SA-404 FILE NO. 1-0005

AIRCRAFT ACCIDENT REPORT

LOS ANGLES AIRWAYS, INC.

SIKORSKY S-61L, N303Y

PARAMOUNT, CALIFORNIA

MAY 22,1968

ADOPTED: DECEMBER 18, 1969

NATIONAL TRANSPORTATION SAFETY BOARD
Bureau of Aviation Safety
Washington, D. C. 20591

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NATIONAL TRANSPORTATION SAFETY BOARD
DEPARTMENT OF TRANSPORTATION

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SYNOPSIS

About 1751 P.d.t., May 22, 1968, Los Angeles Airways Flight 841, a Sikorsky S-61L, N303Y, crashed and burned at Paramount, California. The flight was en route from the Anaheim, California, Heliport, which serves Disneyland, to the Los Angeles International Airport, All 23 persons aboard the aircraft were fatally injured and the aircraft was destroyed by impact and fire after impact.

The overall evidence, coupled with numerous tests and failure analyses, indicates that the accident sequence began while the aircraft was in cruising flight about 2,000 feet above the ground and about 2 miles east of the accident site. The black, yellow, and blue main rotor blades, followed by the red and white blades, underwent a series of extreme over-travel excursions in their lead/lag axis.

During the extreme excursions, the yellow main rotor blade over-traveled in the lead direction and, as a result, its pitch change control road was subjected to downward and rearward loading many times its design operating strength. Under these forces, the rod became detached at its lower trunnion end where it is normally secured to the attachment lugs of the main rotor rotating swashplate. With this separation, the blade went out of control and struck the right side of the aircraft diagonally across the baggage loading door. The other four main rotor blades then struck and penetrated both the aft and forward portions of the aircraft. The blade strikes destroyed the main rotor blades and separated major portions of the fuselage, including the tail rotor pylon and tail rotor assemblies. The aircraft, completely uncontrollable, crashed in a near-vertical descent.

The initial malfunction, failure, or condition which precipitated the accident sequence was probably a loss of main rotor blade damper integrity caused by either failure of the black main rotor blade damper, or a loss of effective damper action by the white main rotor blade damper. An important portion of the black damper and a portion of the black blade horizontal hinge pin to which the damper attaches were not recovered.

The Safety Board determines that the probable cause of this accident was the loss of main rotor blade damper integrity due to either a failure of the black blade damper or a loss of effective damping action by the white blade damper. This resulted in uncontrolled excursions of the main rotor blades in their lead/lag axis, an overload detachment of the yellow main rotor blade pitch change control rod and destruction of the structural integrity of the aircraft by blade strikes. The precise reason for either of the possibilities for the loss of damper integrity is undetermined.

1. INVESTIGATION

1.1 History of the Flight

Los Angeles Airways Flight 841 of May 22, 1968, was a scheduled air carrier passenger flight from the Anaheim, California, Heliport, serving Disneyland, to the Los Angeles International Airport, a distance of about 25 miles. It was the return portion of Flight 441/841, one of approximately 30 flights made daily by the airline between the two points.

Flight 441 departed the Los Angeles International Airport at 1723 1/ under visual flight rules in clear weather and proceeded to Anaheim, where it landed about 1737. The crew made normal radio contacts and the passengers noted nothing unusual about the flight. At Anaheim, the aircraft was not serviced and the crew gave no indication of any difficulties with the helicopter. Because of the short turnaround, 3 minutes, the pilots remained in the cockpit and only the No. 2 engine was shut down while the 20 passengers boarded for Flight 841.

Flight 841 departed Anaheim at 1740 and gave its departure time and a wind advisory for other company pilots to Los Angeles Airways Flight Control. At 1742, it gave Fullerton Tower its route of flight and, at 1747, a position report. Both reports were routine.

About 1750, the pilots of Los Angeles Airways Flight 742, which was inbound to Anaheim, saw Flight 841 as the two aircraft passed about one-half mile apart, flying in opposite directions. The pilots of Flight 742 reported the flights passed approximately over Pioneer Intersection, located about 3 miles east of the crash site and Flight 841 was at an altitude of about 2,000 feet. They reported that at this time Flight 841 was on a normal westerly heading, on course, and appeared completely normal. Within 30 to 60 seconds later, a third pilot aboard Flight 742 and Los Angeles Airways Flight Control heard a radio transmission which was subsequently determined to be, "LA, we're crashing, help us." Flight 841 could not be contacted thereafter.

Witness observations indicate that during the final 2 to 2-1/2 miles of flight, Flight 841 descended from its cruising altitude of about 2,000 feet, as observed by Flight 742, to between 600 and 800 feet above the ground. They indicate that during this descent, the aircraft slowed and there seemed to be an erratic action of the main rotor blades. Several witnesses then saw the aircraft turn left from a westerly to a southwesterly heading. One witness, with a helicopter maintenance background, stated that this turn was a violent yaw to the left of nearly 90^{o}, and when it occurred, he saw one main rotor blade extremely out of tract on the high side.

1/ All times herein are Pacific daylight, based on the 24-hour clock.

Immediately thereafter, there were several sharp sounds from the aircraft and two witnesses saw the main rotor blades striking the front and back areas of the aircraft fuselage. Parts identified as pieces of main rotor blades, fuselage, and the tail rotor were seen to separate from the aircraft and the helicopter immediately fell in a near-vertical trajectory and crashed. Fire followed the ground impact.

Ground witnesses saw no large aircraft in the area of the accident when it occurred. Two helicopters, one apparently Flight 742 and the other a small Bell, were seen but neither was close to Flight 841. Witnesses said the weather was clear with unlimited visibility at the time and place of the accident.[2/]

1.2 Injuries to Persons

There were three crewmembers and 20 passengers aboard the aircraft. All received fatal injuries.

1.3 Damage to Aircraft

The aircraft was destroyed by impact and the ensuing ground fire.

1.4 Other Damage

Falling pieces of the aircraft damaged several buildings and a truck. There were no injuries to persons on the ground.

1.5 Crew Information

The captain, copilot, and flight attendant were properly certificated and qualified for the operation involved. (For detailed information see Appendix A.)

1.6 Aircraft Information

The aircraft was a Sikorsky S-61L, N303Y, with serial No. 61060 and company No. 44. It was manufactured by the Sikorsky Aircraft Division of United Aircraft Corporation[3/] in June 1962 and an Airworthiness Certificate for the helicopter was issued to Los Angeles Airways on August 18, 1962. (For other aircraft information see Appendix B.)

The main rotor head installed on the aircraft had accumulated a total of 9,102 hours,[4/] of which 1,175 were logged since overhaul. The overhaul period for the head was 2,400 hours.

2/ The accident occurred in full daylight.
3/ Present corporate designation.
4/ Nearest full hours are used in this report.

The black main rotor blade damper assembly was installed on the rotor head on May 17, 1968, with 6,680 total hours, of which 430 hours were accumulated since overhaul. The piston component of the assembly, which was life limited to 3,400 hours, had accumulated 3,342 hours.

The maximum allowable gross takeoff weight for helicopter N303Y was 19,000 pounds. Computations indicate that, at departure from Anaheim, its gross weight was 16,809 pounds and at the time of the accident, about 16,700 pounds. At 16,500 pounds gross weight, the forward center of gravity limitation was 255.0 inches aft of the reference datum 5/ and the aft limitation was 273.3 inches, At 17,000 pounds gross weight, the forward limitation was 256.0 inches aft of the reference datum and the aft limitation 278.7 inches. Based on this information, and depending on the seat position of an infant passenger who was either in seat 5 or 24, the center of gravity was 264.83 or 263.63 aft of the reference datum, respectively. Both were well within limitations.

The aircraft was serviced to a total fuel load of 1,000 pounds of JP-4 fuel prior to departure for Anaheim and it was not serviced at Anaheim.

1.7 Meteorological Information

At the time and place of the accident, the weather conditions were clear, with visibility more than 15 miles. Full daylight existed.

1.8 Aids to Navigation

Not involved.

1.9 Communications

Communications with Flight 841 were normal until the final emergency transmission from the flight indicating that it was crashing.

1.10 Aerodrome and Ground Facilities

Not involved.

1.11 Flight Recorders

The aircraft was not equipped with either a flight recorder or a cockpit voice recorder. Neither is required on transport helicopters by current Federal Aviation Regulations.

1.12 Wreckage

The main body of the aircraft, including both engines, the main rotor head, pieces of main rotor blades, and most of the fuselage struck the

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5/ Reference datum is 267.4 inches forward of the centerline of the main rotor hub.

ground in a near-vertical fall. It struck the ground in a nosedown attitude on its left side. At impact, the main rotor was turning very slowly, if at all, and there was little power development from either engine. Fire followed the ground impact and caused extensive additional damage to this wreckage.

