CIVIL AERONAUTICS BOARD

AIRCRAFT ACCIDENT REPORT

OPTED: January 11, 1963 RELEASED: January 15, 1963

AMERICAN AIRLINES, INC., BOEING 707-123B
N 7506A, JAMAICA BAY, LONG ISLAND, NEW YORK,
MARCH 1, 1962

SYNOPSIS

On March 1, 1962, American Airlines Flight One, a Boeing 707-123B, U. S. Registry N 7506A, crashed into Jamaica Bay slightly less than two minutes after takeoff from New York International Airport, Jamaica, New York. The aircraft was totally destroyed. All occupants, 87 passengers and the crew of 8, sustained fatal injuries.

Flight One was cleared for takeoff from Runway 31L on a regularly scheduled nonstop flight to Los Angeles, California, and became airborne at 1007 e.s.t. The takeoff and initial climb appeared to be normal and a gentle turn to the left was started about 8,000 feet down the runway near taxiway AA, at an altitude of about 100 feet. Straightening out from this turn the aircraft continued to climb for several seconds on a magnetic heading of 290 degrees, and started a second turn to the left, apparently in compliance with radar vector directions given by Departure Control. In the second turn the airplane continued to climb. After initiation of the second turn the angle of bank increased until the airplane rolled through 90 degrees of bank at a peak altitude of about 1,600 feet m.s.l. It then entered an inverted, nose-low attitude and plunged earthward in a nearly vertical dive. The airplane struck the shallow waters of Pumpkin Patch Channel of Jamaica Bay approximately three miles southwest of the Idlewild Control Tower at 1008:49 Floating debris and fuel ignited a few minutes later and burned fiercely.

Probable Cause

The Board determines that the probable cause of this accident was a rudder control system malfunction producing yaw, sideslip and roll leading to a loss of control from which recovery action was not effective.

INVESTIGATION

On February 28, 1962, the aircraft, a Boeing 707-123B, U. S. Registry N 7506A, was flown from Tulsa, Oklahoma, to New York International Airport, Jamaica, New York, as American Airlines Flight 2098, a second section of the regularly scheduled Flight 98. The airplane arrived at Idlewild at 0007[1] on March 1, 1962. The crew of this flight later testified that the aircraft performed normally throughout the flight.

1/ All times herein are Eastern standard time based on the 24-hour clock

After arrival at Idlewild and in preparation for its scheduled departure as Flight One at 0945 on the same day, a layover check and an origination check were accomplished on the airplane. To correct pilot-reported discrepancies, a VHF receiver and the cabin pressure auto controller were changed. Cracks were found in the inlet guide vanes of engine Nos. 1 and 2 and were welded. A scheduled main oil screen change on engine No. 3 was performed, the flight engineer's instrument panel light rheostat-transformer was replaced, and the airplane was serviced. Investigation of the maintenance and servicing performed on N 7506A during its layover at Idlewild showed that these tasks had been properly completed and signed off in accordance with American Airlines procedures before N 7506A was released for dispatch.

The crew involved in this accident departed Los Angeles, February 27, as the crew of American Flight 36, flying a Boeing 720B, arriving in Boston at 0608 February 28. After a layover of about 25 hours the flight crew, operating a Boeing 720B aircraft, American Flight 117, departed Boston on March 1 at 0720 and arrived at Idlewild at 0813. The airline dispatcher, who assisted the crew in preflight preparations for Flight One, testified that they "appeared to be rational and normal." All crew members were currently certificated in compliance with FAA regulations and qualified in accordance with the carrier policy and procedure. A ground crewman involved in preparing the aircraft for departure testified that each member of the flight crew occupied his normal crew position for the flight. The crew consisted of Captain James T. Heist, First Officer Michael Barna, Jr., Second Officer Robert J. Pecor, Flight Engineer Robert J. Cain, and Stewardesses Shirley Grabow, Lois Kelly, Betty Moore, and Rosalind Stewart.

A study of the dispatching procedures utilized for Flight One revealed that dispatching was normal and in accordance with standard company procedures. The aircraft was released from Idlewild with a total takeoff gross weight of 247.038 pounds and the center of gravity at 24.4 percent MAC (Mean Aerodynamic Chord), both within prescribed limits. According to takeoff and climb computations the following performance factors were to be used during the flight:

Stabilizer trim - 1-1/2 units nose-up

Takeoff ground roll - 4,400 feet

Time to 100 knots - 23 seconds

V_1 2/ - 136 knots

V_R 3/ - 144 knots

V_2 4/ - 157 knots

2/ V_1 - The speed below which an outboard engine failure dictates discontinuance of the takeoff and at, or above, which the takeoff may be safely continued.

3/ V_R - The lowest speed at which the pilot should apply force on the control column to rotate the airplane for lift-off.

4/ V_2 - Takeoff Safety Speed: The minimum speed permissible at a height of 35 feet, assuming engine failure at V_1 and rotation at V_R speed.

Pertinent normal operating procedures after takeoff and gear up prescribed by the then current AAL 707-123B Operating Manual are given below for reference purposes in reviewing this report:

Establish positive climb in straightaway flight, and accelerate to $V_2 + 20K$

Order "Flaps 20" (if 30° used for takeoff)[5/]

If maneuvering is required during climb, increase speed to $V_2 + 30K$ before initiating turns.

After reaching 2,000 feet AFL: Accelerate to $V_2 + 50K$, then order "Flaps 0°."

Complete terminal area maneuvering at $V_2 + 50K$ with Flaps up. As soon as conditions permit, accelerate to normal climb speed (300K IAS).

Section 3, page 34 stated: "Close adherence to this procedure will result in attaining the desirable 2,000-foot altitude over critical noise areas as quickly as possible while maintaining approximately a 30 percent margin above stall speed (at least 40 knots) even though maneuvers involving banks as high as 25-30° may be required while clearing terminal area."

Item 4 of the After Takeoff Cockpit Checklist required the yaw damper to be turned ON.

In the Boeing 707-123B aircraft the outboard ailerons are operative during flaps down flight only. A flap system operated mechanism gradually locks out, (fixes at neutral) the outboard ailerons when the wing flaps are retracted from 20 to zero degrees, and gradually brings them into play as the flaps are extended to 20 degrees. The purpose of this is to provide increased lateral control during low speed operation and decreased wing torsion at high speeds when lateral control. produced by spoilers and inboard ailerons, is ample with the outboard ailerons deactivated. It is pertinent also to note that activation of the speed brakes lever to the 20-degree position increases the maximum possible spoiler displacement on the desired wing from 40 to 60 degrees, providing additional lateral control.

A typical departure flightpath chart (Attachment 1) was prepared based on the carrier's normal operating procedures and the computed performance capabilities of the 707-123B. Based on a correlation of witness statements, the control tower transcript and flight recorder data (Attachment 2), the following sequence of events occurred during the flight of N 7506A.

At 0954 Flight One, with 95 persons abroad was given taxi instructions to Runway 31L. The flight was issued an IFR clearance nonstop to Los Angeles International Airport at 1002. The clearance contained local departure procedures and included the then prescribed statement ". . . in the interest of noise abatement do not delay turn to heading two niner zero." In the runup area adjacent to Runway 31L, at 1005:05, American One advised Idlewild Tower, "ready for takeoff," and was immediately cleared for takeoff. Flight One then taxied

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5/ The Boeing Operations Manual permitted flap retraction from 30 to 20 degrees at $V_2 + 10K$.

to the runway and was aligned with it at 1006:29. At 1006:51 the tower requested Flight One to "advise rolling" and one second later, while in the takeoff roll, the flight so advised the tower. What appeared to be a normal takeoff was executed and N 7506A was seen by one of the tower controllers to lift off in the vicinity of taxiway "M", approximately 5,000 feet down the runway, at 1007, as recorded by the controller.

At 1007:37 the aircraft started a gentle turn to the left in the vicinity of taxiway "AA" (approximately 8,000 feet down the runway) at an altitude of about 100 feet, and was established on a heading of 290 degrees at 1007:42. The departure controller made radar contact with the aircraft and observed it roll out on the 290-degree heading. At 1007:48 the local controller advised American One to contact Departure Control on 123.9 mcs. This call was acknowledged.

At 1007:54 the airplane started a second turn to the left, and at 1008:01 American One transmitted his call sign, indicating that he was standing by on Departure Control frequency. At 1008:02 the controller advised Flight One to continue a left turn to one four zero degrees and to report out of 2,000. This advisory was acknowledged at 1008:09. According to the controller the flight continued in what appeared to be a normal left turn, as seen on the radar scope, which gives no indication of aircraft attitude. The target was last observed in the vicinity of the crash site. The controller continued to issue normal radar vectoring advisories, but received no further replies from American One, and the target did not reappear.

Company personnel familiar with the voices of the flight crew, after listening to the control tower recording of transmissions from Flight One, believed that they were made by the second officer. No indications of alarm or any abnormality on the part of the crew were discernible during any of Flight One's transmissions.

At 1008:23 an unmodulated signal of one-half second duration was received on the Departure Control frequency. The sound of this signal was very similar to the unmodulated carrier associated with previous transmissions from Flight One.

The aircraft struck the earth in the shallow waters of Pumpkin Patch Channel of Jamaica Bay during low tide when the depth of the water varied from several inches to several feet and higher elevations of the bottom were exposed. The geographical position of impact was fixed at 40°37.1' North Latitude and 73°50.1' West Longitude, approximately three nautical miles southwest of the Idlewild Control Tower. Impact was made at an angle of approximately 78 degrees nose down, on a magnetic heading of 300 degrees. Readings of the Fordham University Seismographic Station established the impact time as 1008:49. Floating debris and fuel ignited a few minutes later and burned fiercely. All persons on board were fatally injured.

The weather at the time of takeoff was: 15,000 feet scattered; visibility 15 miles; wind northwest at 10 knots; temperature 30 degrees F; dewpoint 11 degrees F; altimeter 30.30 inches Hg. Although the carrier's meteorologist testified that from the air soundings and known winds there could have been occasional moderate turbulence, testimony of pilots who had flown in the area at the approximate time of the accident indicated that they had experienced negligible amounts. However, the vertical acceleration trace made by the flight recorder on N 7506A indicated that Flight One did encounter light friction turbulence. This type of turbulence was also indicated on the flight recorder traces of other aircraft departing Idlewild at the approximate time of Flight One's departure.

