NATIONAL TRANSPORTATION SAFETY BOARD
Washington, D. C. 20591

SA-None File No. 1-004

AIRCRAFT ACCIDENT REPORT

UNITED AIR LINES, INC.
Boeing 737-222, N9005U
PHILADELPHIA INTERNATIONAL AIRPORT
Philadelphia, Pennsylvania
July 19, 1970

Adopted: DECEMBER 29, 1971

E R R A T A (Number 2)

Please make the following corrections to subject report:

Page iii - Table of Contents - under 3. Recommendation;

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Appendices	~~16~~	~~18~~
F. Various Engine Parameters	30	31
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H. Fuel Pump Shaft Test Simulation	37	38

Page 2, column 2, line 22 - change the word "titled" to "tilted"

May 8, 1972

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16. Abstract

At approximately 1907 e.d.t., on July 19, 1970, United Air Lines, Flight 611, a Boeing 737-222, N9005U, crashed shortly after taking off from the Philadelphia International Airport, Philadelphia, Pennsylvania. There were no fatalities. Among 55 passengers and six crewmembers, 17 passengers were injured, one seriously, and one crewmember received minor injuries.

Examination of the left (No. 1) engine revealed that a first-stage turbine blade failed. Disassembly of the right (No. 2) engine and functional testing of its components revealed that the engine was operating in the air, during the thrust reversing cycle, and until the engine impacted the ground.

The National Transportation Safety Board determines that the probable cause of this accident was the termination of the takeoff, after the No. 1 engine failed, at a speed above V2 at a height of approximately 50 feet, with insufficient runway remaining to effect a safe landing. The captain's decision and his action to terminate the takeoff were based on the erroneous judgment that both engines had failed.

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SYNOPSIS

At approximately 1907 e.d.t., on July 19, 1970, United Air Lines, Flight 611, a Boeing 737-222, N9005U crashed shortly after taking off from the Philadelphia International Airport, Philadelphia, Pennsylvania. There were no fatalities. Among 55 passengers and six crewmembers, 17 passengers were injured, one seriously, and one crewmember received minor injuries.

Flight 611 made its takeoff from Runway 9. The takeoff roll and lift-off were reported normal in every respect. At the point in the climb where the landing gear is normally retracted, the flightcrew heard a loud explosion, following which the aircraft veered right. The captain stated, "I advanced power on both engines without any response and then made the decision to attempt to land on the remaining runway." The aircraft touched down hard on the departure runway and continued off the end and across the blast pad. The aircraft came to rest, 1,634 feet past the end of the runway, on a magnetic heading of 70°.

No. 1 engine failed in flight. Disassembly of the engine revealed a contained failure within the turbine area. A first-stage turbine blade failed in flight which caused cessation of engine rotation prior to ground contact.

Disassembly of the right (No. 2) engine and functional testing of its components revealed that it was in an operable condition at the time of the accident. All the evidence developed during the investigation demonstrated that the engine was operating in the air, during the thrust reversing cycle, and until the engine impacted the ground.

The National Transportation Safety Board determines that the probable cause of this accident was the termination of the takeoff, after the No. 1 engine failed, at a speed above V_2 at a height of approximately 50 feet, with insufficient runway remaining to effect a safe landing. The captain's decision and his action to terminate the takeoff were based on the erroneous judgment that both engines had failed.

1. INVESTIGATION

1.1 History of the Flight

The aircraft was a Boeing 737-222, N9005U, powered by two Pratt & Whitney JT8D-7 turbofan engines. The aircraft had operated without incident with the same flightcrew from Washington, D.C., to Philadelphia, Pennsylvania, with en route stops at Rochester and Buffalo, New York. An en route maintenance check was completed prior to takeoff from Philadelphia.

At 1850 e.d.t.[1] Flight 611, a scheduled passenger flight from Philadelphia, Pennsylvania, to Rochester, New York, was taxied from the United Air Lines' gate for takeoff on Runway 9.

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

The first officer, who was controlling the aircraft from the right pilot's seat, initiated the takeoff roll at 1905.

The takeoff proceeded normally until the point in the climb where the landing gear is normally retracted, when a loud explosion was heard and the aircraft veered to the right. The captain immediately joined the first officer on the controls. According to the first officer, they "both worked together to control the aircraft until it came to a complete stop."

During an interview with the captain, he stated that he thought the noise came from the right side and he observed the right engine instruments begin to "spool down." He further stated that he advanced power on both engines without any response and then made the decision to attempt a landing on the remaining runway.

The first officer stated, "shortly after the loud noise was heard, it seemed to me that the left engine began to 'spool down' and the aileron and elevator controls felt as if we were in manual reversion." The first officer further stated, "The aircraft became unable to sustain flight at this time."

According to the second officer, the captain "moved both throttles forward at this time but there was no response and the aircraft began settling." Also, according to the second officer, the captain "made an instantaneous decision to set the aircraft back on the runway." He further stated, "the touchdown was smooth approximately 1,000 feet from the end..." and the captain "attempted to place the throttles in reverse to no avail."

The three stewardesses in the cabin heard the loud noise and all agreed that it came from the left engine. One of the stewardesses observed sparks coming from the left engine.

The aircraft touched down on Runway 9 at a point 1,075 feet short of the far or upwind end of the runway. After the touchdown, the aircraft continued off the end of the runway and across the blast pad. It then crossed a field and passed through a 6-foot-high aluminum chain link fence into an area covered with high grass, weeds, and brush. The aircraft came to rest beside a pond, 1,634 feet past the end of the runway, on a magnetic heading of 70°. (See Appendix C.)

Pre-Impact Observations

There were nine eyewitnesses to the accident. The average of these witnesses maximum estimates indicates that the aircraft reached an altitude of 122 feet above the ground. Three of the witnesses heard a banging noise after lift-off. One saw fire and one saw smoke from the left engine. Two saw exhaust smoke coming from the right engine only. Five witnesses described their observation of the wings rocking in flight.

Statements were received from 33 passengers. Eleven passengers heard a loud bang after lift-off. Two saw fire coming from the left engine. One passenger stated that they seemed to lose power on the left engine. Another passenger stated that he saw the right engine slip out of its mount and hang there. Eleven passengers stated that the wings titled or wobbled in the air. Six passengers stated that the lights in the aircraft blinked off and on.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	0	0	0
Minor	1	17	0
Serious	0	1	0
None	5	37	

1.3 Damage to Aircraft

The aircraft sustained major damage.

1.4 Other Damage

An aluminum chain link fence was damaged. Three runway threshold lights and the instrument landing system localizer were damaged.

1.5 Crew Information

The flightcrew of Flight 611 were properly certificated and qualified for the flight. (For detailed crew information, see Appendix B.)

1.6 Aircraft Information

The aircraft was a Boeing 737-222, registration No. N9005U, serial No. 19043. It had accumulated a total flying time of 3,956:04 hours, and 447:53 hours since the last line maintenance inspection was performed.

Two Pratt & Whitney JT8D-7 turbo-fan engines were installed on the aircraft. The left engine, serial No. 655899, and the right engine, serial No. 656069, had operating times of 2,942:00 and 1,846:00 hours, respectively.

No uncorrected maintenance items which were related to the airworthiness of the aircraft were recorded in the aircraft flight logs and maintenance records.

United's maintenance records indicated that the fuel control unit had 713 hours operational time since overhaul, 642 hours of which was on the right engine of this aircraft. Total time on the unit was 1,846 hours.

The maximum certificated gross takeoff weight for this aircraft is 100,000 pounds, and the center of gravity (c.g.) limits are 30.3 percent maximum aft and 8.3 percent forward, mean aerodynamic cord (MAC). The takeoff gross weight for this flight was 90,040 pounds. The center of gravity was computed to have been 24 percent MAC.

The aircraft had been fueled with 7,100 pounds of jet A-1 fuel at Philadelphia. The takeoff fuel load was computed to have been 17,100 pounds.

1.7 Meteorological Information

The surface weather observation at Philadelphia International Airport at 1909 was: High thin scattered clouds at 25,000 feet, visibility 10 miles, temperature 84°F., dew point 69°F., winds 150° at 1[illegible] knots, altimeter setting 29.98. Weather conditions are not considered to have been a factor in this accident.

1.8 Aids to Navigation

Not applicable.

1.9 Communications

Communications were normal.

1.10 Aerodrome and Ground Facilities

Runway 9 at the Philadelphia International Airport is 9,491 feet long and 150 feet wide. A concrete blast pad 150 feet long is located at the takeoff end of Runway 9 (approach end of Runway 27). Runway construction consists of a concrete base surfaced with a bituminous all-weather material. The airport elevation is 14 feet above mean sea level. At the touchdown point of the aircraft on the runway, there was a heavy concentration of black rubber deposits.

1.11 Flight Recorders

N9005U was equipped with both a flight data recorder and a cockpit voice recorder. The flight data recorder was a Fairchild model F 5425-601, serial No. 1934. The cockpit voice recorder was a United Control model V-557, serial No. 2018.

Detailed examination of the flight data recorder foil by the Safety Board revealed the following: The indicated airspeed trace was abnormal which precluded determination of accurate airspeed information. The time trace showed 52 seconds from the start of the takeoff roll until the aircraft came to rest. The altitude trace showed the airborne time to be approximately 13 to 14 seconds. The maximum altitude shown on the altitude trace was 50 feet. The heading trace showed a deviation to the right of 7° immediately after lift off. This change of heading occurred over a time period of 5 seconds. Then the aircraft deviated to the left 3° over a time period of 3 seconds, followed by a sharp deviation to the right of 9° in 2 seconds, 7° of which was in the last second. The trace then returned to the runway heading, which required a total change of heading of 13°, in 2 seconds, at which point the altitude trace indicated 14 feet. On the off-runway roll, the vertical acceleration trace showed a g load factor from minus 2 g to plus 6 g. (See Appendix D.)

The cockpit voice recorder was recovered intact and the tape was examined by the Safety Board. A transcript of the pertinent portion of the tape is contained in Appendix E.

1.12 Wreckage

The aircraft's right main landing gear touched down approximately 1,075 feet from the end of the runway and approximately 46 feet to the right of the centerline. The nose landing gear touched down approximately 776 feet from the end of the runway and 44 feet to the right of the centerline. The blackened condition of the runway prevented the determination of the touchdown point of the left main landing gear. However, the track of the left main landing gear was visible after it came out of the blackered area. It was also evident that the nose gear wheels tracked toward the right main gearwheel tracks throughout the landing and overrun. The aircraft came to rest 1,634 feet beyond the end of the runway beside a pond on a heading of 070° magnetic. During the overrun, the aircraft had passed over three runway threshold lights, through a chain link fence topped with barbed wire, and then impacted two mounds of earth and rubble before it came to rest. (See Appendix C.)

The right wing and right engine were over the surface of the pond. The right engine thrust reverser buckets were fully deployed. The outboard thrust reverser bucket had been punctured by, and contained, a piece of aluminum fence-post.

Pieces of chain link fence and barbed wire were around both main landing gears and in the inlet cowls of both engines. One section of fence-post had penetrated the radome and was lodged in the nose gear well. Barbed wire had entangled around the left engine pylon and extended over the left wing.

The only visual damage on the empennage was two small dents. All the control surfaces were attached and intact. The position of the horizontal stabilizers corresponded to the setting on the cockpit controls.

The left wing sustained major structural damage. The forward trunnion attach fitting of the left landing gear and had been fractured resulting in fuel leakage. The left aileron and tab were not damaged. All leading edge slats and flaps were attached and fully extended. The ground spoilers were intact and fully extended. The inboard and outboard trailing edge flaps were damaged. Measurements of the flap jackscrews indicated a trailing edge setting of 5° flap extension.

There was no major structural damage to the right wing. All the leading edge flaps and slats were attached and fully extended as were the ground spoilers. The flight spoilers were intact and in varying degrees of extension. The inboard trailing edge flap was damaged. The outboard trailing edge flap was intact. The flap jackscrews were extended the same amount as those on the left wing.

The lower fuselage structure was substantially damaged. The right main landing gear was separated from the aircraft. The left main landing gear was attached to the aircraft by the outboard walking beam attachment. The nose landing gear had folded aft and was lodged in the electronic and electrical compartment of the fuselage.

The right air conditioning pack, hydraulic system components in the right wheelwell area, and some of the navigational units located in the electronic and electrical compartment sustained impact damage. Examination and functional testing of the removed systems components revealed no discrepancy that could have contributed to the accident.

The cockpit area was intact. All instruments and control panels were in place. The pertinent cockpit documentation was as follows: Both throttles were fully retarded; left engine reverser was in the idle detent and stuck; right engine reverser lever was near the full reverser position; the flap selector was at 5°; anti-skid was "OFF"; the landing gear lever was down; and the speed brake was in the flight detent.

No. 1 Engine

The engine was separated from the pylon and was lodged beneath the left wing. The engine was deflected in an outboard direction of approximately 45° and had rotated approximately 90°, such that the bottom of the engine was facing towards the left wingtip.

The thrust reverser was attached and in the full stowed position. The thrust reverser was functionally tested and was found to operate normally. The lower portion of the inlet cowl was heavily crushed rearward toward the engine inlet case. The cowlings were all generally intact and displayed no evidence of ground or in-flight fire damage. The external engine cases displayed no evidence of internal to external punctures.

The front and rear covers of the accessory drive gear case were broken open in the area of the fuel pump fuel control mount pad and the hydraulic pump mount pad. The right side cross shaft mount flange was broken off at the accessory drive gear case housing. This damage precluded determination of power lever position at impact.

All engine accessories were intact and attached except for a separated fuel filter housing assembly. This assembly was subsequently recovered. A section of the filter housing was found broken. The filter was intact. No obvious metal particles were present within the filter.

Disassembly and examination of the engine revealed that the engine was not rotating on impact. With the exception of the turbine area of the engine, all the damage was a result of impact.

The turbine front case had a hole, 1-1/2 inches by 2-3/4 inches, at approximately the 12:30 o'clock position, and the rear flange was also ruptured at this location. The ruptured flange remained attached to the case. The case was uniformly circumferentially bulged outward from rotational contact of first-stage turbine blades at the rear flange of the turbine front case.

Nine first-stage turbine blades (P/N 564901)[2] were broken off at the blade inner platform. The remaining blades were broken 1 to 2 inches above the inner platform. The first-stage outer air seal had a 12-inch piece broken out. The inside diameter of the air seal was heavily rubbed. Eigh[illegible] percent of the antirotation lugs were broken or worn off at the leading edge. Three first-stage turbine nozzle guide vanes were broken into pieces. Two vanes were at approximately the 6 o'clock position and the third was at the 11 o'clock position. At the 6 o'clock position adjacent to the two missing vanes, one additional vane was broken into two sections. The remainder of the turbine nozzle guide vanes were heavily damaged, but in their proper position. Vane damage occurred primarily at the trailing edge.

