NATIONAL TRANSPORTATION SAFETY BOARD

WASHINGTON, D.C. 20594

AIRCRAFT ACCIDENT REPORT

NEW YORK AIRWAYS, INC.
SIKORSKY S61-L, N618PA
NEWARK, NEW JERSEY
APRIL 18, 1979

NTSB-AAR-79-14

UNITED STATES GOVERNMENT

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16. Abstract

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Metallurgical examination revealed that the tail rotor blade failed after a fatigue crack propagated across 90 percent of the blade's leading edge spar and about 2 in. of the blade skin, which weakened the blade structure.

The National Transportation Safety Board determines that the probable cause of this accident was the separation of the tail rotor assembly and gearbox from the aircraft at an altitude which made further controlled flight impossible. The rotor assembly and gearbox separated because of severe vibrations in the rotor assembly which were induced by the loss of a tail rotor blade due to fatigue failure.

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Abstract Continued

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SYNOPSIS

About 1823 e.s.t., on April 18, 1979, New York Airways, Inc., Flight 972 crashed on Newark International Airport while attempting an emergency landing. The flight had just taken off from the airport and was at an altitude of 1,200 ft about 1 mile to the east when one of five blades broke and separated from the tail rotor. As a result, severe vibrations in the tail rotor assembly caused the tail rotor gearbox and rotor assembly to separate from the aircraft when it had descended to about 150 ft above the ground. Without a tail rotor to maintain its stability, the helicopter entered a rapid nosedown, right turn to the ground. Of the 18 persons on board, 3 passengers were killed; 10 passengers and 3 crewmembers were injured seriously.

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1. FACTUAL INFORMATION

1.1 History of Flight

On April 18, 1979, a New York Airways Sikorsky S61-L helicopter (N618PA) was being operated as Flight 972 from Newark International Airport, New Jersey, to La Guardia Airport, New York. Flight 972 departed gate 21 at Newark International Airport at 1820 1/ with 3 crewmembers and 15 passengers on board. The first officer was flying the aircraft from the left cockpit seat; both flightcrew members were qualified as captain. At 1821:37, the flight was cleared for takeoff, to climb to 1,400 ft, and to depart eastward to La Guardia Airport. Visual meteorological conditions prevailed at the airport, and the wind was 010° at 15 kns.

At 1822:56, Flight 972 climbed to 1,200 ft and was about 1 mile east of the airport when there was a loud bang, similar to an explosion, followed by severe vibrations in the flight controls. The cockpit voice recorder (CVR) revealed that at 1823:04 the first officer transmitted, "Mayday, mayday, mayday, New York Five is landing on the runway." At 1823:10, the captain transmitted, "I don't know what we've got but we're going to make an emergency landing at Newark." These transmissions were not recorded on the Newark air traffic control tower tape recording, since the first transmission was made on the company frequency and the second was cut out by another transmission. At the same time, the flight attendant cautioned the passengers to insure that their seatbelts were fastened tightly. She later instructed the passengers to remove their eyeglasses.

At 1823:16, the local controller transmitted, "New York Five (Flight 972), you got a problem?" When the first officer responded that they had a control problem, the local controller stated, "Okay you set it down anywhere you want as long as it's not on twenty nine." At 1823:23, the first officer stated, "Yes we're going straight ahead, sir." The captain had instructed the first officer not to make any turns.

The flightcrew stated that as soon as they heard the explosion and felt the vibrations they knew there was a problem with the tail rotor. At first, they thought the drive shaft had failed. The first officer stated that his first instinct was to enter autorotation 2/ and land immediately. However, since that maneuver would have increased rotor rpm and rate of descent, he elected not to autorotate. Additionally, since the maneuver would have intensified the severe vibration, he believed he would not be able to control the aircraft. Since the helicopter was controllable, the crew elected to reduce rpm, slow the airspeed, and attempt to land on the airport.

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

2/ A term for a flight condition during which no engine power is supplied and the main rotor is driven only by the action of the relative wind. It is a means of landing safely after engine failure or certain other emergencies.

According to the first officer, as Flight **972** approached the airport on a heading of about 360°, the airspeed was about **60** kns and the rotor rpm was below **100** percent. The first officer stated that he planned to land beyond the threshold of runway 22R, or near runway **29.** Witnesses who first saw Flight **972** near the east boundary of the airport at **an** altitude of 500 ft stated that the helicopter was "swaying from side to side," and "the tail moved erratically from side to side." (See Appendix D.) As the helicopter passed through about **150** ft, an object—described by witnesses as "a suitcase, or a body or a mailbag--fell from the left side of the aircraft, and the helicopter's flightpath and attitude changed abruptly. Simultaneously, the helicopter began to turn, according to witnesses, from 90° to **270°;** one witness described a turn of two revolutions. The witnesses reported both right and left turns. According to the first officer, at **150** ft the vibrations ceased.

As the helicopter began to turn, its nose went down 35° **to 60'** and it descended. At that point, the first officer lost all directional control and the aircraft would not respond to rudder pressure. It struck the ground in a nosedown attitude on the right side **of** the fuselage, it bounced once and fell on its left side. There was no fire or explosion. The helicopter crashed on the grassy area adjacent to runway **22L** and came **to** rest on the west edge of runway 22L, about **610** ft from the threshold.

At 1823:39, the **CVR** had recorded two loud noises. ,At 1823:41, the first officer exclaimed, "All right," and at 1823:43 the captain stated, "No tail rotor." **As** the aircraft descended in a right turn, the captain closed the fuel levers for both engines. Just before the helicopter hit the ground, the first officer attempted to flare it by applying back pressure to the cyclic and pulling up on the collective. The latter maneuver did raise the nose slightly. The first officer stated that the indicated airspeed was about **60** kns when the captain declared, "No tail rotor." Also, before that event, the helicopter's rate of descent never exceeded **600** to **800** fpm. He could not estimate the helicopter's rate of descent near impact, but he stated that its forward speed over the ground was low. The **CVR** recorded the loss **of** electrical power at 1823:47.

The accident occurred during the hours of daylight at position **40'** 42'N latitude and 74° **10'W** longitude. The elevation of the accident site was about **10** ft mean sea level.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	**0**	**3**	**0**
Serious	**3**	**10**	**0**
Minor/None	**0**	**2**	**0**

1.3 Damage to Aircraft

The aircraft was destroyed.

1.4 Other Damage

Not applicable.

1.5 Personnel Information

The crewmembers were properly certificated and qualified for the flight. (See Appendix B.)

1.6 Aircraft Information

The aircraft was certificated and maintained in accordance with Federal Aviation Administration (FAA) requirements. (See Appendix C.) There were no recent maintenance entries in the aircraft logbook related to the tail rotor assembly, tail rotor gearbox, or any tail rotor blades. The crew reported no unusual vibrations during the three previous flights on April 18.

Flight 972's gross takeoff weight was 16,989 lbs, 1,000 lbs of which was jet A fuel. The maximum allowable gross takeoff weight was 19,000 lbs. Flight 972's center of gravity was 265.3 in. which was within limits.

1.7 Meteorological Information

Surface weather observations taken at Newark International Airport by National Weather Service personnel before and after the accident were:

1751 5,500 ft scattered; visibility -- 30 mi.; temperature -- 58°F; dewpoint -- 20°F; wind -- 340° at 17 kns; altimeter setting -- 30.04 inHg.

1826 7,500 ft scattered; visibility -- 30 mi.; temperature -- 56°F; dewpoint -- 21°F; wind -- 360° at 16 kns; altimeter setting -- 30.06 inHg.

1.8 Aids to Navigation

Not applicable.

1.9 Communications

No communications difficulties were reported.

1.10 Aerodrome Information

Newark International Airport, elevation 18 ft mean sea level, has four hard surfaced runways. The New Jersey Turnpike parallels the airport on the east side. Flight 972 declared the emergency at Port Newark, an area east of the turnpike which contains docks, a container loading facility, container storage and semitrailer parking areas, and several warehouses. (See Appendix D.)

1.11 Flight Recorders

Flight 972 was equipped with a Fairchild cockpit voice recorder. The recorder was not damaged. The tape was transcribed, and the quality of the recording was good. (See Appendix E.)

There was no flight data recorder nor was the recorder required.

1.12 Wreckage and Impact Information

Although Flight 972 approached the airport on a northerly heading, the helicopter crashed on a heading of 242°. A 23-ft-long by 8 in.- deep gouge marked the initial impact point. There was little indication of forward movement. Flight 972 struck the ground in a nosedown upright attitude; the nose section, the right landing gear, and right side of the fuselage hit first. The helicopter rolled over and came to rest on its left side. The impact site was about 35 ft west and 1,000 ft beyond the threshold of runway 22L.