Testimony of eyewitnesses indicated that the takeoff and climb appeared to be normal until the bank to the left steepened to an angle beyond that usually expected of a departing aircraft. As it continued its flight, however, witnesses observed N 7506A continue its roll to the left through a 90-degree bank, enter an inverted nose-low attitude and plunge downward in a dive which was almost vertical. During the vertical portion of the dive, which was estimated by a witness to start at approximately 900 feet, little, if any, rotation of the wings about the longitudinal axis of the airplane was observed. Two witnesses described a slight, abrupt bank to the left, followed by a momentary levelling of the wings, immediately preceding the airplane's final steep roll and nose-over. Several witnesses testified they observed cessation of smoke trails from two or more engines when the aircraft was nearing its peak altitude. Only one of the witnesses registered any impression of smoke and/or flames from unusual points of origin during the flight; this witness observed the airplane only one or two seconds before impact. None of the witnesses believed they saw or heard an explosion prior to impact. The majority did not hear any unusual engine sounds, and none saw any object separate from the aircraft during its flight. Those who paid particular attention to the airplane's configuration were of the opinion that landing gear and wing flaps were retracted. Except for one witness who testified that the aircraft appeared to stall just before nosing over, most of the witnesses stated that the entire maneuver was characterized by a smooth, continuous movement, with no indication of recovery action being discernible.

Runway 31L is 14,600 feet long and 150 feet wide, with a gradient of minus .01 percent. It was dry at the time of takeoff. The field elevation is 12 feet m.s.l. The northwest shore line of Jamaica Bay is about 200 yards to the left of and parallel with the runway. Heavily populated areas lie directly beyond the end of Runway 31L.

Noise abatement procedures in effect at the time of the accident required that flights from this runway not delay in making a left turn to a heading of 290 degrees after takeoff and continue the climb to 800 feet on this heading. After reaching 800 feet of altitude, in the interest of traffic separation, flights were turned further left to a heading of 160 degrees and the climb was continued on this heading for two minutes. This departure procedure therefore accomplished two purposes: avoidance of densely populated areas, and separation of Idlewild Runway 31L departures from inbound traffic to LaGuardia. The 20-degree turn from Runway 31L to 290 degrees is within the limits of the jet transport performance criteria developed for noise abatement maneuvers by representatives of the Air Transport Association, the Air Line Pilots' Association, and Aerospace Industries Association, and promulgated by the Federal Aviation Agency in their Manual of Noise Abatement Procedures. Since the accident the FAA restricted the commencement of the first turn until the aircraft reaches an altitude of 300 feet and also eliminated the advisory, "In the interest of noise abatement, do not delay turn to 290 degrees" from the departure clearance for Runway 31L. As of December 25, 1962 the procedure was changed to require a climb on a 290-degree heading to 1,000 feet before further turns are made.

Insofar as possible during the investigation attention was directed to pathological, histological, and toxicological studies of the bodies of the flight crew which might reveal some indication of physical incapacitation. The results of the toxicological studies were conclusive in ruling out the possibility of incapacitation due to toxic gases, alcohol, and drugs. However, the massive destruction of the bodies and the lack of vital portions of tissue made it impossible to obtain results which would give irrefutable positive or negative proof of incapacitation insofar as the pathological and histological examinations were concerned. Studies were also made of the medical histories of the flight crew, but no evidence could be found to indicate that any member had physical characteristics likely to result in an incapacitation of any kind. Examination of the bodies of the passengers did not yield information of any significance as to the cause of the accident except to support other evidence that there had been no fire or explosion in flight.

Despite extensive damage to the flight recorder foil, exhaustive efforts resulted in restoring it to a condition whereby an accurate readout could be obtained, with two exceptions. Both exceptions involve the heading trace as it passed through an area of tears, wrinkles and abrasions; where the heading changed from 245 to 225° M, and again from 214 degrees through an indicated reversal at 332 degrees. Only the first exception is of significance to this investigation and will be discussed later in the report.

A chart of the flight recorder data, with added information, is attached for reference (Attachment 2). In discussing the recorder readout it must be recognized that absolutely precise time correlation of the four traces was impossible and errors of one or two seconds between them may exist. An additional error is introduced by the friction and play in the recorder, resulting in short-period (up to two seconds) aberrations of airspeed, altitude and heading traces from the smoothly varying changes made by the airplane. Characteristic of these are the numerous steps in the airspeed trace for the takeoff run and acceleration to 190 knots. Although times, speeds, altitudes and headings are usually expressed in terms of exact values throughout the flight recorder discussion, it must be understood that they are approximate but generally accurate to within plus or minus one second. The time scale used in Attachment 2 is based on the time of impact as established by the Fordham University seismograph. The computed aircraft performance factor, "Maximum time to 100K," indicates the maximum number of seconds allowed for the aircraft to accelerate to 100 knots IAS and is a measure used by crews as acceptable performance to this point.

Except where otherwise specified, the stall speeds referred to herein are based on a takeoff gross weight of 247,000 pounds and a gross weight at impact of approximately 246,000 pounds, with a CG position of approximately 24 percent, and were determined in accordance with applicable type certification requirements of Civil Air Regulations Part 4b and Special Civil Air Regulation SR 422B.

Study of the four traces indicates that the flight was normal during the takeoff and the first part of the climb. The acceleration trace variation from lift-off to 1008:29 indicates that N 7506A was operating in lightly choppy air, characteristic of mechanical friction turbulence.

During takeoff, dips in the airspeed and altitude traces indicate that rotation occurred at 1007:24 when the indicated airspeed was 145 knots (V_R = 144 K computed). Lift-off occurred at 1007:28 when the indicated airspeed was 155 knots (V_2 = 157 K computed). The variation in heading trace for six seconds after lift-off corresponds with normal correction for wind drift.

The heading trace indicates that the airplane started its first turn to the left after takeoff at 1007:37 at an indicated altitude of about 80 feet and an airspeed of 180 knots. At this indicated altitude the airplane would still be in its ground effect and the actual altitude would be 100 to 120 feet.

The turn continued to a heading of 290 degrees, an altitude of 260 feet and an airspeed of 190 knots at time 1007:42. The rate of turn to this heading corresponds to a bank angle of about 30 degrees in a coordinated turn. However, the median acceleration trace corresponds to a bank of 20 to 25 degrees, indicating a slight skid. With flaps at 20 degrees, the airplane would then be more than 40 knots above stall. A 290-degree heading was maintained for 12 seconds (1007:42 to 1007:54) to an altitude of 700 feet with airspeed still at 190 knots.

At 1007:54 a second climbing turn to the left was started. This turn continued at a nearly constant rate of 2.34 degrees/second, to a heading of 275 degrees, an altitude of 920 feet and an airspeed of 192 knots at time 1008:01. In a coordinated turn, this heading change rate corresponds to a bank angle of 22 degrees with the airspeed 50 to 60 knots above stall. Almost coincidental with the end of this segment of the turn, Departure Control advised Flight One to continue its turn to 140 degrees.

During the same segment of the second left turn it is most likely that the wing flap retraction from 20 to zero degrees was initiated at about 1007:57, at an altitude of 800 feet and airspeed of 190 knots. Retraction normally requires 12 seconds and this operation would therefore have ended at 1008:09 at an altitude of 1,350 feet and an airspeed of 200 knots. When flaps are retracted from 20 to zero degrees it is common for the recorded airspeed to increase slightly. During this period the trace indicates such an increase from 190 knots to 200 knots. In addition, Boeing flight tests to simulate Flight One as closely as possible indicate that later retraction of flaps to zero degrees would result in lower climbout performance than reflected in Attachment 2. Coincidental with completion of flap retraction, Flight One's last recorded message was received by Departure Control.

Returning to consideration of the second left turn, the heading change becomes slightly wavering from 1008:01 to 1008:07, and in the turn the heading changed from 273 to 250 degrees, averaging a rate of 3.83 degrees/second. Damage to this area of the flight recorder foil produced a slight displacement and rotation with the result that the heading change rate indicated is probably higher than actual. However, the difference is so small as to be of negligible importance in this discussion. With the airspeed increasing from 193 to 198 knots at the same time, the bank angle would be about 35 degrees and the increase in normal load factor due to the turn would be about 0.2 g in a coordinated turn.

This is roughly in agreement with the mean value of the decreasing acceleration trace during the same period. Under these conditions the airspeeds would have been 50 knots or more above stall with 20 degrees of flaps and 35 knots or more above with flaps fully retracted.

Starting at time 1008:07 the heading trace began to record changes the significance of which cannot be precisely defined. From this point in time, therefore, the recorded heading changes will be described but the possible meanings to be ascribed to them, except where the reason for a change is clear, will be discussed under analysis.

At 1008:07, within six-tenths of a second, a heading change from 250 to 243 degrees is recorded; a change rate of 12 degrees/second, amounting to an instantaneous tripling of the 3.83 degrees/second change rate recorded just prior to this time. From 1008:08 to 1008:16 the heading trace passes through the badly wrinkled, abraded and torn portion of the foil, during which time the heading changed from 245 to 225 degrees. At time 1008:16, the heading trace emerges from the badly damaged protion of the foil and for the next one and one-half seconds indicates a much lower rate of heading change than that immediately preceding entry into the damaged area. The heading trace shows a momentary cessation of turn at 1008:19 followed by a higher turn rate continuing until 1008:31, when a sharp reversal in the recorded heading is noted. Such an abrupt change from a left turn to a right turn is beyond the aircraft's capability and is an indication of gimbal error in the directional gyro of the airplane as the angle of left bank approached 90 degrees. From this time to about 1008:42 the recorded headings are similarly in error due to the high roll and pitch angles of the airplane as it inverted and dove to the ground.

At time 1008:18, 1008:25 and 1008:29 the airspeed and altitude traces indicate sharp simultaneous increases which were beyond the aircraft's performance capability, and the highest peaks recorded indicate an airspeed of 230 knots and 2,000 feet of altitude at 1008:30. The precise significance of these recordings cannot be completely defined. However, a single static port low on the left side of the forward fuselage is connected to the airspeed and altitude sensors of the flight recorder. As a result, nose left sideslip (relative wind from the right) and high angles of attack cause appreciable plus errors in the recorded airspeed and altitude.

The median acceleration trace shows a rise from 1.0 to 1.8 g from time 1008:26 to 1008:30 at which time an abrupt change is recorded in a manner indicative of heavy stall buffet, which intensifies and continues until impact. Traces from the same make and model of flight recorder made during Boeing Company test flights in a 707-131B airplane substantiate this observation.

Immediately following the recorded peak airspeed and altitude, both parameters drop abruptly, and the airspeed trace records its lowest subsequent speed of 170 knots at 1008:37. This rapid peaking and decrease in the airspeed and altitude traces indicates pronounced sideslip effects coupled with increased drag resulting from prolonged heavy buffeting.

The abrupt changes of the airspeed, altitude and acceleration traces pinpoint the time of impact within plus or minus one second. As previously mentioned the impact time was established at 1008:49 by the Fordham University seismograph.

An extensive review of the maintenance records of N 7506A was made in search of information which might have some bearing on the accident. This review was conducted at the facilities of both American Airlines and The Boeing Company and, when deemed necessary, maintenance and inspection personnel were interviewed. The flight logs and the maintenance, inspection and overhaul records were studied with special attention directed to the flight control systems. Records of major modifications performed on the airplane were also investigated. One instance of improper maintenance was found. During intended compliance with an American Airlines Engineering Change Order, an outboard bellcrank was erroneously installed at the inboard bellcrank position of the spoiler controls in the right wing. No functional or operational check was required or performed upon completion of the Engineering Change Order. On the following flight, the crew reported difficulty with unsymmetrical spoilers, whereupon maintenance personnel adjusted the speed brake and spoiler control rods to correct the condition. After 13 additional flights, during which no flight discrepancies were logged concerning the spoilers, inspectors discovered the error. A correct installation and re-rigging were then accomplished on February 25, 1962, with no subsequent complaints concerning the spoiler system as a result of the intervening three flights prior to the accident. Except for this instance, all records examined reflected that the aircraft was continuously maintained in an airworthy condition in accordance with FAA-approved company policies and procedures.

