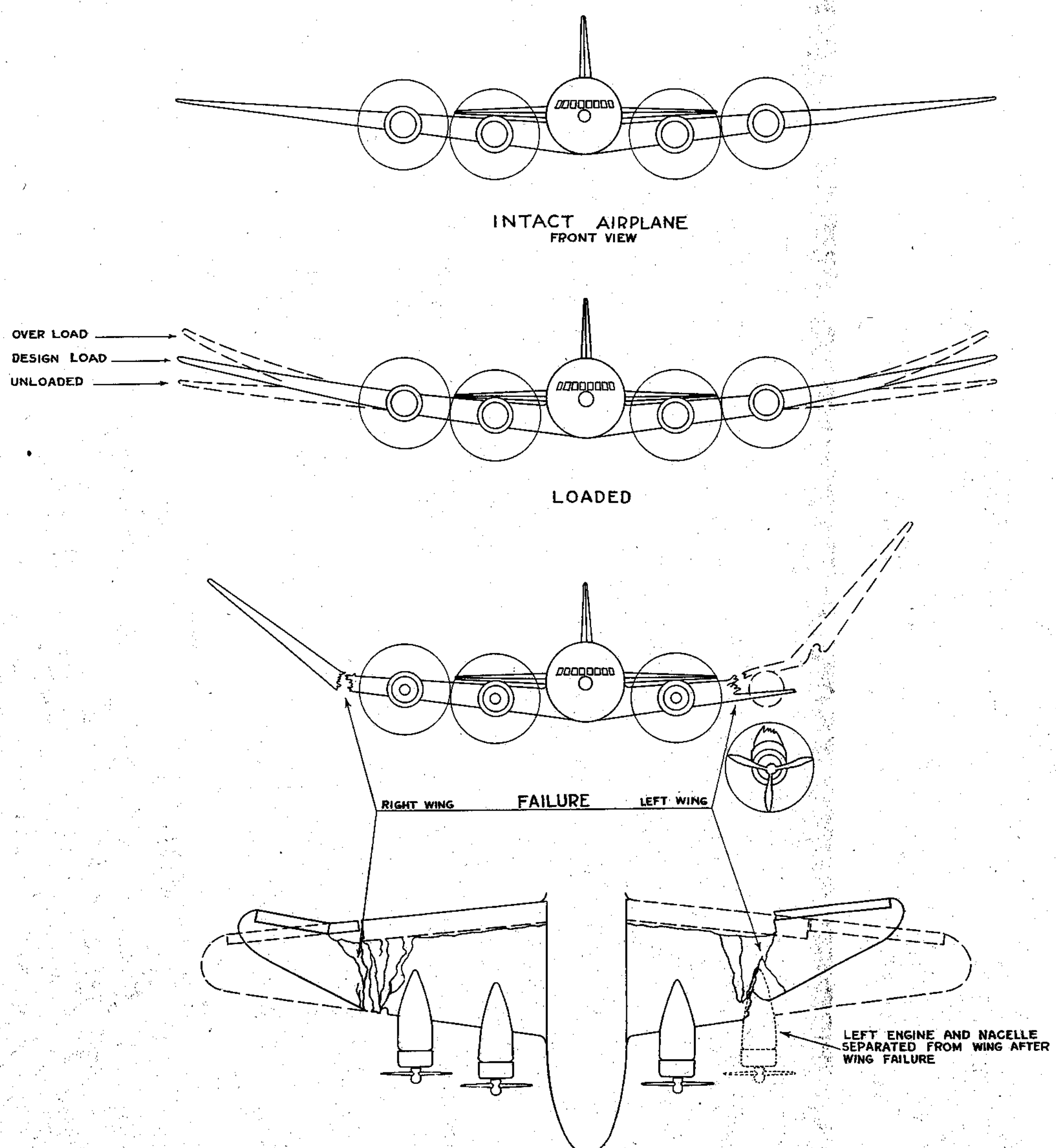
The design of this aircraft for accelerated flight was based upon an ultimate design load factor of 4.29 with 45,000 pounds provisional gross weight and 4.56 with a standard gross weight of 41,000 pounds. The limit loads for the wings, upon which the ultimate design loads were based, were such as to enable the aircraft to withstand 30 ft/sec. up or down gusts at sea level high speed in level flight of 240 m.p.h., and a 15 ft/sec. up or down gusts at a design gliding speed of 303 m.p.h.

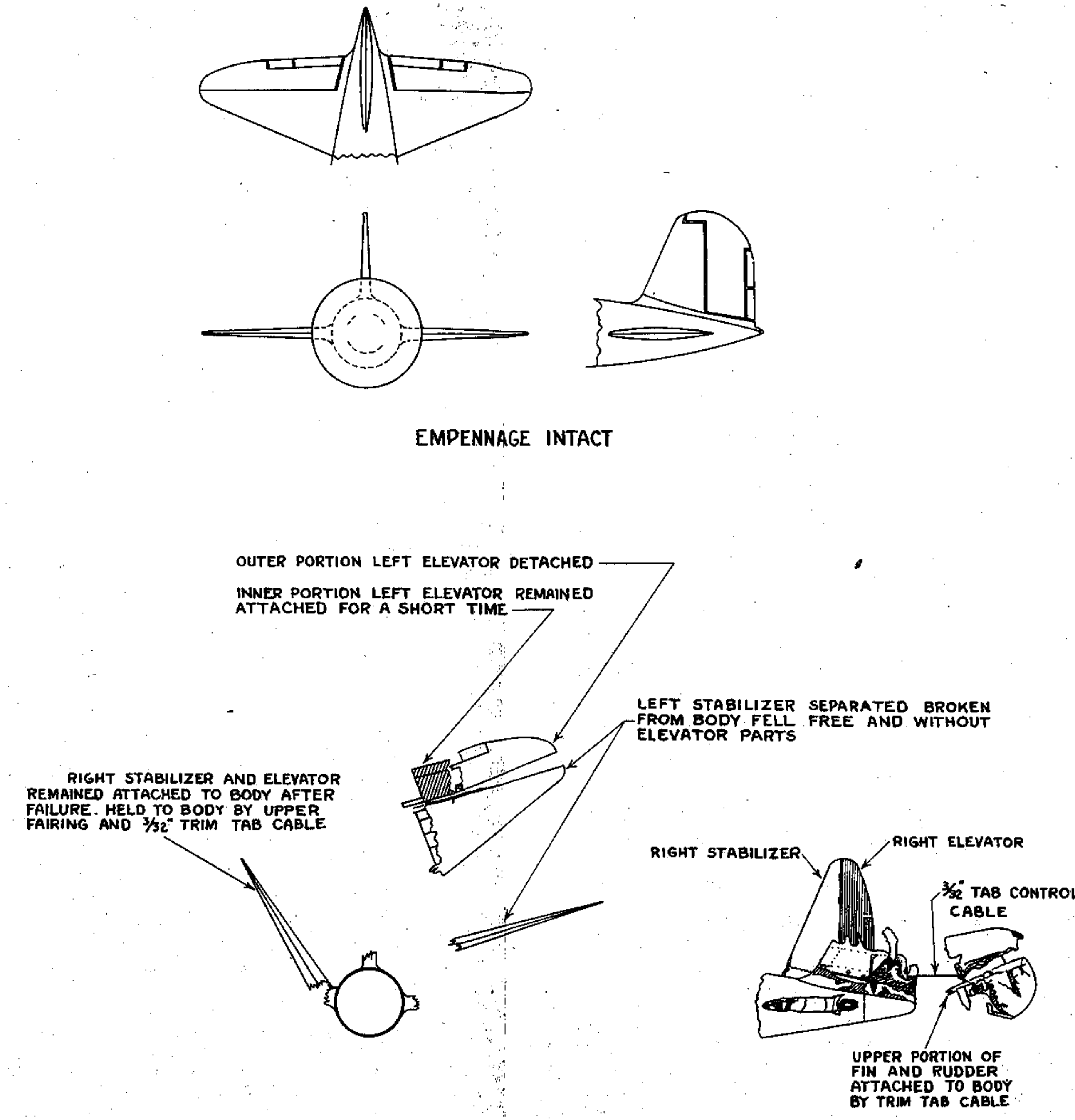
Wings

The wing was of semi-monocoque construction and derived its primary strength from the box formed by two wing spars and the smooth and corrugated aluminum alloy sheet of the top and bottom surfaces between the spars. The portions of the wing ribs between the spars served to maintain the airfoil shape for this part of the wing and to provide the proper restraint for the smooth and corrugated wing covering between the spars at each rib location. The wing spars were of the trussed type. The chords were aluminum alloy square tubing, and the web members were aluminum alloy barrel and rectangular section tubing. The spar web members were joined to the spar chords by means of aluminum alloy gusset plates which were riveted and bolted to the web members and the sides of the spar chords. The corrugated sheet of the top and bottom wing surfaces between the spars was of aluminum alloy, and the smooth skin covering, which was riveted to the corrugated sheet at each corrugation pitch, was Alclad aluminum alloy. The attachment of the smooth and corrugated covering between the spars was made through aluminum alloy spar cap plates riveted to the tops of the upper chord members and the bottoms of the lower chord members of the spars and to the covering by means of aluminum alloy rivets. With the exception of the special wing ribs in the vicinity of the wing fuel tanks, the wing ribs were all of the trussed type. Their chords were hat sections formed from aluminum alloy sheet and the web members, which were attached to the rib chords by means of aluminum alloy gusset plates riveted to the chords and web members, were made from aluminum alloy round, square and rectangular tubing.

The leading edge of the wing consisted of Alclad aluminum alloy sheet reinforced with corrugated sheet, which was attached to the nose ribs by means of aluminum alloy rivets. The attachment of the flat sheet to the corrugations was made by means of flush type aluminum alloy rivets. The portion of the wing aft of the rear spar was metal covered. The wings on either side of the body were made in three sections, consisting of a short tip section, an outboard panel, and an inboard panel incorporating the two engine nacelles on each side. The wing tip sections and the outboard panels were joined by means of aluminum alloy terminal fittings bolted to the spar chords of the outboard panel and by heat treated steel fittings bolted to the wing tip, the terminals being joined by means of heat treated steel bolts. The continuity of the wing tip and outboard panel covering was maintained at this joint from the rear to the leading edge on both top and bottom surfaces.

The attachment of the outboard wing panels to the inboard wing panels was made by means of aluminum alloy terminal fittings machined from forged blocks and bolted to the spar chords. S.A.E. 2330 steel taper pins were incorporated at the hinge. The corrugations and smooth skin between the spars at this connection were made continuous by means of a specially designed joint, which enabled the outboard and inboard surfaces to be connected with special screws and bolts. Though the continuity of the spar web members was also maintained at this





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FAILURES IN DESCENT

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point, no attempt was made to preserve the continuity of the leading edge of the wing at this connection, other than to cover up the gap. The inboard wing panel was connected to the body by means of special heat treated S.A.E. 4345 steel terminals. These terminals were bolted to the spar chords and the mating members of the supporting bulkheads in the body, while heat treated steel taper pins were employed at the hinge. The continuity of the spar web members was carried through into the main wing attachment bulkheads in the body.

The bending moments in the beam direction were resisted by the spar chords and the smooth and corrugated skin between the spars. The beam shears were carried by the trussed spars. The bending moments, due to chord loads, were carried by the front and rear spar chords, and the shears, due to chord loads, were supported jointly by the wing covering between the spar chords and by the spar chords. The torsion on the wing was resisted by the box formed by the two spars and the top and bottom covering between the spars. The wing was so designed that no permanent set or failure of the material between the spars was to be expected before permanent set or failure resulted to the spar chords. With the exception of a few minor variations, due mainly to differences in equipment installations, this wing was identical to the wing used on the Boeing Model B-17B aircraft, which was designed for 45,000 pounds gross weight and an ultimate design load factor of 4.5.

Though the wing used on aircraft NX 19901 was neither proof tested nor strength tested, the stress analysis methods were substantiated by a strength test of the outer wing panel of a Boeing Model XB-15 aircraft of the same general type of design. In addition, static tests of wings on other of the commercial and military aircraft previously constructed by the Boeing Aircraft Company had been analyzed in detail for the purpose of refining the stress analysis methods employed for this wing. The allowable loads employed for the numerous structural members were substantiated by means of numerous supplementary static tests where deemed necessary by the Boeing Aircraft Company during the course of the aircraft's construction.

Body

The fuselage of this aircraft, which was of the all-metal, semi-monocoque type construction, was shaped in such a manner that every transverse section from the extreme nose to the tip of the tail was circular. The shell of this body consisted of aluminum alloy sheet supported by a system of transverse rings of "C" section and by closely spaced continuous longitudinal stiffeners of extruded aluminum alloy bulb angle sections. The only exception to this construction were the main bulkheads which were hat sections.

The body was designed to be air-tight from its nose, including the pilots' cockpit, back to a pressure bulkhead at the rear of the passenger cabin. The part of the fuselage to be subjected to pressure was designed for a maximum internal pressure of six pounds per square inch, acting alone, and 3.5 pounds per square inch pressure combined with the maximum expected accelerated flight loads.

The normal maximum operating pressure was to be 2.5 pounds per square inch controlled as follows: To be built up at a uniform rate after a pressure altitude of 8000 feet was attained so as to reach the maximum internal pressure differential of 2.5 pounds per square inch at 14,600 feet; this differential pressure of 2.5 pounds per square inch to be maintained up to a pressure altitude of approximately 19,000 feet, at which altitude the equivalent pressure altitude inside the pressure cabin was to be approximately 12,000 feet. No provision was made in the design of this pressure system for operating at heights greater than a pressure altitude of 19,000 feet.