The other major portions of the aircraft fell along a groundpath approximately 1,100 feet wide and 2,100 feet long to the east, or back along the final flightpath of the aircraft. These portions included the aft part of the fuselage, tail rotor drive shaft, the pylon assembly including the intermediate gearbox, tail rotor assembly, pieces of main rotor blades, and pieces of cockpit and fuselage structure. These pieces and components showed no evidence of fire. By virtue of their distances from the main wreckage, it was evident that they had separated in flight along the final flightpath of the aircraft.

The main rotor system of the S-61L helicopter has five main rotor blades (see Attachment 1, Explanatory Diagram, Main Rotor Head). Looking down on the head, they turn counterclockwise. Clockwise they are color coded for identification, as the red, black, white, yellow, and blue blades. The blades are identical and, in consonance with the fully articulated design concept of the rotor head, are free to move individually and independently, within controlled limits, about three axes, lead/lag, flap, and pitch.

A vertical hinge pin for each blade permits the lead and lag motion and a damper and bumper assembly cusions and limits the motion. A horizontal hinge pin for each blade allows its blade to flap or move up and down about the hinge point in addition to the normal flexibility of the blade itself. A third hinge, the pitch hinge, permits each blade to change pitch from control inputs. A pitch change control rod for each blade is the medium through which blade pitch changes are made. It attaches at its lower end to the rotating swashplate to which control inputs are applied from the stationary swashplate. At its upper end, the pitch control rod attaches to the pitch change horn of its respective blade.

The rotating swashplate is coupled to the stationary swashplate by a bearing on a ball. This design allows the rotating swashplate to rotate, tilt, and move up and down. The rotating swashplate is attached to the rotor hub by a scissors and rotates with the rotor hub. The hub is driven by engine power through the main gearbox. The rotor head can also continue to rotate in the event of a loss of engine power enabling an autorotational landing. The stationary swashplate is kept from rotating by another scissors attached to the main gearbox.

Control inputs for the rotor blades are applied to the stationary swashplate by a hydraulically operated dual servo system. Inputs to the stationary swashplate cause the rotating swashplate to move up and down, or tilt, thereby changing the pitch of the rotor blades collectively, or in tilt with each rotation of the rotor hub. Since pitch changes of the rotor blades are made through the medium of the pitch change control rods, if a pitch change rod were to become detached for any reason, its blade would become uncontrollable in its pitch axis and, indirectly, in its flap axis about its horizontal hinge pin. Normal operating loads on the pitch control rods are 250 to 300 pounds.

Wreckage examination revealed that all five main rotor blades were broken into two or more pieces. Sections from each consisting of either blade and cuff or of the cuff alone remained attached to the rotor head. Numerous pieces of the red, black, white, and blue blades were found well back along the wreckage path among pieces from the cockpit, aft fuselage, and pylon assembly structure. However, the pieces found farthest back along the wreckage path were pieces of the yellow blade. These were a 13-foot section inboard of the tip and several blade pockets. Pieces of the yellow blade were found some 2,100 feet from the main wreckage and some 800 feet farther than other dense pieces of aircraft structure.

The nature of the damage to the red, black, white, and blue blades was similar. However, with respect to the yellow blade, the damage was distinctively different.

Examination of the red, black, white, and blue blades revealed that all four had struck and penetrated the cockpit and the aft fuselage of the aircraft. By matching of blade with cockpit and fuselage structural damage, it was determined that, except for the red and black blade strikes in the cockpit area, which were reversed, all of the other blade strikes on the aircraft were in the order of blade rotation and the penetrations were progressively deeper into the aircraft structure. Strikes by the white and blue blades separated the aft fuselage and tail rotor pylon. One strike in the cockpit was in the area of the radio control panel and another penetrated the area of the engine controls.

Of particular importance, there was no evidence found to indicate that the yellow blade inflicted any of the strikes in either the cockpit or aft fuselage areas of the aircraft. This, therefore, indicated that the yellow blade was neither in its rotational sequence position between the white and blue blades nor in the rotor disc at the time the blade strikes were made.

The yellow blade was broken into five major pieces. In one piece from the middle portion of the blade, there was an upward curved bend and, near its cuff end, there was a sharp upward bend of about 10°. On the upper surface of a section from near the middle part of the blade, there were paint marks. There were also numerous wavy longitudinal scratches found on the top surface of another blade section from near the hub end. On an outboard section was a distinct rivet pattern impressed in the upper surface of the blade.

The above-described marking and damage were matched with an impression made by an extremely heavy blade hit which ran diagonally forward and downward across the aircraft baggage door. The door was located on the right side of the fuselage below and forward of the main rotor head and just behind the pilot compartment. The curved bend in the mid-portion of the yellow blade matched the curvature of the chine or bottom fuselage line of the aircraft. The rivet pattern found on the blade matched the rivet pattern where a repair patch had been riveted to the bottom of the fuselage. Also, paint found on some pieces of blade matched the paint scheme on the fuselage, and paint in the hit impression on the baggage door matched paint used on the blade. Thus, this overall evidence showed that the yellow blade had hit the baggage door and fuselage area flat with its upper surface. Also, the various distances of the marks on the pieces of the blade from its attachment point to the rotor head showed that the blade was attached to the head when the major blade hit occurred. The longitudinal scratches and gouges on the blade and matching marks on the front of the cockpit area and rotor head components revealed that, after the major strike, the remaining inboard part of the blade was dragged around the front of the cockpit and partially wrapped around the rotor head.

The numerous fractures of all main rotor blades were determined to have been the result of gross overloads. 6/

Inspection of the yellow main rotor blade pitch change control rod disclosed it was detached at its lower or trunnion end where it normally attaches to the rotating swashplate attachment lugs or ears. At this attachment, each end of the trunnion fits into a bearing which is press fitted into a trunnion bearing cap. The trunnion bearing caps are installed in the leading and trailing ears of the swashplate and secured by an upper and a lower bolt installed through each cap and ear. The bolts used are made of steel and the caps and ears are made of aluminum.

The yellow blade pitch change control rod was also broken off near its upper end. The fracture was in the clevis area, just below the clevis which attaches to the yellow blade pitch control horn eyebolt. The shaft portion of the rod revealed no deformation.

Investigation and tests 7/ revealed the yellow blade pitch change control rod was detached at the rotating swashplate end as the result of extremely high loading imposed downward and rearward through the control rod in a manner which broke off the swashplate trailing attachment ear. The fracture occurred through the lower bearing cap bolthole in the ear. The bearing cap remained with the ear. Both parts had been subjected to ground fire and the ear was partially consumed. The bearing cap was attached to the remaining portion of the ear by the upper of the two securing trunnion bearing cap bolts. The lower bolt was missing, although its hole in the cap was intact but elongated and narrowed by heat to the extent that a proper bolt for the hole would not fit through it.

6/ See Section 1.15, Tests and Research, for blade examinations and material tests.

7/ See Section 1.15, Tests and Research, for failure loading analysis.

Examination of the bearing cap revealed it had been pushed out of the ear. The major deformation was in bending in the area of the upper bearing cap bolt. The bearing was missing from the cap; however, curved strike marks were found on the inside rim area of the bearing cap. These marks were found to match the radii and arcs of the inner and outer races of a bearing of the type used in the trunnion bearing cap. 8/

On the leading side of the shaft portion of the yellow blade pitch change control rod there was an interference mark. With the rod positioned to an extreme overtravel lead position, this mark mated with another on the top side of its leading attachment ear of the rotating swashplate.

Examination of the stationary swashplate revealed no major damage and no evidence was found of binding or interference between it and the rotating swashplate. The scissors attaching the stationary swashplate to the main gearbox was intact, but its lower link was bent forward.

On the surface between the leading and trailing ears of the left lateral arm, where one of the control input primary servos attaches to the stationary swashplate, there were several scratches and gouges. 9/ During rotation of the rotor head, the lower attach point for the pitch change control rods passes directly over the primary servo attachment points, with close tolerances between the components of the two attachments.

A number of bearings, such as the one missing from the trunnion bearing cap, are used throughout the aircraft, and, while a number of these bearings or portions of them were found, it was impossible to determine if one was from the yellow bearing cap. All other similar bearing cap bolts were accounted for.

Following examination of the rotor head and associated components at the accident scene, the head was moved to the manufacturer's facilities and mocked up. This was done to study the wreckage in an attempt to document the entire damage in minute detail and to establish a failure sequence analysis. This work took several months.