All four powerplants suffered extensive and similar damage characteristic of a high-velocity, nose-down impact of the aircraft. The extent of torsional damage in the four engines indicates that each was operating at approximately 60 percent r.p.m., which is equivalent to flight idle thrust, at time of impact. Although detailed examinations were performed on each of the shattered engines, no evidence of in-flight damage or failure could be found.

The crater made by the aircraft in the bottom of the bay was approximately 130 feet long and 8 to 10 feet deep. On impact the wings were fragmented and the fuselage crushed accordion-like, breaking into many sections. Part of the horizontal stabilizer and elevator with the tip of the fuselage attached was the largest piece of structure recovered. The fire, which ensued shortly after impact, heavily damaged the above-water portions of the airplane structure. Impact and fire damage was so extensive as to preclude examination of numerous components of the aircraft which might possibly have yielded important information. No evidence could be found to indicate that there had been an in-flight fire, an explosion, structural fatigue, or overload failure.

The cockpit area suffered the most extreme fragmentation of the entire fuselage, the degree of fragmentation gradually decreasing toward the tail of the aircraft. The horizontal stabilizer broke loose at FS (fuselage station) 1505, with the tail cone intact and still attached. The vertical stabilizer tore out of its structural attaching bulkheads at FS 1440 and FS 1507 at impact.

The landing gear was determined to be in the fully retracted position. There was no indication of defective treads on any of the tires, nor was there any evidence of a tire blow-out in any of the wheel wells.

The skin and most of the remaining structure of both wings suffered severe fragmentation. Examination revealed numerous indications that all wing flaps were in the fully retracted position.

"Reconstruction" of the wreckage was made in a hangar for detailed study. As minute an examination as possible was made of the lateral control system considering the extensive damage it had sustained. Of the eight control cables in the right wing, only the right wing down trim cable was missing and all others were in place. Of the eight cables in the left wing, all were in place except for the bus cable ABSB, which was missing from WS (wing station) 460 outward. In the right wing, most of the inboard aileron was burned away; the outboard aileron was complete. In the left wing, the inboard aileron was fairly well intact despite heavy fire damage; the outboard aileron was extensively damaged by the impact and fire. The aileron lockout mechanism of the right wing was determined to be in the locked-out position. Major portions of the left wing lockout mechanism were not recovered, but the actuator screw was found extended to the locked-out position. Impact damage to three of the four aileron bus quadrants disclosed that the right inboard aileron was 10 degrees UP and the left, 10 degrees DOWN at impact. No physical evidence of in-flight failure in either the right or left aileron system was found. The components of the right wing spoiler system were damaged extensively during impact. Examination indicated that at impact the right wing inboard spoilers Nos. 5 and 6 were 28 and 31 degrees UP respectively, and the outer panel of the right outboard spoiler was 40 degrees UP. The left wing spoiler system sustained severe impact damage and heavy fire damage, and the spoilers were found in the full DOWN position. A tear-down inspection of the eight spoiler actuators and four spoiler control valves revealed that these components were capable of normal operation prior to impact. Severe damage to the lateral control system due to impact and fire precluded the examination of some essential parts. Both cockpit control wheels were determined to be slightly beyond the position for a full right wing down control command.

Examination of the horizontal stabilizer revealed no evidence of any malfunction. Measurement of the position of the nut on the stabilizer jackscrew corresponded to a cockpit indication of 2.3 units nose UP trim.

On impact the vertical tail tore completely loose from the fuselage, landed on its right side in the area of the most intense fire, and the rudder and tab were almost completely destroyed by fire. The upper portion of the vertical stabilizer was severely damaged by impact and fire but the remaining identifiable parts included the rudder boost unit, the tab damper, and portions of the Q bellows assembly. The latter was partially consumed or melted but was still attached to structure; all cables and fittings were in place down to the Q rod front fitting where the rod had melted away. The aft rudder control quadrant had melted, but the cable attach ends, with attach brackets and bolts still installed, were found in a mass of solidified metal. However, it was impossible to differentiate the remaining portions of the control cables from those of the servo cables. The aft fitting of the rod attaching to the rudder boost control valve was properly bolted and safetied. No other part of this rod or any portion of the attaching ratio bellcrank, tab control rod, or rudder quadrant control rod, could be found. The rudder control (compound) bellcrank was found with all the rod ends properly attached. The rudder trim torsion rod was in place with its upper end pushed up into the sleeve, and thus disengaged, as a result of impact deformation. The trim drum and gears were recovered and appeared to be in normal condition. The rudder centering spring mechanism showed no evidence of cables climbing out of the pulley grooves.

After removal of the main wreckage to a hangar, the accident site was combed with hand rakes. U. S. Army personnel, with mine detecting equipment, later assisted in the search for wreckage. Because of adverse weather conditions and exceptionally high tides, recovery of the wreckage was difficult and slow. A hydraulic dredge was employed to recover pieces believed to be imbedded in the muck. This operation was conducted for a period of three to four weeks during which time numerous pieces of wreckage were recovered. The search was continued using a crane with clam-shell digging equipment, resulting in recovery of additional wreckage. Some of the wreckage recovered was in the form of metal masses resolidified after having melted. These were given X-ray examination and in some cases chipped apart for study.

Examination of the airplane communication equipment revealed frequency selection in the recovered equipment appropriate for the period of flight in question.

The electrical system was studied for any indications of an electrically caused fire in flight or the malfunction or failure of any system due to electrical faults. Although the thoroughness of this study was restricted by impact and fire damage, no evidence was found to indicate that an electrical arc, short or overload had existed in the electrical system prior to impact. Numerous indications obtained from the wreckage disclosed that electrical energy was present at impact.

The hydraulic system, also damaged by fire, yielded evidence that it was operating until time of impact. The previously mentioned positions of the inboard and outboard spoilers in the right wing, which require both utility and auxiliary system hydraulic power, is one such indication. Ultraviolet examination of the face of the rudder system hydraulic pressure gage, located in the first officer's instrument panel, revealed an outline of the hand at 3,800 p.s.i. The pressure transmitter, which actuates this gage, checked normal when tested.

The rudder hydraulic pressure control valve, electrically operated, was determined to be in the electrically deenergized, or 3,000 p.s.i., position normal for airspeeds below 245 knots. The electrically operated rudder pressure shutoff valve was also determined to be in the pressure ON, electrically deenergized position. When disassembled, it was found that the nickel plating had partially peeled or flaked away within both of these valves. However, subsequent tests by Boeing indicated that any nickel particles thus released into the system would not adversely affect the operation of the rudder boost unit due to a filter at the boost unit inlet.

Rudder damper measurements corresponded to a rudder position of 17.5 degrees LEFT. The piston of the hydraulic actuator of the rudder power control unit was extended 3/4 inch from neutral in a direction corresponding to a position of 9 or 10 degrees RIGHT rudder. Due to various factors the heat of the ground fire would not tend to produce any changes in this position. This is the rudder travel which can be produced by 200 pounds of pedal force at 200 knots airspeed with boost OFF. The actuator control valve spool had moved beyond the position normal for right rudder movement and had been driven through the rear end of its housing. Severe fire damage was evident at this point. However, the over-driven position of the spool was obviously a result of impact forces.

The autopilot disengage switches on the captain's and first officer's control wheels were found with the buttons in a depressed position beyond normal travel. A special study was conducted to determine the significance of this finding. Examination revealed that both operating button plungers had thrust completely through the bakelite bottom of the switch. Normal operating pressure on the disengage button is 3 to 4 pounds. Tests showed that a load of about 90 pounds is required for the plunger to break through the bakelite bottom, a force beyond the physical capability of a pilot's thumb to produce, considering that most of the thumb pressure would be applied to the switch plate rather than to the button. No marks on the top of either plastic button were found to indicate they had been struck by harder aircraft structure during impact. Tests prior to disassembly of the switches and visual examination afterwards showed that the electrical contacts were open in both switches, corresponding to autopilot disengagement. However, the positions of the contactors were abnormal in a manner consistent with the overdriven condition of the plungers. As a result, the condition of electrical discontinuity is not indicative of the switch positions immediately prior to impact.

The automatic flight control system was extensively investigated. This system provides automatic coordinated control of the airplane, and a damper control mode is available to augment yaw stability when the airplane is controlled manually.6/ The autopilot is an electronic, electromechanical device that converts small electrical input signals into mechanical movements of the control surfaces. Sensing devices generate signals that are amplified and converted to electrical power used to energize servo motors which actuate control surfaces directly or through hydraulic actuators. The voltage generated by a sensing device, as it is applied to the amplifier input, is modulated by other voltages generated by a sensor of control surface movement. These modulating voltages originate in surface position transmitters and in units that signal the rate at which a surface is moving. The combined input signal level is in the order of one volt.

The control panel for this flight control system was not recovered from the wreckage though portions of some of the controls were. The autopilot engage switch was recovered and was found jammed in a position slightly toward the autopilot engage position; this direction was also that of impact forces.

The control unit contains two rate gyros that sense rotation rates in pitch and yaw. Transmitters actuated by these gyros generate electrical signals, the strength and sense of which are governed by angular velocity of the aircraft. These are fed to the input of the main autopilot amplifier, and if the autopilot or yaw damper is engaged, certain control surface movements will result.

The rate control unit was recovered with the cover of the housing collapsed inward, but no marks to account for this damage could be found. The yaw gyro was not on its mounts. The pitch and yaw gyros are identical except for the direction of spin axis, and there was a definite difference in damage to these gyros. The

6/ The autopilot engage switch located on the autopilot control panel is a toggle type marked AUTOPILOT-DAMPER and is moved forward to engage the autopilot for three control channel operation (AUTOPILOT position) and aft for rudder channel only operation (DAMPER position). The switch is spring-loaded to the (unmarked) OFF position. The other two positions are solenoid held in the desired position.

frame holding the yaw gyro had a large section broken from the side and the missing section was not found. The frame holding the pitch gyro was whole and intact.

The surface servo units are electromechanical devices which convert electrical signals into proportional mechanical forces that adjust the aircraft control surfaces. Each servo contains an electric motor which is geared to a pulley through a clutch. In this airplane the pulley drives a cable which is connected to a tab on each of the primary control surfaces and to the power unit in the rudder system only. The electrical signal from the amplifier drives the motor and eventually the flight control surface. As the control surface moves to the desired position a follow-up autosyn generates a signal in proportion to surface displacement which opposes the original initiating signal and stops the surface at the desired displacement.