Because of this design for internal pressure in the cabin, the main entrance doors, the cargo loading doors, the accessory compartment door and all emergency exits opened inward. Also, all control cables were equipped with air seals where they passed through the body and pressure bulkhead.

The pressure bulkhead at the rear of the passenger cabin consisted of a convex diaphragm 102 inches in diameter, with a radius of curvature of 70.6 inches and was constructed of formed aluminum alloy sheet reinforced by aluminum alloy radial stiffeners.

The body of an identical aircraft (Serial No. 1995) had been proof tested to an internal pressure of 3.5 pounds per square inch acting simultaneously with the proof loads for the critical accelerated flight conditions. The body also had been proof tested for the critical flight condition loads without internal pressure in the pressure cabin.

Cockpit Windows

The 12 cockpit windows were mounted approximately flush with the contours of the body and in a continuous horizontal row of 6 on each side, starting from the very forward tip of the body back to points opposite the pilots' seats. Each pane had a vertical height of about 14 inches while the actual length of the panes depended upon the slope of the body contour. The panes were so installed that the only interruption to the pilots' view were the frame posts between the window panes. The front three panes on each side of the cockpit were fitted with 5/8 inch thick laminated glass (3 laminations of glass and 2 of plastic) while the three rear panes on each side were fitted with 3/8 inch "Plexiglass."

All panes were mounted permanently in a rigid stainless steel frame for the purpose of reducing deflections and avoiding magnetic compass deviations, with the exception of the third pane from the front on each side. These two panes were mounted in separate frames which were capable of being pulled inward by means of a handle to be provided on the rear edge of the frame. After being pulled inward, the windows were designed to be moved to the rear by means of an inside groove so installed as to be free from ice at all times. These movable panels were held in place when the windows were closed by four latches, one located at each corner of the frame. The two forward latches were operated together by a common handle and the two rear latches were also operated in the same manner. No handles for operating the windows had been installed on aircraft NX 19901.

These particular windows were designed to move directly inward before being moved to the rear so as to pull the glass directly away from any ice which might be present and break its seal by putting it in tension. This design also locates the slide guides to the inside of the aircraft where no ice can form and jam them. These windows were chosen as the ones to be opened for the reason that they permitted forward vision from the pilots' seats and yet were considered for enough aft from the nose of the body that the airstream would not enter the cockpit when they were opened.

The design of the heating system for this aircraft provided means wherein all the hot air from both the cabin heating systems could be directed onto the inside surfaces of the cockpit window panes. The effectiveness of the design in preventing ice from forming thereon or to loosen and melt any ice which had already accumulated had never been tested, however, as this equipment had not yet been installed in aircraft NX 19901.

Horizontal and Vertical Tail Surfaces

The horizontal tail surfaces were identical with those incorporated on the Boeing Model B-17B aircraft. The stabilizers were all-metal, stressed-skin surfaces, incorporating front and rear spars of the built-up "I" beam type, the metal covering being reinforced by "T" section aluminum alloy stiffeners. The stabilizer attachment to the body was accomplished by means of special heat treated steel fittings riveted to the stabilizer spar chords and bolted to the body attachment bulkhead members. The hinge attachments were made by means of heat treated steel taper pins. The elevators were of a conventional all-metal frame construction with fabric covering. The control and trim tabs on the elevators were of conventional all-metal construction. The two elevators were interconnected by means of a rigid aluminum alloy torque tube framed into the nose sections of the outboard portions of each elevator.

The fin and rudder of aircraft NX 19901 were of the same type design as the fin and rudder incorporated on the Boeing Model B-17B aircraft, though both were somewhat larger. The fin was of the same type all-metal structure as the stabilizers, and its attachment to the fuselage supporting bulkheads was made with the same type of fittings. The rudder frame, constructed of aluminum alloy, was fabric covered, while the construction of the rudder control and trim tabs was the same as that employed for the elevator control and trim tabs. At the lower portion of the rudder was a rigid aluminum alloy torque tube to which the rudder controls were attached. This torque tube was framed into the upper nose section of the rudder in a manner similar to the way the interconnecting tube between the two elevators was framed into the outboard sections of the elevators.

The horizontal and vertical tail surfaces were of the same type design as those employed on other large commercial and military aircraft manufactured by the Boeing Aircraft Company.

The tail surfaces were designed for ultimate loads either equal to or in excess of those required by the Civil Air Regulations. The stress analysis of these surfaces was supplemented by proof tests of the surfaces, including the control and trim tabs, for all critical loading conditions.

Ailerons

The ailerons, which were fabric covered and of the conventional Frise type, incorporated aluminum alloy tubular spars to which were attached conventional formed aluminum alloy sheet metal ribs. The left aileron incorporated a trim tab of all-metal construction.

The ailerons and the aileron trim tab were designed for ultimate loads either conforming to or in excess of those required by the Civil Air Regulations. Proof tests were resorted to for supplementing the stress analysis of the ailerons.

Control System

The control systems for the elevators, rudder and ailerons for aircraft NX 19901, were conventional, with the exception that the elevators and rudder were controlled for the normal range of their operation by means of the aerodynamic forces obtained from control tabs which were attached to the trailing edges of these surfaces, and worm gear and quadrant assemblies which were incorporated in the control system for each aileron.

The pilot's and co-pilot's elevator and aileron controls in the control cabin were of the conventional control wheel and column type, while the rudder controls were of the conventional stirrup type pedals, incorporating integral brake controls. The control systems for the elevators and rudder incorporated extra flexible steel cables from the control cabin to the surfaces and the control systems within the surfaces. The aileron control systems incorporated extra flexible steel cables from the control wheel in the cabin to the worm gear and quadrant units in the wing and push-pull rods from these mechanisms to the ailerons. All trim tabs were controlled by means of irreversible mechanisms located near the tabs, the irreversible mechanisms being controlled by the pilot and co-pilot by means of trim wheels operating extra flexible steel cables.

All control systems were designed for loads either conforming to or in excess of those required by the Civil Air Regulations. The stress analyses for all systems were supplemented by appropriate proof tests.

Landing Gear

The landing gear was of the conventional type, utilizing a tail wheel and with the front wheels mounted forward of the center of gravity. All wheels were retractable, the front wheels moving forward and upward into the inboard engine nacelles and the tail wheel moving rearward and up. The retraction was accomplished by screw struts operated by electric motors.

The shock absorbing system consisted of a combination oil and air strut for each wheel. The front wheels were mounted on overhung axles with the bending moments and torque being carried through the shock strut. The tail wheel was mounted on a cantilever knuckle which in turn was carried by a treadle supported by a combination oil and air shock absorbing strut.

Drop tests to which the landing gear and tail gear had been subjected indicated that the shock absorbing chracteristics of both met the pertinent requirements of the Civil Air Regulations.

Engine Naceles

The nacelles were of semi-monocoque construction from the engine mount attachment and firewall aft to the wing. This structure consisted of aluminum alloy sheet stiffened by means of a system of transverse and longitudinal aluminum alloy stiffeners. In addition to these stiffeners, four longerons were incorporated in each inboard nacelle to compensate for the large cut-out in the nacelle necessary to accommodate the landing gear when retracted

The engines were mounted on steel tube truss type engine mounts which in turn were bolted to the semi-monocoque nacelle structure.

With the exception of the attachments of the lower portions of the inboard nacelles to the wings in the vicinity of the cut-outs for the landing gear, the nacelles were attached to the wing leading edge and the upper and lower surfaces of the wing in the conventional manner.

That the strength of the nacelles and their attachment to the wings met the pertinent requirements of the Civil Air Regulations, had been substantiated by stress analysis and by supplementary proof tests.

Provisions for the Prevention of Flutter

Consideration was given in the design of aircraft NX 19901 to the provision of structural rigidity for its various component parts in excess of that required for structural strength so as to obtain structures with natural frequencies as high as it was feasible to obtain with the sizes and types of structure employed.

Complete vibration tests were conducted with the aircraft on the ground to determine the natural frequencies of the different modes of vibration for the following component parts of the aircraft; wings; body; stabilizers; elevators; elevator control tabs; elevator trim tabs; fin; rudder; rudder control tab; rudder trim tab; ailerons; aileron trim tab; and, all control surface control systems. The elevator vibration tests were conducted with the 74.3 pound elevator mass-balance weight in place forward of the elevator hinge line in the plane of symmetry of the aircraft. These vibration test data were studied for evidence of possible trouble due to inadequate separation of the natural frequencies of certain pairs of the component parts.

The mass properties of the control surfaces were as follows:

(a) The elevators, with the 60 pound elevator mass balance weight located 21.4 inches ahead of the elevator hinge line installed for Test Flight No. 19 on

March 18, 1939, had 55.2 percent static balance. The coefficient of dynamic balance for the elevators with respect to the hinge line and the center line of the aircraft was 0.10. The elevator control tabs were static balanced by means of lead weights placed ahead of their hinge lines.

(b) The rudder had 104 percent static balance, and a dynamic balance coefficient of 0.028 with respect to the hinge line and the center line of the aircraft. The mass balancing was realized by means of a 17.0 pound weight in the tip overhang, placed 24.8 inches ahead of the hinge line, and a 50.0 pound weight on an arm at the body center line 24.2 inches forward of the rudder hinge line. The rudder control tab was not statically balanced.

(c) There were no mass balance weights incorporated on the ailerons, therefore the mass properties given for them were based on the actual structural weight distribution of the ailerons. The ailerons were 18.7 percent statistically balanced. They had a coefficient of dynamic balance of 0.292 with respect to the aileron hinge line and the center line of the fuselage. The semi-irreversible aileron control units employed for each aileron were considered to be a protection against flutter, as their purpose was to prevent or retard lagging tendencies.

(d) No mass balancing was employed for the trim tabs because of the rigidity of their structure and irreversibility of their control systems.

The aircraft, with the foregoing provisions for prevention of flutter, was dived during a test flight to an indicated air speed of 303 m.p.h. with no evidence of any flutter trouble. For Test Flight Nos. 16, 17 and 18 of aircraft NX 19901, a 50-pound elevator mass balance weight was installed in place of the 60-pound elevator mass balance weight, which installation provided 47 percent static balance for the elevators. During Test Flight No. 18 of March 17, 1939, made with the 50-pound elevator mass balance weight, trouble was encountered with flutter at an indicated air speed of approximately 240 m.p.h. As the result of this experience, the 60-pound elevator mass balance weight, providing 55.2 percent static balance, was installed on the elevators for Test Flight No 19 of March 18, 1939.

TEST FLIGHT HISTORY *

Flight testing of NX 19901 was begun immediately subsequent to the issuance of the experimental certificate on the subject aircraft by the Civil Aeronautics Authority. As chief test pilot on these flights, the Boeing Aircraft Company employed Edmund T. Allen, a test pilot and aeronautical engineer.

These flights, exclusive of certain tests not considered here pertinent, were conducted from Boeing Field, Seattle, Washington, and consisted of the following tests:

The first test, consisting of ground tests as to functioning of the brakes, including effectiveness, forces required and cooling, was made on December 30, 1938, with a crew of 6, gross weight of 30,884 pounds, and center of gravity location 25.5 percent. The test was concluded with checks as to ground handling characteristics at all speeds up to take-off speed, and the functioning of the engines and propellers.

Following the completion of certain adjustments recommended after this initial test, the second test was made the following day with a crew of 6, gross weight of 30,605 pounds and center of gravity location 25 percent. After determination as to the effectiveness of the adjustments which had been made, the speed of the aircraft was gradually increased until take-off speed was reached. Three runs were made down the length of Boeing Field, during each of which the wheels were slightly off the ground for ten or fifteen seconds. The controls, flaps and landing gear shock absorber system were tested during the remainder of the test and determination made as to the balance and stability of the aircraft. Minor adjustments of the engines carburetor air temperature control and oil temperature control were recommended by Pilot Allen after the completion of the test.

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* BOARD NOTE: All references to speeds in this report are in terms of indicated air speeds unless specifically otherwise noted. All reference to center of gravity locations are in percent of the mean aerodynamic chord and referenced to its leading edge.

Later the same day, December 31, Test Flight No. 3, was made with a crew of 5 and a gross load of 32,000 pounds. This test was concerned with brake testing at high speeds and more severe application of the brakes as might be expected after landing. After testing the directional stability of the aircraft on the ground and finding it satisfactory, a take-off was made. During the flight, both static and dynamic longitudinal and yaw stability, control forces and balance were tested, and the operation of various control instrument readings was checked. Stalls, which were approached with power off and power on and flap positions varying from zero degrees to 30 degrees for the purpose of checking controllability and stability during landings, occurred at 65 m.p.h. indicated air speed with flaps down and 73 m.p.h. indicated air speed with flaps up but without any indicated tendencies to stall abruptly.