One of the most significant results of the failure analysis was that the damage indicated that the black, yellow, and blue, followed by the other main rotor blades, had initially undergone extremely large lead/lag overtravel excursions. The excursions were so severe that in some instances one blade had overlapped the one next to it, The work also revealed that in addition to actions already described, the yellow blade experienced high coning and extreme leading and lagging. Some of the significant damage was as follows:

8/ See Section 1.15, Tests and Research.

9/ See Section 1.15, Tests and Research, for identification of these various markings.

1. Horizontal Hinge Pins

The red, black, white, and yellow blade horizontal hinge pins were broken off, All of these hinge pins were recovered with the exception of a major portion of the one for the black blade. Metallurgical examination revealed no evidence of fatigue in any of failures of these pins. All failures were in gross overload in the blade lag direction, or in the direction opposite to rotor rotation.

2. Dampers

The damper piston threaded shafts were broken from the black, yellow, and blue blades. The yellow and blue shafts were recovered and all of the red, yellow, white and blue damper assemblies were recovered for metallurgical examination.

In the case of the black damper assembly, the damper piston-threaded shaft portion was missing. Also missing were the clevis end of the damper assembly, which consists of two trunnion bearings and bearing caps, the damper trunnion, which is held in place by the bearings and caps, and a small portion of the horizontal hinge pin to which the trunnion attaches. Also missing was the nut which secures the trunion to the horizontal hinge pin. The black damper body was recovered but extensive efforts made to find the missing portion and components of the black damper were unsuccessful.

The available fractured end of the black damper piston and the fractured end of the horizontal hinge pin were given metallurgical laboratory examination. Also, the available body portion of the black damper and all four other dampers were examined and given various tests. 10/

All damper cylinder housings exhibited various degrees of damage. The aft outboard sides of the red and white damper cylinder housings were penetrated, exposing the pistons. These dampers had been penetrated by the black and yellow blade horizontal hinge pin stubs when these blades were in an extreme lag position. The blue blade damper exhibited a deep gouge in its aft outboard side, and this gouge mated with the horizontal hinge pin of the red blade in an extreme lag position.

The front end plates of the damper cylinder heads for the red, white, and blue blades exhibited gouge marks. Marks on the outboard side were associated with damage on the red blade spindle ear, and on the inside with the failed blue blade trunnion end. For these contacts to have occurred, the red blade had to be in the extreme lead position and the blue blade in an extreme lag position. The white damper front end marks were matched to damage on the blue blade spindle ear, with the blue blade in the extreme lead position. The red damper had been struck by the black blade horn when the latter was in its extreme lead position.

10/ See Section 1.15, Tests and Research, for detail of damper examination and testing.

3. Pitch Change Control Rod Horns

Physical evidence indicated that the black blade horn pitch change control rod eyebolt boss had contacted the red damper trunnion assembly, causing separation of the black blade horn from the black blade sleeve assembly. Deep gouge marks on the top of the white blade horn in the area of the pitch change control rod eyebolt boss were matched with damage on the black blade trunnion assembly, with the black blade in an overtravel position.

Examination of damage observed on the yellow blade pitch change control rod horn eyebolt showed it was struck by the white blade damper assembly, causing spreading of the white damper trunnion clevis. In order for this contact to have occurred, the white blade would have had to move from a lag to a lead position and then to a lag position of between 0° to 7°. Damage indicated that the yellow blade had reached angles of 24° in upward flap, 20° to 60° in lead, and minus 2° in pitch.

Damage on the blue blade pitch change control rod horn in the eyebolt boss area showed that it had contacted the yellow blade horizontal hinge pin after the hinge pin had been broken. This contact required the yellow blade to be at a high flap angle and a near-neutral lead/lag angle, and for the blue blade to be in a lead position. Damage showed that subsequent lag motion of the yellow blade with the high flap and low pitch angles and lead position of the blue blade resulted in contact between the broken yellow blade horizontal hinge pin and outboard flange of the blue blade pitch change control rod horn. This contact resulted in shearing off the inboard flange of the blue blade horn in an outboard direction. Shearing of the horn then allowed the blue blade sleeve assembly to overlap and rest on the back side of the yellow blade vertical hinge pin and on the top of the yellow blade horizontal hinge pin.

Evidence indicates that the red blade went into a lead direction at low pitch angle and its pitch change control rod horn eyebolt contacted the crotch of the lower plate. This tore out the red blade horn from its sleeve assembly.

The inputs of the manual and automatic flight control systems (AFCS) are applied to the stationary swashplate which can move up and down and through angles of pitch and lateral deflections. Movement of the stationary swashplate is accomplished by the operation of three primary servos connected to the swashplate at three positions around its circumference. Each primary servo is fixed at the lower end to the main gearbox.

The primary servos are hydraulic actuators controlled by mechanical inputs which attach to the pilots' controls. A redundant hydraulic control for the stationary swashplate is provided by the auxiliary servos, which are mechanically in series between the primary servos and pilot controls. Each servo system, primary and auxiliary, is capable of operating the stationary swashplate in response to pilot inputs. Each servo system is independent hydraulically, each having its own fluid reservoir, pump, and plumbing. In normal operation, both servo systems are in operation and each responds to the pilot input which passes through the auxiliary servo to and through the primary servo to the stationary swashplate.

The APCS input is parallel to, but separate from, the pilots' input. It enters the system through the auxiliary servo. The control valve of each auxiliary servo (except altitude) can be magnetically positioned in either direction by electrical signals from the AFCS. It is at this valve that the pilots' input is combined with the AFCS input. However, the AFCS input is restricted to approximately 7.5 percent of the pilots' input capability in pitch, 10 percent in roll, and 5 percent in yaw. 11/ The pilots' control is mechanically attached to the valve, but the AFCS moves a component of the same valve magnetically.

Control1 of the rotor head is accomplished through either or both the auxiliary servo system or the primary servo system. A complex linkage system around the auxiliary servos transfers pilot control movements to the primary servos whether or not the auxiliary servos are in operation. When operation of the stationary swashplate is only controlled by the auxiliary servo, the primary servos become mechanical links to the swashplate. An interlocking pressure sensing system prevents the shutting off of either system when the other has no hydraulic pressure.

The above-described flight control system positions the stationary swashplate, which in turn, through the rotating swashplate, positions the main rotor blades so they function as a disc. There is no individual blade input from the pilot or AFCS through either or both servo systems. The AFCS can be disconnected by either pilot by depressing a disconnect cutoff button switch on his collective control.

Examination of the flight control systems of the aircraft accounted for all major components of the systems. A short piece of control rod, about 2 inches in length, and about 40 inches of control cable were not recovered; however, the breaks on each side of the missing portions were overload failures.

All hydraulic lines of the primary servo flight control system were recovered. The hydraulic reservoir, although empty, showed evidence that fluid had been in it before it was subjected to ground fire. The hydraulic pump was functional and the pressure sensing switch was operable and in the "on" position. The function of this switch is to prevent shutdown of the auxiliary control system if the primary system loses hydraulic pressure. X-ray of the control manifold showed that its valve was in the "on" position.

11/ The civil version of the S-61L helicopter does not have AFCS control input to the collective control valve.

Each of the three primary servos which provide control inputs to the stationary swashplate was X-rayed and their internal components were found in place and saftied. Two of the primary servos functioned normally in checks, without alteration or repair. The third functioned properly after clearly defined crash damage was repaired.

The auxiliary servo flight control system had received greater impact and fire damage than the primary system. However, all hydraulic lines were accounted for, there was fluid in the auxiliary hydraulic reservoir, and the hydraulic pump was functional. The auxiliary servo flight control manifold would not function because all seals were blown due to impact and fire. Its pressure sensing switch was so severely damaged that no functional testing could be performed. The control manifold valve was found in the "on" position.

Study of the cockpit control switch for both the primary and auxiliary systems, which is located on the Captain's collective control, indicated it was in the "both on" position. This would be the position for normal flight.

Because of the bulk of the auxiliary servo housing, the four auxiliary control valves 12/ for roll, pitch, yaw, and collective had to be removed from the servo housing to be X-rayed. The X-rays showed the internal condition of all of the control valves was good, with their components in proper position and secured. The valves were then put back on the servo housing for functional testing. Each of the valves proved functional; however, the mechanical centering required of the valve after control input appeared to be incorrect. Also, when control input from the AFCS was applied to the roll control valve, a "forcible hardover" 13/ occurred. This meant there was insufficient motion in the feedback linkage 14/ to permit the required recentering of the roll channel control. The other control valves responded to the AFCS inputs without forcible hardovers, but mechanical recentering continued to appear off in all of the control valves. This improper condition meant one of three things: that the control valves had not been reinstalled on the auxiliary servo housing in precisely the same position they had been in before removal for X-raying; that the servo housing was distorted by impact; or that, prior to the crash, there was improper adjustment in the feedback

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12/ These valves are also called MOOG valves and dual input valves.