The tail rotor gearbox and the attached tail rotor assembly were found about 410 ft west of the main wreckage; an 8-in. section of the tail rotor control rod was found 650 ft west of the main wreckage; two blade skin sections were found about 500 ft west of the main wreckage; and a 35-in. outboard section of one of the tail rotor blades was found 5,100 ft south of the main wreckage site. All other major components were found at the main wreckage site.

The bottom of the fuselage from the nose aft to fuselage station (FS) 391 was crushed upward about 9 in. to waterline (WL) 70. Both main landing gears had separated aft and upward. The tail cone fuselage structure from FS 493 aft was relatively undamaged. The engine cowlings, engines, transmission fairings, and transmissions were not damaged.

Except for one main rotor blade, which separated about 40 in. from its attach point, the remaining four main rotor blades were attached to the rotor hub. The blades showed no evidence of either preimpact distress, damage, or failure. The main rotor head assembly was in good condition, and there was no evidence of any preimpact damage or failure. There was no evidence of any preimpact damage or failure to the control system between the cockpit and the main rotor stationary swashplate and from the rotating swashplate to the rotor blades.

The fuel, hydraulic, and electrical systems showed no evidence of preimpact damage.

The power train, from the engine high speed shafts into the main transmission, and the main transmission were intact. There was no evidence of any internal failure in the transmission, and the main rotor shaft was undamaged.

The tail rotor drive shaft was intact from the main transmission through the intermediate gearbox — which was not damaged -- to the laminated coupling located at the input to the tail rotor gearbox. This input coupling had separated in torsion when the tail rotor gearbox separated.

The tail rotor gearbox separated from its mount on the tail pylon, and three of the four attachment lugs separated from the gearbox. The left aft attachment lug separated from the tail pylon but remained attached to the gearbox. Except for the lug separations, the housing was in good condition. The left aft attachment bolt was found **30** ft behind the tail boom at the impact site. When examined, the gearbox rotated freely and the chip detector plug was clean.

The tail rotor assembly consisted of five rotor blades, labeled red, black, blue, yellow, and white and attached to the tail rotor hub. The outboard **35** in. **of** the black blade was found about **5,100** ft south of the crash site. The remaining four tail rotor blades were attached to the tail rotor assembly. The tail rotor blades which remained attached to the hub had been bent, scraped, and broken during impact; there was nq indication of preimpact failure. The black blade fracture surface appeared to be a progressive-type fatigue failure through the blade spar.

1.13 Medical and Pathological Information

A review of the flightcrews medical records revealed no evidence of medical problems that might have affected their performance.

The three crewmembers received serious compression fractures of the lower back, multiple abrasions, contusions, and lacerations. In addition, the captain received a moderate cerebral concussion.

Although seatbelts and shoulder harnesses were installed on the cockpit crew seats, neither pilot was wearing his shoulder harness. The flight attendant's seatbelt was fastened securely.

The three passengers who were killed were seated in seat rows **3** and **4,** which were in the area above the main landing gear. Two of these passengers received cervical fractures while the third suffered a hemothorax. Injuries to the other passengers included fractures of the lower extremities, a fractured skull, fractured ribs, multiple abrasions, contusions, and lacerations.

1.14 Fire

Although fuel spilled and formed puddles around the wreckage, there was no fire. The first officer attributed the absence of fire after impact to the fact that the battery switch and fuel had been shut off just before impact.

1.15 Survival Aspects

The accident was survivable. The cabin area near rows 3 and 4 was damaged most severely; the floor and sidewalls were distorted. Only three of the occupied passenger seats remained intact and all of those seats showed some degree of distortion. All of the remaining occupied seats were either totally or partially separated from the sidewall attach points and legs. Typical failures involved the separation of the tubing from the seat frame support structure. The failures were usually located adjacent to the point at which the seat leg attachment bracket was welded to the seat frame support tubes. The failures were usually accompanied by the separation of one or both fuselage sidewall attachment points.

The captain's seat remained attached to the floor with the top of the seatback tilted to the right. The rear of the seat pan was cracked and the seat pan support tube and corner brace were separated from the vertical tubes of the seatback. The first officer's seat remained attached to the floor. The right corner brace was bent slightly.

The captain's feet were trapped in the wreckage. In addition, he suffered a severe back injury and had to be extricated by rescuers. The first officer also suffered a back injury but was able to escape through a broken cockpit window.

. The passengers reported that the flight attendant's announcements reassured them and that there was no panic before or after the impact. Most of the passengers in the forward compartment were trapped under debris and other passengers. Only one man was able to free himself from the area. At least three passengers, who were seated in the rear of the cabin, were able to open the right rear window and escape without assistance.

Although four passengers reportedly took the brace position just before impact, the flight attendant did not specify or recommend the brace position as a preimpact, emergency procedure. The company had no procedure that required the flight attendant to instruct the passengers to assume a brace position.

The crash/fire/rescue (CFR) efforts were coordinated by the Port Authority of New York and New Jersey Airport CFR organization. The airport units arrived at the accident site within 2 minutes of the alarm; the fire station was about 1/2 mile away. Units from the Newark fire and police departments also responded with 15 CFR units and 12 ambulances.

1.16 Tests and Research

1.16.1 Metallurgical Examination of Black Tail Rotor Blade

The tail rotor blade was composed of a leading edge spar, honey-combed interior and an aluminum outer skin which was bonded to both the spar and the honeycomb.

Fatigue cracks were found through significant portions of both the leading edge spar and the bonded skin of the separated tail rotor blade. No defects which could have started the fatigue were found at either origin.

The black tail rotor blade separated about 15 in. from the butt end of the blade. An area of disturbed paint extended along a line from a point on the trailing edge at the fracture to a point near the leading edge at the tip of the blade. The inboard surface contained a similar, but smaller, area of disturbed paint. (See Figure 1.) The inboard portion of the blade had been damaged substantially after the outboard portion separated from it.

The fracture on the outboard portion of the blade was examined with the aid of a bench binocular microscope. Markings indicative of preexistent fatigue cracking were found in both the blade spar and skin. The cracking in the spar began at the outboard corner of the aft face of the spar. (See Figure 2.) The fatigue extended through 90 percent of the spar cross-section before the spar failed.

Fatigue in the skin appeared to initiate from multiple origins in an area on the bonded, interior surface of the skin. Fatigue propagated away from the origin area in opposite directions toward both the leading and trailing edges. The skin fatigue crack extended around the leading edge out to the area indicated by the arrow T in Figure 2. The skin fatigue in the other direction extended to a point about 1 1/2 in. from the leading edge. The total length of the skin fatigue crack was less than 2 in.

An area of skin at the fracture had been gouged on the outboard face of the blade near the leading edge. The gouge was about 0.2 in. long. (See Figure 2.) A portion of the bond material beneath the gouge had been squeezed out from between the skin and spar, indicating that the skin had been gouged after separation of the blade.

The area of the spar in which fatigue originated is shown in Figure 3. A dark stain was found in the immediate origin area. The most probable location of the origin would be at the center of this stain area as shown by the arrow in Figure 3.

The radius of the corner of the spar near the origin was about 0.009 to 0.011 in. The radius could not be measured precisely because of the uneven surface of the spar. The engineering drawing for the spar specifies that the radius at this location must be between 0.025 in. and 0.050 in. (See Figure 4.)

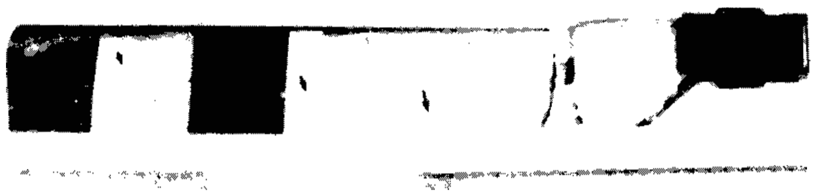


Figure 1. Overall view of the outboard face of the fractured tail rotor blade. (Arrows denote line of disturbed paint.)