Test Flight No. 4 was made on January 4, 1939, with a crew of 7, gross weight of 37,000 pounds, center of gravity location 27.7 percent with wheels down, and 26.9 percent with wheels up. The first tests made on this flight were measurement of take-off distance and initial rate of climb, followed by engine, oil and brake cooling tests and air speed meter calibrations by means of a trailing bomb. A check was then made to determine longitudinal stability oscillations at gradually increased rearward locations of the center of gravity, following which a determination of rudder forces under extreme conditions of unbalanced engine operation was made in addition to a check of rate of climb with two engines inoperative. A check was also made to determine elevator forces in changing from power on to power off and from flaps up to flaps down position. Further checks were made to determine spiral stability, control forces in maneuvers, and minimum landing speed and length of landing run. Additional checks on this flight also included a test of rudder adequacy with No. 1 engine completely throttled and then with both Nos. 1 and 2 engines on the left side completely throttled at various airspeeds, side-slip angles and trim tab positions.

The rudder force was reported to be lighter than the Boeing Model Y1B-17 and approximately as light as the rudder forces of the Boeing Model 314. The rudder force increased smoothly as thrust was simultaneously increased on one outboard engine and decreased on the opposite outboard engine. When engine Nos. 1 and 2 were throttled and the propellers left windmilling in low pitch (the worst condition from the standpoint of increasing torque reaction) and engines No. 3 and 4 were operated at full throttle (1900 r.p.m. at 32 inches Hg. manifold pressure), the rudder force increased to 100 pounds with full assistance of the rudder control tab. A side-slip of a few degrees relieved the rudder load required to keep the aircraft from turning under these conditions. It was apparent that there was no distinct yaw period measurable thus far. The elevator forces were reported by Pilot Allen as being so light as to create the likelihood of passengers being inadvertently lifted from their seats.

During the flight the windshield became badly iced both inside and outside.

The next test flight, No. 5, was made on January 11, 1939, with a crew of 9, gross weight of 40,990 pounds and center of gravity location 26 percent. Although changes had been made in the aircraft since the last test flight, the majority of such changes were relatively unimportant adjustments which did not affect the general operation of the aircraft. Tests as to three-engine take-off performance of the aircraft were the first to be conducted on this flight. The procedure on each of these tests was to begin the take-off run with all engines operating and to attempt the actual take-off after No. 1 engine had been cut at a predetermined distance down the runway. During the first of these tests No. 1 engine was throttled after a run of 1250 feet. Lack of directional control because of insufficient airspeed, however, necessitated the throttle of No. 1 engine being reopened on this particular test. On the second attempt No. 1 engine was throttled after a speed of 90 m.p.h. indicated airspeed had been reached at a point 1650 feet from the start of the take-off. Though sufficient rudder control was present to permit continuation of the flight under this condition, the directional control was marginal.

Landing tests conducted with various center of gravity positions indicated that as the C.G. was shifted progressively forward, it became increasingly difficult to get the tail down. The tail wheel was not quite touching the ground during the landings made with the 22 percent or the 20 percent C.G. locations. The indicated airspeeds during these landings increased from 60 m.p.h. at 26 percent C.G.; to 64 m.p.h. at 24 percent C.G.; to 67 m.p.h. at 22 percent C.G.; and, to 70 m.p.h. at 20 percent C.G. Climbs at rated power to determine optimum speeds for climbing at various altitudes were also made during this test.

Test Flight No. 6, which was made on January 12, with a crew of 8 and gross weight of 41,000 pounds and 27 percent center of gravity location with wheels down, included saw tooth climbs, dives at 240 m.p.h., and longitudinal stability tests under varying conditions of power and locations of wing flaps and landing gear. All propellers overspeeded approximately 100 r.p.m. at an indicated airspeed of 140 m.p.h. shortly after take-off. A climb was started at 2000 feet, using rated power and continued to 8000 feet altitude. At this altitude the throttles were closed and a descent was made to 5500 feet, from which altitude saw-tooth climbs were started at 140 m.p.h. indicated air speed. This flight test was discontinued during this climb because the oil temperature of No. 2 engine began to rise and reached 102° Centigrade.

Test Flight No. 7 was made on January 13 with a crew of 8, a gross weight of 41,050 pounds and center of gravity locations of 26.75 percent with landing gear down and 26 percent with landing gear up. Following the take-off and climb, a dive was made to determine the action of the controls and to check for flutter and vibration on reaching a speed of 250 m.p.h. indicated airspeed. Neither flutter or excessive vibration were encountered during the dive. Following this test, the level flight high speed at sea level was determined. Longitudinal stability tests were made to find the most rearward center of gravity position at which the aircraft was longitudinally stable. Rated power climbs made to determine cooling and rates of climb, completed the flight.

The determination of longitudinal stability of the aircraft under varying conditions of power and wing flaps and landing gear positions constituted the purpose of Test Flight No. 8, which was made on January 14, with a crew of 8, gross weight of 41,000 pounds, and center of gravity location of 27 percent with wheels down and 26 percent with wheels up. On completion of these stability tests, high speed tests using rated power were conducted at 10,000 feet altitude.

Prior to Test Flight No. 9, which was made on January 15 with a crew of 8, gross weight of 41,000 pounds and center of gravity 26 percent, the elevator mass balance weight was changed from 74.3 to 60 pounds. After take-off on this flight, a steady climb was made at rated power to 12,000 feet, followed by saw-tooth climbs to 16,000 feet. A high speed run was made at 15,000 feet using rated power, following which high speed and cruising speed runs were made at 10,000 feet. A dive was then executed during which an indicated airspeed of 260 m.p.h. was attained.

Longitudinal stability tests were conducted to determine the rearmost center of gravity positions under varying power conditions and positions of wing flaps and landing gear. Propellers No. 1 and 2 were feathered and the rate of climb was checked at an altitude of 10,000 feet.

Test Flight No. 10 was made on January 17 with a crew of 8, gross weight of 45,004 pounds, and with center of gravity location of 27 percent. The first test on this flight was the measurement of the take-off distance required to reach 85 m.p.h. indicated airspeed and of determination of altitude reached one minute after take-off. The tests which followed were a climb using rated power at 120 m.p.h. indicated airspeed, speed runs in level flight at 10,000 feet with various amounts of power, determination of dynamic longitudinal stability using rated power at 10,000 feet with the landing wheels in both the up and down positions with Nos. 3 and 4 engines operating at rated power and with Nos. 1 and 2 engines inoperative and their propellers feathered. Fuel dump tests were conducted in a glide at 110 m.p.h. Gentle turns were made during this test and a slight amount of spray touched the stabilizer while the aircraft was in a turn. Further fuel test dumps were made with the aircraft in a climbing attitude at 110 m.p.h. during which tests the spray stayed about 8 inches below the tail surfaces.

Test Flight No. 11 was made on January 18 with a crew of 8, gross weight of 45,000 pounds, and with the center of gravity located at 27.6 percent. A climb was made from 2000 to 12,000 feet altitude for the purpose of determining the average rate of climb using rated power, following which the aircraft was climbed from 12,500 feet to 14,480 feet altitude with No. 1 engine inoperative and its propeller feathered for the purpose of gathering information from which various ceilings could be determined. After descending to an altitude of 10,000 feet, high speed runs were made using rated power and cruising speed runs were made using cruising power. Longitudinal stability tests were conducted at this same altitude with the center of gravity located at 28 percent. The test flight was concluded with fuel dumping tests: (1) in a glide 110 m.p.h., zero flaps, spray 3 feet below the horizontal tail surfaces; (2) 110 m.p.h. glide, 15° flaps, spray 6 feet below horizontal tail surfaces; (3) 110 m.p.h. climb, zero flaps, spray 6 feet below horizontal tail surfaces; (4) 110 m.p.h. climb, 15° flaps, spray 9 feet below horizontal tail surfaces; slight amount of spray touched the flap.

The following day, January 19, Test Flight No. 12 was made with a crew of 8, center of gravity locations 28 percent with wheels up and 28.67 percent with wheels down, and a gross weight of 45,000 pounds. After an altitude of 4000 feet had been reached, a climb to 8000 feet was made at approximately 125 m.p.h. indicated airspeed using cruising power to obtain rate of climb data. A high speed run was then made at 7000 feet using rated power, following which an investigation of the longitudinal stability characteristics was made, with the center of gravity at 28 percent, under varying trim and power conditions, with landing wheels up and down and with wing flaps 0 degrees and 45 degrees.

Following these longitudinal stability tests, another cruising power climb was made from 6000 feet to 8500 feet and additional speed runs were made with various power conditions at an altitude of 10,000 feet. Tests were also made to determine the time required to feather and unfeather the propellers for Nos. 3 and 4 engines. The aircraft was then stalled on five successive occasions, three of which stalls were with power off and two with power on. The wing flaps were fully retracted during all five of these stalls, with the exception of two of the power-off stalls, during which 45 degree flap settings were used. On the first of these tests, which was made with power off and flaps at 0°, the aircraft stalled at an indicated airspeed of 80 m.p.h. and lost 600 feet alitude before recovery was effected at an indicated airspeed of 90 m.p.h. The test was repeated with power on and the aircraft stalled at an indicated airspeed of 66 m.p.h. and lost 200 feet before recovery was made at a speed of 110 m.p.h. The third stall was a duplication of the second and the aircraft stalled at 68 m.p.h. indicated airspeed and lost 500 feet altitude before recovery was made at the same speed as in the preceding test. Flaps were then lowered to 45° for the next two tests, both of which were made with power off. In the first of these tests at this flap setting, the aircraft stalled at 68 m.p.h. and in the second at a speed of 69 m.p.h. Recovery was effected in the first test at a speed of 80 m.p.h. indicated airspeed after the loss of 400 feet of altitude, and in the second test at indicated airspeed of 90 m.p.h. after the loss of an identical amount of altitude. Determinations, made by means of tufts fastened to the upper surfaces of both wings, indicated that during power-off stalls with no flaps, the fillets at the intersections of the wings with the fuselage and the inboard sections of the wing were completely stalled, and that the outboard sections of the wings were unstalled. In the power-off, flaps down stalls, the wings were only partially stalled at the fillets, though the portions of the wings just inboard of the ends of the ailerons were completely stalled. These same conditions existed during the power-on stalls with no flaps.

Later the same day, January 19, with a crew of 3, gross weight 41,000 pounds and center of gravity position 26.2 percent with wheels up, test flight No. 13 was made to determine whether the aircraft had any tendency to flutter during dives up to its design gliding speed. After an initial climb to 13,000 feet, a dive was made from this altitude to 9,000 feet with the propellers set for 1900 r.p.m. and with the full throttle manifold pressure reaching 33 inches. Though an indicated airspeed of 285 m.p.h. was reached during the course of the dive, there was no excessive vibration of any of the control surfaces or any part of the aircraft, except for a very fine engine vibration, which had previously been noticed at 2200 r.p.m. at 140 m.p.h. indicated air speed. Using the same setting on engine controls, with the exception of partial closing of the throttle at the bottom of the dive to prevent the manifold pressure from exceeding 34 inches, the second dive was made from 11,000 feet to 8,000 feet, during which an indicated airspeed of 301 m.p.h. was reached. Another dive was made from an altitude of 9,500 feet to 6,000 feet though it was started at a slightly steeper angle for the purpose of maintaining a dive angle of less than 30° when maximum speed was reached. No excessive vibration was noted in either this or the preceding dive, though the indicated airspeed was 301 m.p.h. in the first dive and 303 m.p.h. in the second. An indicated airspeed of 142 m.p.h. was reached in the fourth dive which was made with 45° flaps. Elevator forces were then measured with the aircraft trimmed at various speeds.

The only change made in NX 19901, prior to Test Flight No. 14, which was made on January 20, 1939, with a crew of 7, gross weight of 41,000 pounds and center of gravity of 22 percent, wheels down, was a change in the dump valve chute on the right main fuel tank. The first part of this flight was devoted to take-offs and landings at forward center of gravity positions in order to determine limited forward positions for satisfactory landings. Maximum elevator angles of 17°, at which point the controls hit the stops, were used on both landings, which were made with 41,000 pounds gross weight and a center of gravity location of 22 per cent. Landing was effected on the first test at 70 m.p.h. indicated airspeed with 45° flap angle. On the second test, the landing was made at 67 m.p.h. indicated airspeed with 30° flap angle. Following the third take-off, speed tests were run at sea level under varying power conditions, and at 7,500 feet altitude with rated power.

Power-on stall tests were then made with 45° flaps. The approaches to these stalls were gradual, as had likewise been true in all stall tests which had been made up to this time, and the wings and ailerons were never at any time completely stalled. It was the custom of Pilot Allen to place the controls in position for recovery as soon as the nose began to drop, and at no time were the elevators held in the up position until the nose had dropped below level flight position. The aircraft had shown no tendency to fall off on either wing during any of these stall tests.

Spiral stability tests were made under varying power conditions, followed by a fuel dumping test, during which dyed water was dumped during a glide of 110 m.p.h. with flap 0° and landing gear up. No spray went within two feet of the stabilizer during the test.

Further landing tests were made with 33,350 pounds gross weight and with a center of gravity location of 19.8 percent. On the first of these tests, which was made with 45° flaps at 65 m.p.h. indicated airspeed, a pull of 80 pounds was required to force the elevator against the stop and a two-point landing was made with the tail wheel two to three feet above the ground. An identical pull was likewise required on the second test which was made with 30° flaps at a speed of 67 m.p.h. and with the tail wheel two to three feet off the ground. Pilot Allen reported that the aircraft was nose heavy under this loading condition to such an extent that it was very easy to raise the tail during these tests through the use of the brakes.