13/ This is a control force to the pilot's cyclic control which the pilot would be unable to overcome. A hardover is a control force within the limits of the AFCS which can be overcome with no difficulty. A hardover is used in pilot training and is induced through a hardover panel in the aircraft for this purpose.

14/ This is also referred to as the sloppy linkage.

linkage which prevented proper recentering of the roll control valve. The first reason was subsequently determined to be the cause of the improper condition. 15/

During the examination of the AFCS, a second electrical fault was discovered. This was an electrical short in the control valve for pitch control input to the auxiliary servo. More specifically, the short was caused by the flow of solder from the "A" pin joint to the frame of the control unit. The unit had been subjected to the postimpact ground fire thereby posing the question of whether the solder flow was caused by heat generated by the ground fire or was the result of a preaccident manufacturing fault, It was subsequently concluded that the most probable reason for the short was the postcrash exposure to fire. 16/

Examination of the engines from N303Y revealed both were developing little or no power at impact. The lack of frictional discoloration on bent compressor blades and the lack of overall damage to the engines indicated they were at or near idle r.p.m. at impact. Examination of the engines, however, showed no evidence of operational distress, malfunction, or failure prior to impact. A lack of metal fusion on the turbine nozzles and turbine showed these parts had cooled for several seconds and, at impact, were below operating temperatures. According to the manufacturer, the engines will cool in a windmilling condition to about 750° F. in about 3 to 4 seconds. The engines will slow to idle from normal cruising power in 10 to 15 seconds.

The engine inlet guide vanes are controlled by an actuator which is scheduled by the fuel control. These guide vanes and the actuator, in the instance of both engines, were closed.

The throttle connection at the fuel control shaft consists of a rack and gear. Movement of the throttle cable repositions the rack which rotates the gear to open or close the throttle. In the case of both engines, each throttle cable was found in the engine shutoff position.

Examination of the main gearbox showed no evidence of operational distress, malfunction, or failure.

1.13 Fire

There was no evidence to indicate in-flight fire was involved in this accident; however, an intense ground fire occurred as the result of ground impact. The Paramount, California, Fire Department was notified and responded to the crash about 1801.

15/ See Section 1.15, Tests and Research, for the manner in which this determination was made.

16/ See Section 1.15, Tests and Research, for the manner in which this determination was made.

1.14 Survival Aspects

This accident was not survivable.

1.15 Tests and Research

Due to the complex nature of this accident and the numerous unexplained factors, after the public hearing the entire rotor head, main rotor blades, and the stationary and rotating swashplates were taken to the facilities of the aircraft manufacturer for further examination and failure analysis. The work involved numerous tests and research requiring over 1 year.

Through metallurgical examination and Barcol hardness tests, it was determined that all critical fractures of the main rotor blades were caused by overload. No evidence of fatigue was found in any of the failures, and metal hardness of the blades was equal to OK in excess of specifications. In addition, it was determined that the yellow main rotor blade was subjected to extreme upward bending near its inboard end and downward bending over its mid and outboard areas. Metallurgical examination of these areas revealed no evidence of fatigue or material deficiencies. The work during this phase also served to further verify that the yellow blade struck the side of the aircraft as previously described. It further verified that the blade was dragged around the front of the aircraft and rotor head, and that the blade did not participate in the blade strikes either in the front or rear areas .of the aircraft.

Because the bolt which secures the bearing cap to the trailing ear of the rotating swashplate where the trunnion end of the yellow pitch change control rod attaches, was missing and the ear broken off, extensive testing was made to determine if the absence of this bolt under normal loading 17/ on the control rod allowed the bearing cap to be pushed out, thereby failing the trailing ear and releasing the pitch control rod.

The test setup simulated a pitch change control rod in its proper position, with the bearing cap at the trunnion end secured by the upper bolt in a normal manner and with the lower bolt installed, but with its nut only finger tight.

Under the first test using this setup, the control rod shaft was not attached and loading was applied directly to the trunnion end. Under a loading of 2,000 pounds, the trunnion support plate deformed but did not fail. The deformation of the bearing cap was not enough to eliminate the looseness of the lower bolt installation.

Under a second test, with the control rod shaft installed on the trunnion end and the rod in a 4° lead position, compression loading of

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17/ As previously stated, normal loading is about 250 to 300 pounds.

7,800 pounds was applied. Under the downward and rearward loading imposed in this manner, there were no failures and the bearing cap, where the loose bolt was installed, was only displaced about 1/8 of an inch.

The fracture at the upper end of the yellow blade pitch change control rod in the area of upper clevis was examined for evidence of fatigue and for proper metal composition and metal hardness. It was determined that the fracture was caused by bending overload, and the material in the fracture area met or exceeded the specifications for hardness, case depth, and material composition.

Laboratory examination of the bearing cap from the broken trailing ear of the yellow pitch change control rod swashplate attachment revealed it was deformed away from the trailing ear in the area where the bearing cap bolt was missing and there was bending in the same direction in the area of the bolt, which remained securing it to the trailing ear. On the inside rim area of the cap, there were several strike marks. One was identified as having been made by the trunnion end of the control rod which normally fits into the bearing of the bearing cap. The other strike marks were matched with the radii and arcs of the inner and outer races of the bearing itself. This indicated the bearing was intact when the strikes occurred, the bearing in fact came out of the cap, and the trunnion came out of the bearing.

The remaining bolt which still secured the bearing cap to the trailing ear was examined. It was found to have been bent under loads in the same direction as those which deformed the cap outward from the ear. The washer installed under the nut was also deformed from the same loads.

The bolthole in the bearing cap for the missing bolt was examined. The hole was intact but elongated by heat from the ground fire to an extent that a proper size bolt for the hole would not go through it, showing that a bolt was not in place when the bearing cap was exposed to the fire.

Laboratory examination of the trailing ear broken from the yellow blade rotating swashplate pitch change control rod attachment showed it had been subjected to intense ground fire. An approximate 120° segment was missing. This portion had been most probably burned away, as both sides of the missing portion showed they were exposed to heat which caused the aluminum to begin to flow. The fracture of the ear through the lower bearing cap bolthole revealed no evidence of fatigue or material deficiency. Also, a section cut out of the ear for testing revealed no fatigue or material deficiency. Metallurgical examination confirmed that the loading which failed the ear was in a trailing or rearward direction. It was also applied to the lower area of the ear. The failure loading was similar to that which acted on the bearing cap.

As previously indicated, there were several scratches and heavy gouge marks found on the surface area between the ears of the left lateral primary servo attachment point of the stationary swashplate. Identification of these marks was important because it was considered possible they indicated that the missing bearing cap bolt had come out and jammed between the rotating swashplate trailing ear of the yellow blade control rod attachment, and the forward or trailing ear of the stationary swashplate left lateral primary servo attachment and, in this manner, caused the failure of the trailing ear.

To identify the marks, a plastic cast was made of them. It was found that the marks matched the trunnion end of the yellow blade pitch control rod with the rod at an angle in an overtravel position in the yellow blade lead direction. With the control rod in this position, marks on the leading ear of the control rod rotating swashplate attachment matched marks on the shaft portion of the pitch control rod. In addition, a bolt of the missing kind could not be positioned in any manner where it could jam between the aforementioned components and could make the scratches and gouges where they were located. Lastly, there was no heavy tear-out on the aluminum components which would be expected if a steel bolt, being much harder, jammed between them.

The four complete main rotor blade damper assemblies and the available piston and cylinder body of the black damper were given laboratory examination and testing. X-rays of the piston assemblies, which contain all the adjustable components affecting their performance with exception of the differential check valve package, disclosed no serious defects. The orifice flow for each piston was measured within limits at 250 p.s.i.g. (pounds per square in. gage) differential pressure. The instroke and outstroke relief valves revealed no significant discrepancies in adjustment, no excessive leakage, and the valves were properly seated. All damper lines, passages, quick disconnects, and orifices were clear of any blockages.

Examinations of the differential check valve packages were made and each was tested for flow and leakage. The results revealed no discrepancies which would cause any problems to normal flight.

The failure of the black main rotor blade damper piston rod was given careful metallurgical examination to ascertain the nature of the failure and whether it was of the proper metal composition and hardness. In addition, five other damper piston rods were intentionally failed to find the approximate loading under which the rod from the accident aircraft should have failed.

Examination of the failure showed the missing threaded end of the damper piston had been torn out by loads applied at a slight angle off the rod axis. The fracture fact evidenced considerable tearing and shearing from near axial tension forces. There was some secondary damage in the form of impressions and gouges on the face of the fracture. Using stereomicroscopic magnifications and electron microfractography examinations, it was determined that there was no fatigue condition in the fracture.