Mounted on the end of the servo motor shaft is a rate generator which develops a voltage in proportion to the motor speed, and in opposition to the initial input signal. The electrically energized clutch of the servo is engaged when the autopilot (or yaw damper in case of the rudder servo) position is selected. The motor, rate generator, follow-up autosyn, clutch and gearing are all enclosed in a cast aluminum alloy housing. The rate generator and end bearing for the servo motor shaft are enclosed in a cylindrical projection of the bell housing on the servo motor assembly. This projection is about 5/8 inch long and about 1-1/8 inch in diameter. A large disc-shaped flange of the end bell is recessed below the end of the motor case approximately 3/16 inch and is retained in this position by a snap ring. The gap between the inner diameter of the motor case and outside diameter of the cylindrical projection is approximately 5/8 inch. One of the wire bundles in the servo assembly which contains the 8 wires to the servo motor and rate generator is wrapped completely around the end bell projection and is snugly held against it at the corner formed by the projection and the disc.

All of the servo units from N 7506A were recovered and examined with the other components from the autopilot system. Each servo was visually checked for damage and none of the servos was found to have been damaged by fire, but all were corroded from exposure to salt water and marked or broken to some extent. Continuity and visual checks were made of the wiring and components. Nothing could be found in the examination of the aileron and elevator servos to indicate the existence of a malfunction prior to the accident.

A large portion of the housing of the rudder servo was missing, partially exposing the servo motor. The rate generator end of the motor was completely exposed. The continuity check of the rudder servo wiring showed an "open" in the rate generator circuit. The rate generator was then disassembled but no faults were found until the protective sleeving covering the wiring to the rate generator and motor was removed. The wire with brown insulation and the wire with orange insulation were found to be severed. The blue wire was holding together with only one strand. Some of the strand ends of these wires bore the appearance of having been cut by a sharp edge, some were pinched and flattened, and some were necked down, as disclosed during microscopic examination by the Board's metallurgist.

The brown wire connects the output of the rate generator to the input of the autopilot amplifier. The blue wire connects 18 volts AC to the rate generator input and the orange wire is the ground or return side of the 18 volts input. The separations in the wires were adjacent to each other. The protective sleeving was then examined and a transverse separation having the appearance of a cut or slice, with a puncture-like indentation at one point in the sleeving separation, was seen to be adjacent to the severed wire ends. When repositioning the wires about the rate generator a series of radial scratches and/or gouges was noted on the end bell of the servo motor under and in line with the wire bundle damage. The sleeving separation extended approximately three-fourths of the way around its circumference, one end being below center on the extreme outer periphery of the bundle as wrapped around the end bell projection. The separation extended from this point up around the top of the bundle and down the side adjacent to the projection approximately one-third of the distance to the extreme inside diameter of the bundle, as wrapped around the projection. Just below this end of the separation there are three indentations in the sleeving, two of them connected by a linear depression in line with the separation. The lower two of these indentations are on the sleeving surface normally in contact with the surface of the projection. Approximately 1/2 inch from the separation area and on the surface of the sleeving adjacent to the surface of the projection, there are several additional indentations and punctures. It was also observed that the wire and sleeve damage and the scratches on the end bell were in an area somewhat protected by the end bell projection and the projection of the motor case above the end bell disc. No marks could be found on this assembly in the area of the wire damage which would indicate that the wire bundle had been struck by some object or otherwise damaged during the break-up. On the end of the servo motor case at the point nearest the clutch housing, the outer surface of the case is flattened and scratched by interference with some other object.

To determine, if possible, whether the damage found on the rudder servo was unique, the aileron and elevator servos were reexamined. The end bell surface of the elevator servo motor was found marked and, although corroded, the surface of the end bell also appeared to have been scratched or marked. Eight spare servo unit motors from the American Airlines stock were then examined, and six of these had the same type of scratching or gouging as found on the rudder servo from N 7506A. Some of them also had similar indentations or imprints on the sleeving enclosing the wires.

In each of the cases examined the marks on the end bell and sleeving were observed to be adjacent to each other and it was also noted that this damage occurred at the same radial position on the end of each motor. In each case where the sleeving was marked or indented, the scratch marks appeared on the end bell. One of the scratched units from the American Airlines stock still bore the manufacturer's seal indicating that it had never been disassembled since last leaving the factory.

As a result of these findings, inspection of servo units was made on the production line at the manufacturer's plant. Board investigators enlisted the aid of the FAA manufacturing inspectors who found six unsatisfactory units.

Marks, indentations and electrical wire damage within the sleeving were found which was similar to the damage previously mentioned. FAA inspectors determined that this damage had occurred as a result of improper use of tweezers when tying the wire bundles to the motor housing. Additional units were found to have marks and damaged protective sleeves, but no wire damage within the sleeves.

At the request of Board investigators, the damaged rudder servo unit from N 7506A was subjected to a searching examination and analysis by the manufacturer at his plant. In a report of this examination, which was later submitted to the Board, the manufacturer concluded that it would be highly improbable for the unit to pass the electrical requirement of the final assembled servo if a lead or leads within the protective sleeving were severed during assembly. The manufacturer further concluded that his examination indicated that the damage could have been the result of flying fragment damage to the sleeving and leads at the moment of impact.

Discovery of damage to the wire leads to the autopilot rudder servo from N 7506A dictated studies to determine what effects such damage could have had on the performance of the airplane. The Boeing Company was requested to perform the necessary tests.

As a result, bench tests were first conducted to determine what autopilot system degradation or malfunctions could occur. These tests showed that a hot wire failure of the 18-volt excitation lead, i.e. blue or orange, making contact with the brown signal lead, produced a "yaw damper hard-over" 7/ to the left or right, depending on the particular connections used. They also showed that loss of excitation voltage as a result of the severed 18-volt lead produced only insufficient servo damping.

Flights were also conducted in an attempt to duplicate the bench test malfunctions while simulating approximately the flight conditions of N 7506A. A Boeing 707-131B was used in the flight tests. Duplication of the crossed wires malfunction in one maneuver, starting from a 30-degree banked turn to the left at constant altitude produced in eight seconds a left rudder deflection of 7 degrees at 210 knots IAS, causing the airplane to sideslip and to roll to the left. Although the hard-over signal was continuously applied throughout the maneuver and recovery action was delayed for four seconds, sufficient aileron control was available to stop the roll at 56 degrees in one and one-half seconds and then to level the wing. Measurements showed that a rudder pedal force of 75 pounds 8/ was required to move the rudder back to neutral against the hard-over servo force. Duplication of the open wire malfunction produced a small amplitude oscillation which was hardly perceptible in the response of the airplane. These flight tests were conducted in conditions of 1.0 g flight loads.

7/ Previous failure analysis of the autopilot system had resulted in the conclusion that electrical/electronic failures can occur which would result in a maximum servo force being exerted continuously to drive the control surface against its opposing air forces. The condition of unwanted maximum rudder servo force being applied constitutes what is commonly termed "yaw damper or autopilot hard-over." FAA-type certification requires that application of such a malfunction will not produce hazardous airplane attitudes or loads, assuming the pilot will initiate corrective action three seconds after the malfunction manifests itself. The rudder servo motor is capable of commanding only 7° to 8° rudder deflection in the speed range of 200 to 210 knots IAS.

8/ The slippage torque of the servo motor used in this test was approximately 15 percent below maximum allowable. A force of 90 to 110 pounds would be necessary to overcome a yaw damper hard-over with a servo motor developing maximum torque. Boeing certification data shows that with the maximum torque value of 118 inch-pounds a rudder deflection of 8 degrees is possible.

Other studies and flight tests were made by the Boeing Company in an effort to pursue various leads in the investigation. One of these tests involved known incidents of the binding of a primary flight control system due to failure of a cable pressure seal which prompted tests to obtain more data on the effects of this type of malfunction. These tests showed that, though increased pressure on the controls was required, continued control movement to overcome the binding was well within pilot capability. Other Boeing studies concerned the possibilities during the flight of N 7506A of: a rudder boost hard-over; jammed control wheel; jammed outboard aileron; jammed spoilers; stall during wing flap retraction; engine failure; and pilot inattention, distraction or incapacitation. As a result of these studies Boeing concluded that no single one of the airplane malfunctions considered should cause loss of control, as evident in this accident.

The carrier conducted a study of the flight recorder traces based on energy gain and loss throughout the flight. By consideration of total head 9/ and specific energy 10/ histories derived from the altitude and airspeed traces, and consideration of the median acceleration trace, "actual" altitude, speed, lift coefficient and sideslip histories were deduced. However, since energy is greatly affected by engine thrust, three variations of total thrust during the critical phases of the flight, from time 1008:14 to 1008:30, were considered. These are: continuing full thrust, a 25 percent reduction in thrust, and a 50 percent reduction in thrust. These separate conditions produce significant differences in the sideslip histories. With no reduction in power the altitude and airspeed trace variations correspond to a sideslip history of low magnitude wavering through zero and reaching a maximum of 3 degrees nose left slip at 1008:30. The cases of power reduction result in a shift of the sideslip history to nose left with greater resultant slip angles, the maximum being approximately 7 degrees nose left for the 50 percent power reduction at time 1008:30. Insofar as the energy analysis alone is concerned the two cases of power reduction apply equally to either symmetric or asymmetric engine thrust. The lift coefficient histories change with the assumed variations of total engine thrust and reach values of approximately 1.00, 1.06 and 1.11 at 1008:30 for the 100 percent, 75 percent and 50 percent thrust conditions respectively. From this analysis the carrier then reasoned "that it is most improbable that the significant disturbance occurred on the yaw axis, either through thrust asymmetry or through rudder action." Further, the carrier stated "to the contrary, we conclude that an initial and critical disturbance on the roll axis through the ailerons is a much more likely possibility . . ." The lift coefficient from the carrier's energy analysis corresponding to this belief of the carrier is approximately 1.00 at time 1008:30 when the flight recorder acceleration trace (Attachment 2) reached its first abnormally high peak.

A program of flight tests known as "Project RACE" was originated by the Federal Aviation Agency in an effort to shed light on the cause of the accident. Organizations participating in this effort to varying degrees included the National Aeronautics and Space Administration, American Airlines, Boeing and the Civil Aeronautics Board. All tests were flown in an FAA-owned Boeing 720 with FAA pilots controlling all flights and performing the maneuvers in all instances. These tests had three main objectives including: provision of flight recorder traces for comparison with those made by Flight One by attempting to simulate possible flight conditions of N 7506A; measurement of the response of the airplane to various pilot opposed malfunctions, particularly of the rudder control system; and measurement of the effects of slips and skids by means of extensive test instrumentation.

9/ Total head is equal to the sum of the dynamic pressure measured at the pitot head and the static pressure measured at the static port.