Test Flight No. 15, the purpose of which was to dump water from the right main fuel tank in a 110-mile per hour glide, was made on January 20th with a crew of five. The aircraft was loaded to 34,261 pounds, with a center of gravity location of 25.2 percent at the time of take-off. The report of the flight indicated that some of the dyed water went on the flaps when they had been lowered to 15 degrees after most of the water was out of the tanks, but that no water touched the tail surfaces of the aircraft. Both the landing and take-off on this test were made by Co-pilot Barr, who was officially approved as First Pilot on this model aircraft by Pilot Edmund T. Allen immediately following this flight. Mr. Barr, who had served as co-pilot on the majority of the test flights made with aircraft NX 19901 to this date by Mr. Allen, had not, according to Mr. Allen, been at the controls during any of the stall tests made with this aircraft.

Julius Barr was in command of the aircraft as pilot for the first time on January 21, 1939, when, with a gross weight of 45,000 pounds aboard, taxying characteristics of the aircraft were tested as a part of test No. 16. The report of Pilot Barr on this test indicates that the results of such test did not differ materially from the original taxying tests conducted by Pilot Allen. At the conclusion of this test, during which the aircraft did not leave the ground, it was returned to the Boeing Aircraft Company plant for changes and further installations.

After a lapse of almost two months, test No. 16 was resumed on March 16 with a crew of six men, gross weight of 35,360 pounds, center of gravity location of 21 percent, and with Julius Barr as pilot and Earl Ferguson as co-pilot. During the period the aircraft was in the Boeing plant the following major changes had been made: heating and ventilating and cabin pressure systems had been installed; a 50-pound mass balance weight was installed on the elevators instead of the 60-pound weight that was used on all flights subsequent to test flight No. 9; elevator travel stops had been reset to permit 25 degrees "up travel" instead of 17 degrees, which had been available on previous tests; long exhaust channels were installed in No. 1 and No. 4 engine nacelles. Accelerometers, including a visual accelerometer, were installed; and, an additional vacuum selector valve was installed. The cabin heating and pressure systems were not, however, connected for operation.

Julius A. Barr received his flight training in the United States Army Air Corps in 1926 and 1927 and had accumulated a total flight time of approximately 5,000 hours prior to the time of taking off on this test flight, which was his first flight on NX 19901 as first pilot. Approximately 2,030 hours of this time was in single engine aircraft, while approximately 2,240 hours was in twin engine aircraft and 765 hours in three-engine aircraft. Since his employment by the Boeing Aircraft Company as a test pilot on November 16, 1938, Mr. Barr had accumulated a total of 3 hours 5 minutes as an observer and 9 hours and 17 minutes as co-pilot on the Boeing Model 314 (Pan American Flying Boat); and a total of one hour 52 minutes as observer and 17 hours 55 minutes as co-pilot in NX 19901, Boeing Model 307.

Co-pilot Earl Ferguson had accumulated a total of approximately 1400 flying hours since his first flight training in 1932 as a reserve student officer at the Naval Air Station, Pensacola, Florida. His flying experience, in addition to various types of single engine and twin engine aircraft, included, while employed as a test pilot by Boeing Aircraft Company, 9 hours and 15 minutes as co-pilot of the Boeing Model Y1B-17A (United States Army Flying Fortress); 27 hours 57 minutes as observer, 18 hours 1 minute as co-pilot, and 16 hours 39 minutes as pilot of the Boeing Model 314 (Pan American Flying Boat); and 2 hours 46 minutes as co-pilot of NX 19901, Model 307.

Following the take-off on test flight No. 16, a climb was made to 4000 feet using 53 percent rated power. At this altitude No. 1 engine was throttled and tests were conducted to ascertain the minimum flying speeds at which directional control could be maintained with flaps at zero degrees. Propeller feathering tests were then conducted, following which a landing was made with No. 2 engine inoperative and its propeller feathered.

Test Flight No. 17, a very brief flight, was made on the morning of March 17 with a crew of 10, gross weight of 41,000 pounds and center of gravity location of 21 percent with wheels down. The aircraft was held with the brakes at the start of the take-off until all engines were at 1500 r.p.m. The brakes were then released and the throttles were gradually opened, as the aircraft proceeded down the runway, until take-off power was attained. The aircraft left the ground at an indicated airspeed of 78 m.p.h. Checking of the feathering operations of No. 2 propeller was the only test conducted during this flight.

Later the same day, March 17, NX 19901 took off on test flight No. 18 with a crew of 10, gross weight of 45,000 pounds and center of gravity location of 25 percent. The 50-pound elevator mass balance weight was still installed on the aircraft. The pilot's report of this flight was never completed by Pilot Barr, although notes taken during the flight by Aerodynamist Cram and statements made subsequently by crew members established with some degree of completeness the details of the flight. Julius Barr was in command of the aircraft as first pilot, although Mr. Hull, Chief Pilot of Transcontinental and Western Air, Inc., was a member of the crew and flew the aircraft from the co-pilot's seat during various stalls, side-slips and other maneuvers.

During a dive at an indicated airspeed of approximately 240 m.p.h. subsequent to a stall, the right rear emergency hatch blew into the cabin and a mode of flutter involving the elevators (apparently due to improper mass balancing) was encountered. Inspection revealed that the emergency hatch had not been properly secured in place. The aircraft was again placed in a power glide, after the emergency hatch had been securely replaced, and the same mode of flutter was experienced at an indicated airspeed of approximately 240 m.p.h. The 50-pound elevator static balance weight was replaced following the completion of the flight with the 60-pound weight which had been used on test flights Nos. 9 to 16 (first part) inclusive.

On the evening of March 17, subsequent to the completion of Test Flight No. 18, Mr. Pieter Guillonard, technical director of Royal Dutch Airlines, and Mr. A. G. von Baumhauer, of the Dutch Air Ministry who had just arrived in Seattle and who were inspecting and considering the Boeing 307 for possible commercial operation by the Royal Dutch Airlines, conferred at length with representatives of the Boeing Aircraft Company concerning the flight characteristics of the aircraft and certain tests of such characteristics in which the two Dutch representatives were particularly interested. Tentative plans were made to include the tests suggested by Mr. von Baumhauer and Mr. Guillonard in the tests to be conducted on test flight No. 19 the following day. The first of these tests in which Mr. von Baumhauer expressed great interest, was that of cutting No. 1 engine during a 100 to 135 m.p.h. climb following take-off and observing the reaction of the aircraft if no correction whatsoever was made with the rudder. The second test consisted of stalls under varying conditions, i.e., power-on with flaps up and landing gear up; power-off with flaps up and landing gear up; power-on with flaps down and landing gear down; and, power-off with flaps down and landing gear down. Mr. von Baumhauer emphasized, during the course of the conference, his keen interest in "complete stalls", emphasizing his conception of such maneuvers as embracing a complete breakdown of lift upon the wings as distinguished from stall tests from which recovery was made as soon as good indications of the stalled condition became apparent.

Mr. von Baumhauer, who had made a special study of stability and control of aircraft, had previously delivered a lecture before the Royal Aeronautical Society on the subject "Testing the Stability and Control of Aeroplanes." An article appearing in the March 15, 1939 issue of "The Aeroplane," an English publication, gives a summary of such lecture, in which Mr. von Baumhauer sets forth his views concerning aircraft stability and control, particularly with regard to certain tests he considered as necessary to prove the desired requirements. The tests required by him as set forth in this article are as follows:

GROUP NO.1. (a) "The first test is made in rectilinear symmetrical flight. The throttle is fixed but the tests are repeated with the Centre of Gravity in its most extreme forward and backward positions. The elevator position is measured for a series of speeds. This test is especially important, as it deals with the static longitudinal stability of an aeroplane."

(b) "The next test is side-slip. The positions of the aileron and rudder are measured for a series of banks which show whether there is positive weathercock stability and whether the rolling couple which results from side-slip has a restoring tendency."

(c) "The third test deals with steady turns. The positions of the three controls are read for left and right-hand turns with little and steep bank without side-slip. The control forces are estimated. Abnormal behaviour in turns may thus be found."

(d) "The fourth test is to find the behaviour in a steady turn with the three controls held fixed."

(e) "The fifth and last test in the first group gives an impression of the reaction of the aeroplane when a given force is applied to the elevator control."

GROUP No. 2. "The second group of tests are to study the effects of quick control movements from the position of balance. These are made for rectilinear flight with elevator and rudder."

GROUP NO. 3. "The third group of tests, for behaviour when going into and coming out of a turn, are made for a few values of bank. They are done, first with all the controls, next with the rudder only and then with ailerons only."

GROUP NO.4. "The fourth group of tests are to find out the effect of throttling or opening up one or more motors. These are done while gliding first symmetrically with all the motors, then symmetrically with one or all motors on one side only."

GROUP NO. 5. "The Dutch requirements provide that aeroplanes shall continue in straight gliding flight when the control column is fully back and the other controls are kept central. Some aeroplanes on the market could not fulfill this requirement, so the requirement was altered so that heavier machines were accepted if they gave adequate warning before their stall."

GROUP NO. 6. "The sixth group of requirements covers various miscellaneous items which must be looked into, including the behaviour of the automatic pilot."

In addition to the above, Mr. von Baumhauer expressed his views, during the course of the lecture, to the effect that control forces should not be lightened to such an extent that a pilot would not have a definite reaction to the response of the aircraft to the controls by forces required in the application of the controls.

Mr. von Baumhauer held a private pilot's license issued on November 28, 1931, by the Dutch Air Ministry, the records of which show that his total flying time as pilot amounted to 116 hours, and that he had no experience as pilot or co-pilot of 4-engine aircraft, but had been observer on trial flights of 4-engine Fokker F-22 and F-36 aircraft.

On the morning following the conference between Mr. Guillonard, Mr. von Baumhauer and representatives of the Boeing Aircraft Company, March 18, 1939, a flight plan concerning detailed tests to be conducted on test flight No. 19 was prepared by Boeing personnel and approved by supervisory officials of the Boeing Aircraft Company. Included in the flight plan as adopted were the following items:

"Gross Weight 43,000.
C. G. 26.8%

Aux tanks full
Main tanks <u>water</u>.

"Changes since last flight.
a - Elev. static bal. - 59 lb. 13 oz., installed.

b - Repair flap autosyn.
c - Co pilots airspeed lines checked.
d - All emergency hatches realigned.
e - Alternator replaced.
f - Main gas tanks outlets plugged.
g - No. 4 prop governor set for 2200 r.p.m.

"1. <u>Take-off</u> - 15° flap, accelerate to 100 m.p.h. and hold climb for 1 min. at 100 m.p.h. with T.O. power.

"2. <u>Climb</u> to 10,000 ft. using M.E.T.O. power.

"3. Longitudinal stability at 28% c.g.

"4. Side slips.

"5. Directional stability.

"6. Throttle No. 1 engine, hold rudder in neutral and hold for 3 to 5 sec.

"7. Repeat with Nos. 1 and 2 throttled.

"8. Stalls - gear up - flaps 0 - power on & off.
" down " 45 " " " "

"9. Simulate overshooting of airport
a. Trim for glide, flaps 45° gear down.
b. Give full power.

"10. Landing. 26% (center of gravity location) 45° flap."

It appears, on referring to the above flight plan approved for test flight No. 19, that the tests were to be conducted along the lines outlined by Mr. von Baumhauer before the Royal Aeronautical Society.

Two special instruments were installed in the aircraft prior to take-off on Trip 19 for the purpose of obtaining special data desired by Mr. von Baumhauer from the flight. The first of these was a special instrument installed on the co-pilot's control column for measuring the elevator forces in pounds required to operate the elevators during specific maneuvers. This instrument was installed to be operated from the co-pilot's seat. The second instrument was the property of Mr. von Baumhauer and was mounted on the top of the instrument panel in front of the co-pilot. Its purpose was to indicate the travel of the control cables of the elevators, rudder and ailerons while specific maneuvers were being executed, and from these data to obtain information concerning the positions of these control surfaces in the various maneuvers. The indication of the control cable travels were accomplished by means of light cords extending from the instrument and connected to the control cables so that any movement of the controls would be registered without interfering with the operation of the controls.

NX 19901 took off from Boeing Field, Seattle, Washington, at 12:57 p.m. (PST), Saturday, March 18, 1939, on test flight No. 19 with a gross weight of 43,000 pounds, center of gravity location of 27.8 percent, and the following designated crew members aboard:

Julius A. Barr, First Pilot

Earl A. Ferguson, Co-Pilot

Benjamin J. Pearson, Alternate Co-Pilot

Harlan Hull, Alternate Co-Pilot

Ralph L. Cram, First Aerodynamist

John Kylstra, Assistant to Mr. Cram in recording data during the flight

Albert G. von Baumhauer, Assistant Aerodynamist

Pieter Guillonard, Recorder and Photographer

William C. Doyle, Operator of the Oscillograph for taking vibration measurements following take-off

Harry T. West, Engineering Officer

Pilot Julius Barr, at the time of take-off on the flight, had a total experience of 17 hours and 55 minutes as co-pilot and 2 hours and 6 minutes as pilot of the Boeing Model 307; while Co-Pilot Ferguson had a total of 4 hours and 52 minutes as co-pilot on this model aircraft.

Permanent seats had not been installed in the aircraft though seven temporary seats were installed in the cabin and two temporary seats, in addition to the pilot's and co-pilot's seats, were installed in the pilot's compartment on test flight No. 19.