Examination was made for proper metal composition of the remaining portion of the piston rod. Of the six elements composing the rod, all were within percentage specifications. The hardness of the rod in the fracture area was equal to the requirement for the rod, Rockwell value C *42*.

The tensile failure tests performed on the five damper pistons revealed an average breaking load of over 53,000 pounds.

The quick disconnect at the damper fluid reservoir for the hose to the white blade damper assembly was found disconnected and it could not be established if this occurred at impact, during the wreckage movement and examination, or whether it was a condition which existed prior to the accident. Because of this, a test was run to determine if damper fluid would drain out of the damper assembly if the quick disconnect became disconnected in flight prior to the accident. The suspect quick disconnect was installed on a rotor head without blades and the rotor head was run for 1 hour at 100 percent normal rotation with the quick disconnect disconnected. The leakage of fluid from the quick disconnect was found to be insignificant at 6 cc.

Another test was run to determine if, under the most adverse flight condition which would result in the main rotor blades going to the greatest lead position, the pitch change control rods could be made to contact the forward ear of the rotating swashplate pitch control rod attachment. To effect this test, .062 diameter soldering wire was taped to the forward or lead side of a pitch change control rod of a test aircraft. The aircraft was then flown into one-engine-out autorotation at 120 knots to produce the maximum rotor blade lead position. The test result was that the pitch control rod did not go into a lead position sufficient to cause the solder to touch the forward ear.

As previously indicated, a forcible hardover occurred when AFCS control input was applied to the roll control valve of the flight control system. This was attributed to three possible causes: improper repositioning of the valve on auxiliary servo housing after it and the other control valves were X-rayed, distortion damage to the auxiliary servo housing, or an improper adjustment of the valve prior to the accident. To resolve the possibilities, the servo housing was subjected to ultraviolet (black light) to disclose the signature imprint of the valve position prior to its removal. When this was done, it was shown that the valve had been positioned slightly off when it was put back on the servo housing. When the valve was positioned on the housing according to the signature imprint revealed by the ultraviolet light, the forcible hardover-condition was eliminated, although the valve centering was slightly out of proper adjustment. The servo housing was not distorted.

Tests were run to determine if the shorted AFCS pitch control input valve to the auxiliary servo was the result of the postimpact fire or a preaccident manufacturing fault. In the tests, new valves were heated until components of the valves became discolored to approximately the same discoloration of the suspect unit. At this temperature, 350%. , to 400°F., the solder flowed in a similar manner as it did in the suspect unit. Examination of new units also showed that voids did not exist in the potting compound around the soldered joints as those found in the suspect unit. Additional tests also indicated that if the shorted valve existed prior to the accident, it would have been apparent during preflight checks and normal flying. The valve had been on the aircraft for several hundred hours. In view of these factors and tests, it was concluded that the shorted valve resulted from the fire after impact.

At the request of the Safety Board, Sikorsky Aircraft conducted an analytical study to determine the effect on the S-61L main rotor blade motion from a loss of damping. The analysis recognized that fore and aft (lead and lag) travel of a helicopter rotor blade can sometimes be induced because of the relationship of the rigid body lag and flapping modes of an articulated rotor when kinematic coupling in the control system produces a decrease in blade pitch as the blade lags back (positive blade angle). If insufficient damping is available due to some failure in the lag damper, excessive blade travel can build up.

The pitch-lag kinematic coupling of the S-61L main rotor blade was found to be such that lag damping is necessary to prevent excessive pitch-lag blade travel. For those rotor systems in which an increase in lag angle (blade lagging hack) causes a decrease in blade pitch, the coupling produces a negative damping force as a result of lag motions at the lag natural frequency. This phenomenon involves Coriolis coupling between pitch and lag. Vibratory edgewise motion in the lag direction at the lag natural frequency produces a negative pitch change which causes a nearly in-phase response in flapping. For positive coning angles, the resulting edgewise Coriolis force is proportional to flapping velocity and is therefore in phase with and in the direction of the lag velocity. This is a negative damping force; to counteract and maintain stability the available lag damping must be greater. Since aerodynamic damping inplane is small, the lag damper must provide the necessary damping.

A second step in the analysis was to determine theeffect of lag motion of the undamped blade on the response of the other blades. An analysis was developed which considered on the lag motions of the blades and the response of the airframe as a rigid body. The analysis was based on the development used by Coleman 18/ for ground resonance analysis, but the resultant linearized equations of motion were solved to determine the response of all blades and the airframe to a unit amplitude of motion of one blade at the lag natural frequency. The analysis showed that the two blades opposite the exciting blade would respond with amplitudes of approximately 60 percent of the amplitude of the exciting blade.

The analysis was then expanded *to* include the effect of blade flapping. The pitch and roll components of hub inplane response of the original analysis were determined, and equations written to determine flapping response due to pitch and roll of the hub. This flapping produces a

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18/ Coleman R.P. "Theory of Self Excited Mechanical Oscillations of Hinged Rotor Blades" NACA ARR G 29, 1943 reissued as NACA Rpt. 1351, 1958.

Coriolis force in the lag equation. By combining relations, the flapping effect can be reduced to a set of additional coupling terms between blade lag motion and hub motion. These terms were found to be insignificant. Only minor changes in response of the other blades were found.

It then appeared that, to have interference between the blades, or even between the controls (pitch horn/pitch change rod connection and the adjacent blade damper clevis/horizontal pin connection) of adjacent blade, within practical flapping and control limits, the limits imposed by the damper stops would have to be eliminated. In other words, a damper would have to be broken loose or the connection between it and its associated blade separated. In the instant case, the stop limitations were broken out.

As a result of this study, the following observations were made:

(1) Loss of damping action on one blade, through some factor such as loss of fluid or separation of a damper shaft, can cause large lag oscillations of the undamped blade at its lag natural frequency.

(2) The lag oscillations of an undamped blade can cause a similar lead-lag response in the two opposing blades, with amplitudes of approximately 60 percent of the exciting blade.

2. ANALYSIS AND CONCLUSIONS

2.1 Analysis

From the overall physical evidence, damage patterns, tests and research, and failure analyses that were conducted in connection with this accident, the sequence of events following the initial cause is reasonably clear. However, despite extensive investigative efforts, the initiating cause is more obscure.

The overall evidence indicates that the sequence began while the aircraft was in cruising flight about 2,000 feet above the surface and about 2 to 2-1/2 miles east of where it crashed. From the physical damage found in the components of the rotor head and patterns of this damage, it is evident that the black, yellow, and blue main rotor blades, followed by the red and white, underwent a series of extreme excursions in their lead/lag axis. These excursions are clearly reflected by the shearing of blade horizontal hinge pins, failures of the blade damper pistons, and damage to damper housing assemblies. The overall damage patterns not only showed that multiple excursions had occurred, but also that some were so extreme that one blade would even overlap the blade next to it. In one instance, the pitch change horn of the yellow blade jammed under the horizontal hinge pin of the white blade while the yellow blade was in an extreme lead position. It is very probable that it was at this time the forward side of the yellow blade pitch change control rod made contact with the leading ear of its attachment to the rotating swashplate. It is also probable that at this time the detachment failure of the yellow pitch change control rod occurred.

For several reasons, the Safety Board concludes that the lead and lag excursions preceded and caused the detachment of the yellow blade pitch change control rod at its attachment to the rotating swashplate.

The first reason is that the damage and damage patterns to the components in the lead and lag areas of the yellow blade were of the same type and extent as those found on the same components of the other blades. This showed that the yellow blade must have participated in the lead/lag excursions and was, therefore, in the rotor disc at the time they occurred. Conversely, structural examination revealed clear strike damage by the red, black, white, and blue blades in both cockpit and aft portions of the aircraft, with the strikes progressively deeper in the structure in the order of blade rotation. There was no evidence of strike damage by the yellow blade on either portion of the aircraft. It is reasonable to believe that had the yellow blade been in the rotor disc and in its normal position between the white and blue blades, it would have inflicted some of the damage.

A second reason is that the eyewitness information clearly suggests the pilots were able to descend the aircraft under partial control from about 2,000 feet above the ground to between 600 and 800 feet before the series of blade strikes destroyed the integrity of the aircraft and it fell uncontrollably to the ground. Had the yellow blade become detached and hit the side of the aircraft as the initial occurrence, the aircraft would have gone out of control immediately, making the descent of some 1,200 feet over about 2 to 2-1/2 miles impossible. Eyewitness information indicates the aircraft fell, near vertically, almost immediately after the rotor blades struck the aircraft separating pieces of structure from it.