The second-stage turbine blades were broken off approximately 1 to 2 inches above the inner platform. Five second-stage turbine nozzle guide vanes were broken off 1/2 inch to 2-1/2 inches out from the outer shroud at random locations. The remainder of the vanes were heavily damaged, but remained in their proper positions.

All third-stage turbine blades were broken off in a fairly uniform pattern approximately 3-1/2 inches above the blade inner platform. The third-stage turbine nozzle guide vanes were intact, but heavily damaged from foreign debris.

The fourth-stage turbine blades were all broken off from 1 inch to 6 inches above the blade inner platform. The fourth-stage turbine nozzle guide vanes were basically all intact, but the individual vanes were heavily damaged from passage of upstream debris.

The fourth-stage turbine air sealing ring was rotationally forced into the exhaust case. It was necessary to drill out the 17 attaching screws which retain the fourth-stage turbine air sealing ring to the nozzle case so that the ring could be removed with the exhaust case.

[2] Unless otherwise specified, the part numbers (P/N) cited herewith are engine manufacturer's designation.

No. 2 Engine

The engine remained attached to the right wing and was partially submerged in a pond. The engine inlet was completely packed with mud and was lodged against a small tree.

The thrust reverser was in the "full reverse" position. The upper inboard and outboard reverser links were twisted slightly. The reverser was functionally tested and was found to operate properly, except for the effects of impact damage. Examination of the turbine from the end of the exhaust duct disclosed no visible damage to the fourth-stage blades. The visible portion of the cowl panels displayed no evidence of in-flight or ground fire.

Mud, metal, and two sections of aluminum fence rail were removed from the inlet cowl. One large section of fence rail, bent into a "U" shape, was found jammed into the engine inlet at the 2 o'clock position, as viewed from the front of the engine. The end of the rail had the appearance of being milled or ground down from contacting the rotating first-stage fan blades. The other end of the rail did not contact the fan blades. One 3-inch section of chain link fence pipe was found between the first-stage blades and stators. This pipe was formed over the stator.

The leading edge of the intake cowl evidenced minor impact damage from the 3 through the 9 o'clock position. The bottom of the inlet cowl and the cowl panels were scraped and buckled rearward for approximately 3 to 4 feet. This area also exhibited several large tears. Mud was forced into the inlet duct "blow in" doors. The cowl was also totally packed with mud up to and including the accessory drive gearbox front cover. The mud totally engulfed the fuel control/fuel pump module.

There was some damage evident on the leading and trailing edges of the inlet guide vanes. The first and second-stage compressor blades evidenced heavy foreign object damage primarily from the midspan outboard. This damage consisted mainly of broken and gouged blade tip sections, mostly at the leading edge of the blades. Some trailing edge blade damage was also present. Several of these blades were also bent rearward with respect to engine rotation. Significantly less damage was prevalent to these blades inboard of the midspan.

The N_1 compressor and its turbine rotated freely.

The inlet of the engine and the debris removed from the inlet were closely examined for evidence of a bird strike. No such evidence was found.

All engine accessories, including the power lever control linkage, were intact. Continuity and full throttle movement were established when the linkage was activated from the cockpit.

The engine was disassembled and the following was noted:

a. A broken fuel pump drive shaft (TRW P/N 208235)[3], a mispositioned fuel pump quick disconnect coupling, and other damage attributable to impact and/or foreign object ingestion.

b. The engine front and rear compressor/turbine (N_1 and N_2) assemblies rotated freely after impacted mud was removed from the inlet.

c. Vegetation was found throughout the secondary gas path, in the primary gas path as far back as the diffuser case, and in the sixth and eighth-stage air bleed system as well as the eighth-stage aircraft bleed system. Mud coating was found adhering completely through the engine from the inlet to the exhaust.

d. All of the fuel nozzles exhibited varying degrees of dirt contamination on the external and the internal surfaces of the nozzle assembly. The fuel exit holes were

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[3] Fuel pump manufacturer part number.

clear and were without an accumulation of dirt to restrict the flow.

e. There was no visual evidence of overheat in the combustion area, turbine blades, or on the nozzle guide vanes.

f. Thirteenth-stage air conditioning modulating valve was closed.

g. Particles of nonmagnetic material were found between the sixth and seventh-stages of the high (N_2) compressor and the combustion chamber support assembly.

h. Aluminum metal splatter was evident on the first-stage turbine nozzle guide vanes.

i. The blades of the fan assembly, the low compressor (N_1), and the high compressor (N_2) all evidenced random foreign object damage.

j. The reed valve installed within the fuel drain valve and the fuel drain manifold was found to be free of contamination. The reed valve was in the open position, indicating normal operation.

k. Both ignitor plugs were covered with dirt and there was no evidence of metal splatter. The cooling air passages were plugged with dirt.

l. The EGT[4] probes were covered with dirt and aluminum metal splatter. The metal splatter had oxidized showing that the melting temperature range of the aluminum (1160° 1250°F.) had been exceeded.

1.13 Fire

There was no evidence of fire on any part of the aircraft or on the ground in the impact area.

[4]Exhaust Gas Temperature.

1.14 Survival Aspects

This was a survivable accident. Seventeen passengers received minor injuries and one was seriously injured. The seriously injured passenger received a fractured ankle when her foot struck the ground at the bottom of the evacuation chute.

No warning was given by the cockpit crew of the emergency to the cabin crew or the passengers. The three stewardesses were aware from the excursion of the aircraft that there was a good possibility that the aircraft would crash. One stewardess at each extreme end of the cabin called for the passengers to grab their ankles.

Evacuation

After the aircraft came to a stop, evacuation was initiated immediately by the cockpit crew and stewardesses. The first officer and the second officer immediately entered the passenger cabin and opened the left main door and the right forward buffet exit door and deployed the slides. The first officer had difficulty in opening the buffet door. He was unable to move the door handle to the full open position due to a partially open galley drawer. He quickly recognized the problem, kicked the drawer to its closed position, and opened the door with no further difficulty. Individual passengers opened the right and left single overwing exits with no difficulty. The two stewardesses seated on the aft jump seats opened the left aft door with some difficulty. One of the stewardesses described the difficulty as, "the plane was slightly tilted to the right which made it a little harder to open the door." (The aircraft at rest had a 5° right roll altitude.) The slide for this door deflated sometime during the evacuation.

After all passengers were clear of the cabin, the first officer discovered four passengers standing on the right wing which extended over the pond. He helped these passengers back in the cabin and they exited by the left forward and aft exits. The evacuation was orderly and the elapsed time was approximately 45 seconds.

There was no evidence of seat or seatbelt failures.

The aft left slide, which deflated during the evacuation, was found to have about an 8-inch tear on its bottom surface. The aft right slide was found to have about a 15 by 8-inch "L" shaped tear on its bottom surface. Examination of the slides disclosed that the tears were caused by foreign objects.

1.15 Tests and Research

In view of the evidence found during the investigation concerning the fracture of the No. 2 engine fuel pump drive shaft P/N 208235, a visual metallurgical examination of this part was conducted by the Safety Board. This examination revealed that there was a complete transverse fracture in the shaft in a 3/8-inch diameter reduced section about 1-7/8 inches from the larger (7/8-inch major diameter) splined end. The appearance of the mating fracture faces, as viewed through a binocular microscope, indicated that the fracture resulted from the propagation of fatigue cracks from the surface of the reduced section of the shaft.

As a result of this examination, the Safety Board requested Pratt & Whitney Aircraft to study this fracture to develop the following technical data:

(1) An analysis to determine the type of fracture and type of loading that induced the failure.

(2) Also, if the fracture was caused by fatigue, the type and level of loading involved in the initiation and propagation of the fatigue crack.

(3) The type of loading that caused the final separation of the shaft and whether low or high cycle fatigue was involved.

(4) An estimation of the number of load cycles and time required to fail the drive shaft.

(5) Analysis of the wear pattern on the fuel pump drive shaft splines, and examine the quick disconnect coupling assembly in terms of possible misalignment and any other pertinent factors.

The Safety Board also requested experimental test data and/or analytical evaluation that would demonstrate the amount of time required to fracture a fuel pump drive shaft.

Pratt & Whitney reported that a laboratory examination of the broken fuel pump drive shaft indicated that the fracture was caused by fatigue, which resulted from a rotating beam-type loading.

Pratt & Whitney also reported that the loading induced in the drive shaft was due to an extreme misalignment between the fuel pump and the spur gearbox drive gear shaft.

In order to study the effects of this misalignment, Pratt & Whitney performed a simplified rig test wherein test specimens were subjected to a known stress range and the number of rotational cycles to failure were counted.

The stress ranges and number of cycles required to fail these specimens are listed below:

Stress Range (p.s.i.)	Rotational Cycles
120,000-125,000	12,000
140,000-145,000	5,300
160,000-165,000	3,500

A technical background literature review[5] by Pratt and Whitney substantiated the validity of the above tests. The literature indicates that a bending stress of high magnitude results in the ratchet marks, which indicate multiple fatigue origins, their quantity increasing with the degree of stress concentration and stress level. Also, the number of cycles to induce failure is primarily dependent on the level of this bending stress.

For this investigation, Pratt & Whitney assumed that the number of cycles to induce the shaft failure was low. Inspection of the fractured

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[5] "Characteristics of Fatigue Fractures" American Society of Metals Journal abstracted from "How Components Fail," by Donald J. Wulpi, copyright 1966.

fuel pump drive shaft revealed an extremely high number of fatigue origins--at least 50. Inspection of the test specimen that was subjected to a stress range of 160,000 to 165,000 p.s.i. revealed a minimum of 40 fatigue origins.

An extrapolation of the above cited rig test and fatigue origin data, coupled with a comparison of the similarity between the fracture surface of the broken fuel pump drive shaft (Appendix G) and the test specimen that was subjected to a stress range of 160,000 to 165,000 p.s.i. (Appendix H), led to the estimation by Pratt & Whitney that a stress in excess of 180,000 p.s.i. was required to fail the broken fuel pump drive shaft.

The manufacturer indicated that it was not possible to determine the specific number of cycles from the initiation of the fatigue crack to the final failure since the stress level or time exposure to the stress necessary to initiate the crack is not known. However, based on the aforecited rig test, there is an indication that the failure could have occurred in fewer than 2,000 cycles from initiation of a crack. This is evidenced by the similarity between the rig failed shaft and engine failed shaft in that the rig failure had at least 40 fatigue origins. These origins were very similar in appearance.

Pratt & Whitney specifications for the shaft required that it be made of AMS[6] 6415, heat treated to a hardness range of 44 to 48 Rockwell C. Laboratory examination showed that the shaft material complied with these specifications. The actual hardness was 44 to 45 Rockwell C, which indicated that the ultimate tensile strength of the shaft material was approximately 210,000 p.s.i

The fuel pump drive shaft splines bore evidence of normal as well as abnormal wear. The normal wear was obvious, due to its pattern and axial location, on the driven side of the spline. The abnormal wear was particularly evidenced by the wear on the nondriven side. This degree of abnormal wear further substantiated a high misalignment and resultant high bending stress. This was a result of a severely misaligned shaft.

Research of Pratt & Whitney records showed that there has not been a failure of the fuel pump drive shaft at the shear section in the 30 million plus hours of operation of the JT8D engine. However, there have been two failures in the retaining ring groove. These failures occurred at the base of the retaining collar groove and were the result of sharp corner stress concentration. No fuel pump drive shafts have been rejected from service as a result of cracking over the past 2 years.

Examination of the fuel pump rear coupling assembly, P/N 473860, and quick disconnect nut assembly, P/N 522702, determined that the wear on the serrations of both was typical of wear which would be encountered during normal service use. The coupling wear pattern also indicated that the fuel pump shaft had not been subjected to a significant degree of preexisting misalignment. The fuel pump drive shaft spline wear could not be directly correlated to the wear pattern on the coupling assembly serrations. The separation found at the interface of the fuel pump quick disconnect and the accessory drive gearbox might have accounted for the second wear pattern on the driven side of the fuel pump drive shaft spline. However, this separation would not have created the high bending loads necessary to produce the type of fracture found in the failed shaft.

Disassembly of the spur gearbox drive gear shaft and associated bearing at Pratt & Whitney revealed the following:

> Location of O-ring rub indicated that the coupling was operating in an improperly aligned position. The roller bearing on the gearbox drive shaft nearest the fuel pump had pieces of the backside of the inner rail broken out. The only plausible explanation is severe distortion of the gearbox housing and the resultant thrust load from the outer race having been transferred through the rollers to the inner race.

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[6]Aerospace material specification.

Four first-stage turbine nozzle guide vanes removed from the No. 2 engine, foreign metal from between the sixth and seventh stage of the compressor and a sample of fence from the Philadelphia Airport, were sent to the Federal Bureau of Investigation for examination.

Examination disclosed that there were numerous small bright-appearing areas on the surface of the turbine nozzle guide vanes. Minute foreign deposits of metal were removed from some of the bright areas on the vanes. An instrument analysis determined that these foreign metal particles consisted essentially of aluminum. These particles were too contaminated and limited in quantity for further compositional analysis.

Instrumental analysis of the metal fragment and the fencepost sample revealed them to be the same in composition.

Therefore, the analyzed metal deposits on the turbine nozzle guide vanes and the fragments could have originated from the aluminum fencepost.

A first-stage turbine nozzle guide vane was sectioned by Pratt & Whitney in order to determine if any aluminum particles from the partially ingested fencepost could have entered the turbine air cooling chambers. By virtue of this examination, it was determined that aluminum particles entered the air cooling chambers. These particles were observed adhering to the inside core of the vane. A particle was also found lodged in a cooling hole of the inlet guide vane spigot.

Research of the operation of the thrust reverser disclosed that the engine running switch installed in the isolation valve control circuit would open at an engine oil pressure of 35 ± 2 p.s.i. When the engine oil pressure was below that value, the thrust reverser would not deploy.

Pratt & Whitney Aircraft was requested to furnish the following technical data:

> Given a sea level day, an ambient temperature of 84°F., a 140 knot true airspeed takeoff power, at this point in the takeoff assuming a rapid fuel cut off, would you provide analytical and/or graphical data that demonstrates for the above conditions, the rate of change of the following parameters as a function of elapsed time from takeoff conditions to both idle and windmilling condition.
>
> 1. Percent of takeoff thrust decay.
> 2. Engine pressure ratio decay.
> 3. Decay in percentage of takeoff r.p.m., both N_1 and N_2.
> 4. Decay in engine oil pressure.
> 5. Decay of turbine inlet temperature.

Graphs were prepared by Pratt & Whitney for the above parameters. These graphs are shown as Appendix F and depict the results of a single engine run on a test engine with all the parameters recorded on this run. Fuel shutoff as depicted on the graph was accomplished by shutting the fuel off to the engine at the source, with the thrust lever in the full power position.