Weather conditions from Seattle to Portland at the time of take-off and during the period involved in the flight, as reported by the United States Weather Bureau in sequence weather reports, were as follows:

12:41 P.M. (PST):

"SEATTLE:	C-special	Broken clouds
	Visibility	3 miles - smoke
	Temperature	54
	Dew point	45
	Wind	South Southwest 9 m.p.h.
	Barometer	30.09
"FT. LEWIS:		Broken clouds
	Visibility	60 miles
	Temperature	61
	Dew point	44
	Wind	Southwest 3 m.p.h.
	Barometer	30.03
"CHEHALIS:		Broken clouds
	Visibility	30 miles
	Temperature	63
	Dew point	51
	Wind	Southwest 4 m.p.h.
	Barometer	30.05
"CASTLE ROCK:		Clear
	Visibility	30 miles
	Temperature	65
	Dew point	48
	Wind	South Southwest 13 m.p.h.
"PORTLAND:		Clear
	Visibility	12 miles
	Temperature	66
	Dew point	46
	Wind	North 2 m.p.h.
	Barometer	30.06

1:41 P.M. (PST):

"FT. LEWIS:		Broken clouds
	Visibility	60 miles
	Temperature	65
	Dew point	47
	Wind	West 12 m.p.h.
	Barometer	30.05
"CHEHALIS:		Broken clouds
	Visibility	50 miles
	Temperature	65
	Dew point	47
	Wind	Southeast 5 m.p.h.
	Barometer	30.04
"CASTLE ROCK:		Scattered clouds
	Visibility	30 miles
	Temperature	66
	Dew point	48

"PORTLAND:		Clear
	Visibility	15 miles
	Temperature	69
	Dew point	48
	Wind	East Northeast
	Barometer	30.05"

Through the use of a pilot balloon, released at the Seattle station of the United States Weather Bureau at 1:42 P.M. (PST) on the same day, the following upper air winds were recorded:

"ELEVATION	DIRECTION	VELOCITY
Surface	West Northwest	6 MPH
1,000'	West	6 MPH
2,000'	West Southwest	8 MPH
3,000'	West Southwest	12 MPH
4,000'	West Southwest	24 MPH
5,000'	West Southwest	30 MPH
6,000'	West Southwest	30 MPH
7,000'	West Southwest	32 MPH
8,000'	West Southwest	33 MPH
9,000'	West Southwest	33 MPH
10,000'	West Southwest	37 MPH
10,380'	West Southwest	38 MPH"

A pilot of the United States Army Air Corps Reserve, who was flying in the general vicinity of Alder, Washington, at an altitude varying from 4,000 to 9,000 feet during the early afternoon of March 19, 1939, reported the weather in that vicinity during the course of the flight as being clear with unlimited visibility. The air was reported as being very smooth at all altitudes.

Notes made and recorded by Mr. Cram, subsequent to take-off, include the following data concerning the flight:

The aircraft took off with zero flaps at an indicated airspeed of 85 to 100 m.p.h. and climbed to 1,300 feet in one minute at a speed of 105 m.p.h. From this point the aircraft continued to climb with maximum, except take-off, power with a manifold pressure ranging from 35 inches to 33.5 inches and an indicated airspeed of 118 to 127 m.p.h. until an altitude of 2,200 feet was reached. The aircraft climbed from this point to 10,000 feet in 7 minutes and 25 seconds, and at this altitude was trimmed for level flight at 140 m.p.h. indicated airspeed, with cruising power at 1,900 r.p.m. and 23 inch manifold pressure. Tab setting for this condition was 5° nose down and the control force was zero. The aircraft was then pulled up to 117 m.p.h. indicated airspeed, at which speed a control force at 8 pounds was necessary to hold the aircraft in this attitude.

(The data to this point indicates that the purpose of the test above described was one of static longitudinal stability determination as distinct from the dynamic longitudinal stability tests which were apparently then started. The difference between the static stability test and the dynamic stability test is that in the static stability test the stick forces and the elevator angles are measured at various airspeeds above and below trimmed speed in static condition; whereas, in the dynamic stability test, determinations are made as to the action of the aircraft when left free to oscillate when trimmed in that condition.)

The controls were released and as the aircraft oscillated freely in a longitudinal sense, readings were taken at 10-second intervals during such oscillations to determine the flight path and the damping of the longitudinal oscillations. The zero position was recorded as the aircraft passed 100 m.p.h. indicated airspeed on its downward oscillation. At 10 seconds the indicated airspeed was recorded as 115 m.p.h.; at 20 seconds, 120 m.p.h.; at 30 seconds, 110 m.p.h.; at 40 seconds, 100 m.p.h.; and, at 50 seconds, 110 m.p.h.

A static longitudinal stability test was then conducted, during which readings were taken of the elevator control forces and positions. The speeds listed were 85, 80, 78, 75, and 72 m.p.h., with readings being entered after 85, 80 and 75. Nothing was written after 78 or 72. The elevator position indicator read 395 at 85 and 80, and 400 at 75 m.p.h. A control force of 9 pounds was entered after 80 and of 10 pounds after 75 m.p.h.

It is to be noted by reference to the flight plan for this flight that sideslips constituted the next test to be conducted.

At 1:12 P.M. (PST) a radio message was transmitted from NX 19901 to the Boeing Aircraft Company radio station located at Seattle, Washington, which message gave the position of the aircraft as being between Tacoma, Washington and Mount Ranier at an altitude of 11,000 feet. Some two or three minutes later, while flying at a comparatively slow rate of speed in the vicinity of Alder, Washington, the aircraft stalled and began to spin in a nose down attitude. After completing two or three turns in the spin, during which power was applied, it recovered from the spin and began to dive. The aircraft partially recovered from the dive at an altitude of approximately 3,000 feet above sea level, during which recovery it began to disintegrate. Outboard sections of the left and right wings failed upward and broke entirely loose from the aircraft. Major portions of the vertical fin and portions of the rudder were carried away by the wing wreckage. The outboard section of the left elevator separated from the stabilizer and both fell to the ground detached. The right horizontal tail surface, being held on by the fairing along the top surface and also by the elevator trim tab cables, remained with the fuselage. The No. 1 engine nacelle also broke loose from the aircraft and fell to the ground separately. The main body of the aircraft settled vertically and struck the ground in an almost level attitude both longitudinally and latterally at a point approximately 1,200 feet above sea level. Watches and clocks aboard the aircraft, which were broken by force of the impact, indicated the time of the accident as approximately 1:17 p.m. (PST).

Major component parts of the aircraft, which broke loose in the air struck the ground with respect to the fuselage as follows:

Left outer wing panel, about 1,000 feet in a southwesterly direction; right outer wing panel, about 530 feet in a southerly direction; left outboard engine nacelle, about 660 feet in a southeasterly direction; left horizontal stabilizer, about 200 feet in a southerly direction; and the left elevator outboard portion, about 500 ft. in a southeasterly direction. The great majority of small parts were found within the distance incompassed by the location of the larger pieces from the fuselage, though a few parts were found at a distance of 2,500 feet in a southeasterly direction.

Parachutes of either the seat or chest type were available for all persons aboard the aircraft though were not being worn at the time of the accident.

The crash resulted in fatal injuries to all persons aboard the aircraft. Julius Barr was seated in the pilot's seat and A. G. von Baumhauer was seated in the co-pilot's seat at

SPECIAL INSTRUMENTS

Accelerometer

During the last four flights of NX 19901 a visual maximum indicating accelerometer, the purpose of which was to enable the pilot to know exactly the loading to which he was subjecting the aircraft during maneuvers and to prevent him from overloading the structure, had been installed on the aircraft. One hand of this instrument indicated at all times the normal acceleration to which the instrument was being subjected while the other hand indicated the maximum acceleration to which the instrument had been subjected since the hand was last set back. This instrument was removed from the wreckage of NX 19901 and forwarded to the manufacturer for analysis, who reported that it had been subjected to an acceleration in excess of 10 g, the exact amount of such excess being indeterminable. (This indicated acceleration obviously occurred at the time of impact with the ground.)

V-G Recorder

A V-G recorder, owned by the National Advisory Committee for Aeronautics and lent to the Boeing Aircraft Company, had been installed on NX 19901 during the early flight testing of the aircraft. This instrument traces, on a piece of smoked glass, a graph of the acceleration encountered parallel to the vertical axis of the aircraft and the indicated airspeeds of the aircraft as the accelerations occur. This graph enables a determination to be made of the maximum positive and negative accelerations to which the aircraft had been subjected and the indicated airspeed at which they occurred since that particular glass had been installed in the recorder.

During lateral stability tests on test flights Nos. 10 and 11 of NX 19901, a maximum acceleration of 1.7 g had been recorded at 230 m.p.h. indicated airspeed and a minimum of 0.2 g at 195 m.p.h. indicated airspeed. The normal value for steady unaccelerated level flight is 1.0 g. During similar tests on flight test No. 12, which also included climb, high speed runs and stall tests, a maximum acceleration of 2.0 g occurred at 228 m.p.h. indicated airspeed and the minimum of 0.4 g at 150 m.p.h. indicated airspeed. A maximum positive acceleration of 2.1 g and a minimum of 0.4 g were recorded during dive tests conducted on test flight No. 13 at speeds in excess of 320 m.p.h. indicated airspeed, which speed is the upper limit of the range of this instrument. A slight discrepancy between this instrument and the airspeed indicator was indicated during this test inasmuch as the latter instrument read 303 m.p.h. at the same time the V-G recorder indicated the speed above mentioned. No acceleration in excess of those encountered in the previous tests were recorded during take-off, landing, speed run and spiral stability tests conducted on test flight No. 14, the only other flight on which the instrument was utilized.

The glass in the V-G recorder during test flight No. 19 was not smoked and consequently no graph was obtained. Exhaustive scientific tests failed to reveal the path of the instrument stylus, and consequently no information whatsoever was revealed by this instrument concerning test flight No. 19.

Horizontal Accelerometer

There was also aboard NX 19901 at the time of impact a horizontal accelerometer, the purpose of which was to record accelerations along the fore and aft axis of the aircraft. This instrument was badly damaged in the crash and subsequent examination did not reveal any readings which would indicate the acceleration at the time the aircraft disintegrated.

Oscillograph

An oscillograph, the purpose of which was to record engine vibration, was installed on NX 19901 during test flight No. 19 but was not in operation on this flight.

AIRCRAFT ENGINES

An inspection of the four aircraft engines subsequent to disassembly revealed that all four had been badly damaged by impact with the ground, and that certain damage had occurred during the flight, as the result of which the engines were in the following conditions prior to impact:

Engine No. 1 (left outboard), Serial No. 25030, which had fallen to the ground separate from the wing and the main body of the aircraft, had 44 hours and 40 minutes total operating time. The thrust nut had scored the oil seal ring bearing surface of the gear case thrust plate flange, dragging metal into the ring grooves and thus sticking all rings. The region of maximum scoring on the thrust plate flange was near the bottom and to the right of the vertical center line, though the scoring continued entirely around the surface.

The sticking of the thrust nut rings and the scoring of the oil seal ring bearing surface on the thrust plate resulted from temporary misalignment of the propeller shaft due to gyroscopic forces induced by precession of the propeller during the spin and at the time of pull-out from the dive. The master rod bearing was burnished about 3/4 inch from each end. (This condition is normal in engines which have run several hundred hours, but due to the few hours that this engine had been operated prior to impact, this condition evidenced operation at excessively high r.p.m.)

There was no indication of malfunctioning or failure of the engine to deliver rated power during normal flight.

Engine No. 2 (left inboard), Serial No. 25031, which had remained with the main body of the aircraft, had 45 hours and 10 minutes total operating time. The front oil seal ring of the thrust nut was slightly stuck in the groove and gear case thrust plate flange was scored in a manner similar to that on Engine No. 1 but to a lesser degree and on the opposite side of the seal oil ring bearing surface. This scoring was also caused by gyroscopic forces due to precession of the propellors during the spin and pull out from the dive. The master rod bearing was in good condition.

There was no indication of malfunctioning or failure of the engine to deliver rated power during normal flight.

Engine No. 3 (right inboard), Serial No. 25032, which remained with the main body of the aircraft, had 47 hours total operating time. All oil seal rings of the thrust nut were stuck in the grooves and the nut had scored the oil seal ring bearing surface of the gear case thrust plate flange. The condition of each was identical to those of Engine No. 2, and had been caused by the same type gyroscopic forces as had been imposed on No. 1 and No. 2 engines.

Some of the teeth on the pinion and stationary gears were scuffed and blued indicating that there had been a short period in which they were out of alignment possibly due to the same gyroscopic forces which had caused misalignment of the propeller shaft. The cam bearing was slightly scuffed in such manner as to indicate high r.p.m. There was no indication of malfunctioning of this engine or failure to deliver rated power during normal flight.

Engine No. 4 (right outboard), Serial No. 25033, which had remained with the main body of the aircraft had 42 hours and 25 minutes total operating time. The thrust nut oil seal rings were stuck in the grooves and the gear case thrust plate flange was scored in the same manner as that in No. 2 engine. The condition of the thrust nut and the thrust plate flange indicated that in the case of these parts the heat had been greater, the thrust plate being cracked around the flange (possibly due to a greater gyroscopic force being applied to this engine).

The crank shaft was burnished and slightly blued on the counter-weight side of the crank pin, which is evidence of heat and excessive r.p.m. The master rod bearing was very slightly scuffed and there was evidence of some etching of the bearing material. The bearing metal of the cam bearing was slightly etched which further evidenced excessive r.p.m. No condition was present which would indicate failure of the engine to deliver rated power during normal flight.

