NATIONAL TRANSPORTATION SAFETY BOARD
Office of Aviation Safety
Washington, D.C. 20594

July 13, 2016

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: CEN15MA290

A. ACCIDENT

Operator:	Air Methods Corporation
Aircraft:	Airbus Helicopters AS350 B3e, Registration N390LG
Location:	Frisco, Colorado
Date:	July 3, 2015
Time:	1339 mountain daylight time

B. AIRWORTHINESS GROUP

Group Chairman:	Chihoon Shin National Transportation Safety Board Washington, District of Columbia
Member:	Matthew Rigsby Federal Aviation Administration Fort Worth, Texas
Member:	Don Lambert Air Methods Corporation West Mifflin, Pennsylvania
Member:	Dennis McCall Air Methods Corporation Englewood, Colorado
Member:	Edward Stenby Professional Helicopter Pilots Association Local 109 Frisco, Colorado
Member:	Vincent Ecalle Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile Le Bourget, France

Member:	Seth Buttner Airbus Helicopters Incorporated Grand Prairie, Texas
Member:	Bryan Larimore Turbomeca USA Grand Prairie, Texas

LIST OF ACRONYMS

AAIP	approved aircraft inspection program
AD	airworthiness directive
AHI	Airbus Helicopters Incorporated
ALF	aft looking forward
AMC	Air Methods Corporation
ATT	aircraft total time
BEA	Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile
CFR	Code of Federal Regulations
CO	Colorado
CRFS	crash resistant fuel system
CWP	caution and warning panel
DECU	digital engine control unit
EBCAU	emergency backup control ancillary unit
EDR	engine data recorder
FAA	Federal Aviation Administration
HAA	helicopter air ambulance
HIGE	hover in ground effect
MOD	modification
NTSB	National Transportation Safety Board
psi	pounds per square inch
RFM	rotorcraft flight manual
RV	recreational vehicle
S/N	serial number
SB	service bulletin
SIN	safety information notice
SUP.23	supplement 23 (to the AS350 B3e rotorcraft flight manual)
TCDS	type certificate data sheet
TSN	time since new
US	United States
VEMD	vehicle engine monitoring display
91CO	Summit Medical Center helipad

C. SUMMARY

On July 3, 2015, at 1339 mountain daylight time, an Airbus Helicopters AS350 B3e helicopter, N390LG, impacted the upper west parking lot 360 feet southwest of the Summit Medical Center helipad (91CO), Frisco, Colorado (CO). A postcrash fire ensued. Visual meteorological conditions prevailed at the time of the accident. The helicopter was registered to and operated by Air Methods Corporation (AMC) and the flight was conducted under the provisions of 14 *Code of Federal Regulations* (CFR) Part 135 on a company flight plan. The airline transport pilot was fatally injured and two flight nurses were seriously injured.

According to AMC, the helicopter was flying to Gypsum, CO, for a public relations mission. Multiple witnesses observed the helicopter lift off from the ground-based helipad, rotate counterclockwise, and climb simultaneously. One witness estimated that the helicopter reached an altitude of about 100 feet before it started to descend. Surveillance videos capturing the initial liftoff showed the helicopter yaw to the left simultaneous to lifting off the helipad. The helicopter continued to spin counterclockwise several times before it impacted a parking lot and a recreational vehicle (RV) located to the southwest of 91CO. A surveillance video capturing the ground impact showed fuel flowing from the wreckage and the onset of a postcrash fire shortly thereafter. The helicopter came to rest on its right side, was damaged by impact forces, and was charred, melted, and mostly consumed by the postcrash fire.

On July 4-5, 2015, members of the Airworthiness Group convened at the accident site to document and photograph the wreckage. The wreckage was recovered and stored at Beegles Aircraft Services in Greeley, CO. On August 20, 2015, members of the Airworthiness Group convened at Beegles Aircraft Services to search for the tail rotor hydraulic isolation valve and solenoid in the recovered wreckage. The tail rotor hydraulic isolation valve and solenoid were not found.

D. DETAILS OF THE INVESTIGATION

1.0 HELICOPTER INFORMATION

1.1 HELICOPTER DESCRIPTION

The Airbus Helicopters AS350 B3e has a three-bladed main rotor system that provides helicopter lift and thrust.[1] A two-bladed tail rotor system provides anti-torque and directional control. The helicopter flight controls are hydraulically assisted by a dual hydraulic system which consists of an “upper” and “lower” hydraulic circuit. Both upper and lower hydraulic circuits provide hydraulic assistance to the main rotor flight controls. Only the lower hydraulic circuit provides hydraulic assistance to the tail rotor flight controls. The helicopter was equipped with a high skid-type landing gear and a Turbomeca Arriel 2D turboshaft engine. The AS350 B3e is a marketing designation; the AS350 B3e helicopter is type certificated under Federal Aviation Administration (FAA) Type Certificate Data Sheet (TCDS) No. H9EU as an AS350 B3 equipped with the Arriel 2D engine.