The graph of main oil pressure as a function of elapsed time in seconds disclosed that the oil pressure dropped from 45 p.s.i. to below 37 p.s.i. in 4 seconds and 35 p.s.i. in 5 seconds during fuel shutoff. In a snap deceleration to idle, the oil pressure dropped off rapidly to 40 p.s.i. in 2 seconds, then gradually to 37 p.s.i. in 16 seconds and then held 37 p.s.i. constantly. (See Appendix F.)

The Boeing Company was requested to determine the approximate engine power level at which the 13th-stage aircraft bleed duct modulating valves would open, i.e., thrust, EPR and N_2 rotating speed.

The Boeing Company stated the 13th-stage aircraft bleed duct modulating valves would begin to open at the following conditions (on decreasing power):

V_{TAS} Knots	N_2 Percent r.p.m.	E.P.R.	Net Thrust-lbs	Percent of Takeoff Net Thrust
0	85.4	1.392	6,660	49.0
102	84.8	1.346	5,170	42.6

The graph showing percent of takeoff thrust decay at time of fuel shutoff disclosed that thrust decayed from 100 percent to 25 percent in 1 second and continued to decline to 11 percent in 2 seconds. In a snap deceleration to idle, the thrust decayed from 100 percent to 28 percent in 2 seconds (See Appendix F)

On the engine pressure ratio graph, a drop from 1.95 EPR to 1.15 occurred in 1 second at fuel shutoff. (See Appendix F)

The graph showing percentage of takeoff N_2 rotor speed showed that an initial decay from 100 percent to 72 percent occurred in 2 seconds, and continued to decay to 58 percent in 4 seconds and to 40 percent in 8 seconds.

It should be noted that this curve was not corrected for horsepower extraction required to operate the aircraft electrical and hydraulic systems; thus, this curve reflects a maximum time required for the engine to decelerate. Since the No. 1 engine failed and the resultant electrical load was transferred to the No. 2 engine, the total horsepower extraction would have been greater than normal and tended to slow the engine at a rate greater than shown on this curve.

Boeing Aircraft Company also stated that laboratory tests have shown that generator transfer occurs at the following values of N_2 during a deceleration of 180 r.p.m./sec. on N_2 speed.

Electrical Load	N_2 Speed
90 amps (30 KVA)	44.5%
45 amps (15 KVA)	43.1%
No Load	42.3%

The Boeing Company indicated that the aircraft fuel boost pump would not have the capability to supply the engine fuel in the event that the fuel pump drive shaft was lost. The high pressure stage of the engine-driven pump is a gear type pump and no fuel will pass through when the pump is not rotating. There are no alternate means for the engine to receive fuel.

2. ANALYSIS AND CONCLUSIONS

2.1 Analysis

There was no evidence of structural failure, malfunction, or abnormality of the airframe, control systems, powerplants, and other components other than the failure of the No. 1 engine. The failure of this engine occurred at a height of approximately 50 feet and above V_2 speed. Failure of this engine would not have caused the accident, as the aircraft at the time of the engine failure was capable of continuing to climb on one engine and to make a subsequent safe landing.

The only causal factors involved in the accident were those directly associated with the powerplants and the operational procedures used by the crew.

In assessing the powerplant factors involved, it was confirmed that the No. 1 engine was not rotating at ground contact. The cause of the inflight failure of the engine was a M082 heatcode first-stage turbine blade failure. This blade failure is typical of other M082 heatcode failures in that the blade material contained a concentration of 1.6 parts per million of the tramp element bismuth. Laboratory examination of previous M082 heatcode blade failures disclosed concentrations of 1.4 to 1.9 parts per million of bismuth.

The airline operators and the engine manufacturer are cognizant of this problem. The engine manufacturer has recommended that the first-stage turbine blades be examined at the next heavy maintenance and that all blades identified with heatcode M082 be removed from service.

To accomplish this, United Air Lines initiated a program on March 13, 1970, to identify and remove these blades from service. United had examined more than 260 engines in the implementation of this program.

The No. 1 engine was scheduled for examination at the next heavy maintenance check, which would normally have occurred at 5,800 hours or less.

It is apparent the No. 2 engine was in an operable condition at the time of the accident. All the evidence reveals the engine was operating in the air, during the thrust reversing cycle, and until the aircraft came to rest.

Results of examination and testing of the fuel pump drive shaft showed that the fracture was caused by fatigue resulting from a rotating beam type loading.

The loading induced on this shaft was due to an extreme misalignment between the fuel pump and the spur gear drive shaft. It was estimated that the misalignment created a bending stress in excess of 180,000 p.s.i. in the fuel pump drive shaft. Such a high bending stress results in ratchet marks, indicating multiple fatigue origins, which increase in number with the degree of stress concentration and stress level. The number of cycles to failure is primarily dependent on the level of this bending stress. In this case, the number of cycles is assumed to have been low. Inspection of the fuel pump drive shaft fracture surfaces revealed an extremely high number (at least 50) of fatigue origins.

It is not possible to determine the specific number of cycles from the initiation of the crack to failure since the stress level and time exposure are not known. However, based on laboratory tests conducted at higher than normal bending load (180,000 p.s.i.), there is an indication that the failure could have occurred in less than 2,000 cycles from initiation of the crack. This was evidenced by the similarity between the rig failed shaft and engine failed shaft in that the rig failure had at least 40 fatigue origins. These origins were very similar in appearance.

This type of fracture could have occurred only at impact when the shaft was subjected to an extreme misalignment over a short period of time. The location of the O-ring rub indicated that the shaft was operating in an improperly aligned position.

The roller bearing on the gearbox drive shaft nearest the fuel pump had pieces of the backside of the inner race broken out. The only plausible explanation is severe distortion of the gearbox housing which resulted in a thrust load from the outer race being transferred through the rollers to the inner race.

Microscopic analyses performed on the four first-stage nozzle guide vanes showed that the metal splatter on the vanes was aluminum. Similar material was found lodged in the cooling air spigot of a vane and was deposited inside the vane core surfaces. The source of the aluminum was the chain link fence that the aircraft passed through, approximately 2,000 feet after touchdown. The melting point range of this type aluminum is approximately 1,150 to 1,250°F.

In addition to the metallurgical analysis findings of the fuel pump drive shaft failure at impact, there are other factors to substantiate the fact that the No. 2 engine was operating at touchdown and throughout the overrun.

The turbine inlet temperature decay graph shows at fuel shutoff the temperature will drop from 1,800°F. to 900°F. in 1 second and down to 600°F. in 2 seconds.

It is recognized that turbine vane cooling rate is slower than the rate of decay of turbine inlet gas temperature. However, the uniformity and degree of adherence of the aluminum splatter observed on the blades and vanes indicated that sufficient heat, pressure, rotation, and air flow were available upstream of the first-stage nozzle guide vanes to melt and fuse the aluminum splatter to the vanes and blades. This finding significantly demonstrates normal No. 2 engine operation after the aircraft contacted the chain link fence.

Additional evidence of No. 2 engine operation was an increase of engine noise level after touchdown and audible on the cockpit voice recorder. (Appendix E.)

The No. 2 engine thrust reverser was fully deployed and was subsequently functionally tested and found to operate properly A further indication that the engine and reverser were operating was the evidence that during the runway and off-runway roll the nose gearwheels tracked toward the right main landing gearwheel track. (Appendix D.)

An engine oil pressure of 35 ± 2 p.s.i.g. is required to deploy the engine thrust reverser. Below this pressure, the reverser will not deploy. If the shaft had failed in flight, the engine oil pressure would have decreased below 37 p.s.i. in 4 seconds and below 35 p.s.i. in 5 seconds.

The 13th-stage aircraft bleed modulating valve was closed. The 13th-stage bleed modulating valve would begin to open on decreasing power at the following conditions:

V_{TAS} Knots	N_2 Percent r.p.m.	E.P.R.	Net Thrust-lbs	Percent of Takeoff Net Thrust
0	85.4	1.392	6,660	49.0
102	84.8	1.346	5,170	42.6

At fuel shutoff, a drop of N_2 percent r.p.m. from 100 to 72 percent would occur in a maximum of 2 seconds. A drop from 1.95 EPR to 1.15 occurred in 1 second. A drop from 100 percent of takeoff net thrust to 25 percent would occur in 1 second. The 13th-stage bleed modulating valve closed position showed the engine was operating throughout the flight and during the reverse cycle until the aircraft came to rest.

Boeing Aircraft Company laboratory test showed that the generator would not carry an electrical load below 42.3 percent N_2 speed. These values of N_2 are for a deceleration of 180 r.p.m./sec. of N_2 speed. At fuel shutoff, the percent of N_2 dropped off to 41 percent in a maximum of 8 seconds.

If the aircraft did have a double engine failure, the normal electrical generating systems would have been lost.

Electrical power on the aircraft was available throughout the overrun. The only question is – was it coming from a normal bus or from a standby bus and the aircraft battery. One transfer of electrical power was evidenced on the voice recorder. This would have occurred when the No. 1 engine failed, since the normal power source for the voice recorder comes from the No. 1 radio bus. There was no evidence on the flight recorder of a power loss and it is powered from the same bus.

The fact that the flight and voice recorders operated throughout the overrun is conclusive evidence that electrical power from a normal generating system was available. The flight and voice recorders cannot be powered from the standby buses. In addition, as the landing gear on the aircraft was down, the electrical circuit for the flight recorder could not have been completed unless an engine was running. The electrical circuit for the flight recorder is completed through either engine oil pressure switch or the landing gear latch relays.

The similarity of the compass information on the pilot's and copilot's Course Indicator (CI) and Radio Magnetic Indicator (RMI) instruments would indicate that normal electrical bus was powered when the aircraft came to rest. If the aircraft had switched to the standby buses, only the pilot's CI and copilot's RMI should have displayed the correct compass headings.

Other factors of particular significance to substantiate that No. 2 engine was operating include:

A. Vegetation was found throughout the secondary gas path, in the primary gas path as far back as the diffuser case, and in the sixth and eighth-stage aircraft bleed system. Mud coating was found adhering completely through the engine from the inlet to the exhaust.

B. All fuel nozzles were coated with dried mud except for the nozzle nut fuel exit holes.

C. There was no evidence of overheat in the combustion area, turbine blades, or on the nozzle guide vanes.

United Air Lines engine-out procedure is as follows:

> If an engine fails after reaching V_1 speed, the takeoff will be continued. The climbout will be at V_2 (if higher speed is already attained at the time of engine failure reduction to V_2 speed is not necessary), with a 15° bank maximum and a maximum deck angle of 15°. On reaching 500 feet accelerate the aircraft to V_2+15 knots and set flap position 1. At 190 knots set flaps 0.

At a gross weight of 90,000 pounds operating from a field elevation of 14 feet and the aircraft in flight at an approximate height of 50 feet and above V_2 speed, a single-engine climb and a subsequent safe landing could have been accomplished if the engine-out procedures had been followed.

Statements by the crew and questioning by the accident investigation group revealed that after a loud explosion was heard none of the three crewmembers checked the engine instruments to ascertain their problem or whether they had lost one engine or both engines. The first officer stated, "numerous amber lights on the overhead panel came on. The B system low quantity light, two or three lights on my master caution panel came on. At this time, it seemed to me that the left engine began to 'spool down'." The captain related that he believed he saw the right engine instruments spooling down. None of the crewmembers could recall what the airspeed was or the altitude at the time of the "loud bang," only that it happened about the normal time of gear retraction.

It is difficult to understand, without a check of the engine and flight instruments, how the captain determined that both engines had failed and why a decision to land was made immediately. This decision to land was made by the captain who stated, "I applied additional power with no response. There was no audible sound of power from either engine, no additional rudder feel, no increase in airspeed." The captain's assumption that both engines had failed must have been based on the decrease in engine noise, no increase in airspeed, and no additional rudder feel. This hasty decision to land must have been based on the captain's sensory faculties rather than on aircraft and engine instrumentations. This conclusion is verified by the second officer's statement, in part, that the captain "moved both throttles forward at this time but there was no response and the aircraft began settling." He "then made an instantaneous decision to set the aircraft back on the runway."

The cockpit voice recorder and flight data recorder showed that the time interval from the loud explosion to touchdown was approximately 12.7 seconds. The maximum altitude shown was approximately 50 feet. Six seconds after the loud explosion the aircraft started descending. The pilot's decision and action taken to land the aircraft occurred in approximately 6 seconds. The cockpit voice recorder revealed no discussion of the problems involved. Immediately after the loud bang, the captain said, "Okay, I got it," and the first officer replied with a question, "Are you flyin' it?" The captain never stated his intention to land but his intentions were made clear with his statement, "Get the gear down quick!"

All the flightcrew's training and experience in this type of emergency would dictate that they continue the flight. With the aircraft at a height of 50 feet and above V_2 speed, the crew should have been cognizant of the fact that it was not possible to land the aircraft and stop it before overrunning the far end of the runway. The captain had satisfactorily accomplished an engine-out takeoff in the simulator on March 6, 1970, in the aircraft on September 27, 1969, and in his rating flight on March 12, 1969. Takeoff with simulated engine failure is required by United Air Lines for captain proficiency flights and by the FAA for type rating flights. The engine cut is made immediately following V_1 and before reaching V_2 speed. The pilot is required to continue the takeoff and demonstrate his ability to fly the Boeing 737 aircraft on one engine.

This accident appears to point up the difficulties that can be encountered when control of an aircraft is shifted from one pilot to another during an emergency. The pilot assuming control, in such a situation particularly, lacks knowledge of control pressures and rates of control pressure changes that were occurring prior to the takeover. In this accident, the crew stated there was an immediate yaw to the right. This yaw was not, aerodynamically, the result of the left engine failure. One is left with the conclusion that such a yaw resulted from application of control pressures by a pilot. It can be assumed that the first pilot may have applied rapid excessive right rudder control in a reflexive response to a left yaw that probably occurred with the sudden loss of left engine power. Then the captain, not realizing this rudder input, took over control of the aircraft, which was yawing to the right after the explosion occurred. The captain's assessment of the engine instruments revealed to him that one set of instruments was spooling down and he interpreted this set as representing the right engine. Since the captain did not note both sets of engine instruments spooling down, and since the left engine suffered a rapid total power failure, one must conclude that he read the left set of instruments and interpreted them to be the right set.

It can be assumed then, that a right yaw, observed by the captain, which may have induced him to transpose the instrument readings to be compatible with a yaw to the right. It is interesting to note that the first officer's impression was that the left engine was "spooling down."

Perhaps the first officer had a better feel for the aircraft prior to the captain's assuming control. If such a misinterpretation as to which engine had failed remained fixed, subsequent subjective "feel" for the aircraft could have been confusing. In this accident, accurate "feel" for the aircraft may also have been compromised by the presence of both pilots on the controls.