Another consideration in the conclusion is that, had the yellow blade pitch change control rod become detached initially, it is improbable that the extreme lead and lag excursions of the other rotor blades would have been the expected result. This is because when the blade became free, it was unrestrained in its pitch and flap axis. With its aerodynamic tendency to descent in the retreating (left half) portion of rotation and rise in the advancing (right half) portion, the blade would not be expected to remain in the plane of rotation for any extended period. Consequently, it must have struck the side of the aircraft very shortly after its detachment. Once it hit the side of the aircraft, was dragged around the front of the aircraft, and was wrapped around the rotor head, the rotation of the main rotor head would have decelerated so rapidly there would have been insufficient time for the multiple lead and lag excursions to have taken place. Further, the forces resulting from the blade hitting the aircraft should have been in the lag direction rather than in both the lead and lag directions.

Another, and probably the most convincing reason, is the manner of detachment of the yellow blade pitch change control rod itself. Physical evidence and tests and research showed that for the trunnion bearing cap to have been pushed out and to have failed the trailing attachment ear, allowing the trunnion to come out of the bearing, the downward and rearward loading on the control rod must have been many times the normal operating loads. In the Safety Board's opinion, the extreme lead and lag excursions were responsible for this loading. More specifically, it seems probable that these forces were generated when the yellow blade pitch change control horn jammed under the white blade horizontal hinge pin. In this situation, the yellow blade pitch change control rod could have rested on the forward ear of the yellow pitch change control rod rotating swashplate attachment. This would create a fulcrum effect, and when the yellow blade, in its jammed position, tried to react aerodynamically and to collective control imputs in its flap axis, the necessary loading was produced to cause the detachment failure.

From an analytical viewpoint, it is probable that the detachment of the yellow blade also caused or contributed to causing the blade strikes on the aircraft. When the yellow blade became detached, it would have been uncontrollable in its pitch and flap axes and free to create an extreme imbalance in the rotor head. Such imbalance could well have caused the main rotor head to displace in such a manner as to deflect the other four rotor blades downward into the front and rear structure of the aircraft.

Based on tests and research, and for the reasons stated above, the Safety Board concludes that the extreme main rotor blade excursions in their lead/lag axis occurred first in the sequence resulting from the initial accident cause, and that the detachment failure of the yellow blade pitch change control rod resulted from overloads created by the extreme blade excursions. Based on metallurgical examinations, it is further concluded there was no fatigue or material deficiencies involved in the detachment failure; and based on tests and research, the absence of the bearing cap bolt at the trunnion attachment point did not cause or contribute to the cause of the yellow pitch rod detachment. The absence of the bearing cap bolt may have resulted from its nut being stripped off during the extreme loading on the bearing cap, or it may be reflective of a maintenance omission.

Concluding, for the reasons stated above, that the resulting sequence of events began with the extreme lead and lag excursions of the main rotor blades, the Safety Board explored numerous possible reasons for the excursions. Much of the testing and research was directed to this purpose, and the aircraft manufacturer assisted to a maximum extent with its facilities and expertise.

Considered were the failure of a main rotor blade, the failure or seizure of the main rotor gearbox, the failure of a main rotor blade spindle, and the failure of a servo connection. All of these possibilities have the potential for creating the excursions; however, they can be dismissed definitively on the physical evidence and metallurgical examination which clearly showed none of them had occurred.

Both engines were developing little or no power at impact, with damage indicating an approximate 15 percent rotational speed. This suggested the possibility of a total power failure situation in which the pilot did not lower collective control for autorotation quickly enough. This could result in a serious main rotor speed loss and blade stall, thus causing the extreme lead and lag blade excursions. This possibility was concluded by the Safety Board as being remote for several reasons. First, there was no evidence of engine malfunction or failure. Each engine has a separate fuel system, and the pilot who was highly experienced could be expected to react promptly and efficiently to a power failure. Additionally, according to the aircraft manufacturer, a serious blade stall would result in erratic pitch attitude changes of the aircraft which are not indicated by ground witness obsevations.

The Safety Board is of the opinion that the lack of engine power development was most probably the result of the pilots' reaction to what they knew was a most critical situation and to reduce, to the extent possible, the crash fire hazard. This reason is supported by the last transmission from the aircraft, "LA, we're crashing . . . . "

There were two severe impact damage marks on the yellow blade at the junction of its tip cap and blade spar, the cause of which could not be identified satisfactorily. This gave rise to the possibility that a foreign object had struck the blade, deflecting it critically or damaging its pitch control components. However, one damage mark was not considered of sufficient magnitude to have created a problem and was dismissed as incapable of causing a critical failure in the rotor head. The other mark, because of other impact marks superimposed over it, was considered to have occurred following separation of pieces of the yellow blade. This latter mark, although severe, did not cause separation of the blade tip cap and was, therefore, also considered incapable of causing a critical control failure of the rotor head.

Another possible cause, carefully considered, for the extreme lead and lag excursions was a malfunction of the AFCS alone, or in conjunction with a maladjustment of the AFCS servo, resulting in a forcible hardover in the yaw, pitch, or roll axes of the aircraft. There are several reasons for dismissing this possibility.

The primary reason is that any control input to the control system affects all five blades nearly simultaneously and no one blade alone. This cannot change if control input was manual or was an input in the form of an AFCS forcible hardover from system malfunction or maladjustment. In this accident, the evidence is clear that the yellow blade was out of the rotor disc when the other blades cut into the front and rear areas of the aircraft. It is therefore evident that the yellow blade must have been affected singularly and before the blade strikes occurred.

Another factor is that properly adjusted, the AFCS has limited authority and, in the event of malfunction of the system, it can immediately be cut off by either pilot by depressing a button switch on his collective control.

Finally, examination and tests and research work on the auxiliary servo control input systems reasonably determined that maladjustment and/or malfunction of the sloppy linkage and input control valves did not exist, Additional verification of this conclusion exists in that maladjustment of the control input systems should have been evident during operation of the control systems. None of the pilot writeups on the AFCS were indicative of maladjustment or malfunction of the control systems.

The Safety Board believes that based on all the evidence, both positive and negative, tests and research, and failure studies, the extreme excursions occurred first as a result of the initiating cause. From this conclusion, it is evident that the initiating cause had to be one which would affect the blades in their lead and lag axis. In this accident most suspect with this capability would be a loss of damper integrity resulting from a failure of the black main rotor blade damper or a loss effective damping action by the white main rotor blade damper. These possibilities will be discussed more fully later in the report.

Based on the analytical study and tests by Sikorsky Aircraft and other technical data (see references in footnotes 18 and 19), the Board concludes it would be possible for the main rotor blades to become unstable in the lead/lag axis as the result of a loss of a blade damper integrity. The study and reference data indicate that in a rotor system with three or more blades, the blades are attached to the rotor hub by a horizontal hinge which permits the blades to move in a vertical plane, i.e., flap up or d m as they rotate. In forward flight, lift increases on advancing blades causing the blades to flap up, which decreases the angle of attack. Lift decreases on the retreating blades causing the blades to flap down, increasing the angle of attack. The combination of decreasing angle of attack on the advancing blades and increasing angle of attack on the retreating blade through blade flapping action tends to equalize the lift over the two sides of the rotor disc.

As a main rotor blade flaps up on the advancing side, it must speed up and as it flaps down on the retreating side, it must slow down. This is because the distance of the center of mass from the axis of rotation (measured perpendicular to the axis of rotation) times the rotational velocity must always remain the same for a given rotor r.p.m. Since the distance becomes shorter when the blades flap up, the rotational velocity must increase; conversely, when the blades flap d m , the distance becomes greater and the rotational velocity must decrease. This phenomenon is an application of Coriolis effect. 19/

19/ For a more detailed discussion of Coriolis effect, as the phenomenon applies to helicopters, the reader is referred to the following publications:

1. Sikorsky Helicopter Flight Theory for Pilots and Mechanics, Sikorsky Aircraft Division of United Aircraft Corporation, USA, 1964, all rights reserved (chapter 6).
2. Basic Helicopter Handbook, 1965, Federal Aviation Agency, U.S.Government Printing Office, (pages 13 & 14).

In addition to the Coriolis effect which produces speed up and slow dam of the blades in their plane of rotation, there is the accentuating effect of the change in angle of attack which accompanies the blade flapping. With airspeed constant, an increase in angle of attack of an airfoil is accompanied by an increase in drag. The drag in this instance tends to further slow down the retreating blade, while the decrease in drag in the advance blade tends to aid the speed up.

The change of blade velocity in the plane of rotation causes lead and lag action about the vertical hinge. This acceleration or deceleration (lead and lag) is absorbed by the dampers.