The terms “left”, “right”, “up” and “down” are used when in the frame of reference of looking forward from the aft end of the helicopter, i.e. aft looking forward (ALF). All locations and directions will be viewed from ALF unless otherwise specified.

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[1] The main rotor blades rotate in a clockwise direction when looking down at the main rotor disc from above.

1.2 HELICOPTER HISTORY

The accident helicopter, serial number (S/N) 7595, was manufactured on March 2013 and had accumulated an aircraft total time (ATT) of about 487.4 flight hours at the time of the accident. According to helicopter records, engine S/N 50309 had accumulated an engine total time of about 480.5 hours at the time of the accident. The helicopter was estimated to weigh about 4,717 pounds at the time of the accident.

2.0 WRECKAGE DOCUMENTATION AT THE ACCIDENT SITE

On July 4-5, 2015, representatives from National Transportation Safety Board (NTSB), FAA, AMC, Airbus Helicopters Incorporated (AHI), and Turbomeca convened at the accident site to document the wreckage. The helicopter came to rest on its right side oriented on a magnetic heading of about 60°. A surveillance video capturing the descent and ground impact showed the tailboom impacting the hood of an RV during the helicopter's counterclockwise rotating descent. The surveillance video also showed fuel flowing from the helicopter wreckage and the onset of a postcrash fire shortly thereafter. The postcrash fire spread and consumed or severely damaged the majority of the helicopter wreckage (**Photo 1**).

Photo 1. The main wreckage site.

2.1 STRUCTURES

2.1.1 STRUCTURES OVERVIEW

The helicopter structure consists of the main fuselage, tailboom, and landing skids. The main fuselage comprises: the [central] body structure, primarily supporting the fuel tank, main transmission, and landing skid; the rear structure, primarily supporting the engine and baggage compartment; the bottom structure, primarily supporting the main cabin; and the canopy, primarily supporting the doors and windows. The tailboom is attached to the rear structure and supports the tail gearbox, horizontal stabilizer, tail rotor drive shafts, and the vertical fin.

2.1.2 ON SCENE OBSERVATIONS

The majority of the main fuselage was consumed or severely damaged by the postcrash fire; however, a portion of the bottom structure was not consumed by the postcrash fire. Excluding the forward right door, the majority of the canopy and doors were consumed by the postcrash fire. The engine was found installed on the steel engine deck, which was located immediately aft of the main transmission. Fractured segments of the landings skids were found at the main wreckage site. An Appareo Vision 1000 onboard image recorder was found installed on the roof of the cabin and was retained for further examination. Due to extensive damage to the Appareo Vision 1000, no useful data could be extracted from the device. For additional details on the Appareo Vision 1000 examination, see the Onboard Image Recorder Specialist’s Factual Report in the docket for this investigation.

The majority of the tailboom was consumed by the postcrash fire. A circular frame at the aft end of the tailboom was observed with the tail gearbox still attached to it. A portion of the horizontal stabilizer spar was found adjacent to the RV. The vertical fin and tail cone were found about 20 feet from the main wreckage, adjacent to a second RV near the main wreckage. The ventral fin exhibited impact marks on its forward surfaces. Both dorsal and ventral vertical fins, along with the tail cone, exhibited heat distress.

2.2 MAIN ROTOR SYSTEM

2.2.1 SYSTEM DESCRIPTION

Power from the engine reduction gearbox is transferred to a power transmission shaft, the forward end of which is connected to a freewheel shaft. The freewheel shaft is connected to the engine-to-transmission drive shaft via a splined adapter. Flexible couplings on both ends of the engine-to-transmission shaft allow for minor misalignment. The engine-to-transmission shaft is connected to the main transmission input pinion pulley flange, which drives the main transmission input pinion, the aft hydraulic pump, and air conditioning unit, the latter two of which are belt driven via the pulley flange. The main transmission contains a single-stage sun and planetary gear system that turns the main rotor shaft. The main rotor shaft is attached to the Starflex via 12 bolts. The main transmission is attached to the airframe via four rigid suspension bars and an anti-torque bi-directional crossbeam with laminated pads installed between the lower transmission housing and the airframe.

The three main rotor blades attach to the Starflex via blade sleeves (two sleeves per blade). An elastomeric bearing connects the inboard end of the sleeves to the Starflex, while an elastomer block (also known as the “frequency adapter”) is located near the outboard end of the sleeves and is attached to the outboard end of each Starflex arm. The blade is secured to the outboard end of the sleeve via blade pins. The elastomeric bearing allows for the blade to move in the flapping, lead-lag, and pitch change directions. The Starflex arms are flexible in flapping, but rigid in lead-lag and pitch change directions.

The frequency adapter is flexible in lead-lag, but rigid in the flapping and pitch change directions. Each set of main rotor blades, sleeves, and pitch change links are assigned a color for identification purposes; the assigned colors are ‘blue’, ‘red’, and ‘yellow’.

2.2.2 ON SCENE OBSERVATIONS

All three main rotor blades were observed attached to the main rotor head at the main wreckage site. All three main rotor blades exhibited heat distress and were not identifiable by their assigned color. The main rotor blade spars exhibited significant damage consistent with high rotation energy. The Starflex and blade sleeves exhibited heat distress and fractures consistent with impact with high rotational energy.