The captain stated he applied full power with no response in airspeed, engine noise, or rudder feel. Very little increase in thrust would have resulted from this action. The increased power lever input would not have given significant increases in the parameters of response for which the captain was looking.

All three crewmen stated there was a steady and substantial loss of engine noise before the attempt to land was initiated. The decrease of engine noise associated with the loss of one engine, located symmetrically with the second engine in relation to the flight deck, could be perceived as no more than a 3 decibel decrease. This decrease, if noted at all, would not be alarming. It also would not have been perceived as a gradual steady loss.

It is difficult to explain the reason the flight deck crew heard a steady and substantial decrease in engine noise. One possible explanation is that there is a substantial decrease of engine noise in the cockpit when an aircraft leaves the ground on takeoff.

The captain's assessment of the emergency was that the aircraft would not sustain flight. He was then forced to make an immediate decision as to where to make the inevitable landing. Since a portion of the runway was still visible, his choice was to land back on the runway. Furthermore, the need for a rapid decision in order to effect a return to the runway greatly compromised the time available to assess the emergency.

2.2 Conclusions

(a) Findings

1. The flightcrew members were properly certificated for the flight.

2. The aircraft was properly certificated and airworthy.

3. The weight and balance of the aircraft was within the allowable limits.

4. At the gross weight at which the aircraft was being operated, it was capable of climbing on one engine.

5. The No. 1 engine failed in flight as a result of a first-stage (N_2) turbine blade failure.

6. The No. 2 fuel pump drive shaft failed at impact of the engine.

7. The No. 2 engine was operating until it impacted the ground.

8. The aircraft was airborne and above V_2 speed at the time of the engine failure.

9. The flightcrew did not properly utilize the engine and aircraft instruments to determine the condition of the engines, altitude, and airspeed.

10. Company procedures and applicable flight manuals dictate that the flight should have been continued with one engine inoperative.

11. The captain discontinued the takeoff and landed back on the runway.

12. The captain erroneously decided power to both engines had been lost.

13. The No. 2 engine reverse thrust was selected and power was applied after touchdown.

14. The captain had satisfactorily accomplished an engine-out takeoff in the simulator and two in the aircraft since March 12, 1968.

15. The first officer remained on the controls after the captain took over the control of the aircraft.

(b) Probable Cause

The National Transportation Safety Board determines that the probable cause of this accident was the termination of the takeoff, after the No. 1 engine failed, at a speed above V_2 at a height of approximately 50 feet, with insufficient runway remaining to effect a safe landing. The captain's decision and his action to terminate the takeoff were based on the erroneous judgment that both engines had failed.

3. RECOMMENDATIONS

During its deliberations, the National Transportation Safety Board found that important safety lessons were evident from the facts, conditions, and circumstances of this and similar accidents. The Board, therefore, recommends to the Federal Aviation Administration the following:

1. Reassess the respective duties and responsibilities of the captain and the first officer during critical phases of flight. In so doing, the "captain in command" concept should be reexamined with its applicability in situations where time may not permit the captain to countermand effectively the decision of the first officer who is flying the aircraft.
2. Reappraise the current training manuals and instructions provided by all airlines with a view toward a positive approach toward emergency procedures. Such an evaluation would include an amplification and clarification of such procedures, including safety margins and the need for prompt and proper sequencing of each action.
3. Reemphasize in training that pilots use the aircraft instrumentation, rather than their physiological responses, to determine the extent and cause of emergencies.

The Board further recommends that the Air Transport Association bring this report to the attention of its training committee.

BY THE NATIONAL TRANSPORTATION SAFETY BOARD:

/s/ JOHN H. REED
Chairman

/s/ OSCAR M. LAUREL
Member

/s/ FRANCIS H. McADAMS
Member

/s/ LOUIS M. THAYER
Member

/s/ ISABEL A. BURGESS
Member

December 29, 1971

INVESTIGATION AND HEARING

1. Investigation

The Board received notification of the accident at approximately 2000 e.d.t. on July 19, 1970, from the Federal Aviation Administration. An investigating team was immediately dispatched to the scene of the accident. Working groups were established for Operations, Aircraft Records, Witnesses, Structures, Systems, Powerplants, and Human Factors. Parties to the investigation were: The Federal Aviation Administration, Air Line Pilots Association, Pratt & Whitney Aircraft, The Boeing Company, and United Air Lines, Inc.

2. Hearing

A public hearing was not held by the Safety Board. A preliminary report was released on October 5, 1970.

CREW INFORMATION

Captain Joseph Lubozynski, aged 46, was initially employed by Capital Airlines August 3, 1956, and continued with United Air Lines after the merger of Capital and United Air Lines. He holds an Airline Transport Pilot Certificate No. 421597 with ratings in the Boeing 737 and the Douglas DC-6/7 and Vickers Viscount 745 D aircraft.

He passed his last examination for a Federal Aviation Administration first-class medical certificate on June 26, 1970, with the limitation noted, must possess corrective glasses for near vision. He had accumulated 11,236 hours of flying time as of July 19, 1970, of which 164 hours were accumulated in the preceding 90 days and 2.55 hours in the preceding 24 hours. He had acquired 517.51 total hours in the Boeing 737 aircraft. Ground school and flight training in the Boeing 737 was completed when he passed his rating flight check on March 15, 1969. His last en route check was completed January 13, 1970, his last proficiency check was completed on March 6, 1970, and his last emergency evacuation review on the Boeing 737 type equipment was completed on March 5, 1970.

First Officer James W. McWilliams, aged 25, was employed by United Air Lines on February 7, 1966. He holds Commercial Pilot Certificate No. 162674 and Flight Engineers Certificate No. 1689418. His commercial pilot's certificate was rated for airplanes single and multi-engine land, instrument and flight instructor.

He passed an examination for a Federal Aviation Administration first-class physical without limitations on August 29, 1970. He had accumulated a total of 2,319 flight hours as of July 19, 1970, of which 180.38 hours were in the last 90 days and 2.55 hours in the last 24 hours. He had acquired 736 total hours in the Boeing 737 aircraft. His first officer training in the Boeing 737 was completed on May 5, 1969. His last proficiency check in the Boeing 737 was completed May 29, 1970.

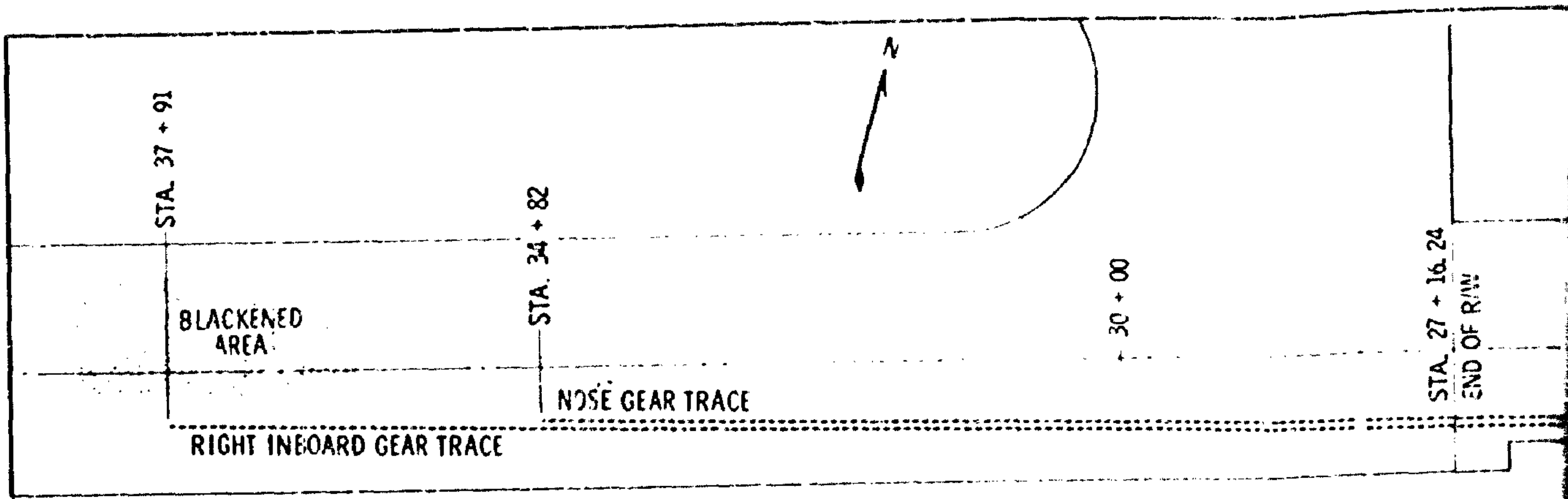
Second Officer Lee H. Hoffer, aged 32, was employed by United Air Lines on July 10, 1969. He holds Commercial Pilot Certificate No. 1639471. His commercial pilot's certificate was rated for airplane single engine and multi-engine land and instrument ratings.

He passed an examination for a Federal Aviation Administration second-class physical without limitations on October 4, 1969. He had accumulated a total of 3,024 hours of flight time as of July 19, 1970, of which 141.00 hours were in the last 90 days and 2.55 hours in the last 24 hours. He had acquired 380 total hours as second officer in the Boeing 737 aircraft. His second officer training in the Boeing 737 was completed in August 1969. His last en route check was completed on May 31, 1970.

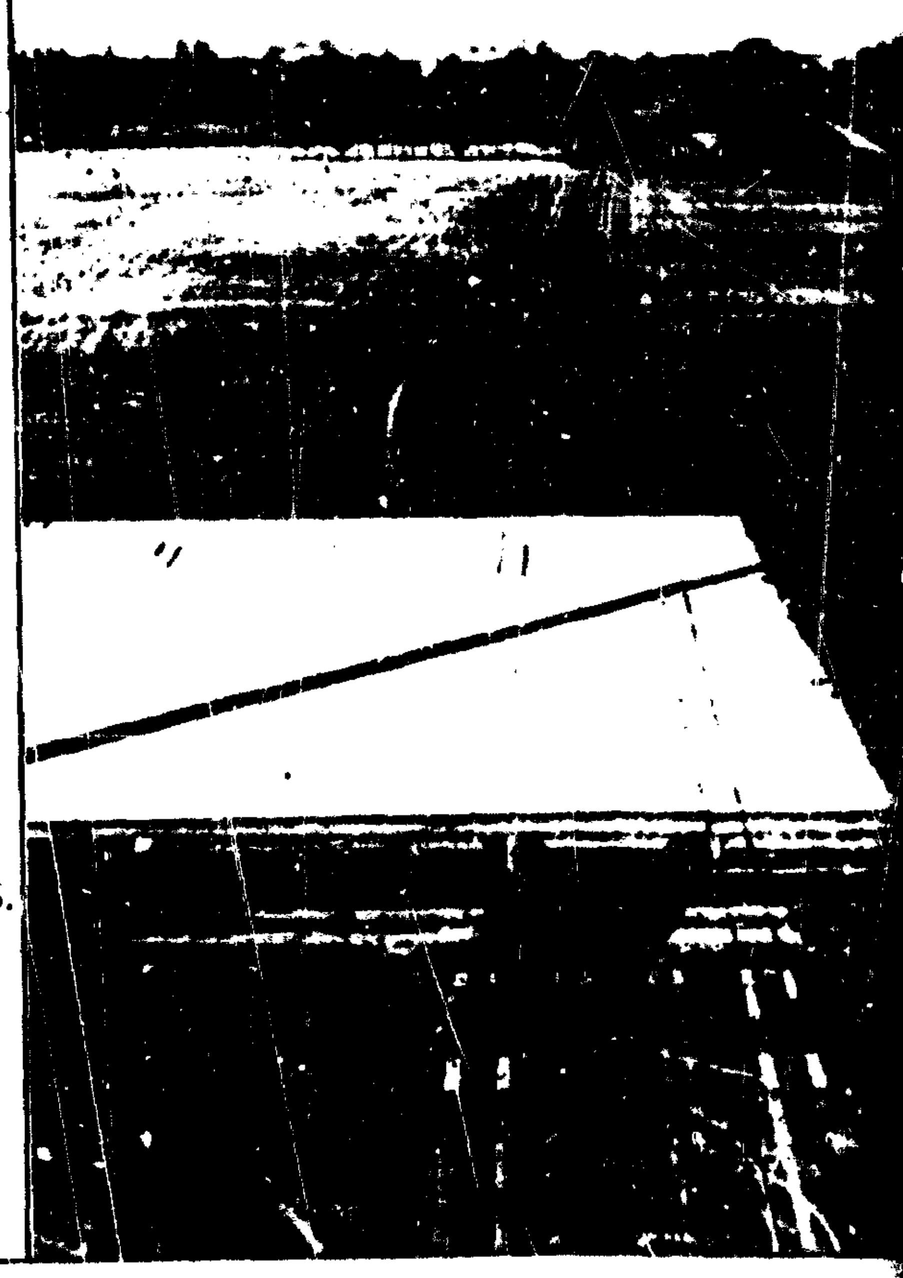
Stewardess Linda Evans, aged 24, was employed by United Air Lines on September 4, 1968, and received her last recurrent training on September 9, 1969.

Stewardess Margaret Powell, aged 22, was employed by United Air Lines on June 26, 1968, and received her last recurrent training on July 16, 1970.

Stewardess Cynthia Holt, aged 21, was employed by United Air Lines on October 26, 1969, and received her last recurrent training May 11, 1970.



STATIONING	DISTANCE SOUTH OF RUNWAY CENTERLINE		REMARKS
	RIGHT INBOARD GEAR	NOSE GEAR	
37+91	46.2'		TOUCH DOWN
34+83		44.0'	TOUCH DOWN
25+65	63.5'	59.2'	END OF BLAST PAD
22+95	71.0'	66.2'	
21+14	77.8'	74.0'	
19+68	86.3'	81.6'	
18+32	94.1'	88.7'	
15+90	109.4'	101.7'	NOSE GEAR INTERSECTS FENCE
15+52		106.3'	PIECE OF NOSE GEAR DOOR: 103.8'
14+31	128.7'	124.0'	RIGHT NOSE GEAR DOOR: 129.5' S. TOP OF DITCH BANK: 158' S.
13+31	140.4'	136.4'	LEFT NOSE GEAR DOOR: 123.4' S. TOP OF DITCH BANK: 156' S.
12+67			MOUND OF EARTH AND RUBBLE: 150' S.
12+05			PIECE OF RT. INBOARD FLAP: 146' S.
11+93		151.3'	
11+66			TOP OF DITCH BANK & CENTER OF FUSELAGE NEAR TAIL: 161.7' S.
10+82			CENTER OF NOSE: 143.8' S. TOP OF DITCH BANK: 165.8' S.