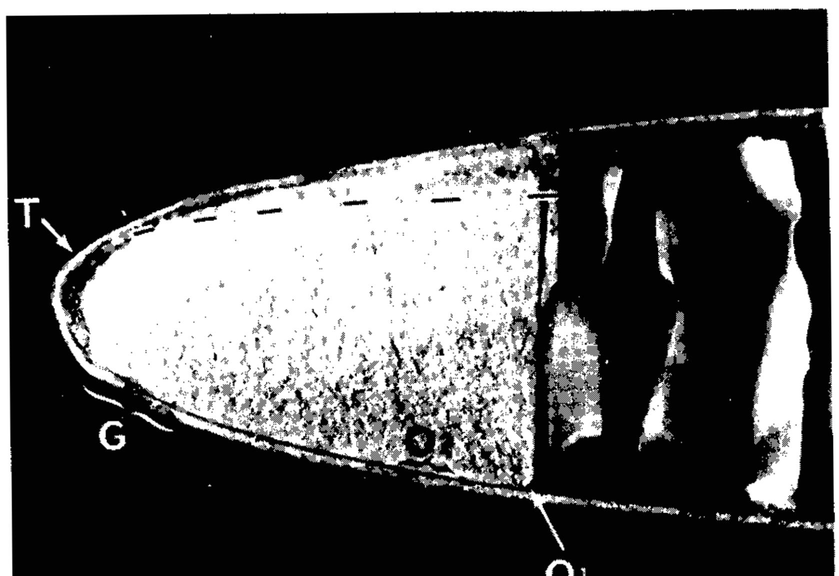


Figure 2. Fracture at the leading edge of the blade. Arrow "01" indicates the fatigue origin in the spar and bracket "02" the origin area in the skin. Arrow T indicates the terminus of the portion of the skin fatigue crack which propagated toward the leading edge. Bracket G indicates a gouge in the skin (X3 approximate).

Figure 3. High magnification visible light photograph of the origin of the fracture on the spar (arrow). X400

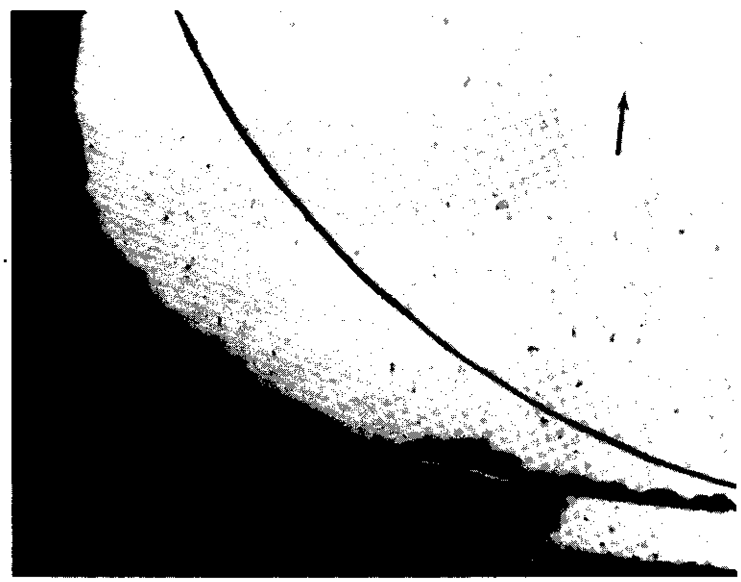


Figure 4. Spar section adjacent to the fracture. Arc represents a 0.025 in. radius. Arrow points in the direction of the leading edge. (X200, as polished)

In order to determine if the radii on the aft surface of the spars on the remaining tail rotor blades met specifications, a section was cut from each of the blades at locations similar to the fracture location on the black blade. Only the outboard radius of the white blade spar appeared to be within the specified limits. The inboard radius on the leading edge spar on the white blade and both radii on the leading edge spars of the remaining three blades were irregular and below the 0.025 specified minimum radius.

The following is a list of the approximate material hardness values obtained from sections cut from the various spars:

Blade	Average Rockwell "E" hardness number[3/]
Black	93.5
Blue	94.5
Yellow	95
Red	94
White	91

1.16.2 Metallurgical Examination of the Tail Rotor Gearbox and Gearbox Mounting Plate

The tail rotor gearbox was bolted to the tail boom through four mounting flanges. Three of the gearbox mounting flanges had fractured and at the the fourth flange, the left aft position, the barrel nut had become disengaged from the bolt. All fractures in the mounting flanges of the gearbox appeared typical of overload separations. (See Figure 5.)

The bolt from the left aft flange position was bent slightly, and the threads were damaged extensively, particularly those threads closest to the bolt head. (See Figure 6.) Most of the thread appeared to have been damaged by transverse contact directly on the crowns of the threads, which resulted in widened crowns and compression bulging of the thread flanks. (See Figure 7.)

Two portions of one of the threads contained a different type of damage. The threads in these areas had been deformed in a direction away from the bolt head by heavy contact along the flank. One of these areas is indicated by bracket W in Figure 7. The second such area was on the same thread approximately 90° around the bolt.

3/ The ASM Metals Handbook, Volume 1, lists an average hardness of Rockwell "E" 85 to 97 for 6061-T6 aluminum (specified material for spar).

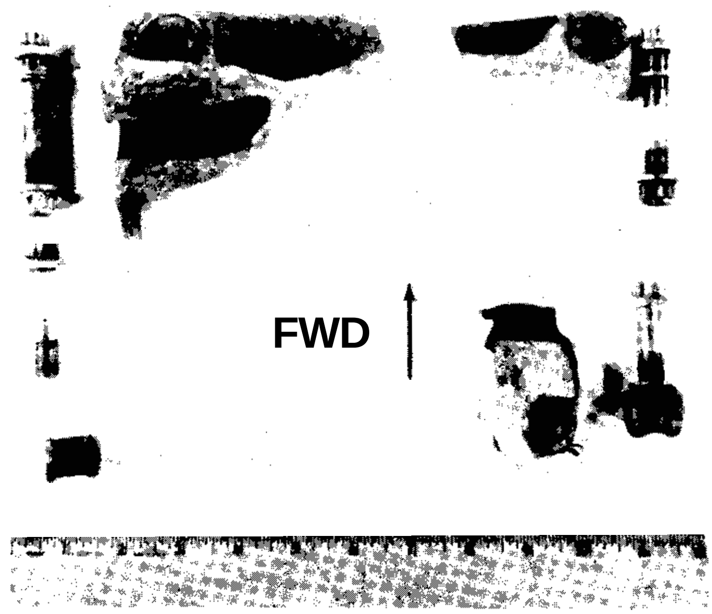


Figure 5. Tail rotor gearbox mounting bolts and nuts with fractured pieces of the gearbox mounting flanges.

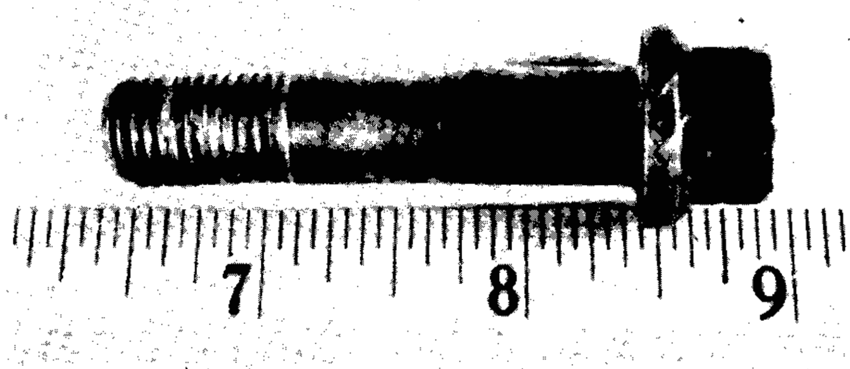


Figure 6. Left aft gearbox mounting bolt (lower left bolt in Figure 5). Note that this bolt in Figure 5 contains tape on the shank just below the head which was used to maintain identification.

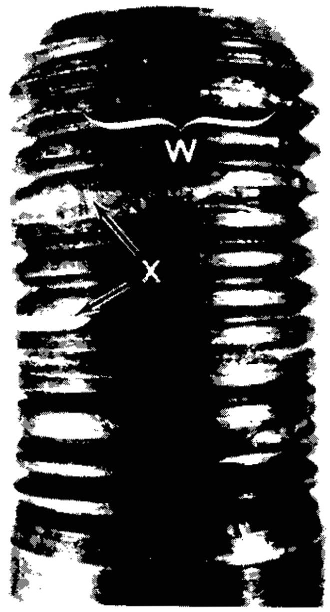


Figure 7. Thread damage on the left aft bolt. "W" denotes area containing deformation. "X" denotes areas where metal deposits were found. (X4)

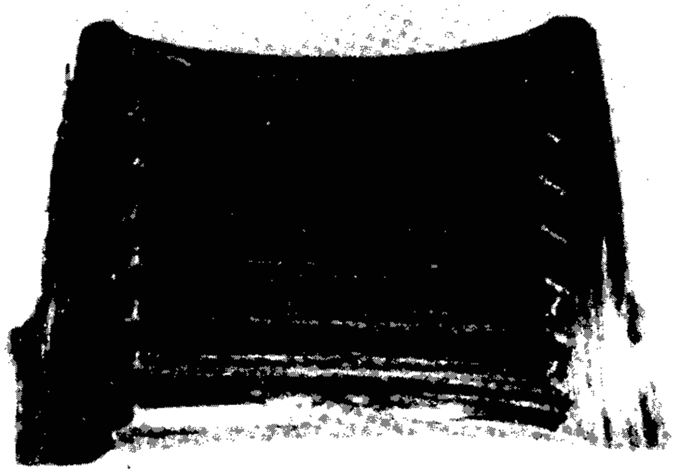


Figure 8. View of interior threads of the left aft barrel nut. (X4)

Foreign metal deposits were found in some areas between the threads. (See Arrow X Figure 7.) This material was rich in aluminum rather than iron, the bolt material. This indicated that the source of the foreign material was probably the aluminum gearbox mounting plate. About 225° of the thread root circumference closest to the bolt shank also contained shallow cracks.