Another manifestation of Coriolis effect occurs when the rotor system is tilted as in forward flight. In this condition, the forward blades are flapping down while the rearward blades are flapping up. The center of mass of the bw flapping blades has moved fartheraway from the axis or rotation, while the center of mass of the high flapping blades has moved in toward the axis of rotation. Again the blades accelerate or decelerate as the center of mass moves closer or farther away from the axis of rotation. These changes in blade velocity also cause lead and lag about the horizontal hinge and are absorbed by the dampers.

Without damping action, the lead and lag motions of the blade would be accentuated. Accordingly, as the excursions of the affected blade continue, they would build up in amplitude. As the amplitude buildup continues, the other blades, beginning with those on the opposite side of the rotor disc, react in response to the motions of the affected blade until lead/lag instability of all blades occurs. It follows that the blade instability would continue to increase, ultimately causing the extreme blade excursions, breaking of the mechanical limiting stops blade over-lapping, and the other results as they have been described.

Concluding that a loss of main rotor blade damper integrity was the cause of this accident, the Safety Board believes such loss of integrity resulted either from a failure of the black main rotor blade damper or the loss of effective damping action of the white main rotor blade damper. The Board finds the substantive evidence insufficient to definitively isolate one to the exclusion of the other.

In regard to damper failure, examination of the red, yellow, and blue dampers and their associated components revealed no evidence to substantiate their involvement in the initial cause. The black damper, however, was found with its piston broken at the radius of its threaded end shaft. As already described, the bumper, trunnion, trunnion bearings, and horizontal hinge pin stub and bushing of this damper were missing and have never been found.

The Board believes it is possible that the black main rotor blade damper, bumper and associated parts separated from the rotor head assembly early in the breakup sequence and fell to the ground somewhere along the flightpath prior to the main impact site. The absence of these parts for examination gives rise to speculation as to their degree of involvement in the cause of the blade excursions. As has already been stated, if a damper were to become separated, the extreme blade excursion could result.

There are three logical reasons to be considered in considering damper separation failure: (1) loss of torque between the damper bumper and the damper piston shaft, (2) horizontal hinge pin bushing failure, and (3) damper trunnion bearing failure.

In the first instance, the damper bumper is screwed to the threaded end of the damper piston, torqued to a given valve, and locked in place by a jam nut. Should this assembly lose its torque, stresses could be set up on the piston shaft in the area of its threaded shank which could eventually lead to failure as was noted by examination of the black damper piston.

Earlier models of main rotor blade dampers were known to have paint on the mating surfaces of the piston shaft and bumper. It was theorized that if the paint worked out of this area, torque would be lost and stresses would be set up in the radius of the piston shaft and its threaded extension. Such stresses could lead to ultimate failure of the shaft.

This possibility must be discounted, however, since the portion of the damper pistion shaft that was recovered showed no evidence of ever having been painted.

In evaluating the second reason, it is noted that the damper trunnion is connected to the horizontal hinge pin stub and rides on a lubricated bushing. Looseness or disintegration of this bushing would allow the stub end of the hinge pin to hammer the damper trunnion and lead to eventual failure of the'stub end, trunnion or piston shaft. Since the black damper bumper, trunnion, piston shaft threaded end and portion of the hinge pin stub end containing the bushing were not recovered this possibility cannot be accepted nor satisfactorily eliminated.

The third reason involves the damper trunnion bearings. There are two in the installation of the ball and race type. Should one of these bearings fail, it is possible for the trunnion to work loose with resultant damper separation. In addition, failure of one of the bearings would easily perpetrate the failure of the other and lead to damper separation from the system.

In the instance of a loss of effective damping action, the white main rotor blade damper is suspect. There is some evidence that the white damper quick disconnect may have been disconnected from its damper fluid reservoir prior to the accident. Although the quick disconnect did not leak fluid of any significant amount in leakage tests, the possibility remains it may have done so if disconnected prior to the accident. Sufficient leakage of fluid from the damper would result in a loss of effective damping action, and the results would be the same as those from a damper failure.

The aircraft manufacturer, based on its analytical studies, tests and research, also reached the conclusion that a loss of damper integrity was the basic mode of the accident. They, however, believe that this was more probably due to the loss of damper effectiveness of the white blade damper rather than a failure of the black blade damper.

The manufacturer cited as their major reason the fact that the black damper failures at the damper piston and at the horizontal hinge pin were conclusively determined to be gross overload. They conclude that a structural failure within the missing portion of the black damper would produce fatigue failures and not gross overload or static failures. In conclusion on this point, the manufacturer considered the black damper failure was part of the result sequence and not the initial cause of the sequence.

The Safety Board recognizes that while it could have been the white damper there are also good reasons that it could have been the black damper. In this connection a failure could have occurred within the missing portion of the black main rotor blade damper allowing the damper to separate at: the point of failure, and the failures at the horizontal hinge pin and damper piston occurred later during the extreme blade excursions in their lead/lag axis.

Another substantive reason the manufacturer concluded the loss of effective damper action by the white damper was more probable than failure of the black damper, is based on their analytical study. This indicated that a loss of effective damper was probably capable of resulting in blade instability in the lead and lag axis, and it was not necessary to have a damper failure which took the component completely out of the rotor system.

As noted above, the Safety Board agrees that a loss of damper effectiveness is considered one of the possible reasons for a loss of damper integrity. It notes, however, that operational history suggests that a loss of damper effectiveness, while producing roughness of flight and passenger discomfort, would not be disastrous.

In summary, the Safety Board respects with high regard the manufacturer's conviction relative to the single initial cause for the loss of damper integrity and believes their corrective measures show their conviction. 20/ In its judgment , however, the Safety Board believes a loss of effective damping action by the white damper should not be concluded to the exclusion of failure of the black damper.

2.2 Conclusions

(a) Findings

1. The flight crewmembers were properly certificated and qualified for the flight involved.

20/ See Section 3, Corrective Measures.

2. Weather conditions for the flight were clear with unlimited visibility, and in full daylight.

3. The weight and center of gravity of the aircraft were within limitations at departure from Anaheim, California, and at the time of the accident.

4. The accident sequence began while the aircraft was in normal cruising flight about 2,000 feet above the ground, on course and heading, and about 2 to 2-1/2 miles from where it crashed.

5. The crash sequence began with the main rotor blades undergoing a series of extreme excursions in their lead/lag axis.

6. The extreme lead and lag excursions caused the overload detachment failure of the yellow main rotor blade pitch control rod at its lower trunnion end attachment to the rotating swashplate.

7. Detachment of the yellow blade made the blade uncontrollable in its pitch and flap axes, and it struck the right side of the aircraft.

8. The four remaining blades in the rotor disc, as the result of imbalance, struck and penetrated the cockpit and aft fuselage of'the aircraft.

9. Strikes by the rotor blades destroyed the structural integrity of the aircraft and it fell nearly vertically to the ground.

10. There was no fire in flight; however, postimpact ground fire occurred.

11. The cause of the extreme excursions of the main rotor blades in their lead and lag axis resulted from a loss of damper integrity. Probable reasons are failure of the black main rotor damper or a loss of effective damping action of the white main rotor blade damper.

12. An important portion of the black damper was not recovered for examination.

(b) Probable Cause

The Safety Board determines that the probable cause of this accident was the loss of main rotor blade damper integrity due to either a failure of the black blade damper or a loss of effective damping action by the white blade damper. This resulted in uncontrolled excursions of the main rotor blades in their lead/lag axis, an overload detachment of the yellow main rotor blade pitch change control rod and destruction of the structural integrity of the aircraft by blade strikes. The precise reason for either of the possibilities for the loss of damper integrity is undetermined.

3. CORRECTIVE MEASURES

As a result of this accident, a second accident involving a Los Angeles Airways S-61L, on August 14, 1968, and an incident involving a control malfunction on June 26, 1968, a number of corrective measures were taken to improve the safety of operations of the S-61L aircraft. The pretakeoff check of the AFCS was expanded to check the system for proper operation.

Prior to the aforementioned accidents and incident, a hardover input intentionally induced through the hardover panel could not be cut off by the cutoff button switch on the collective control of the pilot. An electrical wiring change was incorporated enabling the pilot to cut off any AFCS input from the hardover panel, as well as to cut off the AFCS normal input to the control system. As indicated, the latter capability already existed with respect to the system.

The tests and research conducted in'connection with the subject accident in an effort to determine its cause also served to reconfirm stress loadings in the main rotor head. Loadings were found to be somewhat higher than originally determined but not beyond the margin of safety designed into the rotor head. In the interest of safety, the unlimited life for horizontal hinge pins was reduced to a life limit of 5,000 hours.