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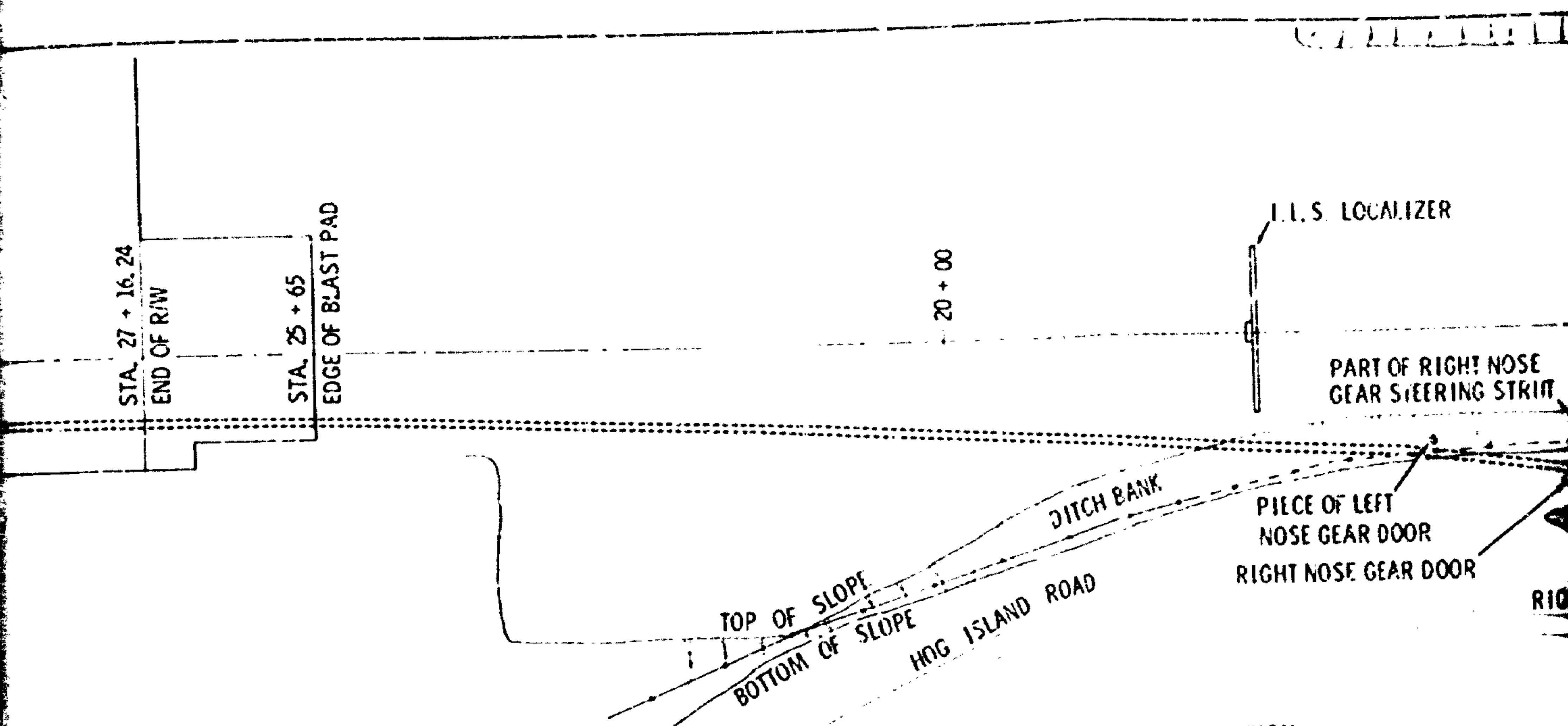
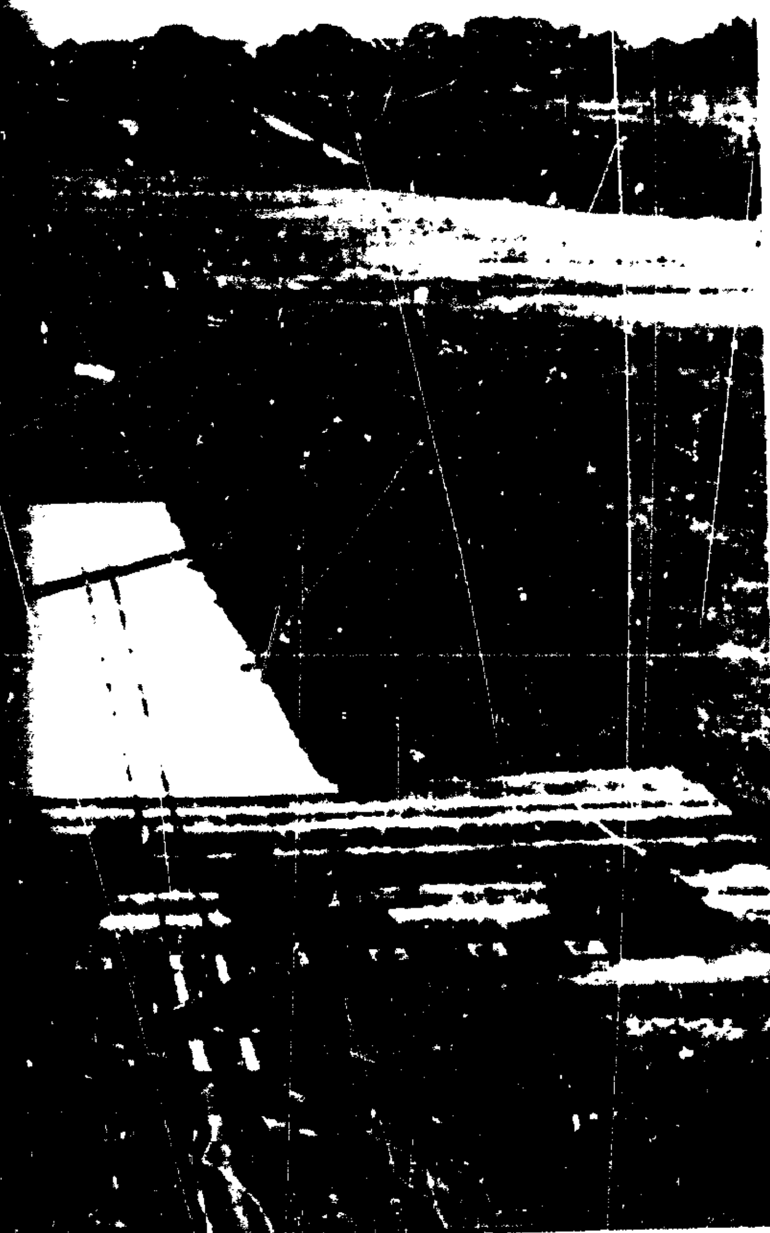
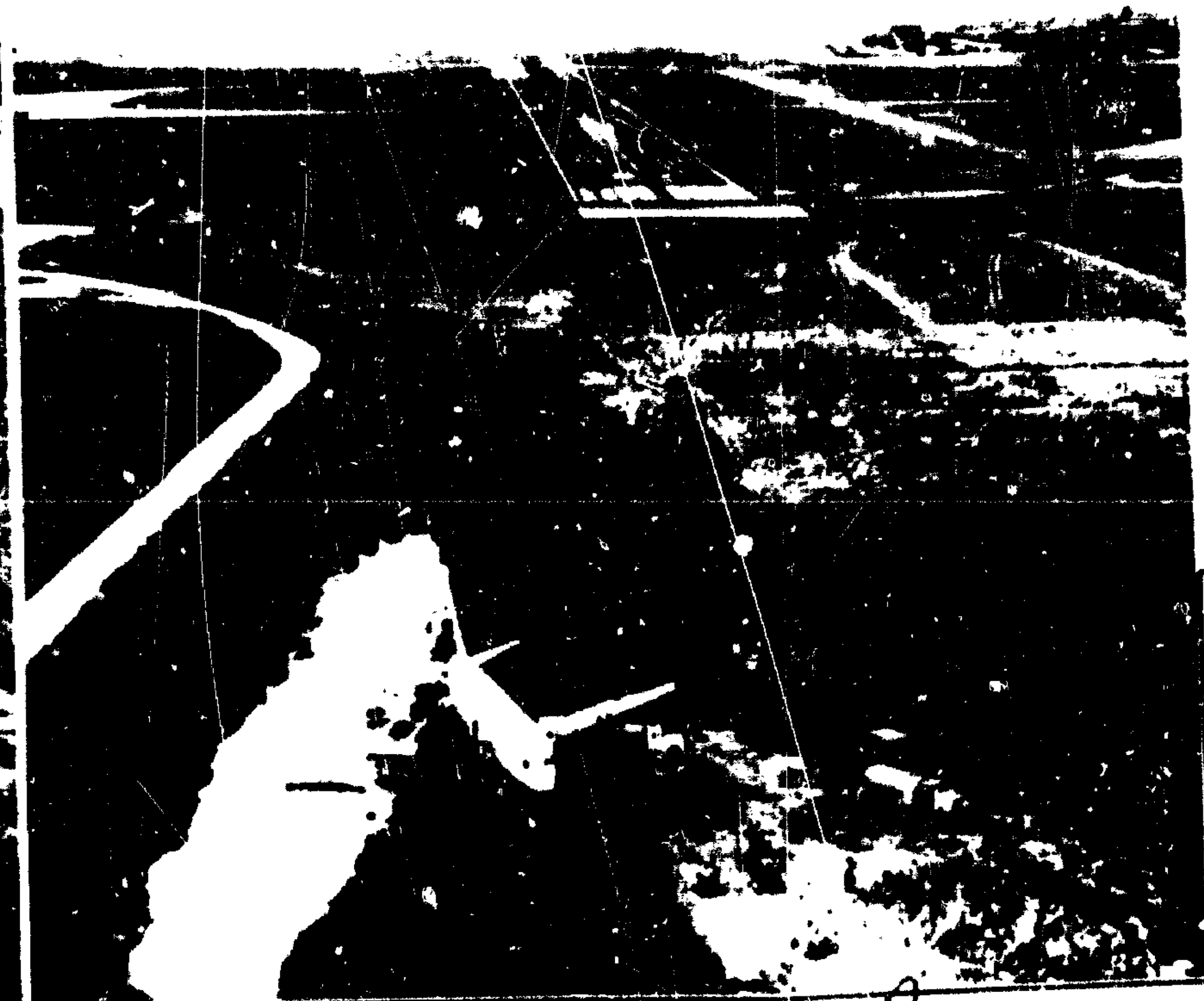


PLAN VIEW OF WRECKAGE DISTRIBUTION

SCALE: 0 50' 100'





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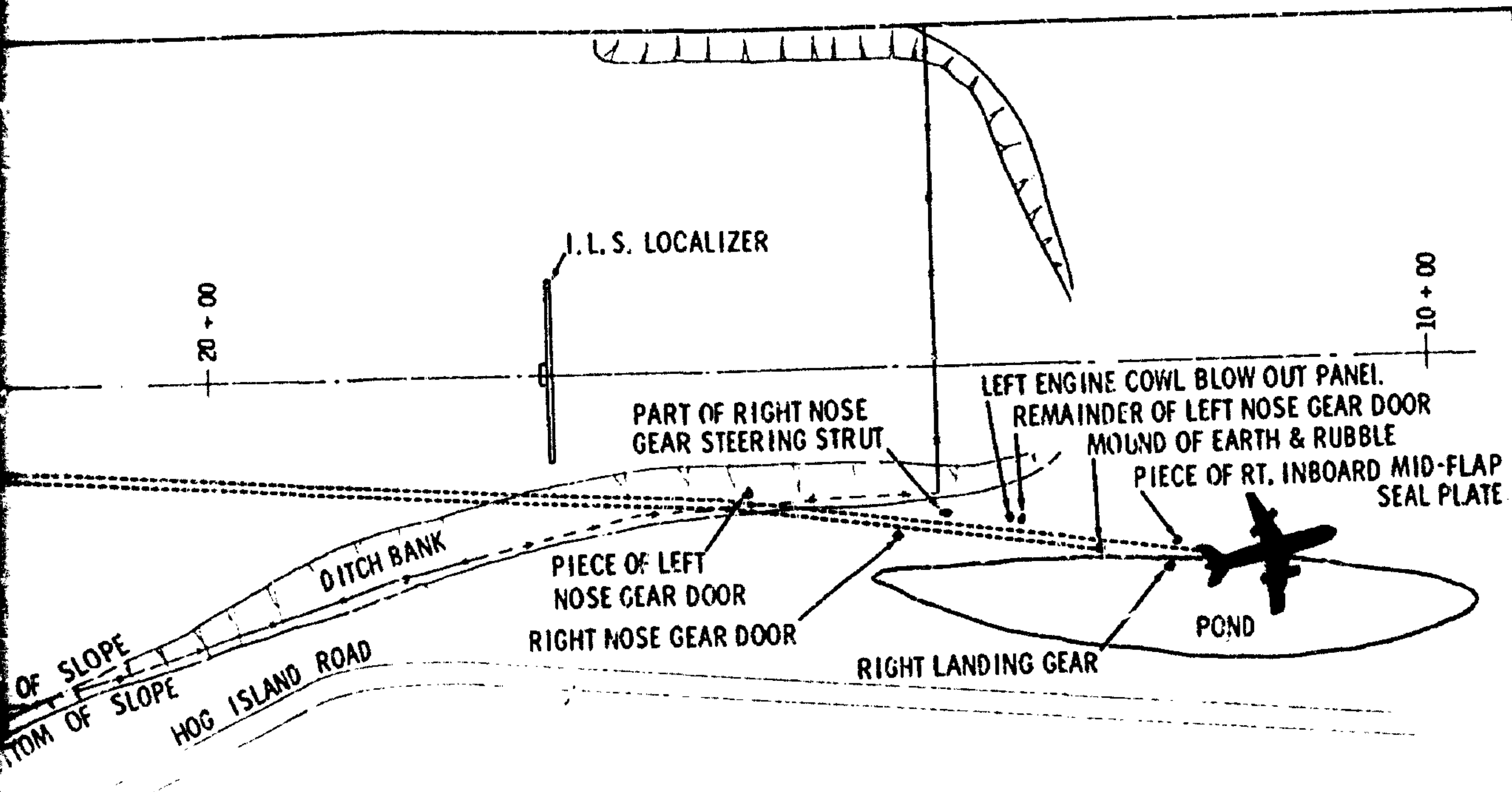
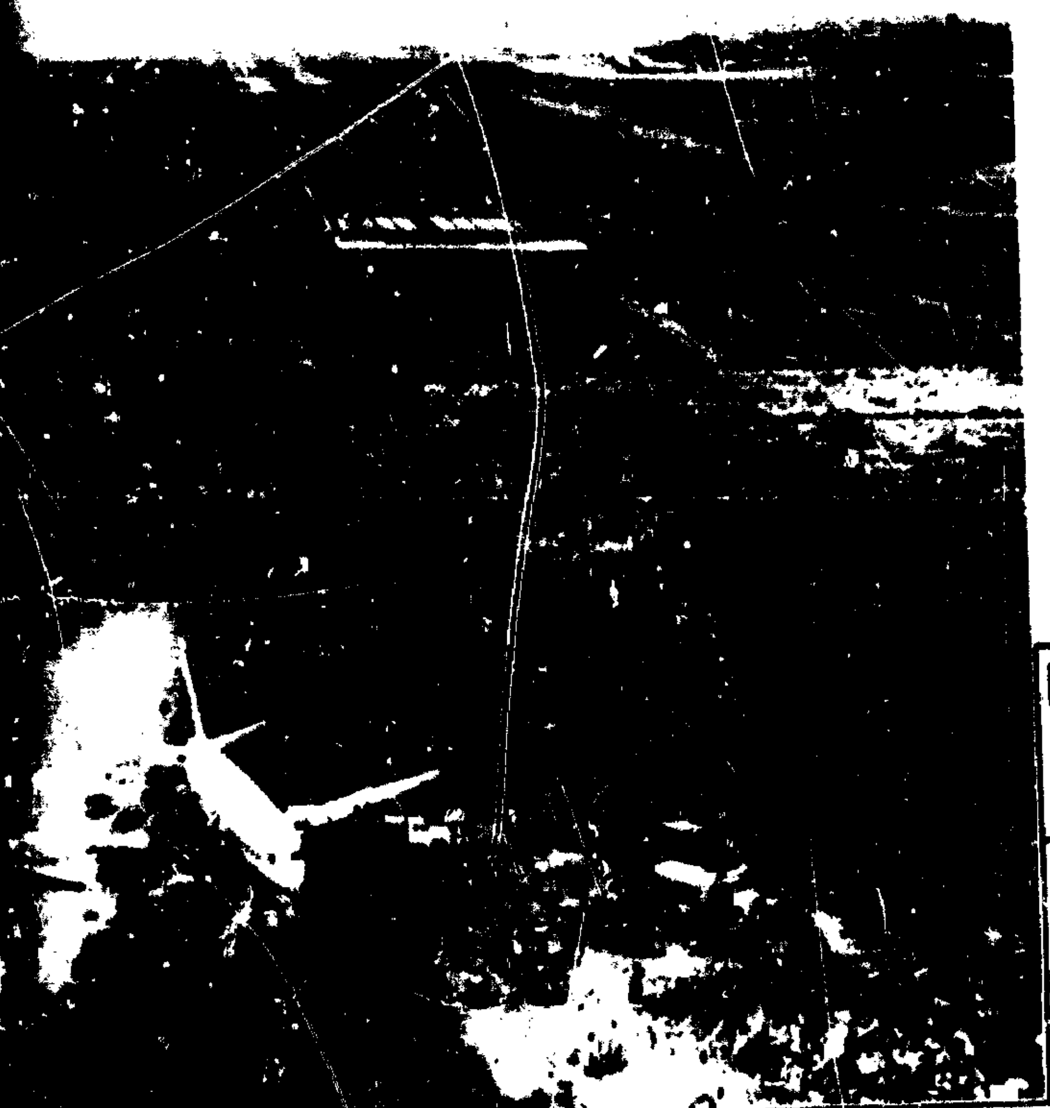