Examination of the threads of the barrel nut from the left aft position of the gearbox mounting plate revealed only one damaged thread. This thread was the one which would be closest to the bolt head when assembled. The thread was deformed in the direction of the bolt head, apparently by hard contact with the thread flank. (See Figure 8.)

Both the left aft bolt holes in the gearbox and in the mounting plate had been deformed into an oval shape. (See Figure 9.) Most of the damage was on the left and right sides of the holes. The damage to these bolt holes appeared to have been produced by relative motion between the gearbox and mounting plate in the left and right directions while the bolt was in place.

1.16.3 Additional Testing of Black Rotor Blade

The outboard section of the blade was tested for evidence of bonding voids and to determine the bonding characteristics between the blade skin and the core. The blade was x-rayed, and coin tap and peel tests were conducted. No discrepancies were noted except in the areas, immediately adjacent to the fractured surface.

In another test, a new tail rotor spar was notched intentionally about 14 in. from the blade attachment holes on the outboard spar. The blade assembly was assembled using the notched spar and subjected to a single combined edgewise/flatwise load level accelerated beyond a normal cruise level in order to initiate a fatigue fracture from the notch.

Once the crack was initiated, the blade assembly was subjected to a spectrum of loads which were similar to loads which would result from a New York Airways operation. The spectrum of loads was repeated every 3 aircraft hours. At the end of each 3-hour block, the blade was inspected ultrasonically. This procedure was repeated until a complete fracture occurred. The fracture occurred about 54 hours after crack initiation.

Additional testing of similar blade assemblies are underway. However, the current 6-hour inspection requirement imposed by the FAA for blade assemblies with more than 1,200 hours appear adequate based on the findings of the first test.

1.17 Other Information

1.17.1 Emergency Procedure For Tail Rotor Malfunction

The Sikorsky Aircraft S-61L Flight Manual Part 1, Section III, Emergency Procedures, states:

3

Figure 9. View looking forward of the left aft bolt holes in the tail rotor gearbox (bracket "1") and the gearbox mounting plate (bracket "2"). Also visible is the barrel nut hole (bracket "3"). (X2/3)

Tail Rotor Malfunction

The most probable type of tail rotor malfunction is a drive system malfunction whereby tail rotor rpm and thrust are lost. This may be caused by fracture of the shaft, coupling, or gearbox, or separation of the tail rotor assembly from the helicopter. Tail rotor separation from the helicopter is usually caused by severe vibration that has been induced by the fracture of a rotating component. A drive system malfunction is the most difficult type to cope with as it is accompanied by the loss of the rotating disc area that normally acts as stabilizing fin in forward flight. In most cases, extended flight is not possible after tail rotor malfunction and a sudden malfunction at high speeds may produce violent aircraft response. If excessive vibration or unusual noise is noted in the tail section during forward flight, airspeed should be immediately reduced to the best autorotational speed, as this may be an indication that a tail rotor malfunction is imminent. An early indication of a tail rotor malfunction is the loss of directional control. When a drive system malfunction in which loss of thrust occurs is experienced, the main rotor torque will yaw the helicopter to the right. The rate and amount of yaw will be governed by the amount of power applied and the airspeed at the time of the malfunction. The yaw tendency can only be reduced by an immediate reduction in power. Extended flight with a nose right yaw is not possible and it is recommended that a full autorotation be entered immediately upon detection of a tail rotor malfunction. Landing without tail rotor thrust is considered hazardous. Autorotation may occur with a nose right sideslip of greater than 45° and a greater rate-of-descent than normal autorotation. This will require a modified side flare to reduce the ground speed to as near zero as possible.

Tail Rotor Malfunction During Flight

Tail rotor malfunction which results in loss of tail rotor thrust during flight will be indicated by a loss of directional control accompanied by a rotation of the nose to the right. The first and most important step is to regain directional control by an immediate reduction of power to the main rotor. Do not attempt to extend flight by pulling power. Immediately perform the following procedures:

1. Reduce power to the main rotor and establish a glide at 65 to 75 knots IAS to regain directional control.
2. Choose the least hazardous landing site.
3. Maintain directional control by lateral movement of the cyclic control stick.
4. Alert passengers and fasten seatbelts.
5. Wheel brakes - OFF.
6. Tail wheel - LOCKED.
7. Make final approach into the wind if possible.
8. Shut down both engines.

APPROACH AND LANDING WITH TAIL ROTOR INOPERATIVE.

1. Accomplish moderate flare at approximately 100 feet to reduce rate-of-descent and airspeed by using aft cyclic.

2. Level off at 30 feet.

3. Collective pitch lever — Increase to cushion landing and touch down in a level attitude.

4. Collective pitch lever — Slowly reduce collective pitch to minimum after ground contact and move cyclic stick slightly forward.

5. As soon as helicopter is firmly on the ground, wheel brakes AS REQUIRED.

6. Secure helicopter.

1.17.2 Inspection Procedure

The tail rotor assembly and tail rotor blades were inspected by New York Airways, Inc., maintenance personnel in accordance with the procedures and at established time intervals.

Sikorsky Aircraft Service Bulletin (SB) 61B15-24B, Tail Rotor Blade Shank Reinforcement Brackets Inspection, required a visual inspection "prior to 1st flight of each day or every **6** hours, whichever occurs first;" SB61B15-1D Tail Rotor Blade Inspection for Cracks, required a visual inspection "before 1st flight of each day and every 6 flight hours." The inspection required, in part:

(1) Clean blade with a clean, dry cloth.

(2) Sight along blade on both sides and inspect entire length of skin for cracks, paying special attention to the inboard 32 in. of the blade.

(3) If a crack is suspected, strip paint in area **of** suspected crack.

Warning: If a crack is found in a blade, remove the blade immediately.

On April 18, the tail rotor blades were inspected twice at New York Airway's facility at John F. Kennedy International Airport. The first inspection was made in the morning before the helicopter departed on its first flight. The second inspection was completed about 45 minutes before the accident. Each inspection was completed according to established. procedures by two mechanics and a maintenance supervisor. The mechanic who conducted the second inspection stated that he wiped each blade individually. He paid particular attention to the white painted areas for cracks, dents, and bulges. He observed no defects.

1.17.3 Black Tail Rotor Blade History

The black tail rotor blade, part No. S6115-30001-044, serial No. 61V-10042-10842, was manufactured on April 28, 1977, and shipped to New York Airway's, Inc., on May 12, 1977. The blade had been installed and removed from two other New York Airway's, Inc., helicopters before being installed on the accident helicopter on September 23, 1978. The total time on the blade was 2,444:01 hrs since manufacture.

A review **of** the blade maintenance records did not reveal any indication of service difficulties, vibrations, or cracks. According to the manufacturer, the blade was the first S-61 tail rotor blade to fail under these circumstances.

1.18 New Investigation Techniques

None

2. ANALYSIS

Failure of the Black Tail Rotor Blade

The accident sequence was initiated by the failure of the 35-in. section of the black tail rotor blade. The blade broke off when Flight 972 was about 1 mile south of the accident site. The loss of the blade caused the loud noises and severe vibrations reported by the crew. A progressive-type fatigue crack propagated across 90 percent of the cross-section of the leading edge spar. The spar fatigue originated at the outboard side of the aft face of the spar. The tail rotor blade failed as a result of preexisting fatigue cracks in the outboard edge spar and skin. The fatigue began in the spar before it started in the skin. (See Appendix G.)

The metallurgical examination of the spar disclosed that fatigue began in a radius which did not meet engineering drawing specifications. However, radii in similar locations on the other four tail rotor blades were of similar dimensions. Therefore, the improper radius alone did not initiate the fatigue crack in the spar. Since no one item could be singled out as the primary initiator of the fatigue crack, the Safety Board was not able to determine exactly what initiated the fatigue failure of the tail rotor blade.