10/ Specific energy is equal to the sum of the potential and kinetic energy divided by the weight of the airplane.

to make possible a more definitive study of the flight recorder traces from Flight One. As a result of this program the FAA concluded, in a January 1963 draft of the Project RACE Report, that the "data from autopilot11/ hard-over rudder tests do not appear to resemble trace characteristics of the accident data. Data from Project RACE tests of pilot hard-over rudder, however, appear somewhat more severe than the CAB read-out of the accident flight recorder tape." This draft report stated also, "It was found that the automatic pilot11/ system could not, within the limitations of its force authority, displace the rudder control sufficiently to develop sideslip angles to the extent that would cause uncontrollable lateral roll.

"During our tests simulating American Airlines Flight 1 configuration it was found that the rudder boost system did have this capability. Should the rudder boost system command full rudder, within the limitations of hydraulic pressure and aerodynamic resistance, the resulting sideslip angle would cause lateral rolling that could only be arrested by reducing rudder boost pressure or assisting lateral control by deploying symmetrical speed brake handle or asymmetric thrust."

Project RACE provided the Board with much information that will prove helpful in future accident investigations. Insofar as investigation of this particular accident is concerned, the Project RACE test data are considered valuable principally as corroboration of more applicable Boeing flight test data in some respects. However, the Project RACE tests were made in an airplane having lower thrust engines, a shorter fuselage, lower moments of inertia about the Y and Z axes, and lower gross weights than necessary for close simulation of the conditions of Flight One. The lower gross weights necessitated power reduction for comparable performance, which resulted in decreased thrust asymmetry with one engine idling, or too much thrust reduction to produce asymmetry equal to the loss of one outboard engine during Flight One. The special tests by Boeing in a 707-131B were closer in these respects. In addition, the Boeing tests to simulate yaw damper and rudder power control malfunctions included more realistic rudder deflection-time histories than those of Project RACE. Both the Boeing and the Project RACE malfunction tests were essentially "1 g" maneuvers which did not approach the high vertical accelerations and lift coefficients experienced by Flight One.

ANALYSIS AND CONCLUSIONS

Throughout the investigation numerous possibilities as to the cause of the accident were considered and the merits of each were carefully examined. With the evidence that was amassed, all possibilities were narrowed down to the following areas which will be discussed in this report: physical incapacitation of the crew; loss of engine power; loss of lateral control; malfunction of the rudder boost system; and malfunction of the rudder servo unit. However, prior to detailed discussion of the causal areas there are several subjects of pertinent interest which must be treated.

It is important to keep in mind that, except where otherwise specified, the stall speeds referred to in this report apply only to coordinated flight

11/ Equally applicable to yaw damper.

conditions at the approximate CG position of Flight One and, that stall speeds for high sideslip angles and extended spoilers are higher. It is also important to note that initial stall buffet is caused by separation of the airflow on the inboard portions of the wings. With increasing angles of attack the stalled area spreads outward until the wing is completely stalled. Initial buffet may occur at even high margins above stall speed due to sudden shock exerted upon the airflow, such as turbulence or a rapid aileron control application. If the speed margin prior to such initial buffet is large and the initiating influence is of short duration, the buffet will cease on removal of the influence. However, if the aircraft was operating very close to the initial stall buffet speed immediately prior to the aggravating influence, the stall may well persist after removal of the input. Under this condition, an appreciable decrease in angle of attack is required to restore laminar flow. The initial buffet felt by the pilot results from the start of a stall in the inboard sections of the wing, and control surfaces in these areas become less effective. With the outboard ailerons locked out, as is the case when the flaps are fully retracted, lateral control is then reduced markedly under partially stalled conditions, reducing further as the stall progresses. However, this lateral control still is sufficient to comply with the CAR 4b stall requirements.

Boeing 707 type aircraft are equipped with a stall warning device in the form of a "stick shaker" which vibrates the control column to warn the pilot of an impending stall. With flaps extended 20 degrees this warning device actuates at speeds seven or more knots higher than noticeable buffet, with little change in the margin during sideslips up to 5 degrees. However, with flaps up and zero sideslip, stick shaker actuation and initial buffet are at the same speed. With 5 degrees sideslip, there is approximately a 5 knots differential between the speeds at which the stick shaker actuates and those at which initial buffet occurs. As shown by the pilot's and copilot's airspeed indicators, the sense of this differential for a given sideslip direction is dependent on the particular airspeed indicator referred to. For nose left sideslip, the pilot's airspeed indicator reads high with the result that stick shaker actuation occurs at lower than stall buffet speed as shown by his airspeed indicator. Nevertheless, with either flaps UP or at 20 degrees the device actuates at speeds 18 to 20 knots above the CAR stall speeds. Later, these relationships will be referred to during discussion of the causal areas.

It must also be borne in mind that swept-wing airplanes are subject to a more pronounced roll-yaw coupling than straight-wing aircraft. This roll due to yaw was referred to as "dihedral effect" on straight-wing airplanes. When a swept-wing airplane with dihedral yaws, not only is the advancing wing at a higher angle of attack but it also presents a greater span to the airstream. Also, the retreating wing is less effective due to the change in airflow to a more spanwise direction. The lift differential developed by the swept wings is therefore higher and produces a greater rolling moment than would be experienced with a straight-wing airplane under similar conditions. It follows therefore that roll due to yaw input of the rudder is much more pronounced on swept-wing than on straight-wing aircraft.

Physical incapacitation of the crew: Unrecoverable body tissue vital to complete medical evaluation, resulted in a lack of conclusive positive or negative proof of physical incapacitation. However, toxicological studies were conclusive in ruling out incapacitation due to toxic gases, alcohol and drugs; and the crew's medical histories also disclosed no reason to suspect incapacitation.

Flight One's last radio transmission at 1008:09 revealed no sign of crew incapacitation. Though not conclusive, an indication that both pilots were alert and conscious at impact was the downward bending of each right rudder pedal, evidence that both were applying pressure to these controls at the time of impact. The fact that the control wheels were found calling for full right wing down is also indicative that at least one of the two pilots was still attempting recovery action at impact. It is apparent, then, that any incapacitation of the crew would have occurred only between 1008:09 and some undeterminable number of seconds before impact, an interval of less than 40 seconds.

The flight recorder indicates the first deviation from normal climbout to start at 1008:12 and that flight conditions at about 1008:30 were beyond the possibility of successful recovery action. The 21-second interval between 1008:09 and 1008:30 is, therefore, the most logical interval of time to be considered wherein crew incapacitation might have occurred.

The possibility of both pilots becoming physically incapacitated simultaneously is so remote as to be eliminated from any consideration whatsoever. The history of incidents involving crew incapacitation during flight has yielded little information to date concerning the effects of such incapacitation on the controllability of an airplane. Involuntary control forces that might be applied could vary from a negligible to a substantial force. However, incidents of operator disablement while driving motor vehicles has indicated that severe pain usually causes the driver to double over or slump forward, and that disablement due to a heart attack is not usually so severe that the driver cannot pull off the road and stop.

It is reasonable to believe that during the departure from Idlewild either pilot was in a position to immediately assume control of the airplane in the event the other was disabled. Furthermore, the second officer and the flight engineer were available to assist in the restraint of an unwanted control input by a disabled pilot.

Within the period of time in question, 1008:12 to 1008:30, there were 18 seconds in which the remaining crew members could have restored control of the airplane had there occurred an incapacitation of one of the pilots during the early part of this interval. It appears highly improbable that any control input during the period 1008:12 to 1008:30 would be of such magnitude and duration as to prevent correction by the other crew members within the time indicated. In view of the foregoing factors, the Board considers it unlikely that physical incapacitation of either the captain or first officer was a causative or contributing factor in this accident.

<u>Loss of engine power</u>: Examination of the engines disclosed no evidence of an abnormality which would affect their operation.

Witness testimony relative to cessation of smoke trails from the engines tends to support one analysis of the flight recorder data which indicated a power decrease near the apex of the climb. Observations of numerous takeoffs of jet aircraft using the same kind of engines as N 7506A indicate that visual impressions of smoke trails are unreliable due to many factors, including: viewing angles, changing flight maneuvers, relative angle and amount of sunlight, and variances in the amounts of smoke emitted by individual engines. Therefore, the true significance of witness observations cannot be properly assessed. However, there very probably was a power reduction in the late stages of American One's flight.

American Airlines energy analysis and flight tests by Boeing indicate that maximum power must have continued until approximately 1008:14 in order to produce airplane performance consistent with the flight recorder traces. The energy analysis indicates further that from 1008:14 to 1008:28 the thrust history could have varied anywhere from continuation of maximum power to a 50 percent reduction. The energy analysis does not, of course, provide any indication as to whether any possible power decrease considered was intentional or unintentional.

The flight tests indicated that a total loss of power from the left outboard engine, the most critical loss to be considered, would not have presented a critical problem in maintaining control of the airplane. The simultaneous loss of two engines on one side is believed so improbable that it does not warrant consideration.

The Board therefore concludes that a loss of engine power was not an initiating or contributory factor in this accident. However, such a conclusion does not eliminate from consideration the probability of an intentional power reduction by the crew in an effort to maintain control of the airplane.

<u>Malfunction of lateral control system</u>: No positive indication of any malfunction in the lateral control system was found during detailed examination of the wreckage. However, many critical parts were either unrecovered or melted down, with the result that there could have been a malfunction in one of these parts.

One area of possible discrepancy found during examination of the wreckage was that marks made on the aileron cable bus quadrants at impact corresponded to the right inboard aileron being about 10 degrees UP at the time, with other impact damage indicating that the control wheels were beyond the full right wing down position, the right inboard spoilers about 28 and 31 degrees UP and the outboard section of the right outboard spoiler about 40 degrees UP. Since the airspeed at impact was about 200 knots, as indicated by the flight recorder, normal operation of the lateral control system with wheels at full throw would have produced 20 degrees UP right inboard aileron, and 40 degrees UP right, inboard spoiler, without use of speed brakes to augment lateral control. This discrepancy tends to lend credence to the possibility of some malfunction in the lateral control system.

A study made by Boeing indicates that if an outboard aileron is jammed, for example by ice on the balance panels, the action of the lockout mechanism on the connecting quadrant during flap retraction from 20 degrees to zero degrees, can actuate the other aileron surfaces through the bus cables. If the left outboard aileron is more than two degrees up when jammed, the resultant left roll from the flap-driven aileron surfaces cannot be overcome by control wheel effort alone. The study indicates further that as the flaps retract through the range of 13 to 9 degrees, the load in the link rod of the binding aileron exceeds its design strength and fails, after which the remainder of the system is freed, permitting nearly normal control of the aircraft.

However, there appear to be additional possibilities in connection with a jammed aileron which could be pertinent to this accident. One of these is that deflections or failure at another point, unanticipated in the Boeing analysis and not disclosed by the ground tests on which it was based, could result in full flap retraction without failure of the link rod. This could result in at least three of the four ailerons being held in deflected positions, the amount of deflection depending on the jamming condition, aerodynamic loads, cable stretch, and other variables. The spoilers would still remain operable through the cable system from the control wheels.