PLAN VIEW OF WRECKAGE DISTRIBUTION

SCALE: 0 50' 100'



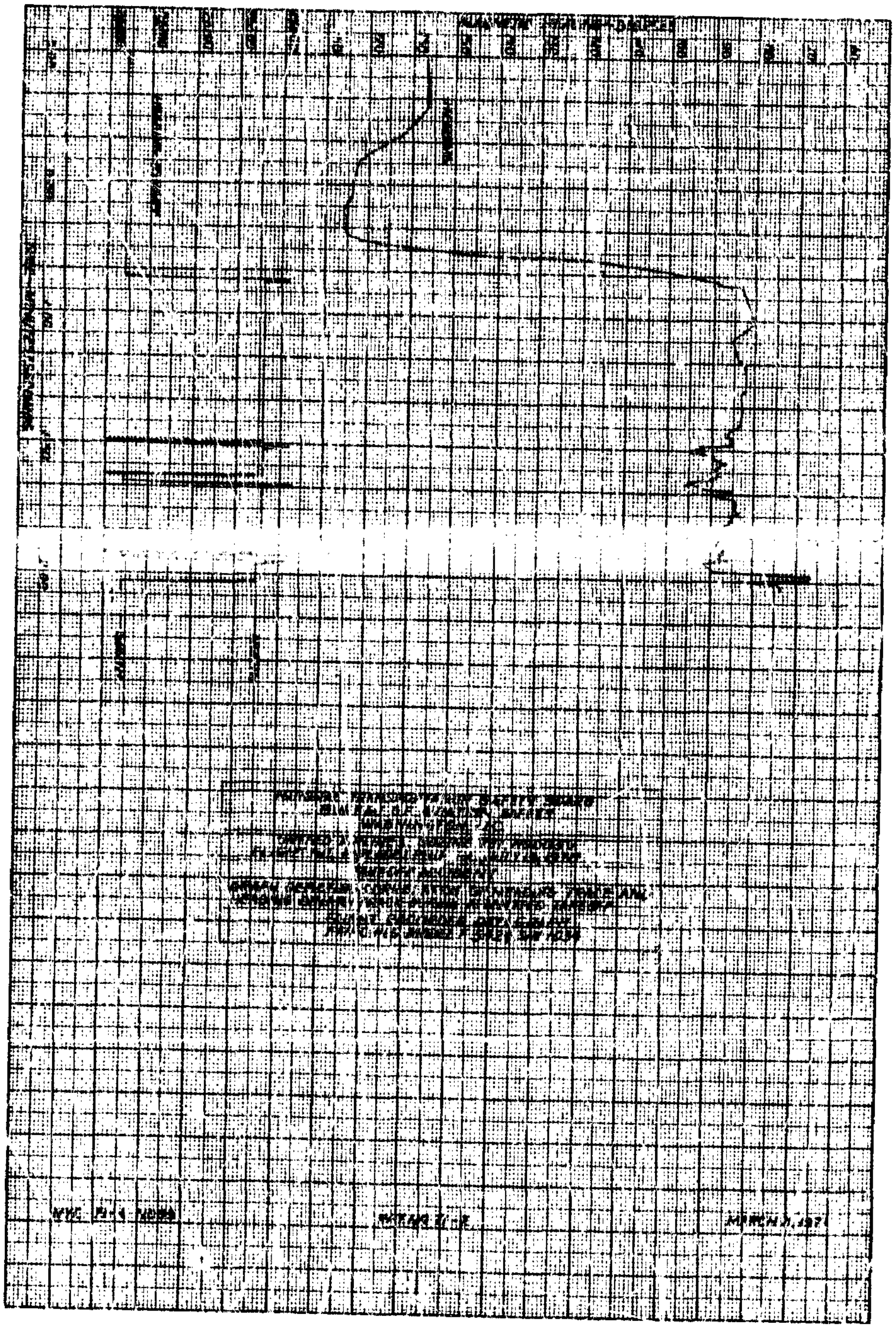
APPENDIX C

NATIONAL TRANSPORTATION SAFETY BOARD

Washington, D.C.

WRECKAGE DISTRIBUTION CHART
UNITED AIR LINES, INC. B737-222, N9005U
PHILADELPHIA INTERNATIONAL AIRPORT
July 19, 1970

APPENDIX D



NATIONAL TRANSPORTATION SAFETY BOARD
Washington, D. C. 20591

March 9, 1971

SUPPLEMENT TO FLIGHT DATA RECORDER REPORT NO. 71-2

A. Accident

Location	: Philadelphia, Pennsylvania
Date	: July 19, 1970
Aircraft	: Boeing B-737, N9005U
Airline	: United Air Lines
Flight No.	: 611
Flight Recorder	: Fairchild F5424-601, S/N 1034
Ident. No.	: NYC 71-A-N009

B. Supplementary Information

The subject flight data recorder foil was reexamined on March 1, 1971. The heading parameter trace and heading north-south binary trace were plotted on a graph to show their correlation.

The design of the subject flight data recorder is such that the heading binary scribes along a line approximately 2.6050 inches above the reference line when the aircraft heading is in the azimuth of 90 degrees through 180 degrees to 270 degrees. This is referred to as the southern hemisphere of the compass. When the aircraft heading is in the azimuth of 90 degrees through zero degrees to 270 degrees, or northern hemisphere of the compass, the heading binary scribes along a line approximately 2.6350 inches above the reference line. The heading synchro rotor, which drives the heading stylus, operates a cam driven switch to electrically energize a D.C. solenoid when the compass heading goes through 90 degrees or 270 degrees toward north. Similarly, the solenoid is de-energized when the heading synchro rotor passes through 90 degrees or 270 degrees going south. A loss of electrical power to the flight data recorder and/or N-S binary solenoid results in the binary scribe arm dropping to the lower or south value due to a spring preload. The cam-switch-solenoid sequence must be set to operate at 90 degrees or 270 degrees plus or minus 2 degrees. Calibration data from United Air Lines for the subject flight data recorder indicated that the binary solenoid activation setting was within tolerances when checked on July 9, 1970. A heading calibration curve was made on March 1, 1971, and the readout of the heading trace was plotted on the attached graph applying this calibration data. In addition, the graph includes the heading binary trace for the same time period.

The attached graph was plotted after applying an offset of +.0210 inches to the "X" axis (time) of the heading parameter reading. Foil examination reflected that the heading binary trace was leading the heading trace by +.0210 inches.

The readout showed that the aircraft heading during the attempted takeoff roll was easterly with the heading varying between 80 degrees and 95 degrees. The heading binary showed 10 binary changes during the readout which covered the time from turning onto the runway to loss of flight data recorder power after the accident. Comparison of the heading information with the N-S heading binary showed no activity on the binary trace except binary shifts due to compass heading information denoting passage through or approach to 90 degrees plus or minus 2 degrees.

John D. Rawson

/s/ John D. Rawson
Air Safety Investigator

Attachment

**TRANSCRIPTION OF LAST PORTION OF COCKPIT VOICE RECORDING
BOEING 737, N9005U, PHILADELPHIA, PENNSYLVANIA, JULY 19, 1970**

LEGEND

PHL	–	Philadelphia Tower Local Controller
RDO	–	Radio transmission from N9005U
CAM	–	Cockpit area microphone sound or voice source
–1	–	Voice identified as Captain
–2	–	Voice identified as First Officer
–3	–	Voice identified as Second Officer
–?	–	Voice unidentified
*	–	Unintelligible word or phrase
( )	–	Words in parentheses are subject to correction

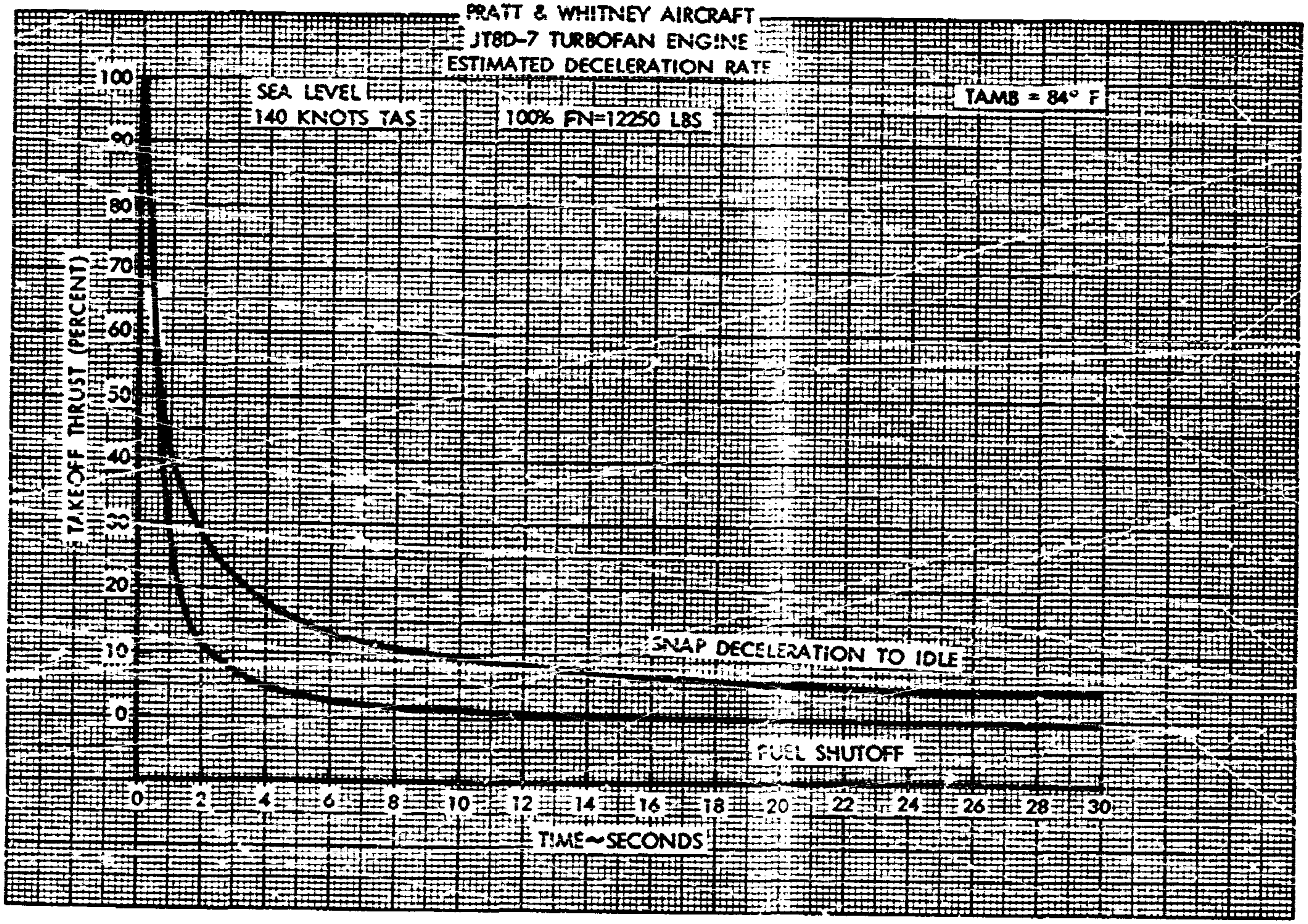
SOURCE & TIME		CONTENT
PHL	–	United six eleven are you ready?
RDO-2	–	Yeah, you were broken up there, we're ready
PHL	–	United six eleven taxi into position and hold runway nine
RDO-2	–	Okay
CAM-3	–	And we need a recall on * departure
CAM-?	–	Checked and out
CAM-?	–	Position and hold
CAM-?	–	Right
CAM-?	–	Give * some gas back there