The Safety Board attempted to estimate the number of cycles necessary for the skin to crack as the fatigue progressed from the origin to the terminus. Through the use of a scanning electron microscope, the number of cycles was determined to be about 150,000. The tail rotor blade would experience 150,000 cycles of stress in less than 2 hours of flight time if each rotation of the tail rotor blade introduced one cycle of stress. (The tail rotor operates at 1,243 rpm.) It is not known whether the fatigue propagated during all stages of flight.

Obtaining an estimate of the number of striations proved to be more difficult for the spar fatigue crack than for the skin fatigue crack. In many areas the striations were unresolvable or obscured, probably as a result of rubbing between the mating fracture surfaces. For these reasons an estimate of the number of striations in the spar was not made.

Initially, the spar crack may have propagated as a result of lower frequency stress cycles, such as startup/shutdown loads, or when the blade was exposed to higher stresses during maneuvers. In this case, the amount of flight time for total crack extension would be significantly more than 2 hours.

The provisions of the various Service Bulletins had been adhered to, and all of the required inspections had been made and documented by New York Airways, Inc. However, since the section of the spar which failed was completely enclosed in an aluminum envelope, a fatigue crack which had begun in the spar could not be seen during a visual inspection and would not necessarily distort the skin of the blade at a specified time before failure. The examination of the failed blade indicated that the fatigue crack in the skin was present when the second inspection was made, about 45 minutes before the accident. At that time the crack was less than 2 in. long. Since a typical fatigue crack is tight, the crack in the skin of the blade was probably difficult to see, especially since it was less than 2 in. long. Consequently, while inspection procedures were adequate for the detection of certain faults in a tail rotor blade, they did not assure detection of a skin fatigue crack which resulted from the fatigue failure of the spar.

Failure of the Tail Rotor Gearbox and Assembly

The failure of the rotor blade generated an unbalanced force in the rotor system which caused three of the four tail gearbox attachment lugs to fail under static overload. As a result, the tail rotor and tail rotor gearbox assemblies separated fram the helicopter and complete directional control was lost.

The fourth tail gearbox attachment lug remained intact and in place on the tail gearbox housing. The lug hole and the corresponding hole in the gearbox attachment plate were elongated. The bolt was bent slightly, and the threads damaged badly. The barrel nut for the bolt was in place in the pylon and relatively undamaged. The lack of damage to the threads of the barrel nut indicated that the bolt had lost torque and partially backed out of the nut before the tail rotor gearbox and assembly separated from the tail pylon.

The damage to the threads of the left aft bolt could only have occurred if the bolt had at least partially backed out and if the gearbox was capable of motion relative to the mounting plate at the bolt hole. Relative motion between these components could occur only if another of the gearbox attach points had been broken. Therefore, the left aft bolt threads were damaged after the black tail rotor blade separated and at least one of the gearbox attachment lugs had been fractured. The Safety Board could not determine if the bolt had partially backed out before the failure of the tail rotor blade, or as a result of the failure of the blade.

Any adverse effects of a partially backed out attachment bolt would probably be shared equally by all five tail rotor blades. However, the investigation did not indicate any additional fatigue failures in the remaining blades. The Safety Board concludes that a loose tail rotor gearbox was not the cause of the fatigue initiation.

Flightcrew Decisionmaking

The flightcrew was certificated properly and was qualified for the flight. They had received the off-duty time required by regulation, and there was no evidence that medical or physiological factors might have affected their performances.

When the tail rotor failed, Flight 972 was at an altitude of about 1,200 ft, and 1 mile east of the airport over the congested Port Newark area. The crew recognized immediately that the tail rotor system had malfunctioned. As a result, they had two options -- either to return to the airport or to autorotate. They chose to attempt an emergency landing at the airport for several reasons: (1) The airport, which was the only suitable forced-landing area, was less than a mile away. (2) Once the appropriate emergency checklist had been accomplished and the helicopter had been slowed to between 60 and 70 kns, the crew was able to control the helicopter in spite of the severe vibrations. (3) There was no yaw to indicate a drive shaft malfunction and resultant loss of tail rotor rpm. (4) They knew that if autorotation was initiated the rotor rpm would increase to about 105 to 108 percent, which would place a greater load on the tail rotor system. (5) The first officer believed that the additional rpm would have increased the vibration and caused complete loss of control of the helicopter. (6) There was no suitable landing area below them.

Autorotation may have resulted in an earlier landing, but it may also have caused the gearbox to separate sooner because of increased rotor rpm's and increased stress on the gearbox mounting flanges. Also, it probably would have resulted in a landing in a hazardous area of Port Newark. Therefore, because extended flight was not necessary, the helicopter was not yawing significantly, and the flightcrew had no reason to believe that the gearbox mounting flanges would fail in the short period of time needed to reach the airport, the Safety Board concludes that the flightcrew's decision to make a controlled emergency landing at autorotational speed was an appropriate decision.

When the tail rotor gearbox and assembly separated from the tail pylon, the loss of the 180-lb unit caused the center of gravity to change immediately to **260** in., **or** within **2** in. of the forward limit, and resulted in an abrupt nasedown change in pitch. Simultaneously, directional control was lost when the tail rotor fell off. The helicopter began an immediate descending right turn **of** about 270° before hitting the ground. Although all directional control was lost, had the aircraft been at a higher altitude, it might have been flared sufficiently to cushion the touchdown on the airport; however, touchdown still would have been accomplished from a spiraling turn. Although the first officer did attempt to slow the rate of descent by initiating a flare before impact, the nosedown attitude. and **the** low altitude of the helicopter prevented him from raising the nose high enough to cushion the touchdown.

Survivability

The accident was partially survivable because the g forces were within the range of human tolerance and because of the minimal disruption of the fuselage structure and minimal reduction in the occupiable volume of the fuselage. (See Figure 10.) The fatalities and severe injuries were caused by failures of passenger restraint systems under comparatively high vertical g forces in the forward portion of the helicopter.

The vertical g loads, which exceeded the 4-g minimum certification standard imposed for helicopter seats by Civil Aeronautics Manual **7.260,** were variable, but they probably were in the range of **15** g's in the forward cabin. This estimate is based on the damage to the helicopter. In the 15-g range of impact force, the U.S. Army Crash Survival Design Guide, TR **71-22,** October **1971,** indicates that only moderate injuries should be expected when adequately restrained persons encounter impact forces of this magnitude.

The impact forces caused the sidewall tiedowns to fail and separate, which caused many of the seats to separate **or** become loose. The sidewall tiedowns probably separated when the fuselage sidewalls flexed on impact. Controlled crash testing of various aircraft and helicopters conducted by the National Aeronautics and Space Administration indicate that fuselage sidewalls flex several inches in crashes similar to that experienced by Flight **972.** As a result, seat pan attachments will separate from the fuselage. Once the seat pan has separated, it will swing through a wide arc until the passengers come into contact with other objects. Extensive sidewall flexion was probable in the forward fuselage of Flight 972 because the landing gear lateral supports were attached to the outside of the fuselage in this area. The forces on the landing gear would have been transmitted directly to the fuselage walls and flooring, causing localized flexion of the walls and distortion of **the** floor. As a result, the passengers in the forward cabin were probably thrown when the sidewall tiedown structure failed.

The Safety Board is aware that similar inadequacies in design requirements for passenger seats for general aviation aircraft exist. Therefore, improvements in general aviation aircraft crashworthiness have been made a special safety objective. We expect to include improvements in design requirements for helicopter passenger seats in this objective.

Four occupants reported taking a brace position before the initial impact. These occupants received less severe upper torso and head injuries than other occupants seated in the same rows who did not assume a brace position. For example, one passenger who was in the brace postion received only minimal injuries although both occupants seated beside him were killed. Another passenger, a former **U.S.** Army helicopter crewman, assumed a brace position and received only minimal head and upper torso injuries while the passenger next to him received a serious open, depressed frontal skull fracture.

The emergency procedures and the passenger briefing cards ,should have specifically required the flight attendant to instruct passenghrs to assume the standard brace position, which would have reduced the possibility of serious injuries during the emergency landing.