Another possibility, one suggested by the control position discrepancy at impact, is that although failure of the link rod is accepted, the captain and first officer could reasonably be expected to apply lateral control effort to the limit of their physical capabilities prior to the link failure. The resulting force would load the aileron control system from the control wheels through mechanical linkage to the tabs on the inboard ailerons and to the spoiler control valves. As a result, abnormal pilot input failures at certain points in the system appear possible, such as deformation of the sleeve between the control wheel and the control column, or the terminal at the bottom of the control column. Such deformations could result in less than normal lateral control being available after the flaps are fully retracted.

If, as discussed earlier in this report, the flaps were retracted from 20 to zero degrees between times 1007:57 and 1008:09, the possible dog leg in the flight recorder heading trace as the result of gimbal error at high bank angles between times 1008:07 and 1008:17 is in general agreement with a left roll produced by binding of the left outboard aileron. If flap retraction did not cause failure of the outboard aileron link rod, or if abnormal pilot effort caused control system deformations, the left roll could continue despite maximum opposing control wheel effort. Rapid application of right rudder could then be expected. This should yaw the airplane nose right and roll it out of the bank, since the flight recorder acceleration trace indicates no probability of the wing being stalled at this time. However, the flight recorder traces do not indicate any right yaw until about time 1008:19, and this in only a small fraction of that which could be produced by rudder effort.

Using the actual speeds from the energy analysis and median values from the flight recorder normal acceleration trace, as indicated in Attachment 2, lift coefficient histories were determined. Comparison of these at time 1008:30 with the lift coefficients for heavy stall buffet as determined by Boeing tests discloses agreement only for the 50 percent thrust condition. This implies the start of a nose left sideslip at time 1008:12. The only apparent logical way in which a nose left sideslip could have started at this time in a manner necessary to satisfy the energy analysis, would be the loss of power from the Nos. 1 and 2 engines as a result of the unwanted roll. However, no reason for such power loss can be seen without assuming other independent failures. As a result, these types of lateral control failure do not appear to be a causal factor.

There has been at least one reported instance involving another 707 which experienced difficulty in rolling out of a 30-degree banked turn. Although it did not involve a lateral control malfunction, it appears pertinent for discussion at this point. After takeoff, while flying in extremely rough air, the airplane levelled off at 2,800 feet. A 120-degree turn to the right was started, flap retraction from 20 to zero degrees initiated at $V_2 \neq 30$, and power reduced. When attempting to roll out on the new heading, left aileron was applied rapidly, but the low wing failed to come up. Left rudder was then applied to bring the wing up, but the bank and turn continued and the descent rate increased. Since the airplane was not responding to normal recovery actions, the pilot applied additional power, reversed aileron control to the right with the turn, and pushed the nose down. Recovery was then effected. The crew reported that there was no evidence of a stall at any time in this sequence of events. Although the crew did not recognize stall buffet, probably due to the very high turbulence, it appears certain that the wing was at high angles of attack and in the buffet region where the flaps-up lateral control becomes less effective and can be insufficient to raise a low wing. Although lateral control deterioration probably occurred on Flight One after approximately time 1008:28 and added to the difficulty, the flight recorder traces indicate that abnormal conditions started at least 16 seconds earlier due to other reasons.

One last and important aspect of the lateral control question not yet discussed lies in the finding that the flaps had been left in the retracted position as indicated by the physical evidence after impact. Had the crew recognized their difficulty as one of lateral control it would be reasonable to expect that they would have extended the flaps in order to regain use of the outboard ailerons. Two other recovery methods were also available: asymmetric power and rudder control. Considering the recovery methods available, as applied solely to a lateral control malfunction, it does not appear likely that such a malfunction occurred.

The Board therefore believes it unlikely that a malfunction in the lateral control system was a causative factor in this accident.

Malfunction of rudder boost system: In this airplane the rudder control system incorporates a boost unit which is in operation throughout normal flight. Pilot forces applied to the rudder pedals operate control cables running to the aft end of the fuselage, and up into the vertical fin structure where they connect to the aft rudder quadrant. Cables extending upward from the rudder servo unit also connect to this quadrant. An actuating rod extending rearward from this quadrant is connected to the ratio bellcrank, which in turn is connected both to the control valve of the power unit by means of one rod; and to the rudder tab by means of another rod, a compound bellcrank in the rudder leading edge, and finally, a rod bolted to the tab horn. A Q-spring assembly supplies modifying forces to the rudder control system, mainly to provide artificial feel when the rudder is deflected more than 17 degrees. The Q-spring system connects to the primary rudder system at the compound bellcrank in the rudder leading edge. When a force is applied to the rudder pedal the resultant rotation of the ratio bellcrank actuates both the tab and the power unit control valve causing both aerodynamic forces and forces from the power unit to move the rudder in the desired direction. The power unit frame is connected directly to the rudder by a pivot bolt, and the piston rod of the actuator of this unit is connected to stationary fin structure by means of another pivot bolt. This results in displacement of the power unit case with movement of the rudder. This movement of the case provides follow-up action which centers the control valve spool when the rudder reaches the desired deflection. The rudder power system normally receives hydraulic pressure from one auxiliary hydraulic pump and during takeoff and climb American Airlines requires both auxiliary pumps to be ON to supply pressure to the rudder power system. At speeds below approximately 250 knots the system pressure is at 3,000 p.s.i. As airspeeds increase through 250 knots an airspeed switch actuates the rudder pressure control valve, reducing the hydraulic pressure to 1,000 p.s.i. The power control unit in N 7506A could be deactivated by operation of a guarded toggle rudder boost switch located in the right rear corner of the overhead panel in the cockpit. Since the accident, the carrier has relocated this switch to the approximate center of the overhead panel, making it more readily accessible to both the captain and the first officer. The forward position of the switch is ON and its aft position is OFF. The guard on the toggle switch protects it in the ON position and must be raised to actuate it to the OFF position.

Damage to various components of the rudder system gave conflicting evidence of rudder position at impact. However, study of various factors indicates that the most reliable evidence of rudder position was that indicative of 9 to 10 degrees right rudder deflection. The impact deformation to the right rudder pedal assemblies, distinctly different from that to the left rudder pedals, was indicative of both the captain and the first officer applying right rudder pressure at time of impact. The fact that the right inboard and outboard spoilers were found extended is indicative of both auxiliary and utility hydraulic pressure to time of impact. This is an indication that the hydraulic quantity was sufficient to supply hydraulic pressures for normal operation of all systems, including the rudder power system. The rudder system hydraulic pressure gage indication of 3,800 p.s.i. is above the normal operating range. This position during impact breakup could have resulted from the needle being displaced from the normal 3,000 p.s.i. range due to hydraulicking effects at impact. It is also possible that immediately prior to impact the needle could have been at zero due to previous actuation of the rudder boost switch to the OFF position, and that distortion of the gage at impact resulted in the hand moving counterclockwise from zero to 3,800. The rudder boost switch guard, the handle and the switch mechanism were missing and it was not possible to determine if this switch was ON or OFF prior to impact.

Any failure in the control valve link rod, the ratio bellcrank, or structure supporting the bellcrank; or disconnect of either the bolt attaching the rod to the bellcrank or the pivot bolt for the ratio bellcrank, would prevent normal application of both control input and follow up action to the control valve. The bolt connecting the actuating rod to the control valve was found still in place. However, as stated previously, the above mentioned parts were not recovered; therefore, no determination could be made in regard to continuity in this area.

The possibility of a disconnect of the bolt attaching the ratio bellcrank to the forward end of the valve actuating rod was given considerable attention during the investigation. It must be noted that this bolt has a countersunk head and is installed head down to avoid interference with a stiff, flexible hydraulic hose connecting to the power unit case. This hose passes directly under the bolt head with approximately 1/4-inch clearance. It is pertinent to note that the hose axis is essentially parallel to control input movement of the bolt in question. Therefore, if the securing nut, normally safetied by a cotter pin, were missing the bolt could drop down and contact the hose where it would ride back and forth with subsequent movement of the controls. It is possible that the bolt could drop out entirely free of the bellcrank and rod end, depending upon the particular aircraft installation. If the sharp-edged bolt head should come to rest on the hose, the resultant rubbing action could cause wear and fouling of the bolt with the hose, either restricting control movement and/or rupture of the hydraulic hose. A worldwide inspection campaign of 707 type aircraft required by the FAA since this accident disclosed that in all aircraft this bolt was properly installed and safetied.

A study of the results from Boeing and Project RACE tests, in conjunction with the flight recorder traces for Flight One, indicate roll effects from sideslips which could possibly result from a malfunction in the rudder boost system caused by any of the control valve disconnects mentioned above. Control valve unporting which may result from such disconnects could be sufficient to cause full hydraulic flow rate to the power cylinder, or it could be at a lesser rate due to the throttling effect of a small uncentering of the rudder control valve. These two variations will be discussed separately in the following paragraphs.

Considering first the case of a full hydraulic flow rate to the power cylinder (maximum rate hard-over) starting at about time 1008:12, the variations of indicated altitude and airspeed shown in Attachment 2 do not correspond to the high sideslip angles which can be predicted as a result of full rudder displacement. The Boeing test data show that maximum rudder deflection would probably occur in less than two seconds with maximum rate hard-over producing extremely violent airplane response. At the probable high rate of rudder deflection, any attempt to correct with normal lateral control alone would not stop the resultant roll and sideslip.

In less than four seconds the sideslip would build up to about 14 degrees which is two times the maximum sideslip reasonably deducible from the flight recorder traces and at a rate of sideslip increase about 8 times greater.

The use also of 20 degrees of speed brakes, with only one second delay in starting the recovery attempt, would produce sufficient control to stop the roll, but not sufficient to decrease the bank angle. However, approximately the same sideslip angle and sideslip rate would remain, which again is not in agreement with the flight recorder traces.

The use of lateral control and maximum asymmetric thrust, with only one second delay in applying both, would counteract the roll and sideslip, but the maximum slip angle and rate would still be much greater than indicated by the flight recorder traces. From a practical viewpoint it appears highly unreasonable to assume that the pilot would accomplish this sequence of corrective actions in the one-second time interval. Any additional delay would make the disparity even greater.

As a result it is concluded that this accident could not have been initiated by a maximum rate rudder hard-over.

Considering secondly the case of a small uncentering of the control valve, the flow rate could conceivably be throttled sufficiently to reduce rudder deflection to produce sideslip effects grossly consistent with the angles and rates indicated by the flight recorder traces from time 1008:12 to 1008:26. This would imply application of asymmetric thrust after a delay of about six seconds, as indicated by the cessation in sideslip increase from time 1008:19 to 1008:22 in the American Airlines analysis for 50 percent thrust reduction.

Such a delay in applying thrust asymmetry appears more reasonable than any lesser time delay, since first attempts to take corrective action with the control wheel are more instinctive. The increasing sideslip after 1008:22 could then result from the increasing rudder displacement caused by the unported control valve, and after about time 1008:28 with decreased lateral control effectivity as the wing angle of attack increased. With maximum aileron effort being applied and nose high stabilizer trim corresponding to that at crash impact, it appears possible that the pitch-up indicated by the acceleration trace could have resulted from an entirely unintentional small change of the elevator control force as a direct result of the high aileron control forces being applied, as the pilot concentrated with great physical effort on lateral recovery. Carrying this possible sequence still further, boost disconnect at about 1008:32 would also tend to result in the nose right sideslip indicated by the flight recorder airspeed trace due to the cessation of the rudder input with power asymmetry and opposite aileron still applied. Cutoff of the remaining two engines shortly afterward still leaves time for the reduced rpm indicated by the torsional damage to all four engines at crash impact.