APPENDIX E

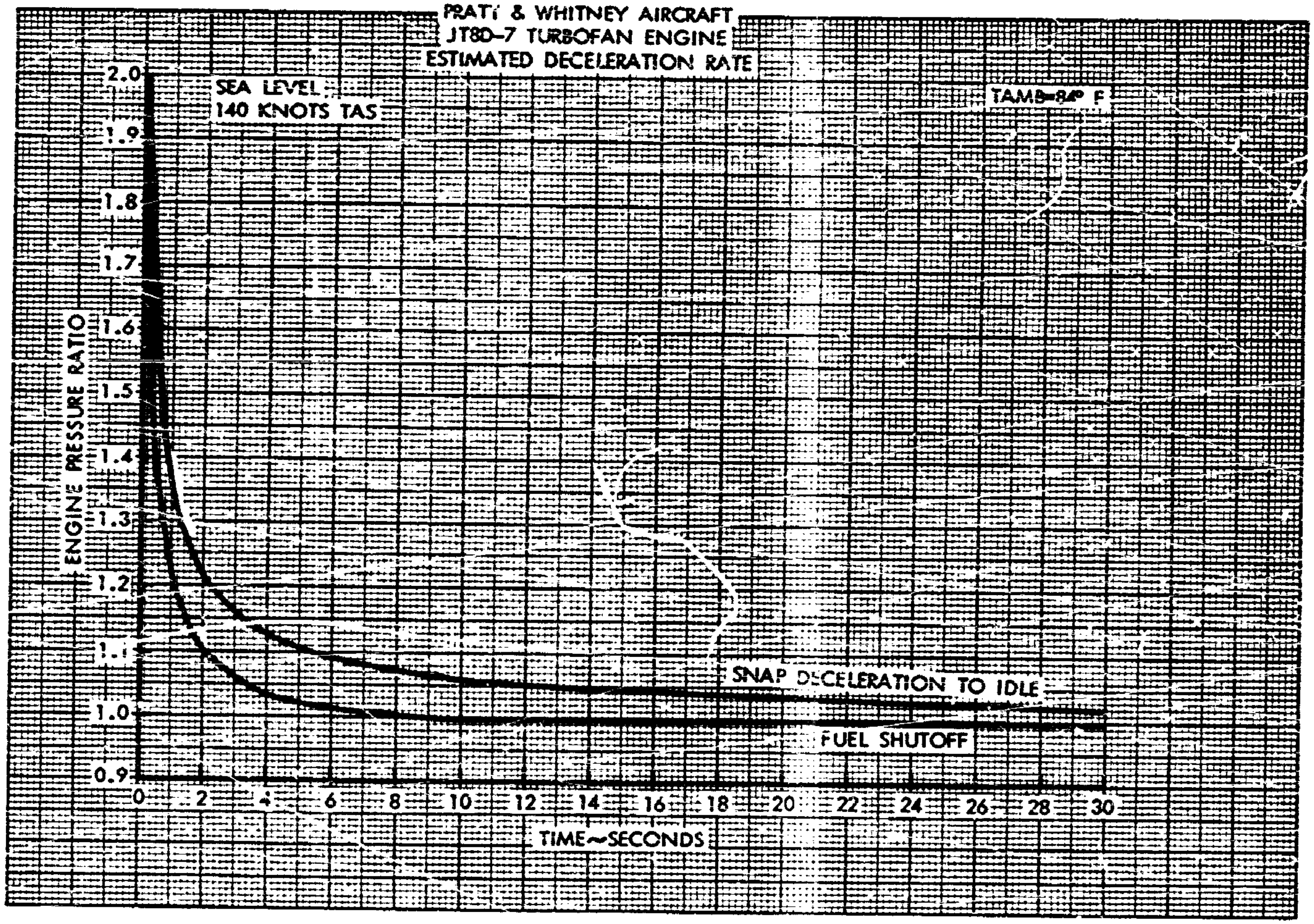
SOURCE & TIME		CONTENT
0:00		
PHL	–	United six eleven after departure proceed direct to Pottstown, cleared for takeoff
RDO-2	–	Six eleven cleared to go
RDO-2	–	What do you want, a right or left?
RDO-2	–	What do you want, a right or left?
PHL	–	A left turn direct Pottstown
RDO-2	–	Okay
0:17.6 CAM-1	–	Let 'er rip!
0:27.2 CAM-2	–	Takeoff power
CAM-2	–	Ahhh!
CAM	–	Sound of rattling
CAM-3	–	* and temps good
CAM-?	–	(trim) handles *
CAM-1	–	* * *
0:48.5 CAM-1	–	Vee one, Vee R, Vee two
0:52.3 CAM	–	Sound of loud bang
CAM-?	–	Hang on!
0:54.5 CAM	–	Sound of electrical power transfer for recorder
0:55.2 CAM-1	–	Okay, I got it

SOURCE & TIME		CONTENT
CAM	–	Sound of loud rattle
0:56.9 CAM-2	–	Are you flyin' it?
CAM-1	–	* * down now
CAM-2	–	Huh?
1:00.6 CAM-1	–	Get the gear down quick!
CAM	–	Sound of loud rattle
1:05.0 CAM	–	Sound of touchdown followed by increase in engine sounds
1:11.0 CAM-?	–	Hang on!
CAM	–	Sounds of impact
1:21.2 CAM	–	Sound of electrical power removal from recorder
CAM-1	–	Everybody out!!
		End of recording

APPENDIX F
VARIOUS ENGINE PARAMETERS



APPENDIX F
VARIOUS ENGINE PARAMETERS



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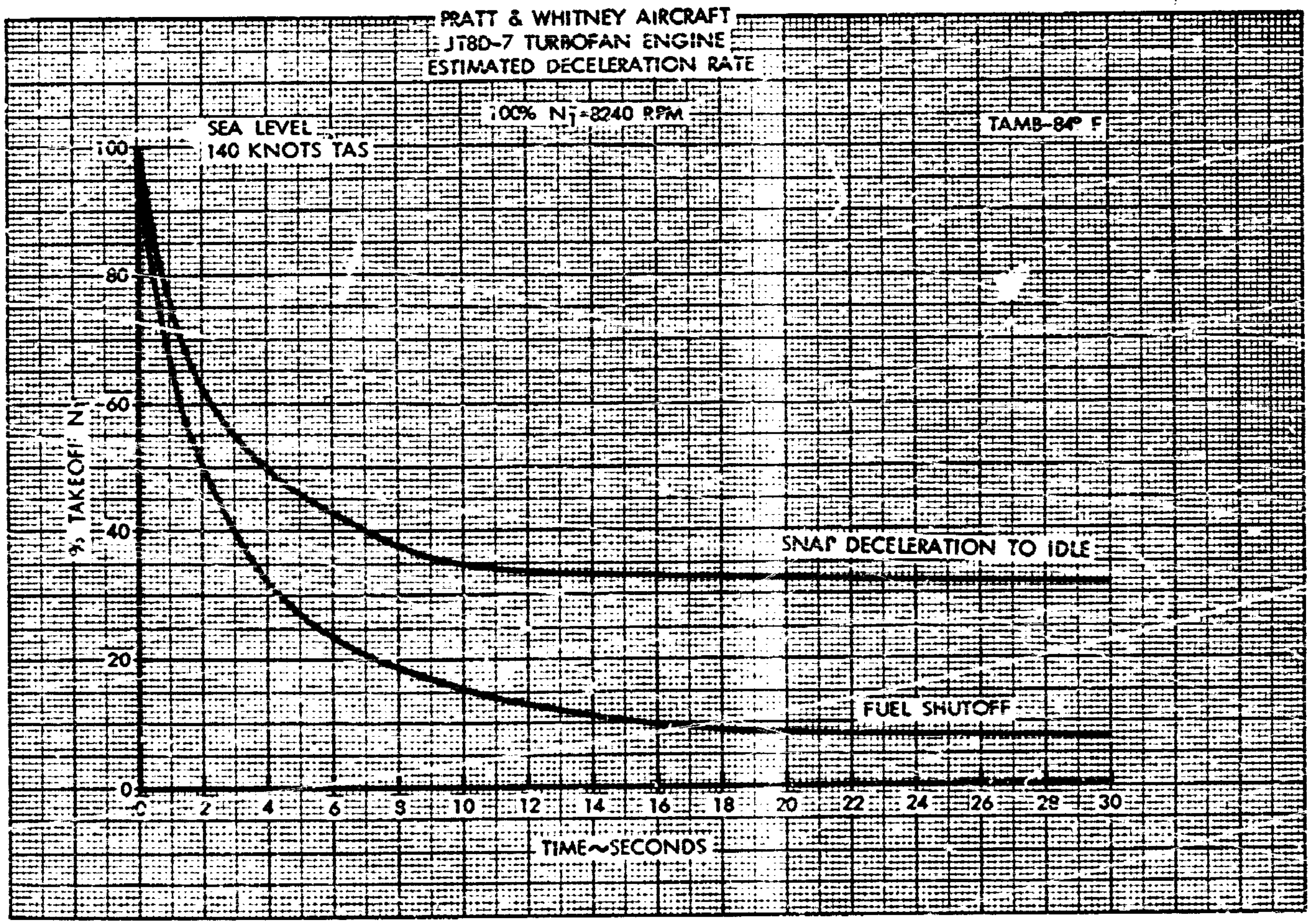
VARIOUS ENGINE PARAMETERS