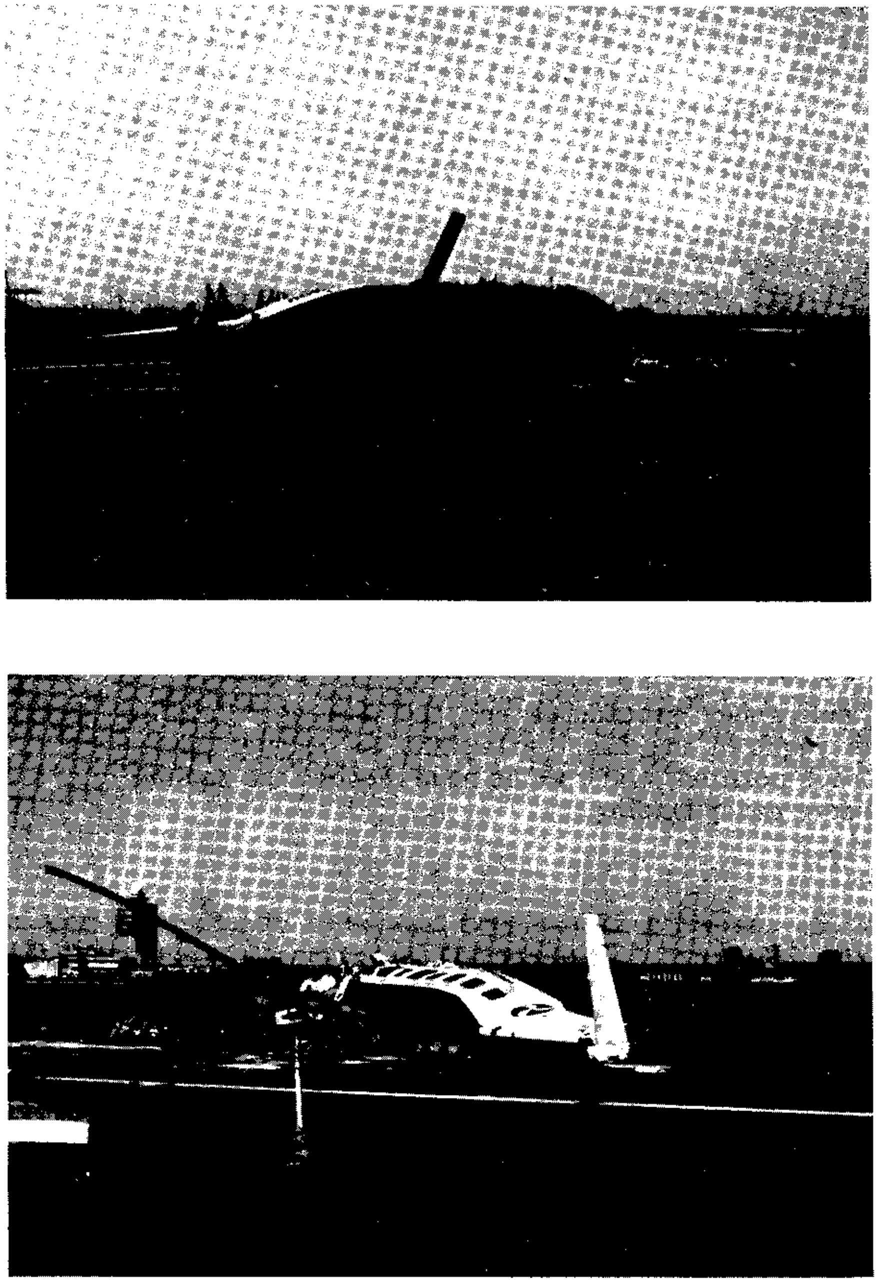


Figure 10. Views of structural damage.

3. CONCLUSIONS

3.1 Findings

1. The flightcrew was certificated properly and qualified for the flight.

2. The aircraft was certificated, maintained, and dispatched in accordance with Federal Aviation Regulations and approved maintenance procedures.

3. The black tail rotor blade broke when the helicopter was at 1,200 ft and about 1 mile east of the airport.

4. The blade broke because it had been weakened by a preexisting fatigue crack in the leading edge spar and the blade skin about 35 in. from the tip of the blade.

5. The fatigue crack had propagated through 90 percent of the leading edge spar and through less than 2 in. of the skin covering the spar.

6. The spar fatigue origin was located at the outboard corner radius of the aft face of the leading edge spar.

7. The area where the fatigue originated in the spar was covered by the aluminum envelope, which made it impossible to detect'during a visual inspection.

8. The fatigue crack began in the spar before it started in the skin.

9. The fatigue crack in the skin developed over a period of at least 2 flight-hours before the failure of the tail rotor blade.

10. The fatigue crack in the spar developed over a period significantly longer than 2 hours.

11. The fatigue crack in the skin of the tail rotor blade was probably present when the blade was inspected 45 min. before the accident. However, the crack could not be detected readily.

12. The visual inspection procedures in effect were not adequate to detect the fatigue crack in the skin which developed from a fatigue crack in the spar.

13. The loss of the 35-in. section of the tail rotor blade did not cause a loss of directional control.

14. The loss of the 35-in. section of the tail rotor blade generated unbalanced forces, which caused three of the four tail rotor gearbox housing attachment flanges to break from static overload and a portion of a thread of the bolt in fourth flange to fail.

15. The failure of the attachment flanges permitted the gearbox and assembly to separate from the tail pylon, which caused the loss of directional control and an abrupt nosedown pitch attitude change.

16. The helicopter's low altitude when directional control was lost and the nosedown pitch change made further control of the helicopter impossible.

17. The flightcrew's decision to attempt an emergency landing at Newark International Airport was an appropriate decision under the circumstances.

18. The accident was partially survivable.

19. The failure of the seat support and tiedown structure contributed to the number of fatalities and serious injuries.

20. Fewer serious injuries would have occurred if the passengers had taken a "brace" position.

3.2 Probable Cause

The National Transportation Safety Board determines that the probable cause of this accident was the separation of the tail rotor assembly and gearbox from the aircraft at an altitude which made further controlled flight impossible. The rotor assembly and gearbox separated because of severe vibrations in the rotor assembly which were induced by the loss of a tail rotor blade due to fatigue failure.

Contributing to the severity of the passengers' injuries were the seat failures which occurred when the deceleration forces exceeded the relatively low design strength of the FAA-approved seats, and the lack of guidance on a passenger brace position for emergency landings.

4. SAFETY RECOMMENDATIONS

On April 19, 1979, the National Transportation Safety Board issued the following safety recommendations to the Federal Aviation Administration:

> Withdraw the airworthiness certificates of Sikorsky S61 helicopters until a means of detecting potential tail rotor blade failures can be devised and implemented. (Class I—Urgent Action) (A-79'25)
>
> Notify foreign operators of Sikorsky S61 aircraft of this action. (Class I—Urgent Action) (A-79-26)

The FAA, in response to these recommendations, issued an Airworthiness Directive on April 20, 1979, requiring S-61 operators with PN 6115-30001 and 6117-30001 series tail rotor blades installed to (1) perform a one-time dye penetrant inspection of the inboard 32-in. section of each blade and of the tail rotor gearbox housing attachment lugs before further flight, (2) conduct a visual inspection of each blade skin every 6 flight-hours for evidence of cracks, (3) ultrasonically inspect each blade spar and skin for cracks within 6 flight-hours, and (4) repeat the ultrasonic inspections every 6 flight-hours on tail blades which have accumulated 1,200 or more flight-hours.

Results of the initial ultrasonic inspections revealed no additional spar cracks. The metallurgical examination of the failed tail blade and skin did not disclose any abnormalities in the material used or the manufacturing techniques. The manufacturer is conducting additional fatigue testing of "notched' tail blade spars to determine propagation rates. The initial results of these tests indicate that the 6-hour ultrasonic inspection interval required by AD is adequate to detect potential tail rotor blade defects.

Also as a result of this investigation the Safety Board has issued the following recommendations to the Federal Aviation Administration:

> "Establish a research project to determine the optimal brace position for various seat designs and seating configurations on aircraft used in passenger-carrying operations. (Class II, Priority Action) (A-79-76)
>
> "Issue an Air Carrier Operations Bulletin on the basis of this study requesting principal operations inspectors to insure that the training of crewmembers includes information on the appropriate passenger brace position for specific' aircraft configurations during potential crash landings. (Class II, Priority Action) (A-79-77)
>
> "Issue an Air Carrier Operations Bulletin requiring principal operations inspectors to instruct their assigned air carriers to describe the appropriate emergency brace position on the 'passenger briefing card and to require that preflight briefings include a reference to the proper brace position. (Class II, Priority Action) (A-79-78)"

BY THE NATIONAL TRANSPORTATION SAFETY BOARD

/s/ JAMES B. KING
Chairman

/s/ ELWOOD T. DRIVER
Vice Chairman

/s/ FRANCIS H. McADAMS
Member

/s/ PATRICIA A. GOLDMAN
Member

/s/ G.H. PATRICK BURSLEY
Member

September 27, 1979

5. APPENDIXES

Appendix A

INVESTIGATION AND HEARING

Investigation

1. The Safety Board was notified of the accident about 1900 e.s.t. on April 18, 1979. The investigation team was dispatched to the scene and working groups were established for operations/air traffic control/maintenance records, witnesses, airworthiness, cockpit voice recorder, and metallurgy.