PROPELLERS

The propellers, all of which were attached to their respective engines, were found to be complete with no blades or parts missing. The blades which were partially buried in the ground were uncovered and the blade angles after impact, as indicated by the hub graduations, were noted as follows:

	Blade No. 1	Blade No. 2	Blade No. 3
No. 1 Propeller	33°	28°	10°
No. 2 Propeller	22°	25°	27°
No. 3 Propeller	33°	70°	170°
No. 4 Propeller	18°	13°	-17° (negative angle)

Each propeller was removed from the engine and later disassembled and examined in detail. All blades of each propeller were bent to varying angles as a result of impact with the ground.

Propeller No. 1, hub Serial No. 33543, had a total operating time of 44 hours and 10 minutes. No. 1 blade of this propeller was sheared from its bushing toward the high pitch angle approximately 25 degrees. Blade No. 2 was not sheared from its bushing. Blade No. 3 was sheared from its bushing toward the low pitch angle approximately 20 degrees. Other than a dent in the dome there was no damage to the hub assembly or its operating mechanism.

Propeller No. 2, hub Serial No. 37347, newly installed, had a total operating time of only 2 hours. There was no failure of parts in the hub assembly and none of the blades were sheared from their bushings.

Propeller No. 3, hub Serial No. 33544, had a total operating time of 47 hours and there was no failure of the hub assembly parts. Blade No. 1 was not sheared from its bushing. Blade No. 2 was sheared from its bushing toward the high pitch angle approximately 30 degrees. Blade No. 3 was sheared from its bushing toward the high pitch angle approximately 130 degrees.

Propeller No. 4, hub Serial No. 33699, had a total operating time of 42 hours and 25 minutes. Blade No. 1 was sheared from its bushing toward high pitch approximately 4 degrees. Blade No. 2 was sheared from its bushing toward the low pitch angle approximately 2 degrees. Blade No. 3 was not sheared from its bushing, but the blade gear segment was broken in such manner that it rotated toward low pitch. Microscopic examination of the break in the gear segment showed that it was due to tension and that there was no sign of fatigue or defect in the material.

No condition was found in any of the propellers that would indicate any malfunctioning during normal flight.

STRUCTURAL FAILURES

The principal failures of the structure of NX 19901 and the nature and extent of such failures, as indicated by exhaustive inspection and analysis of such failures, were as follows:

Right Wing

The initial failures were a compression failure in the upper wing covering, compression and bending failures in the upper spar chords, tension and bending failures in the lower spar chords and tension failures in the lower covering. The outer panel of the right wing failed upward near the joint which connected the inner and outer panels. As the outer panel failed, it carried the outer section of the right aileron with it. The inboard section from the control rod inboard, which was left attached to the wing, later became detached with a section of the wing trailing edge. The section of the upper covering between the spars and between Station 17 and Station 21 were detached from the aircraft, though most of the covering was still attached to the aircraft.

As the outer panel moved upward after failure, the aileron cables, which were just behind the rear spar, cut through the upper trailing edge skin in a straight line parallel to the spar directly into the body, thus removing all support for the upper side of the trailing edge. All airloads on the trailing edge appear to have been upward from the time the cables cut through the upper skin until the aircraft reached the ground. (This is evidenced by the fact that the cut ends of the ribs on this upper surface butted against the spar and the spar cap, and the conclusion that any down load on the trailing edge would have caused it to break away from the aircraft.)

The marks of the aileron cables on the body clearly indicate that as the right outer panel swung upward still attached to the aircraft by means of the two aileron cables, it swung over the body until it occupied a position to the left of and about three foot below the top of the fin.

Left Wing and Outboard Nacelle

The outer left panel failed upward between Station 19 and Station 13, which location is farther inboard than the point of failure on the right wing, and carried the entire aileron with it as it left the aircraft. The failures of the panel were compression in the upper covering and in the upper spar chords, tension and bending failures in the lower spar chords and tension and tearing in the lower wing covering. As the outer panel moved upward, the upper covering, which had not been failed as cleanly as on the right side, peeled upward to a point just inboard of the left outboard engine nacelle.

The upper support for the engine nacelle, known as the upper skate, rests on top of the wing skin and is bolted to the covering and ribs. In peeling upward and inboard the wing covering tore this upper skate loose from the wing, leaving the engine and nacelle supported only by the inboard leading edge angle, the lower skate and the connecting tubes and controls. The nacelle, as indicated by the direction in which the rivets in the leading edge sheared, then failed forward and downward.

The strength of the covering and rivets was greater than the remaining strength of the front spar, and as the covering moved upward it broke a section of the spar into several pieces, the largest being a section of the lower spar chord about 6 feet in length. The rear spar was failed in a similar manner.

The wing covering was broken into a number of pieces, which ranged in size from about three feet by six feet to about one foot square. The flap was torn free and was broken into four major pieces, ranging from four feet to six feet in length.

As the outer panel moved upward the aileron cables cut through the upper skin of the trailing edge parallel to the rear spar. One aileron cable then failed at the body, and the other cable, after failing about 18 feet outboard from the body, rebounded inward, leaving a mark along the body.

The upper surface of the trailing edge and the outboard half of the lower surface of the trailing edge of the left wing were entirely detached from the aircraft. A number of parts from the left wing struck the body of the aircraft, leaving marks which were substantially along a line from the wing to the base of the fin. Many of the marks were covered with aluminum pigmented P-27 primer with which the internal structure of the wing was finished.

The weight of the aircraft at the time of the accident was approximately 42,500 pounds. The acceleration required to cause complete failure of the winges of the aircraft with this weight was estimated to be 4.8g.

Fin

The fin was failed backwards and slightly towards the right as the result of impact by some part of the wing wreckage. Both chords of the front spar had tension failures about one-third of the way up from the body, the failure of the right-hand chord being somewhat lower than the left-hand chord. The rear spar was failed in compression, which failure was considerably greater than the left-hand chord, indicating that the loads which caused the failure were directed partly from the left. (It is believed that the fin could only be failed in this manner by loads acting near the top of the fin in a direction almost straight to the rear. It is difficult to conceive of any airload of sufficient magnitude which would cause the fin to fail in this manner.)

The upper one-third of the leading edge of the fin bore heavy marks, which apparently were made by the aileron cables from the right wing. As stated above, these cables came out of the body just back of the rear spar and extended up and around the body of the right outer panel, which was in a position to the left and below the top of the fin. While in this position, the cables broke at their intersection with the body, snapping outward and striking the leading edge of the fin about one-third of the way from the top. They then slipped up the leading edge of the fin to the point where the radio antenna fairlead, which was attached by means of a 3/16" steel tie rod, extending through the top of the fin from the leading edge to the rear spar, came out through the leading edge of the fin.

The aileron cables, which had an ultimate tensile strength of approximately 10,000 pounds, pulled this tie rod out of the fin about one-half way back to the rear spar and cut away the tip of the fin which is located above this tie rod. The top of the fin between the leading edge and the rear spar was also failed in compression. After passing over the fin, the cables also caught the upper rudder mass balance weight and scored this weight the full length of its leading edge.

The fin also was struck a heavy blow from the left side by the wreckage from the wings. One member of the striking object entered the fin just back of the leading edge, and after passing through the fin, struck the right-hand chord of the front spar, tearing out a section of this spar chord together with its attaching skin. The upper two-thirds of the fin was broken entirely clear of the aircraft but it remained attached to the mainbody of the aircraft by means of rudder trim tab cables which had pulled out about 10 feet. These rudder trim tab cables came back through the body and turned on pulleys, supported by a bracket in the body below the base of the fin, and ran directly up the rear spar of the fin. The bracket in the body was failed and the cables had cut into the bulkheads in the body near this bracket and also had cut into the section of the fin which remained on the body. As indicated by additional marks down the center of the rear fin spar, made by the rudder trim tab cables, this fin spar was bent approximately straight back.

Rudder

As the fin moved backward it forced the rudder, which was attached to the body and fin by means of three hinges on the fin and two on the body, backward. As the rudder was forced back, a failure occurred in the top body hinge (which indicates that the rudder was turned at an angle of 20 degrees to the right at the time this hinge was pulled loose from the body portion). The half of the top body hinge attached to the body was damaged very little, being bent downward about one-half inch (which fact indicates that at the time of contact with the ground, the rudder was not attached to this hinge). The rudder control tab was detached from the rudder and along with a section of the rudder was not found subsequent to the accident. The piece of the rudder which tore loose was one-half the length of the control tab in length by about 18 inches deep and was normally located immediately forward of the top half of the control tab. The rudder apparently had been struck a heavy blow from the left rear by some part of the wreckage, the force of which was carried into the fin. The stub of the fin remaining on the body had been struck twice, the first blow coming from the front and left and bending the stub of the fin over toward the right and denting it badly near the left side of the rear spar. This blow apparently was struck by some failed part of the internal structure from one of the wings, as there were marks on the fin which had been made by the protective coating of P-27 primer with which the internal structure of the wings were finished. The second blow, which came from the rear and the left, also struck the rudder and bent a section of the fin covering forward and over the section which was marked with the P-27 primer. This piece of skin has a general shape of the rudder torque tube and also bore some marks made by aluminumized dope (which fact indicates that the blow probably drove the rudder torque tube into the fin).

The rudder was broken into two pieces on a line between the rudder control and trim tabs, the two pieces being held together only by the fabric on the rudder. The rudder was held to the main body of the aircraft by one of the spring cartridge links of the rudder control tab and rudder control systems at the lower end of the rudder. As the failure of the one spring cartridge link assembly appeared to have been caused when the main body of the aircraft made contact with the ground, it is possible that at the time of such contact with the ground it was held by both of the spring cartridge links. On contact with the ground, the lower end of the rudder torque tube was driven about one foot into the ground, with the top pointing toward the left at an angle of about 60 degrees. The remainder of the rudder was practically flat on the ground along side of the lower portion. The rudder mass balance weights which were installed approximately 12 feet apart, were driven into the ground only four and one-half feet apart, both weights having failed directly at right angles to the rudder torque tube. The rudder trim tab remained attached to the rudder.

Body

As the main body of the aircraft made contact with the ground in an almost level position both longitudinally and laterally in vertical descent, the entire fuselage structure failed downward, breaking completely through in cross-section at three points. The upper portion of the pressure bulkhead in the rear part of the body buckled toward the front portion of the fuselage. As far as could be determined, no damage was done to the body while the aircraft was in the air, except for the control cable marks on the right side of the body as above described. There was no evidence of power being applied by the engines at the time of impact with the ground.

Stabilizers and Elevators

Both the stabilizers and elevators failed upward, and, as was indicated by the symmetry of the failures on both sides of those surfaces, both as to type and location of the failures, these tail surfaces were apparently symmetrically loaded at the time of such failures. The failures in the stabilizer front spar upper chords consisted of the rivets attaching the hinge fittings to the spar chords being sheared by loads acting inward. The upper spar chords made deep symmetrical dents in the body on their respective sides. The lower front stabilizer spar chords failed in tension, the one on the left failing in the steel hinge terminal, and the one on the right failing in the spar chord just outboard of the steel hinge terminal. The failures in the rear spars were similar to the failures in the bottom chords in the front spars.

The elevator torque tube, as indicated by the inward failure of the rivets attaching the outboard flanged couplings to the outboard elevator torque tube, and the outward failure of those on the bottom sides, failed in bending on both sides of the aircraft at the point where the outboard elevator torque tubes are joined to the section of the torque tube which passes through the body of the aircraft. The outboard torque tubes, which fit over the male portions of the flanged couplings, failed by splitting along the tops of the torque tubes lengthwise of the tubes.

The left elevator was broken into two pieces, the outboard failure occuring in the nose section of the elevator formed by the elevator channel section spar and the leading edge skin just outboard of the point where the heavy torque tube ends within the elevator. The failure was due to torsion and bending, with bending predominating. As the stabilizers and elevators were deflected upward under the action of the upward acting loads, the elevators moved outboard with relation to the stabilizers due to interaction of the stabilizers and elevators. This relative motion bent the elevator ends of the hinge brackets attached to the stabilizers outboard, and caused the hinge brackets on the left stabilizer to tear loose from the stabilizer, while those on the right-hand stabilizer showed only a slight evidence of such interaction between the stabilizer and elevator. Though the top half of the leading edge of the left-hand outboard section of the elevator had been cut into deeply by the elevator hinge brackets, there was only a slight indication that the hinge brackets had contacted the bottom half of the leading edge portion of this part of the left elevator. Both pieces of the left elevator were ripped from the left stabilizer. The trim tab of the left elevator was intact on the outboard part of such elevator and was set approximately in neutral, while part of the control tab, including its static balance weight, was found on the inner portion of the left elevator. The torque tube of the left elevator control tab had failed at the side of the body at the coupling. The left stabilizer and both pieces of the elevator were detached from the aircraft.

The leading edge section of the right-hand elevator, at the point just outboard of the end of the elevator torque tube, showed evidence of the same type of failure as that which occurred at this location in the left elevator leading edge section. However, as is evidenced by the fact that the elevator torque tube and both stabilizer spars were driven directly into the ground about 18 inches, the right stabilizer and left elevator were in a vertical position at the time of contact with the ground. There also were permanent wrinkles in the right-hand elevator just outboard of the point where the heavy torque tube ends within the elevator, which wrinkles had been caused by a rotation of the elevator in the up direction due to the action of torsion about the elevator hinge line. The right-hand elevator and its trim and control tabs were intact when the aircraft struck the ground, though the elevator control tab operating torque tube was failed at the side of the body. Holes which had been punched through from top to bottom in the trailing edge of this elevator were apparently caused by the elevator having contacted what was left of the vertical control surfaces after the right stabilizer and elevator had failed upward. The elevator hinge brackets also had cut deep gashes in the lower half of the right elevator nose section at all hinge points. The right-hand stabilizer and elevator were held on to the body of the aircraft at the time of impact by the left stabilizer fairing strip and by two 3/32" elevator trim tab control cables.