The manufacturer of the aircraft issued a Service Bulletin to all operators requesting them to check all main rotor blade dampers for proper torque of the screw-in fitting of the damper piston to the damper trunnion end bumper or shock absorber.

As a result of the accident, on August 14, 1968, an Airworthiness Directive, AD, was issued. This AD required that only new main rotor blade spindles be used on S-61 aircraft. It also placed a life limit of 2,400 hours on the spindles. Prior to the AD, reworked spindles were used and there was no life limit on the spindles.

For a period of time the aircraft manufacturer considered the use of a quick disconnect of an improved design in the lines from the damper fluid reservoirs to the main rotor blade damper or, in the alternative, to eliminate entirely the use of a quick disconnect in the line. By letter dated October 30, 1969, the manufacturer advised it had decided to eliminate the quick disconnect. The Board believes this is an appropriate action in the area of corrective measures.

BY THE NATIONAL TRANSPORTATION SAFETY BOARD:

/s/ JOHN H. REED
Chairman

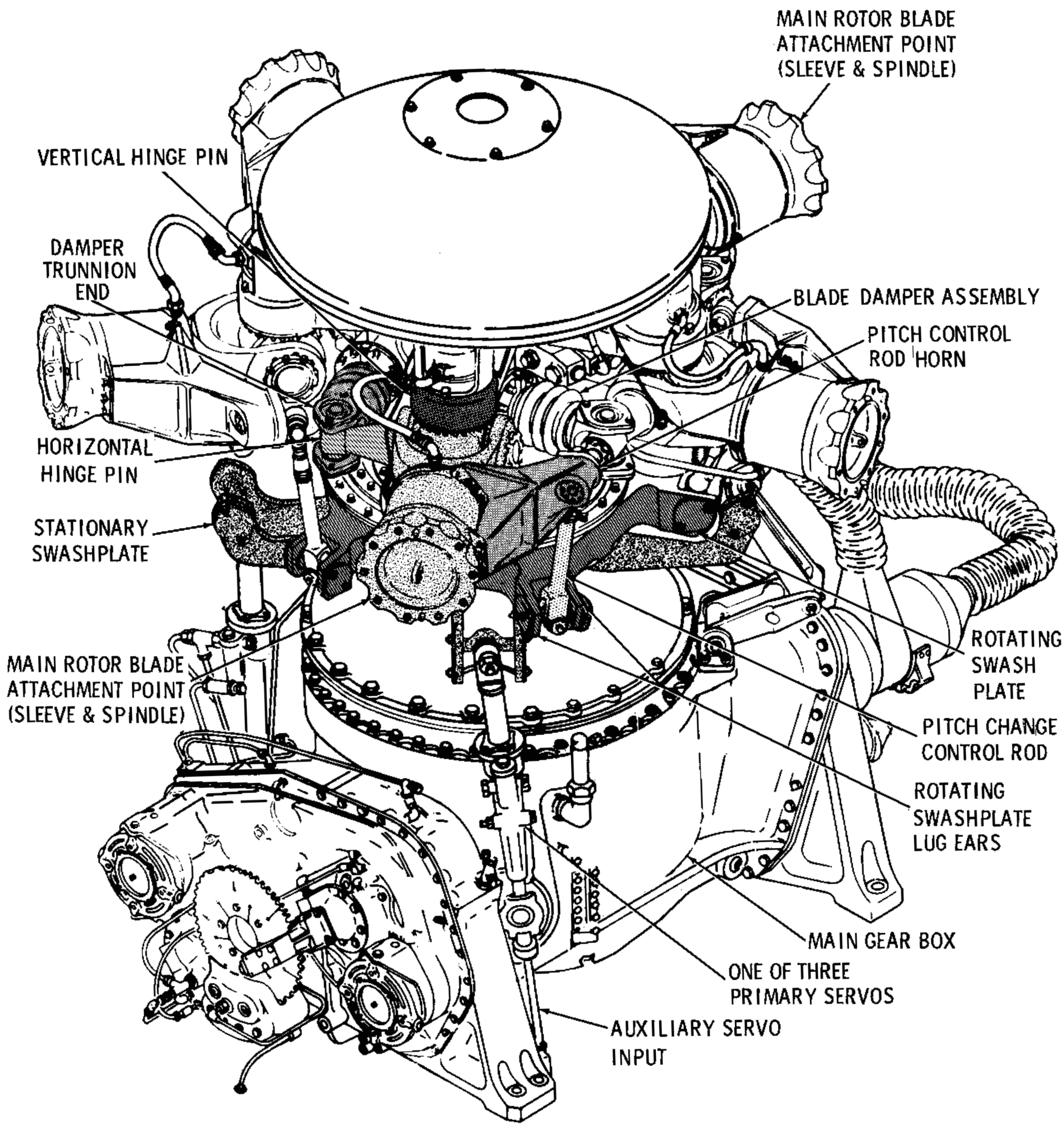
/s/ OSCAR M. LAUREL
Member

/s/ FRANCIS H. McADAMS
Member

/s/ LOUIS M. THAYER
Member

Isabel A. Burgess, Member, did not take part in the adoption of this report.

December 18, 1969.



Attachment 1. Explanatory Diagram of Main Rotor Head

APPENDIX A

Investigation and Hearing

1. Investigation

The Board received notification of the accident at approximately 2200 on May 22, 1968. An investigation team was immediately dispatched to the scene from Washington, D. C. Working groups were established for operations, witnesses, structures, powerplants, aircraft and maintenance records, systems, and human factors. Parties to the Investigation were Los Angeles Airways, the Federal Aviation Administration, Air Line Pilots' Association, and Sikorsky Aircraft Division of the United Aircraft Corporation. The on-scene phase of the investigation was completed in about 10 days; however, extensive tests and research and failure analyses were continued for many months thereafter at the aircraft manufacturer's facilities at Stratford, Connecticut.

2. Hearing

A public hearing was convened September 25, 1968, at El Segundo, California, and lasted approximately 3 days.

3. Preliminary Reports

A summary of the testimony taken at the public hearing was published by the Board on October 15, 1968. A preliminary report was not issued on the accident.

APPENDIX B

Crew Information

Captain John E. Dupies

Captain Dupies, aged 45, had been an employee of Los Angeles Airways since 1953. At the time of the accident, he held Airline Transport Certificate No. 554033, with ratings (VFR) on Sikorsky S-61 S-55, S-62 aircraft and an unrestricted (IFR) rating in the S-61 helicopter. He had a total of 12,096 flying hours, of which 4,208 were in the S-61. In the 60- and 30-day periods before the accident, he had flown 124 and 55 hours, respectively. On the day of the accident, he had flown about 15 minutes.

Captain Dupies had completed satisfactorily his most recent proficiency check on February 28, 1968, his most recent line check on January 5, 1968, and his most recent recurrent training on April 23, 1968. He held a current first-class medical certificate with no limitations, dated December 26, 1967.

Copilot Terry R. Herrington

Copilot Herrington, aged 27, was employed by Los Angeles Airways on January 26, 1968. At the time of the accident, he held Commercial Pilot Certificate No. 1600649, with airplane single- and multiengine land, Sikorsky S-58 aircraft and instrument including helicopter ratings. He had a total of 872 flying hours, of which 589 were in helicopters. In the 60- and 30-day periods preceding the accident, he had flown 118 and 62 hours, respectively.

Copilot Herrington completed satisfactorily initial copilot training on February 18, 1968, and a line check qualifying him to make takeoffs and landings on May 6, 1968. He held a first-class medical certificate with no limitations, dated January 1, 1968.

Flight Attendant Donald P. Bergman

Flight Attendant Bergman was employed by Los Angeles Airways on July 3, 1967, as a utility helper., He became a cargo agent on August 8, 1967, and a flight attendant on August 28, 1967. His most recent refresher training was completed satisfactorily on February 29, 1968.

APPENDIX C

Aircraft Information

Helicopter S-61L, N303Y, serial No. 61060, was manufactured in June 1962 by the Sikorsky Division of United Aircraft Corporation. Its Airworthiness Certificate was issued to Los Angeles Airways on August 18, 1962.

At the time of the accident, the aircraft had accumulated 11,128 total hours. The aircraft was last overhauled on November 14, 1967, by Los Angeles Airways with 9,973 hours. It received a M3-3 (2,400 hour) periodic check on March 13, 1968, 533 hours before the accident. The most recent M2-0 (200 hour) and M1-02 (50 hour) periodic checks were on May 19, 1968. It received a M1-01 (daily) inspection on May 22, the day of the accident.

The aircraft was equipped with two General Electric engines, Model CT 58-140-1. Total time of the No. 1 engine was 6,581 hours, including 1,273 since overhaul. Total time on the No. 2 engine was 6,873 hours, including 1,143 since overhaul.