Participants in the on-scene investigation were the Federal Aviation Administration, New York Airways, Inc., Sikorsky Aircraft Division of United Technologies, International Association of Machinists, Air Line Pilots Association, and Association of Flight Attendants.

2. There was no public hearing or depositions.

Appendix B

Personnel Information

Captain Lee G. Richard

Captain Lee G. Richmond, 49, was employed by New York Airways, Inc., on February 24, 1964. He held Air Line Transport Pilot Certificate No. 644892 with the following ratings: Rotor helicopter BV-107-7, VFR SK-61, commercial airplane single-engine land, and sea glider. His first-class medical certificate was issued April 16, 1979, and he was required to carry corrective lenses for near vision.

Captain Richmond qualified as captain on the Sikorsky 61-L helicopter on November 30, 1970. He passed his proficiency check on January 26, 1979; his last line check on June 8, 1978, and recurrent ground training on January 10, 1979. The captain had flown about 12,000 hrs in helicopters, about 2,500 hrs of which were in S-61 helicopter. During the months of March and April 1979, the captain had flown 42 hrs and 34 hrs, respectively. He had flown 2.8 hrs in the 24 hrs before the accident.

First Officer Lesley G. Carter

Lesley G. Carter, 54, although qualified as captain, was flying as first officer on this flight. Captain Carter was hired by New York Airways, Inc., in October 6, 1952. He held Air Line Transport Pilot Certificate No. 1043280 with the following ratings: Rotor helicopter VFR S-55, S-58, S-61, Vert-44, V-107-2, commercial airplane single-engine land, instruments flight instructor, and CFI rotor instruments.

His first-class medical certificate was issued November 7, 1978, and he was required to wear correcting glasses for near and distant vision while exercising the privileges of his airman's certificate.

Captain Carter was qualified as captain on the Sikorsky 61-L helicopter. He passed his proficiency check on January 16, 1979, his line check April 12, 1979, and recurrent ground training on January 10, 1979. The captain had flown a total of 14,500 hrs, about 5,000 hrs of which were in S-61 helicopters.

During the months of March and April 1979, he had flown 31 hrs and 12 hrs, respectively. In the 24 hrs before the accident, he had flown 2.8 hrs.

Flight Attendant Lannie Chevalier

Flight Attendant Lannie Chevalier, 29, was hired by New York Airways, Inc., in their accounting department on October 30, 1972. She transferred to operations and completed flight attendant training January 30, 1973. She completed recurrent ground training on February 8, 1979, and her line check on February 28, 1979. Flight Attendant Chevalier had flown 1,800 hrs. At the time of the accident, she had been on duty 2 hrs 43 min, 1 hr 45 min of which was flight time.

Appendix C

Aircraft Information

Sikorsky S-61L, N618PA (S/N 61426), was owned by the General Electric Credit Corporation of Georgia and operated by New York Airways, Inc. It was certificated and maintained according to procedures approved by the FAA. At the time of the accident, the helicopter had accumulated about 12,376 hrs of flight time.

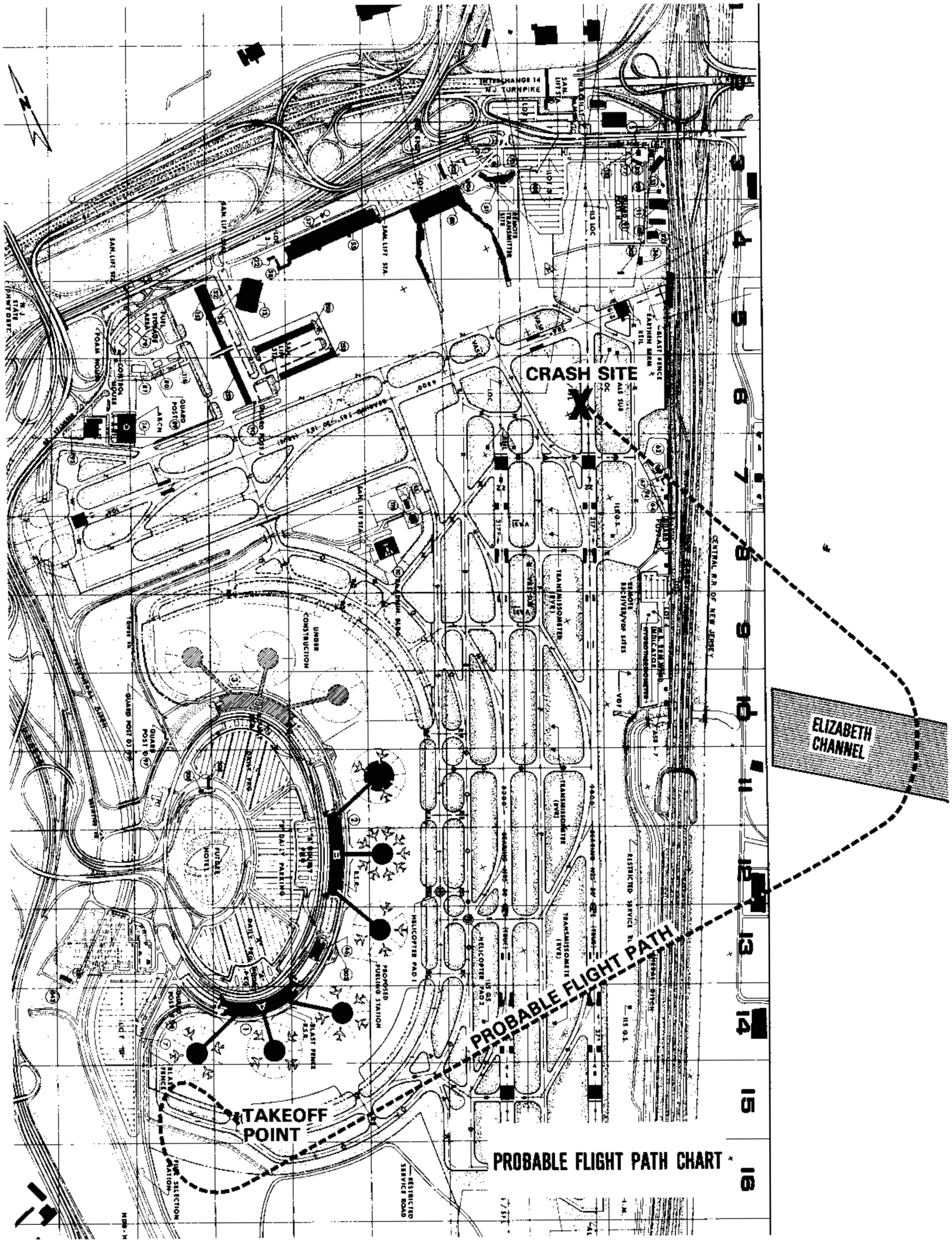
The helicopter was equipped with two General Electric CT 58–140–2 engines.

Position	Serial No.	Time Since Overhaul (Hrs)	Total Time (Hrs)
Engine No. 1	295059C	41:36	7,387:20
Engine No. 2	295065C	2,429:12	7,713:47

Total times for components, serial numbers, and times since overhaul are as follows:

Component	SIN	TSO (Hrs)	T.T (Hrs)
Tail Gearbox	A–16–754	2846:23	12610:28
Int. Gearbox	A–15–762	639:03	6931:46
Tail Rotor Head	A–12–774	61:02	8719:57
Red Tail Blade	61V10020-10930	-	1818:28
Yellow Tail Blade	61V10029-10840	-	3256:18
Blue Tail Blade	61V10001-10800	-	2623:13
Black Tail Blade	61V10042-10842	-	2444:01
White Tail Blade	61V9973-10772	-	2692:54
Yellow Main Damper	72	1079:04	9046:50
Red Main Damper	A073-01757	237:00	2793:08
Blue Main Damper	92	1012.23	8591:41
Black Main Damper	MH–294T	166.43	7277:18
White Main Damper	SP–001	317:23	8842:36

Appendix D


CRASH SITE
ELIZABETH CHANNEL
PROBABLE FLIGHT PATH
TAKEOFF POINT
PROBABLE FLIGHT PATH CHART
INTERCHANGE 14 N.J. TURNPIKE
CENTRAL R.R. OF NEW JERSEY
UNDER CONSTRUCTION