Inspection

A very detailed inspection of the body, all working parts which remained with the aircraft, and all control cables failed to reveal any indication of malfunctioning or failure prior to disintegration of the aircraft, which would have accounted for lack of or loss of control. All fuel lines and the entire fuel system were also thoroughly inspected and no indication of stoppage in the fuel line or failure prior to disintegration of the aircraft was found. All control surface balance weights were accounted for in the vicinity of the main body of the aircraft, and inspection of such weights revealed that all conformed to specifications. The possibility of sabotage was considered and investigated thoroughly, but no evidence was found to indicate that it was involved in this accident.

Spins

It is of interest in connection with the facts hereinabove enumerated that records are available indicating that aircraft of the same general type and design as NX 19901 have been spun on two occasions. In the first of these experiences, an aircraft of the same general dimensions and design as the Model 307, with the exception of the fuselage, while flying with a gross load of about 42,000 pounds at an altitude of 14,000 feet, went into an inadvertent spin and made two complete turns before recovery was effected. During the pull-out from the ensuing dive, permanent distortion occurred in the structure of both wings, necessitating the installation of new wings on the aircraft.

In the second of these experiences, a similar ship was intetionally permitted to enter a spin following a complete stall. The controls were immediately reversed and the aircraft responded promptly, enabling the pilot to effect recovery after three-fourths of a turn in

Evidence indicated that power was used in recovery from the spin in the case of NX 19901. It should be noted that in the two instances above described recovery from spin in similar aircraft was accomplished without the employment of power. In one of these cases, permanent distortion occurred in both wings.

SUMMARY OF FINDINGS

1. Aircraft NX 19901, Boeing Model S 307, met the pertinent requirements set forth in Section 01.3 of the currently effective Civil Air Regulations and was properly issued an experimental certificate by the Civil Aeronautics Authority.

2. NX 19901 had only been operated experimentally by the Boeing Aircraft Company and had never been submitted to the Civil Aeronautics Authority for final inspection and flight tests required prior to certification as to its airworthiness.

3. The currently effective experimental certificate issued on aircraft NX 19901 by the Civil Aeronautics Authority authorized flight of the aircraft at a maximum take-off weight of 45,000 pounds and a maximum landing weight of 41,000 pounds. The certificate prohibited the carrying of passengers and restricted flight personnel to bona fide members of the crew.

4. Aircraft NX 19901 crashed near Alder, Washington, at approximately 1:17 P.M. (PST), Saturday, March 10, 1939, with the resultant destruction of the aircraft and fatal injuries to all persons aboard.

5. The weather in the vicinity of Alder, Washington, at the time of the accident was clear with unlimited visibility. The upper air was very smooth with a moderate west southwest wind prevailing at altitudes above 2000 feet.

6. Pilot Julius Barr held a currently effective Commercial Certificate of Competency issued by the Civil Aeronautics Authority with appropriate ratings for the flight herein involved. Bar had a total flying time on NX 19901 prior to test flight No. 19 of one hour and 52 minutes as observer; 17 hours and 55 minutes as co-pilot; and, 2 hours and 6 minutes as first pilot. The remainder of his flying time in four-engine aircraft consisted of 3 hours and 6 minutes as observer, and 9 hours and 17 minutes as co-pilot on the Boeing Model 314.

7. Co-Pilot Earl Ferguson held a currently effective Commercial Certificate of Competency issued by the Civil Aeronautics Authority with appropriate ratings for the flight herein involved. Ferguson had a total flying time on the Boeing Model S 307 prior to test

flight No. 19 of 4 hours and 52 minutes as co-pilot. The remainder of his flying time in four-engine aircraft consisted of 9 hours and 15 minutes as co-pilot on the Boeing Model Y1B-17A; 27 hours and 57 minutes as observer; 18 hours and one minute as co-pilot, and 16 hours and 39 minutes as pilot of the Boeing Model 314.

8. Albert G. von Baumhauer held a private pilot's license issued on November 28, 1931 by the Dutch Air Ministry, the records of which show that his total flying time as pilot amounted to 116 hours. He had no experience as pilot or co-pilot of four-engine aircraft, but had been observer in trial flights of four engine Fokker F 22 and F 36 aircraft. He occupied the co-pilot's seat and was serving as co-pilot in aircraft NX 19901 at the time of the accident.

9. The performance of aircraft NX 19901 on flights prior to test flight No. 19 had either met or exceeded the manufacturer's estimates.

10. The main control surfaces were mass balanced during test flight No. 19 as below indicated:

The rudder had 104 percent static balance accomplished by means of a 17 pound weight at the top of the rudder, located 24.2 inches ahead of the hinge line and a 50 pound weight on an arm at the body center line 24.2 inches ahead of the rudder hinge line. This gave a coefficient of dynamic balance of 0.028 with respect to the rudder hinge line and the center line of the body. The rudder control tab was not statically balanced.

The elevator had 55.2 percent static balance accomplished by means of a 60 pound weight located 21.4 inches ahead of the elevator hinge line and on the body center line. This gave a coefficient of dynamic balance for each half of the elevators of 0.10 with respect to the elevator hinge line and the center line of the body. The elevator control tabs were statically balanced.

The ailerons were 18.7 percent statically balanced based on their actual structural weight distribution and there were no mass balance weights incorporated. Each aileron had a coefficient of dynamic balance of 0.292 with respect to its hinge line and the center of the body. Semi-irreversible aileron control units were employed.

The aircraft with the foregoing control surface mass balancing was dived in a flight test previous to test 19 to an indicated airspeed of 303 miles per hour with no evidence of flutter, though the adequacy of such balancing had not been determined under varying conditions of flight.

11. The original mass balancing of the elevators incorporated a 74.3 pound weight. During the early flight tests of aircraft NX 19901, this weight was reduced to 60 pounds in order to obtain an increase in the center of gravity range within which the aircraft was longitudinally stable.

A further reduction in the elevator mass balance wieght from 60 pounds to 50 pounds was made on test flight Nos. 16, 17 and 18 in an attempt to further increase the range of center of gravity. During the dive, in recovering from a stall on test flight No. 18, mild flutter developed at an indicated airspeed of 240 miles per hour. Following this experience, the 50 pound elevator mass balance weight was replaced with the 60 pound weight for test flight No. 19.

As 303 miles per hour was the design glidin speed for this aircraft, no tests were made to determine at what speed in excess of 303 miles per hour flutter might occur with the 60 pound elevator mass balance weight.

12. During the stall tests of this aircraft, there was a stall warning consisting of a characteristic buffeting of the wings and control surfaces at indicated airspeeds from 5 and 8 miles per hour with power off and from 1 to 2 miles per hour with power on faster than the speed at which the actual stall occurred. Recoveries from all stalls conducted on flights prior to test flight No. 18 were begun immediately after the nose of the aircraft began to drop. Under these conditions, the nose was not permitted to fall below level flight position before recovery was started, the result of which was the loss of 200 to 600 feet altitude in effecting recovery. The aircraft showed no tendency to fall off on either wing during such stalls.

Stall tests were conducted on test flight No. 18 by Harlan Hull, of Transcontinental and Western Air, Inc. During some of these tests Mr. Hull completely stalled the aircraft and permitted the nose to drop below level flight positions before recovery was started. Under these conditions there was no tendency of the aircraft to spin and recovery was normal.

13. The notes made by Mr. Cram on test flight No. 19 indicated that the take-off was normal and that the climb up to 11,000 feet altitude, using rated engine power, had been made in a normal manner. At this altidude longitudinal stability tests were made and the notes indicated that this phase of the flight test was nearing completion, if not actually completed. The next tests, as outlined by the previously approved flight plan, were side-slip tests.

14. At the request of Mr. von Baumhauer, who occupied the co-pilot's seat of NX 19901 at the time of the accident, two special instruments were installed on the aircraft. One instrument was installed on the co-pilot's control column to be used for measuring the forces in pounds required to operate the elevators during specific maneuvers. It would be necessary to operate the aircraft with the co-pilot's controls in order to obtain readings from this instrument. The second instrument, which was mounted on top of the instrument panel in front of the co-pilot, was connected to the control cables of the ailerons, rudder and elevators for the purpose of indicating the amount of movement of these controls during various maneuvers.

15. The flight plan covering test flight No. 19 was based upon specific requests made by Mr. von Baumhauer for the purpose of determining whether or not this aircraft would meet the flight characteristics desired by the Dutch Air Ministry. Although this flight test was intended to include tests set forth in the flight plan, it was also a demonstration of the aircraft to representatives of the Dutch Air Ministry, Royal Dutch airlines and Transcontinental and Western Air, Inc.

16. All pertinent evidence indicated that the aircraft went into an inadvertent spin subsequent to a stall at an altitude of approximately 11,000 feet. It made two to three

turns in the spin, during which the engines were used to aid recovery. In recovering from the dive subsequent to the spin, the wings and horizontal tail surfaces failed upward apparently due to air loads in excess of those for which the aircraft was designed.

17. There is no conclusive evidence as to which wing failed first or whether the wings failed before the horizontal tail surfaces. The left elevator failed about 3 feet outboard of the side of the body, such failure being due to bending and torsion in the elevator nose section with bending predominating. The outboard section of the left elevator was torn loose from the stabilizer and the condition of the hinges indicated that its movement had been upward and outward with respect to the stabilizer. Indications are that this was the first failure which occurred in the horizontal tail surfaces and that it was followed by the separation of the left stabilizer and the inner portion of the left elevator from the body. The right stabilizer and elevator were subjected to and failed by forces similar to those which failed the left horizontal tail surfaces. These surfaces, however, remained attached to the body by means of the upper fairing and the elevator trim tab cables.

The vertical tail surfaces failed as a result of being struck by parts of the broken wings. The upper portion of the rudder and fin remained attached to the body by the rudder trim tab cables.

18. A vertical accelerometer had been mounted on the instrument panel to indicate to the pilot the acceleration to which the aircraft was being subjected in order that dangerous stresses due to excessive acceleration could be avoided.

19. A V-G recorder, the property of the National Advisory Committee for Aeronautics, was installed on NX 19901 during test flight No. 19. The glass installed in this instrument on this flight had not been smoked, and consequently there was no record of the maximum indicated airspeed attained or the vertical accelerations encountered during this flight.

20. The weight of the aircraft at the time of the accident was approximately 42,500 pounds. The acceleration required to cause complete failure of the wings on the aircraft with this weight was estimated to be 4.8g.

21. An oscillograph was installed on NX 19901 for the purpose of recording engine vibrations, but was not in operation during test flight No. 19.

22. Examination of the engines and propellers subsequent to disassembly disclosed nothing which indicated failure to deliver rated power prior to impact.

23. Each person aboard the aircraft on test flight No. 19 had a parachute of either the seat or chest type available for his use.

24. Flight tests conducted with NX 19901 indicated that the control forces required to maneuver this aircraft were light in comparison with other four-engine aircraft of comparable weight and design.

25. On a previous test flight (No. 13) employing the same control surface mass balance weights used during flight No. 19, there was no indication of flutter at the design gliding speed of 303 miles per hour which was attained during a shallow dive in smooth air. No high speed dive tests were run under varying atmospheric conditions to determine whether flutter might develop at less than the designed gliding speed if executed in turbulent air.

26. Stalls and spin tests conducted with aircraft, comparable to the Boeing Model 307 in size and design, do not indicate with any degree of finality the characteristics of such aircraft in these maneuvers.

* * * *

In addition to the facts, conditions and circumstances presented above, the following known and potential factors may have contributed to the accident although evidence to this effect is not conclusive:

1. Assignment of a first pilot inexperienced in testing and demonstrating aircraft of this type, particularly with respect to the maneuvers called for on test flight No. 19.

2. Inexperience on the part of Mr. von Baumhauer, who was serving as co-pilot, in the handling of this aircraft in maneuvers being conducted at his request.

3. A test flight of the aircraft involving demonstration of maneuvers which had not previously been investigated thoroughly.

4. Inexperience with the radical difference in stall characteristics of the aircraft with 25° up-elevator travel as compared with the 17° up-elevator travel previously used, particularly in view of the probability that the aircraft was not completely stalled during tests conducted with the 17° up-elevator travel.

5. Sensitivity of the elevator and rudder control system such as to enable the pilot to over-control easily, and thus change the attitude of the aircraft so abruptly as to impose stresses exceeding those for which the aircraft was designed or would withstand.

6. Elevator flutter in the dive, following recovery from the spin, which may have developed, because of inadequate mass balancing of the elevators, at an airspeed in excess of the designed gliding speed of the aircraft.

(Flutter developing during this maneuver might have influenced the pilot in recovering from the dive too abruptly.)

7. Locking of the rudder during a sideslip, because of the lateral inflow of the air on the rudder and the loss of the assistance of the control tab, resulting in the inability of the pilot to exert sufficient force upon the controls to prevent spins.

8. Insufficient vertical tail surfaces, as presently designed and installed, to give adequate control of the aircraft under all conditions of flight.