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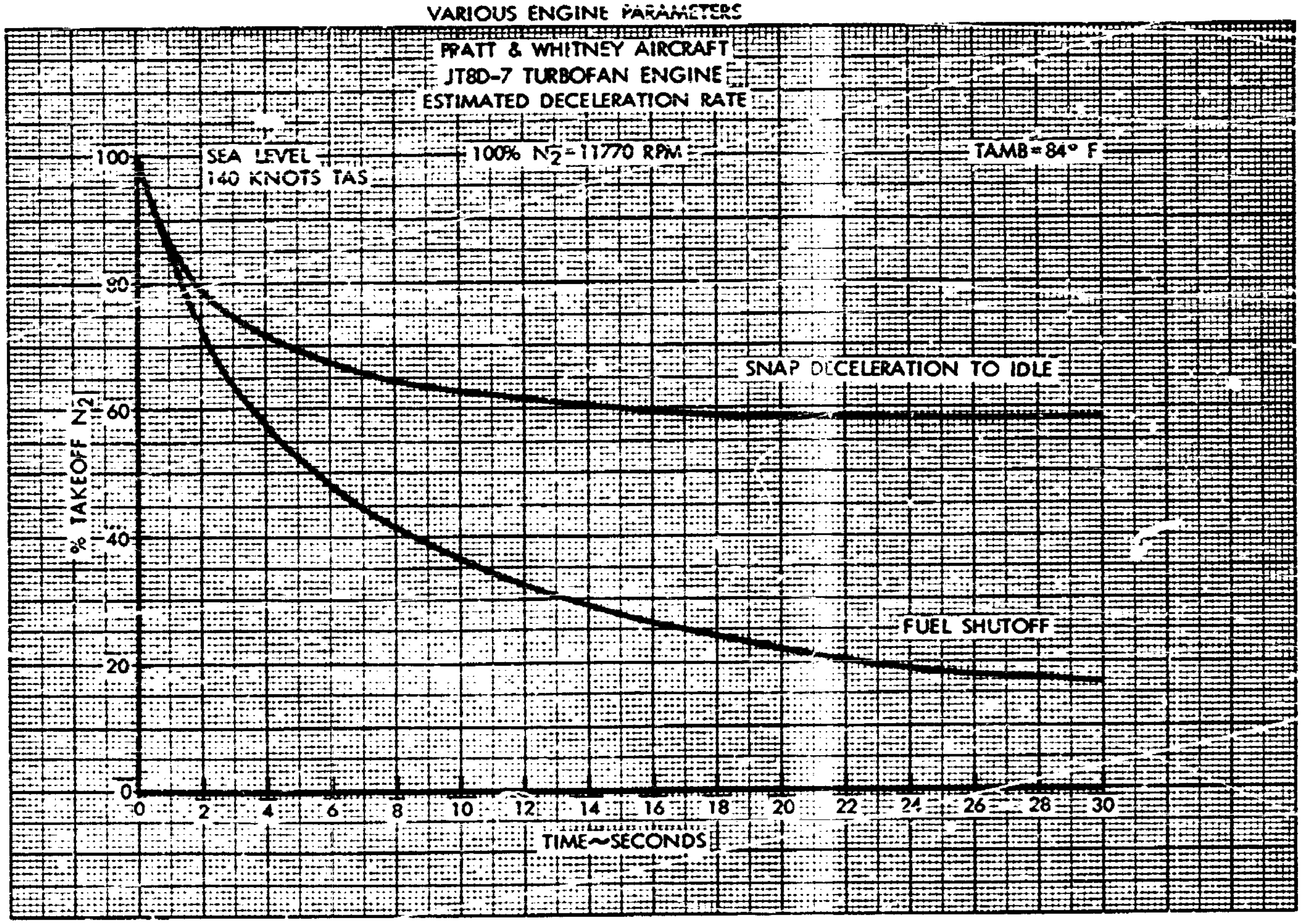
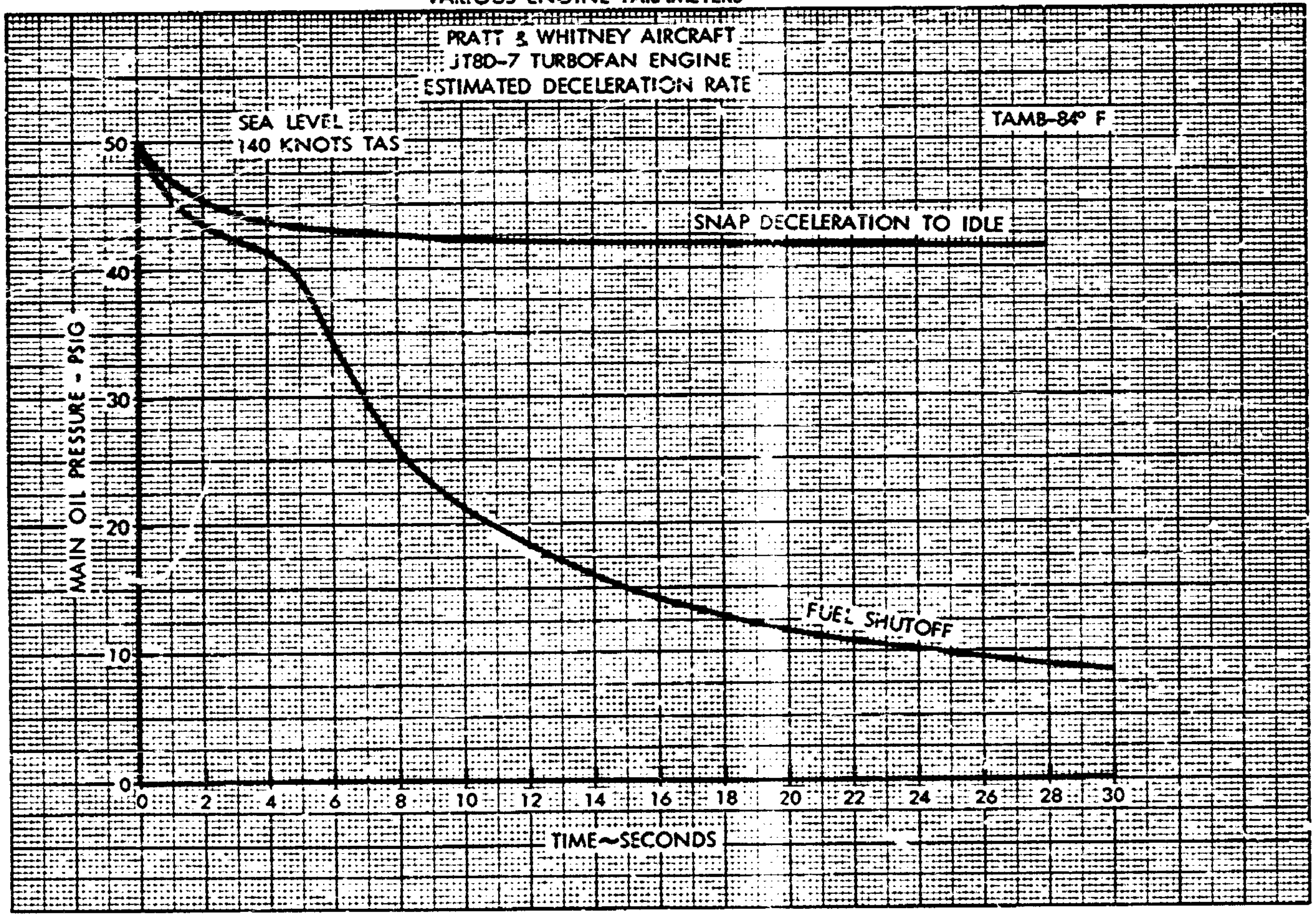
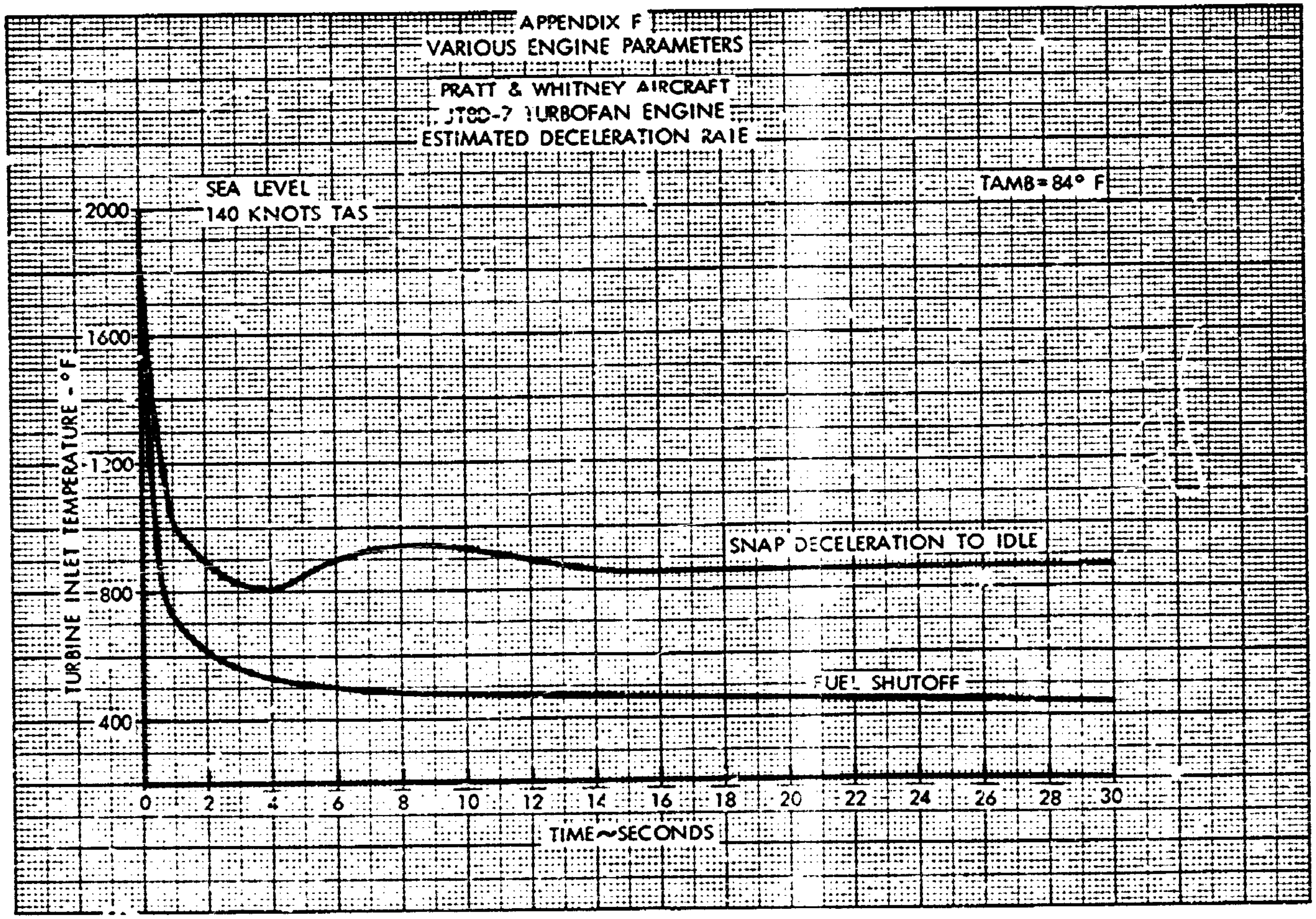
VARIOUS ENGINE PARAMETERS

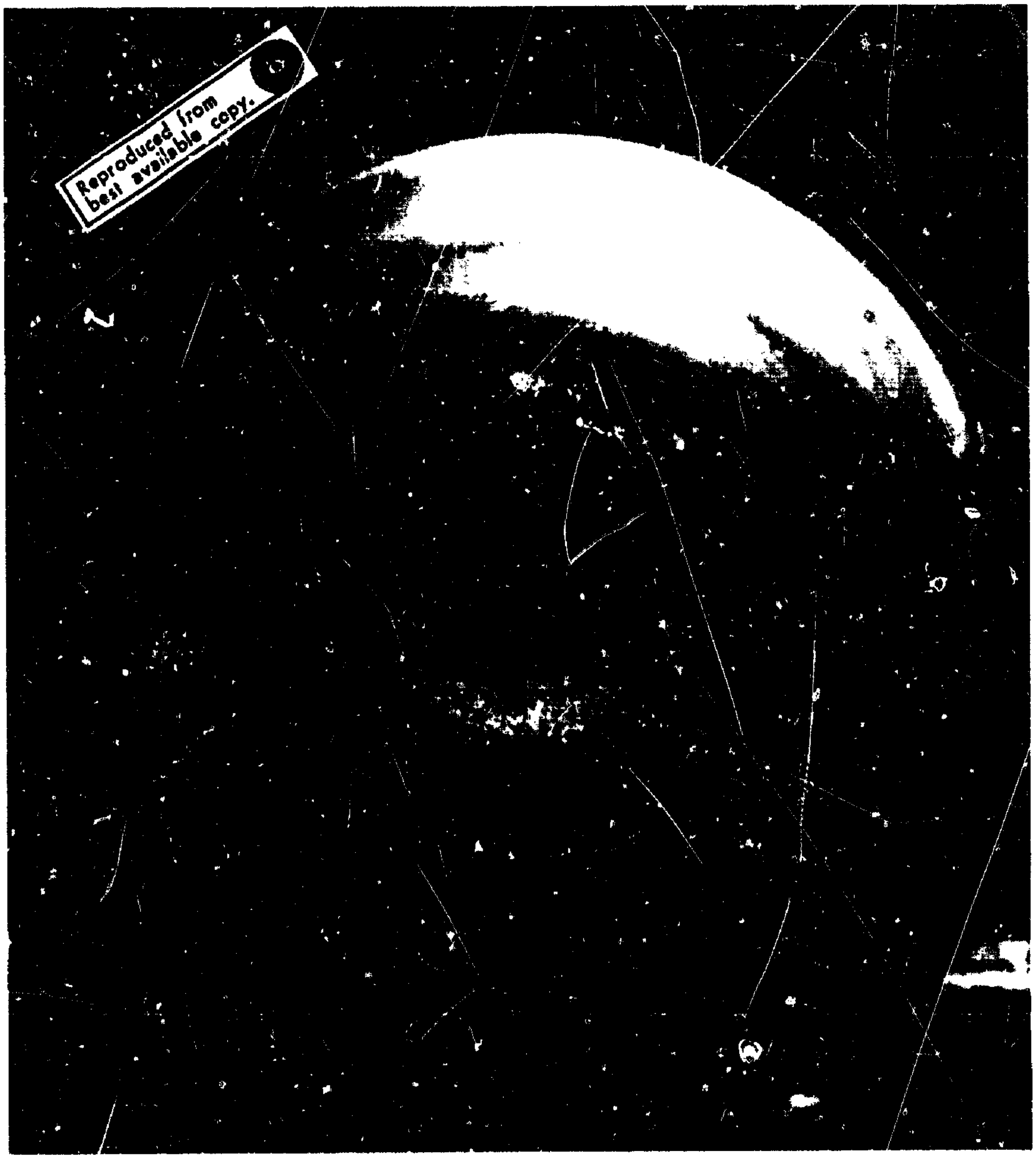


APPENDIX F

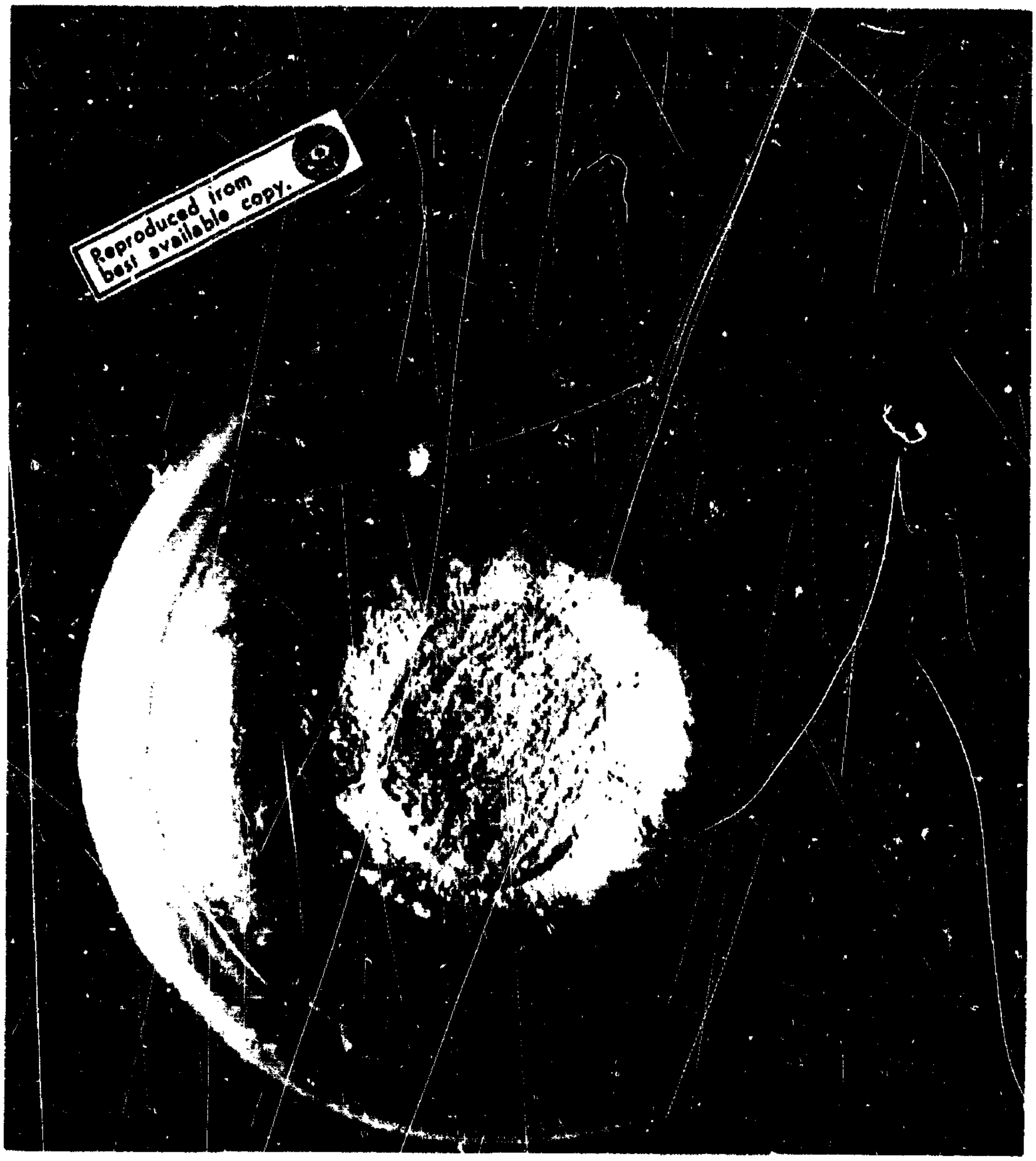
VARIOUS ENGINE PARAMETERS




APPENDIX F
VARIOUS ENGINE PARAMETERS
PRATT & WHITNEY AIRCRAFT
JT8D-7 TURBOFAN ENGINE
ESTIMATED DECELERATION RATE
SEA LEVEL
140 KNOTS TAS
TAMB = 84° F
SNAP DECELERATION TO IDLE
FUEL SHUTOFF
TURBINE INLET TEMPERATURE - °F
2000
1600
1200
800
400
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
TIME ~ SECONDS



Failed Fuel Pump Drive Shaft P/N 208235 from UAL Engine 656059 Involved in Philadelphia Incident. . .Boeing 737 N900 5U, Flight 611, No. 2 Position, 7/19/70. Viewed Toward Gearbox.



Fuel Pump Shaft Test Simulation – Bending Stress 160,000-165,000 PSI – 3500 Cycles