Appendix E

TRANSCRIPT OF A FAIRCHILD COCKPIT VOICE RECORDER S/N UNKNOWN REMOVED FROM THE NEW YORK AIRWAYS S61 HELICOPTER APRIL 18, 1979

LEGEND

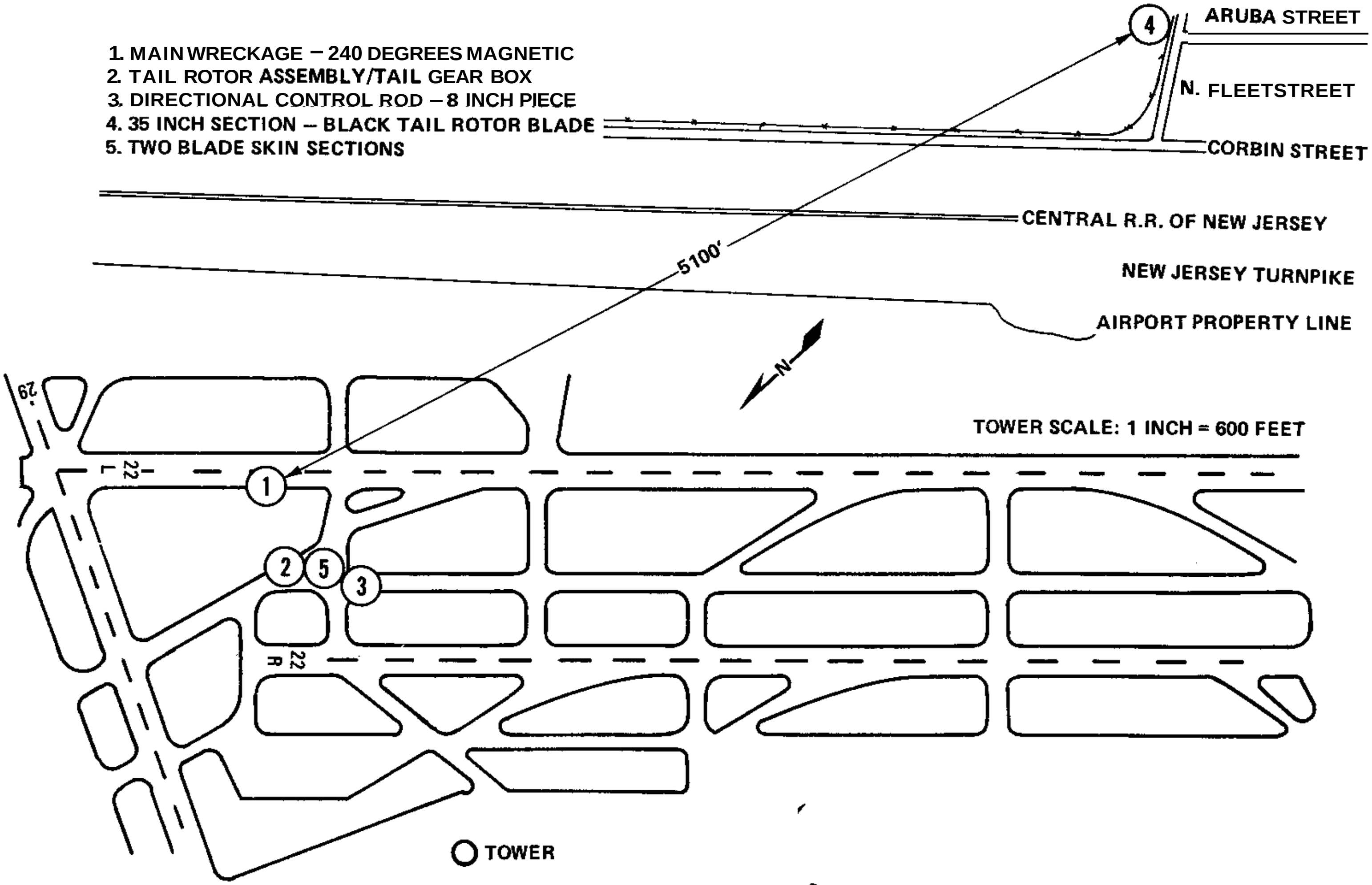
CAM	Cockpit area mircrophone voice or sound source
RDO	Radio transmission from accident aircraft
IC	Intercom
TWR	New York Tower
CO RDO	Company Radio
PA	Public address system
*	Unintelligible word
#	Nonpertinent word
%	Break in continuity
( )	Questionable text
(( ))	Editorial insertion
—	Pause
Note:	Times are expressed in eastern standard time.

AIR-GROUND COMMUNICATIONS		INTRA-COCKPIT	
TIME & SOURCE	CONTENT	TIME & SOURCE	CONTENT
1821:30 RDO-2	Tower New York number five is ready for and eastbound departure, we have the guy on final		
1821:33 TWR	Five eastbound approved, wind zero one zero at one five		
1821:34 RDO-2	Thank you		
	%	1821:53 CAM	((Sound of takeoff))
	%		
1821:58 CO RDO	% to New York		
1822:39 TWR	Okay four you're going to La Guardia why don't you go a little bit to your left until you get north of the approach to twenty nine, you got traffic out there about three and a half miles final for two nine, it a Navaho		

AIR-GROUND COMMUNICATIONS		INTRA-COCKPIT	
TIME & SOURCE	CONTENT	TIME & SOURCE	CONTENT
1822:47 RDO-2	Okay that's for New York seven right ah five rather		
1822:49 TWR	Ah fine yes sir do you want to go to fourteen		
1822:52 RDO-2	Yeah will go to fourteen		
1822-54 TWR	It's approved		
RDO-2	Thank you		
1822:56 RDO	((Sound of severe tape flutter))	1822:56 CAM	((Loud bang followed by increase in sound level))
1823:04 RDO-2	Mayday (hello), mayday, mayday, New, York five is landing on the runway	1C-1	Put her down Les
PA	Mayday, Mayday, mayday		Ladies and gentlemen, please be sure that your seatbelt's tightly fastened
TWR	Stand by one		

AIR-GROUND COMMUNICATIONS		INTRA-COCKPIT	
TIME & SOURCE	CONTENT	TIME & SOURCE	CONTENT
1823:10 RDO-1	I don't know what we've got but we're going to make an emergency landing at Newark		
1823:15 1C-°	Okay you've got it		
1823:16 TWR	New York five you got a problem		
		1C-1	Don't try and make any turns straight ahead
1823:17 RDO-2	Yes we do have a problem ah it's a control problem	PA	Stay calm
1823:21 TWR	Okay you set it down anywhere you want as long as it's not on twenty nine		
1823:23 1C-1	Straight ahead Les		
RDO-2	Yeah we're going straight ahead sir		
1823:27 TWR	Straight ahead, okay		
1823:28 1C-1	No # turns		

AIR-GROUND COMMUNICATIONS		INTRA-COCKPIT	
TIME & SOURCE	CONTENT	TIME & SOURCE	CONTENT
1823:30 XXX	* * a three sixty	PA	(Please) take any glasses off and put them in your pocket just as a safety precaution
1823:31 TWR	Seventy two yes sir a three sixty to the right		
1823:32 XXX	Okay		
1823:37 1C-1	No # turns now	1823:39 CAM	((Two loud noises, ka bong followed by a fluttering sound))
1823:41 1C-2	All right		
1823:43 1C-1	No tail rotor		
1823:47 CO RDO	The other crew is gradual inbound to La Guardia		
RDO	((Loss of electrical power))	CAM	((Sound of loss of electrical power))



WRECKAGE DISTRIBUTION CHART

Appendix G

Metallurgical Information

A single, fatigue origin was found on the spar, indicating rather low initiating stresses since no significant stress risers were found. The fatigue initiation area of the skin contained a large number of origins, indicating higher initiation stresses. Thus, fatigue cracking in the spar probably resulted in higher loads being transferred into the skin.

The first **3/8** in. of the spar fracture was darker in appearance (under visible light) than the remaining fatigue fracture in both the spar and skin. The darker appearance in general indicated older age because of longer time to weather or mechanical rub, or both.

The location of the skin origin area was about halfway between the spar origin and the terminus of the darker area of the spar. This is consistent with propagation of the spar fatigue followed by initiation of the skin fatigue from multiple origins because of the load transferred to the skin from the cracked portion of the spar.

The fracture surface in the vicinity of the spar origin was extremely smooth and few striations were evident. The fracture surface in the vicinity of the skin origin area was rougher and individual striations were evident. This indicated that the propagation rate and propagation stresses were greater at the skin origin than at the spar origin.