The Board therefore concludes that a throttled rudder control valve malfunction could have been the initiating abnormality which resulted in the accident.

Malfunction of the rudder servo unit: The point at which the rudder servo connects to the rudder control system is mentioned in the preceding section concerning the rudder boost system. The method in which the rudder servo operates is discussed in detail in the Investigation section. However, it is pertinent to reiterate some of the salient features of operation at this point. The servo motor drives a cable pulley through a clutch which limits the force authority of the servo. Since the cables from this pulley attach into the rudder system at the aft quadrant, control forces from the servo produce exactly the same effects as equal cable loads from the rudder pedals. However, the clutch in the servo unit is so designed as to permit overpowering of the servo by application of pilot forces to the rudder pedals in the event of any probable malfunction, including false electrical signals. The American Airlines 707 checklist specifies engagement of the yaw damper, of which the rudder servo is a component, shortly after takeoff. The heading trace shown in Attachment 2 changes from a wavering line to a straight line at 1007:38, suggesting yaw damper engagement at this instant.

The investigation disclosed only one instance of unairworthiness of N 7506A at the time of the accident; that of the wire damage to the rudder servo unit. As previously indicated, the nature and protected location of the wire damage precludes the possibility of such damage having occurred at impact. The additional fact that numerous servo units were found on the assembly line with similar damage and markings is considered to be conclusive evidence that the damage to the rudder servo unit of N 7506A was initiated by assembly or maintenance operations. Following the original damage it is believed that tensile strain in the securing of the wire bundle caused wires that were damaged but not completely severed to be necked down and weakened to the extent that vibration and other disturbances over a period of time caused their final separation. The severed wire ends disclosed no evidence of melting or deposits characteristic of arcing; however, the low voltages and high impedances involved would not produce an arc of sufficient intensity to create such evidence.

Flight tests have demonstrated that separation of the wires without shorting results only in a loss of damping which is hardly perceptible to the crew in the speed range under consideration. The final wire separations therefore could have occurred during Flight One or prior thereto. However, a yaw damper hard-over occurs when there is shorting between the proper ends of the damaged rate generator leads. By reference to Attachment 2, this appears likely to have occurred at time 1008:12, where the recorded altitude and airspeed indicate the start of an abnormality. Shorting at this time could have been brought about by the inherent tendency of severed leads to untwist from a twisted bundle, as well as by the loosening of the loop around the rate generator case as a result of the wire separations which makes shifting due to vibratory loads much more likely.

It has been established that shorted rate generator leads can produce a maximum rudder deflection of 8 degrees in 8 seconds, which in turn results in a roll to 56 degrees in 5-1/2 seconds, starting from a 30-degree bank at 210 knots IAS. It is significant to note that maximum aileron recovery action during flight tests was started 1-1/2 seconds prior to the airplane reaching 56 degrees. During this 1-1/2 second interval, the roll increased 13 degrees. Test data establishing the foregoing was based on flight conditions at essentially 1 g acceleration loads. Furthermore, the tests are obviously planned maneuvers under which conditions the pilot is not confronted with the necessity of analyzing the malfunction, deciding what corrective action he will take, and experimenting to produce the desired results. In addition, when considering the operating conditions of Flight One, there were several distracting influences such as departure procedures, radio communications, flap retraction, turbulence, lack of visual horizon reference ahead due to the nose-high attitude of the aircraft, and the excellent weather conditions which would decrease frequency of reference to the attitude instruments. As a consequence it is unreasonable to assume that under the operating conditions of Flight One at this time the pilot, confronted with an unexpected roll, would start corrective action as soon and to the extent characteristic of planned flight tests.

The above is borne out by recorded instances of yaw damper malfunction or mismanagement. In all instances the crew was late in recognizing the yaw damper as being the source of the problem and were slow in initiating corrective action. In some cases, even after initiation of corrective action the dangerously steep banked attitudes increased and persisted well beyond flight test values before recovery was effected. In some instances of yaw damper mismanagement the crew never properly analyzed the difficulty and the flights were completed after application of additional lateral control such as use of speed brakes, flaps extension, etc. There are some instances wherein the crew took advantage of additional lateral control capabilities, recovered to level flight, analyzed the difficulty, and then disengaged the offending yaw damper.

Returning to time 1008:12, the beginning of the nose left yaw damper hard-over, it appears from the flight recorder traces that the airplane was in about a 30-degree bank. It follows then that an unopposed yaw damper hard-over would rapidly increase the bank angle to critical conditions. The first reaction of the crew would be to decrease the bank by gradually applying opposing control wheel force, probably with a greater delay in reaching full aileron deflection than the five seconds experienced during previously mentioned test flight conditions. The pilot may have applied opposite rudder also but with insufficient force to overpower the servo, with little or no benefit.

The flight recorder traces indicate that five to six seconds after the malfunction started, the nose-left slip effect of the malfunction suddenly became greater than the effects of opposing control forces. It can be assumed that the pilot then applied asymmetric power to arrest the roll, producing the indicated drop in altitude and the levelling of the airspeed trace at 1008:21 as a result of decreased sideslip. This power reduction also agrees with the energy analysis. In conjunction with these altitude and airspeed trace characteristics, consideration of the heading trace indicates the possibility of a time mismatch between traces, placing the cessation of heading change about one second early. Throughout this portion of the maneuver the nose-high pitch attitude of the airplane was maintained. Because of late and inadequate application of lateral control the momentarily arrested yaw then resumed and started an increasing nose left slip at time 1008:22, as indicated by the rising altitude trace.

At 1008:25 the median acceleration trace indicates the start of a rapid increase in load factor to 1.8 g's at 1008:30. During this rise the individual deflections of the acceleration trace become higher in frequency than before, indicating the start of stall buffet. The turbulent airflow over the wing during stall buffet further decreases the lateral control capability remaining after lock-out of the outboard ailerons.

It is possible that the increasing load factor progressing to stall buffet could have been brought about by a combination of some or all of the following factors: 1. the basic malfunction of the rudder control system was initially disguised by turbulence and was not quickly identified; 2. the difficulty of recognizing, in the initial stages, the abnormal attitude of the aircraft due to excellent VFR conditions tending to decrease frequency of reference to the attitude instruments; 3. an attempt to maintain the specified flight departure path as evidenced by the 2.3 nose high elevator trim found in the wreckage; 4. inability to effect immediate corrective action due to possible initial reliance on lateral control without application of the additional effect of speed brakes or flap extension; 5. an unintentional nose-high attitude while attempting lateral recovery; 6. the absence of stick shaker stall warning prior to initial stall buffet; 7 the continued operation of a malfunctioning yaw damper.

The flight recorder traces suggest that at about 1008:33 the yaw damper was disengaged, accounting for the sharp decrease in indicated airspeed characteristic of a nose right slip. This leaves sufficient time for retarding the Nos. 1 and 2 throttles, with resultant reduction of the rpm to flight idle prior to impact. It appears likely that the rudder boost was deactivated shortly prior to impact, accounting for the 9 degree right rudder indication found in examination of the wreckage.

After time 1008:30 the airplane was in heavy stall buffet, highly abnormal attitudes, and at altitudes too low for recovery to be effected before crash impact.

The Board therefore concludes that a rudder servo malfunction due to shorted wires is the most likely abnormality to have produced the accident.

Probable Cause

The Board determines that the probable cause of this accident was a rudder control system malfunction producing yaw, sideslip and roll, leading to a loss of control from which recovery action was not effective.

Recommendations

The Board presently has made three recommendations to the Administrator of the Federal Aviation Agency as a result of this accident. The first of these was that an Airworthiness Directive be issued to require a one-time inspection of the servo rate generator motors on all Eclipse-Pioneer Model PB-20D Automatic Flight Control Systems for damaged wire bundles, and that the Agency take measures as necessary to insure satisfactory quality control during manufacture and overhaul. The second was that an Airworthiness Directive be issued to require mandatory incorporation of applicable Boeing Service Bulletins pertaining to replacement of the Gladden solenoid-operated valves in the flight control and hydraulic interconnect systems due to flaking of the nickel plating tending to contaminate the hydraulic fluid. The last was that the current airworthiness requirements for automatic flight control systems in Section 4b.612(d) of the Civil Air Regulations and the related CAM material, as specifically applied to the high speed swept-wing design turbojet aircraft, be reevaluated for the purpose of establishing realistic time allowances for recognition of abnormal airplane motions, decision to take corrective action, and initiation of the proper correction in all pertinent flight regimes; and that necessary changes to the requirements be applied retroactively to turbojet aircraft equipped with automatic flight control systems. As of the date of this report the Federal Aviation Agency has taken appropriate action on the first two recommendations and has the third under study.

BY THE CIVIL AERONAUTICS BOARD:

/s/ ALAN S. BOYD
Chairman

/s/ ROBERT T. MURPHY
Vice Chairman

/s/ CHAN GURNEY
Member

/s/ G. JOSEPH MINETTI
Member

/s/ WHITNEY GILLILAND
Member

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

Investigation

The Civil Aeronautics Board was notified of this accident at approximately 1010 on March 1, 1962. Investigators were dispatched immediately to the scene to conduct an investigation in accordance with the provisions of Title 701(a)(2) of the Federal Aviation Act of 1958. A public hearing was ordered by the Board and held at the International Hotel, New York International Airport, Jamaica, New York, on March 20-23, 1962. The investigation was continued until December 1962.

Air Carrier

American Airlines, Inc., a Delaware Corporation with General Offices at 633 Third Avenue, New York City, New York, operates as an air carrier under currently effective certificates of public convenience and necessity, and an air carrier operating certificate, both issued pursuant to Federal Aviation Act of 1958, as amended. These certificates authorize the transportation by air of persons, property, and mail between various points in the United States, Mexico, and Canada, including points on Route 4. This route includes, among other cities: New York City, New York, and Los Angeles, California.

Aircrew

Captain James T. Heist, age 56, was employed by American Airlines, Inc., on May 1, 1940, and had accumulated a total of 18,300 hours flight time, of which 1,600 hours were in the Boeing 707. He held a currently effective FAA multiengine land airline transport certificate No. 20152 with numerous ratings, among which was the Boeing 707 rating. Captain Heist was issued an FAA rating in the Boeing 707 on April 1, 1960, and was line qualified on April 25, 1960. He received his last proficiency check in the Boeing 707-123B on October 13, 1961, and his last line check on September 20, 1961. Records indicate that Captain Heist satisfactorily passed an FAA first-class flight physical on October 1, 1961, without waivers.