* * * *

Sketches showing the probable flight path and maneuvers of the aircraft during its descent; relative locations of parts of the aircraft subsequent to impact with the ground; and the main structural failures prior to impact with the ground, are attached hereto and marked Exhibits "A", "B", and "C", respectively, and are by reference made a part of this report.

* * * *

PROBABLE CAUSE

Structural failure of the wings and horizontal tail surfaces due to the imposition of loads thereon in excess of those for which they were designed, the failure occurring in an abrupt pull-out from a dive following recovery from an inadvertent spin.

* * * *

RECOMMENDATIONS

As the result of its investigation of this accident, the Air Safety Board recommends that:

1. The Civil Aeronautics Authority make:

a. A further study of the mass balancing of all control surfaces of the Boeing Model 307, as presently designed and installed, for the purpose of determining its adequacy in all respects before issuing a type certificate for this aircraft.

b. Studies for the prevention of flutter in control surface tabs on all aircraft by proper mass balancing of these units or by satisfactory mechanical means in the event of lag or failure in the operating mechanism.

c. Continued studies of the factors which cause or contribute to flutter and vibration of aircraft structures, and that the results of all such studies be made available to the industry and minimum requirements based thereon be included in the Civil Air Regulations.

2. The Civil Aeronautics Authority make a study of the Boeing Model 307 tail group, particularly to determine the adequacy of the attachment fittings of both fixed and movable component parts.

3. The Civil Aeronautics Authority:

a. Make a further study of the stall and spin characteristics of the Boeing 307 by such flight and other tests as may be required for the purpose of determining the effectiveness and adequacy of its design in the prevention of critical stalls and spins, and in recovery therefrom. Should the results of such studies and tests prove the characteristics to be unsatisfactory, it is recommended that corrective measures be taken prior to the issuance of a type certificate on this model aircraft.

b. Require all future civil aircraft to be so designed as to eliminate critical stall characteristics and be inherently spin-proof.

4. The Civil Aeronautics Authority expedite the development of an adequate stall warning device. It is further recommended that such device, when developed, be a required installation on all air carrier aircraft.

5. In addition to any other similar studies, the Civil Aeronautics Authority require extensive wind tunnel tests on scale models of all aircraft designed for air carrier service and submitted for type certificate and that the results of such tests to be submitted to the Civil Aeronautics Authority for review and approval. It is further recommended that the spin characteristics of such aircraft be determined in so far as possible through the use of a spin-tunnel.

6. The Civil Aeronautics Authority thoroughly investigate the tendency in certain large aircraft designs to so reduce the forces required to operate the controls as to permit undue stresses being imposed inadvertently on the aircraft.

7. Thorough study be made by the Civil Aeronautics Authority to determine the adequacy of the mechanism employed on the adjustable cockpit windows of the Boeing Model 307, particularly as to the satisfactory functioning of this mechanism under severe icing conditions and the visibility provided under such circumstances.

8. The Civil Aeronautics Authority consider the advisability of increasing present maneuvering-load-factor requirements for large aircraft.

9. The Civil Aeronautics Authority investigate the adequacy of the differential between the design level speed, the placarded "never-exceed-speed," and the design gliding speed of the Boeing Model 307, to give reasonable assurance that the aircraft can be controlled within these design limits under adverse flying conditions and that loads beyond its designed structural strength will not be imposed under operating conditions in turbulent air.

10. The Civil Aeronautics Authority require determination of the design loading for tail surfaces on a basis which takes into account the definitely calculated effect of gusts and maneuvers on the total loads imposed and their distribution.

11. (a) The Civil Aeronautics Authority require as a provision of experimental certificates that a V-G recorder be installed on all aircraft intended for air carrier use and that the record obtained from it be preserved and made a part of each flight test report. It is further recommended that a complete record of all flight tests be submitted to the Civil Aeronautics Authority prior to test flights conducted by the Authority in connection with the granting of a type certificate.

(b) An accelerometer be installed in full view of the pilot during all flight tests, and that a limit mark be placed on the instrument to represent the maximum acceleration to which it is safe to subject the aircraft. It is further recommended that such an accelerometer be made a required installation on all air carrier aircraft, and that all readings obtained in flight exceeding an amount to be specified by the Civil Aeronautics Authority for each particular type aircraft be reported to the Civil Aeronautics Authority together with a description of the circumstances, (i.e., indicated air speed, altitude, air conditions, etc.) under which the acceleration was obtained.

(c) An immediate study be made looking toward the development of adequate vibration recorders, taking into consideration the size and capacity of various model aircraft and frequencies which would include the entire range of vibration and flutter affecting their structure and control. As such recorders become available, it is recommended that aircraft manufacturers be required to install them and make the data so obtained a part of the record of all phases of the experimental test flight program.

12. The Civil Aeronautics Authority establish by regulation uniform and more adequate test flight procedures for aircraft intended to be used in air carrier service. It is specifically recommended that such test flights include all maneuvers in which the aircraft might become involved inadvertently under turbulent air or other conditions to be anticipated in air carrier service. It is further recommended that the test flight procedure include complete stalls from straightaway flight and high angle sideslips both in turbulent air and under conditions of unsymetrical power.

13. The Civil Aeronautics Authority by regulation establish a special rating for test pilots of large aircraft and require all test flights of aircraft in this category, prior to issuance of a type certificate, to be conducted by pilots holding such certificates.

* * * *

Studies and research are being continued in connection with certain technical phases of this accident which may require a supplemental report and additional recommendations at a later date.

BY DIRECTION OF THE BOARD

Executive Officer

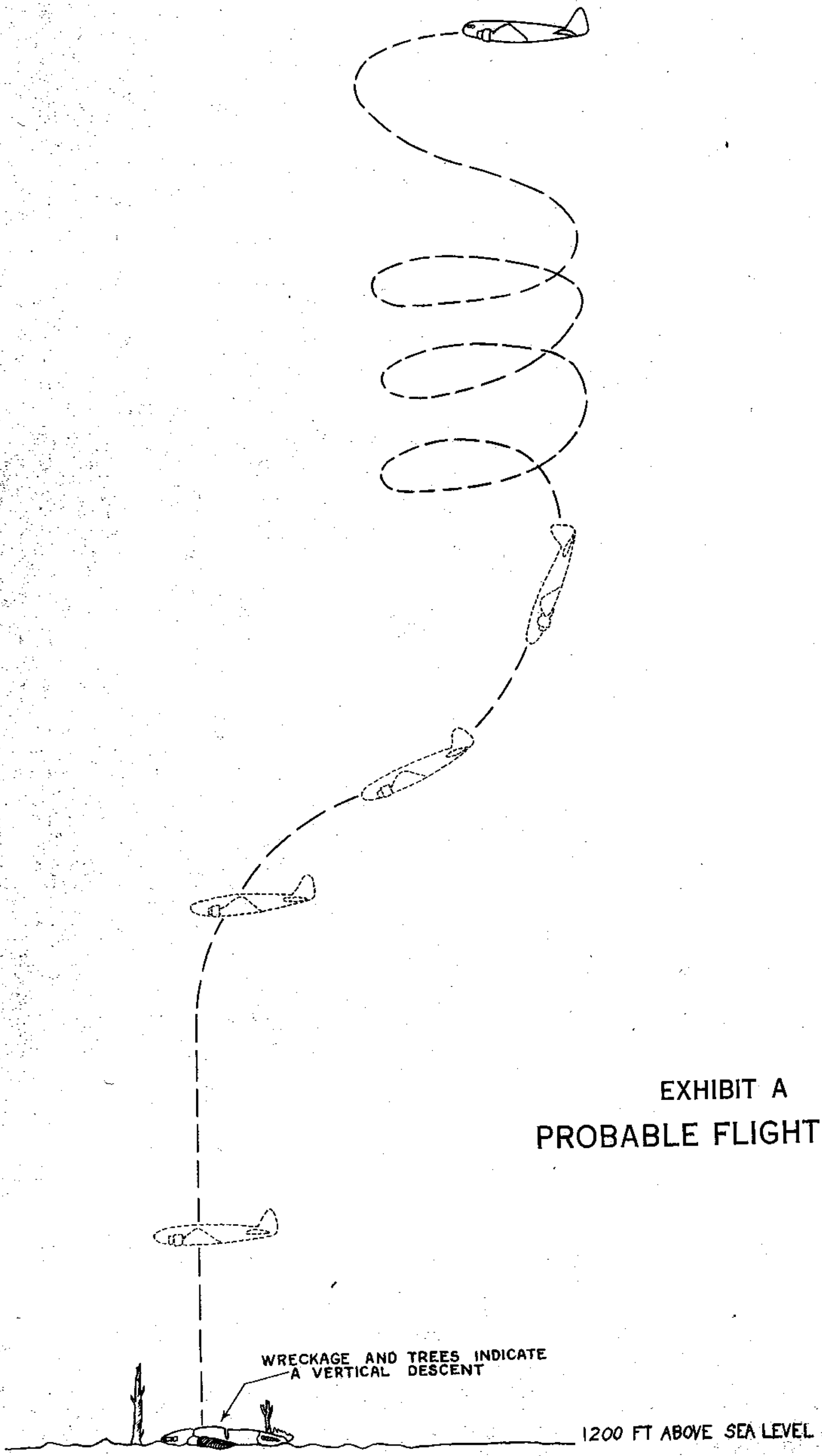

EXHIBIT A

PROBABLE FLIGHT PATH

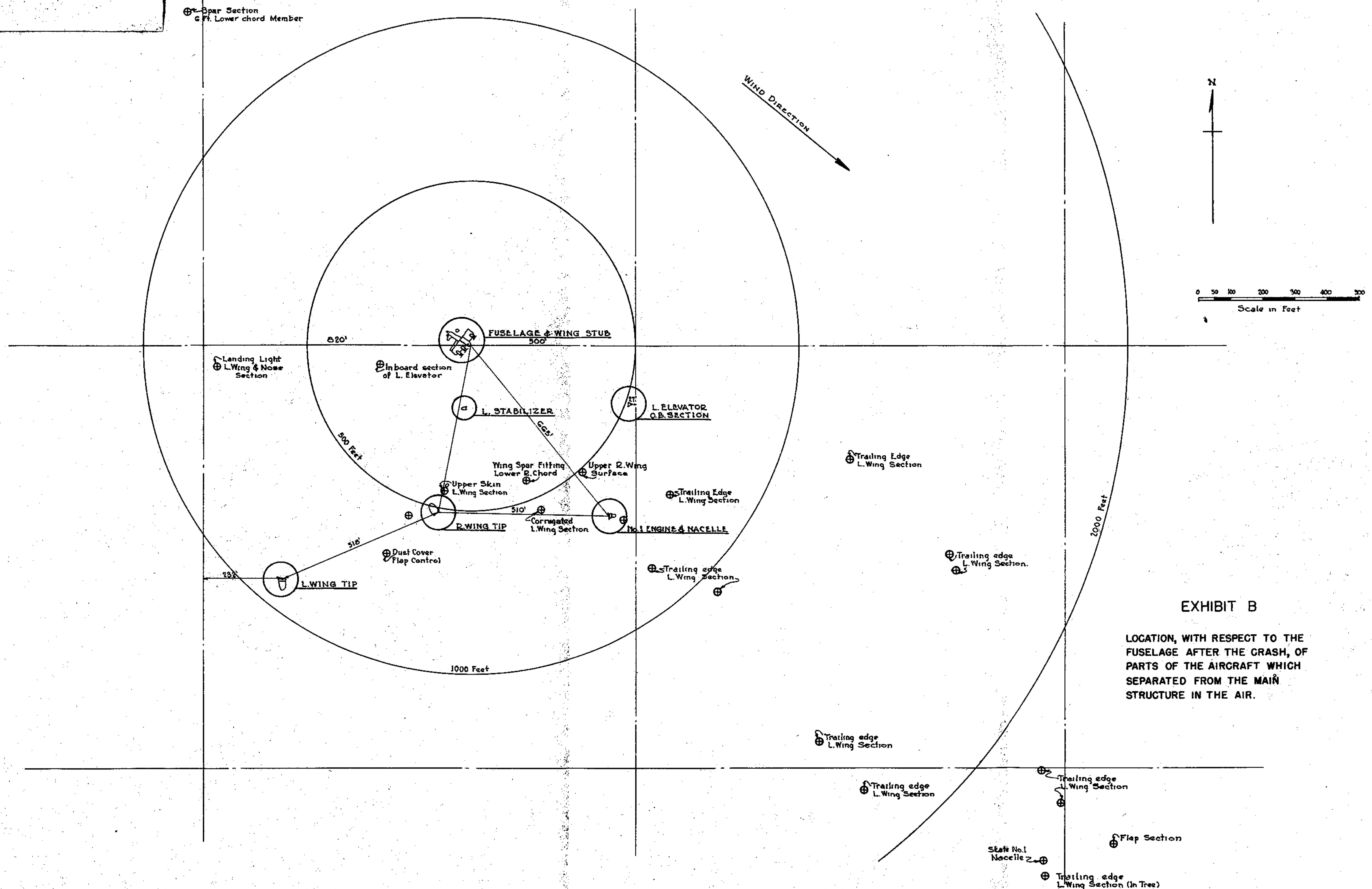

EXHIBIT B

LOCATION, WITH RESPECT TO THE FUSELAGE AFTER THE CRASH, OF PARTS OF THE AIRCRAFT WHICH SEPARATED FROM THE MAIN STRUCTURE IN THE AIR.

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