First Officer Michael Barna, Jr., age 35, was employed by American Airlines on January 12, 1953, and had accumulated a total of 4,800 hours flight time, of which 900 hours were in the Boeing 707. He possessed a valid FAA multiengine land ATR certificate No. 273798 with Douglas DC-6 and DC-7 ratings. Mr. Barna qualified as first officer on the Boeing 707 on September 30, 1959. He received his last proficiency check in the Boeing 720B on December 19, 1961, and his last line check on February 22, 1962, in piston equipment. Records indicate that First Officer Barna satisfactorily passed an FAA first-class flight physical on December 5, 1961, without waivers.

Second Officer Robert J. Pecor, age 32, was employed by American Airlines on April 23, 1957, and had accumulated a total of 3,400 hours flight time, of which 1,716 were in the Boeing 707. He possessed a valid FAA multiengine land ATR certificate No. 1255374. Mr. Pecor received his last proficiency flight check on May 5, 1961, in a DC-6, and his last line check on August 27, 1961, in a Boeing 707-123B. Records indicate that Mr. Pecor satisfactorily passed an FAA first-class flight physical on April 14, 1961, without waivers.

Flight Engineer Robert J. Cain, age 32, was employed by American Airlines on June 16, 1952, and had accumulated a total of 7,500 hours flight time, of which 2,000 hours were in the Boeing 707. He held a valid FAA flight engineer certificate No. 1245069. Mr. Cain received his last proficiency flight check on November 28, 1961, and his last line check on December 15, 1961. Records indicate that Mr. Cain satisfactorily passed an FAA second-class flight physical on June 30, 1961, without waivers.

Stewardess Shirley Grabow, age 28, was employed by American Airlines on December 7, 1960.

Stewardess Lois Kelly, age 23, was employed by American Airlines on February 24, 1961.

Stewardess Betty Moore, age 22, was employed by American Airlines on November 17, 1959.

Stewardess Rosalind Stewart, age 20, was employed by American Airlines on September 12, 1961.

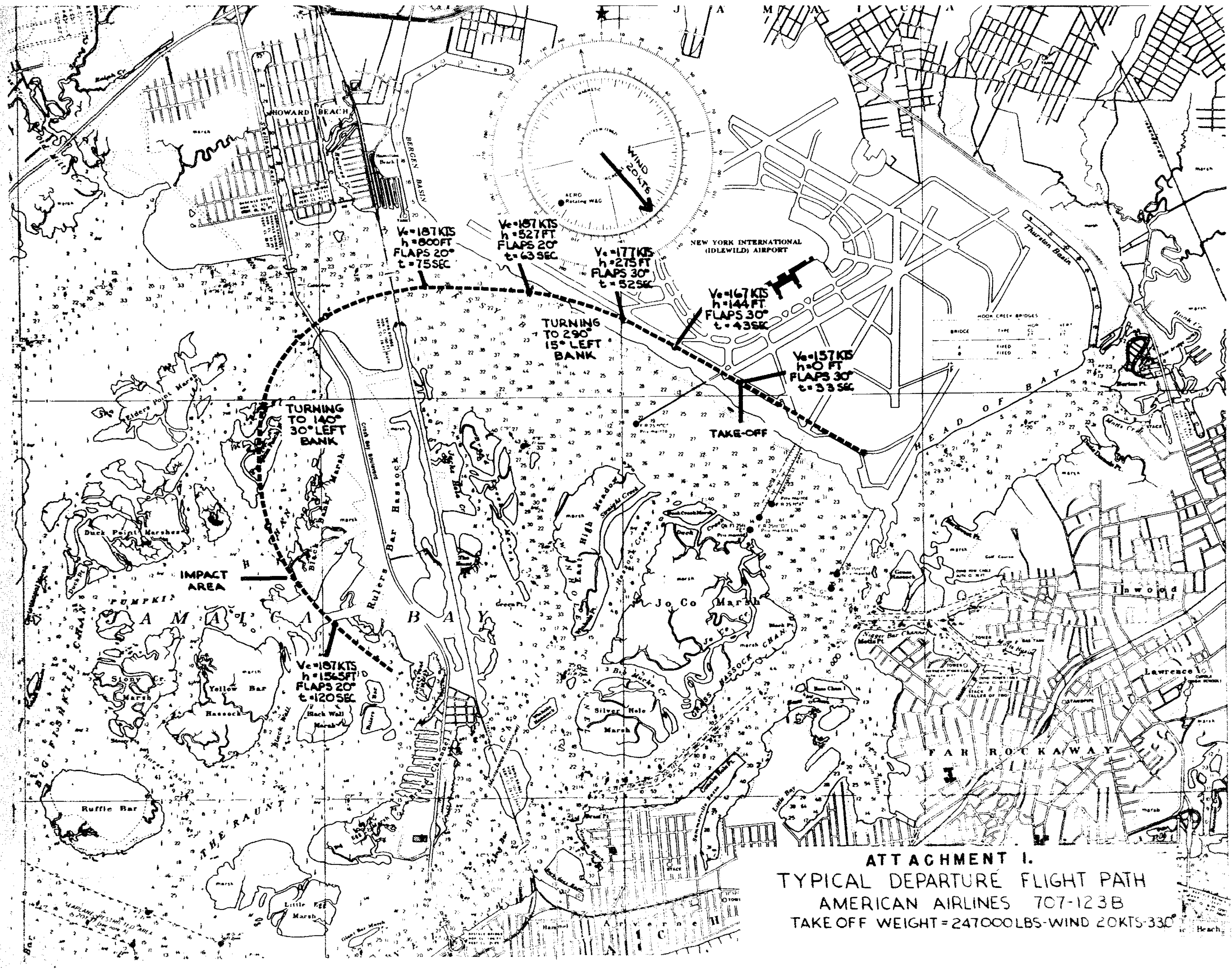
The Aircraft

A Boeing 707-123B aircraft, manufacturer's serial No. 17633, U. S. Registry N 7506A, bore a manufacturer's date of February 12, 1959, and was delivered to American Airlines, Inc., on the same date.

The last periodic inspection (No. 31) was performed January 18, 1962, when the TST (Total Ship Time) was 7,922 hours. The second Main Base Check was accomplished March 26, 1961, with a TST of 5,530 hours. Retrofit, which consisted of installing Pratt & Whitney JT3D series (Fan) engines, was completed March 3, 1961, and Fin Modification was completed on February 8, 1962. TST as of March 1, 1962, was 8,147 hours.

The aircraft was powered with four Pratt & Whitney JT3D1 engines with time since overhaul and total times as follows:

Eng. Pos.	TSO	TT
No. 1	111	4,427
No. 2	726	2,581
No. 3	1,121	5,768
No. 4	367	2,305



ATTACHMENT I.

TYPICAL DEPARTURE FLIGHT PATH

AMERICAN AIRLINES 707-123B

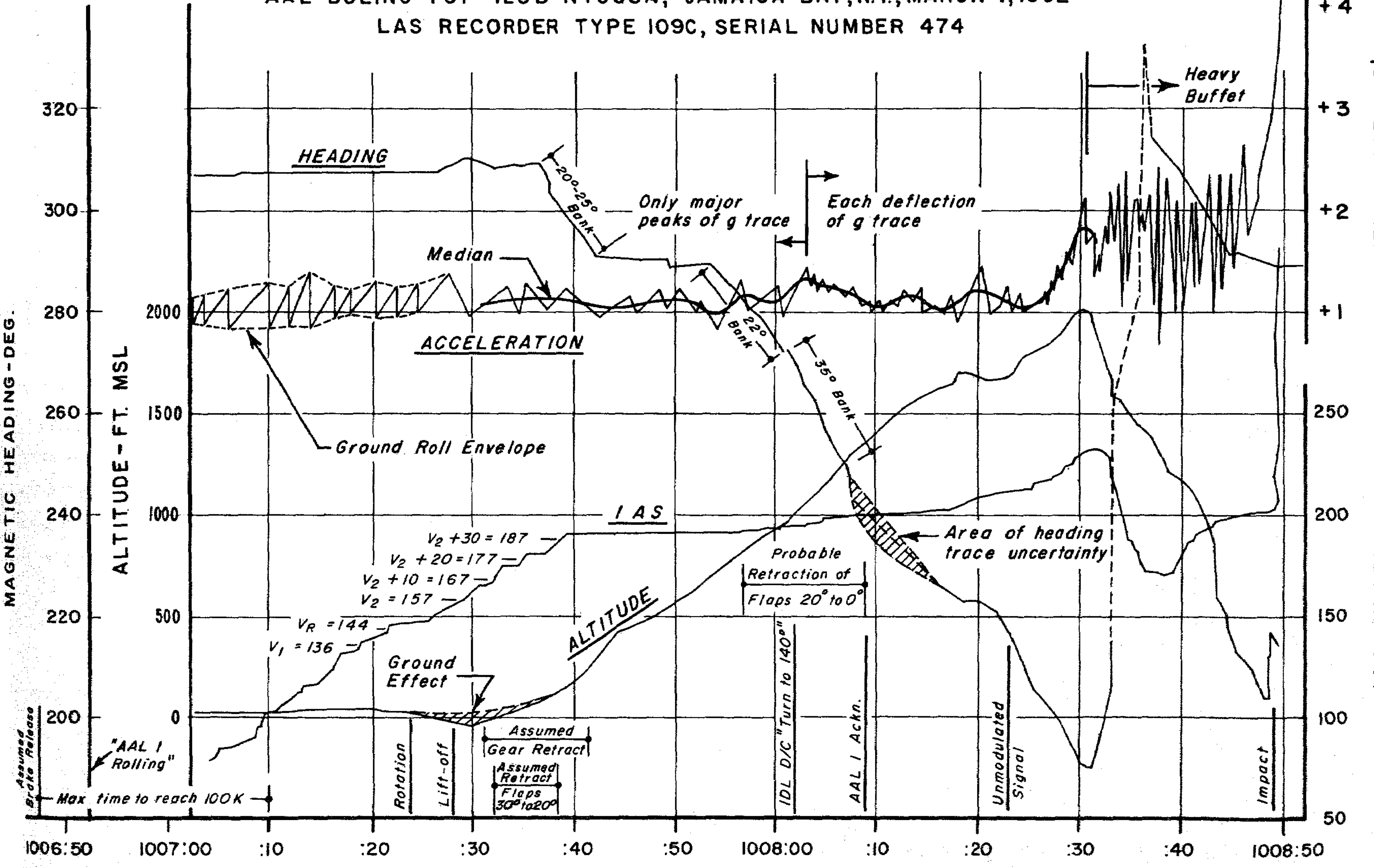
TAKE OFF WEIGHT = 247000 LBS - WIND 20KTS - 330°

ATTACHMENT 2

CIVIL AERONAUTICS BOARD
FLIGHT RECORDER DATA

AAL BOEING 707-123B N7506A, JAMAICA BAY, N.Y., MARCH 1, 1962

LAS RECORDER TYPE 109C, SERIAL NUMBER 474



GPO 838357

JFP - 12/4/62
WEL - 12/18/62