The main transmission was observed in the main wreckage site laying on its right side. The four main transmission suspension bars were fractured and exhibited thermal damage. The anti-torque bi-directional crossbeam was found in its installed position but exhibited heavy sooting and heat distress. The engine-to-transmission drive shaft was found separated from the transmission input pulley flange and exhibited heat distress. Fragments of the engine-to-transmission shaft’s forward flexible coupling and an attachment bolt were found within the wreckage adjacent to the engine-to-transmission shaft. The aft flexible coupling and splined adapter (normally mated to the freewheel unit splines) remained attached to the aft flange of the engine-to-transmission shaft with its respective attaching hardware. The splined adapter did not exhibit evidence of abnormal wear. The freewheel unit was manually rotated and the freewheel mechanism exhibited normal operation. The main rotor shaft remained attached to the Starflex. When the transmission input pinion was rotated manually, a corresponding rotation was observed on the Starflex, consistent with continuity of drive through the main transmission. The pitch change rods remained attached between the rotating swashplate and each pitch change horn.

2.3 TAIL ROTOR SYSTEM

2.3.1 SYSTEM DESCRIPTION

Engine power is transferred to the tail rotor via two tail rotor drive shafts and a tail gearbox. The forward tail rotor drive shaft, made of steel, is connected to a flange connected to the aft end of the freewheel shaft. The aft tail rotor drive shaft, made of aluminum, connects to the forward tail rotor drive shaft via a splined, steel flange adapter. Flexible couplings are located between each drive shaft attachment point to allow for minor misalignment. Five ball bearings (also known as “hanger bearings”), mounted within support brackets along the tailboom, support the tail rotor drive shafts. The tail gearbox provides gear reduction and changes the direction of drive. The tail rotor hub, connected to the tail gearbox output shaft, provides final drive to the tail rotor.

The two tail rotor blades share a common composite spar that is flexible in both the flapping and pitch change (torsional) directions. Two metal half-shells are clamped to the center of the spar. The inboard half-shell connects to the tail rotor hub and allows for the tail rotor to teeter. Blade pitch is changed via pitch change links mounted between a pitch horn and a pitch change assembly (also known as the “spider”). The pitch horns rotate about a set of elastomeric bearings at the root end of each blade. The spider slides along the tail gearbox output shaft and is controlled by a pitch change bellcrank. Each set of tail rotor blades and pitch change links are assigned a color for identification purposes; the assigned colors are ‘red’ and ‘yellow’.

2.3.2 ON SCENE OBSERVATIONS

The tail gearbox was found in the vicinity of the main wreckage and attached to it were remnant portions of the tailboom structure, which had fractured and exhibited evidence of heat distress (**Photo 2**). The aft end of the tail rotor drive shaft remained attached to the tail gearbox input flange via flexible coupling and attaching hardware, but was fractured and exhibited evidence of heat distress. The flexible coupling did not exhibit damage, such as fracturing and splaying of the coupling laminates, consistent with high rotational energy. The spider and the tail rotor hub remained installed on the tail gearbox output shaft. Both tail rotor pitch change link ends remained connected to their respective pitch change horn and the pitch change spider, but were fractured near the pitch change spider ends and exhibited heat distress. The ‘yellow’ tail rotor blade pitch change horn, with its respective boss weights, a partial elastomeric bearing, and a portion of its pitch change link, was found about 130 feet away from the main wreckage (**Photo 3**). The fractured ‘yellow’ pitch change link was not heat distressed and its fracture surface exhibited signatures consistent with overload.

Photo 2. Tail gearbox laid out with both tail rotor blades. (Photo courtesy of AHI)

Photo 3. The recovered ‘yellow’ pitch change link attached to the blade horn and boss weights. (Photo courtesy of AMC)

Both tail rotor blades were recovered at the accident site. The 'red' tail rotor blade was fractured at its root end and exhibited heat distress, but remained partially attached to the tail rotor hub. The balance weights of the 'red' blade remained in the blade tip end. An outboard portion of the 'red' blade's afterbody was found away from the main wreckage and did not exhibit heat distress (**Photo 4**). The 'yellow' tail rotor blade was fractured at its rod end and exhibited evidence of heat distress. The 'yellow' blade was found within debris adjacent to the second RV.

Photo 4. The 'red' tail rotor blade afterbody laid next to the 'red' tail rotor blade. A red arrow points to the afterbody. (Photo courtesy of AHI)

Aside from the aft flange, attached to the tail gearbox input flange, and the forward [steel] splined end, the majority of the aluminum tail rotor drive shaft was consumed by the postcrash fire. All five hanger bearings were recovered in the wreckage and exhibited evidence of heat distress. The aluminum splined adapter, connecting to the forward end of the aluminum tail rotor drive shaft, was recovered but was partially consumed by the postcrash fire. The flexible coupling between the splined adapter and the steel tail rotor drive shaft was recovered with attaching hardware installed, but was mangled and exhibited evidence of heat distress. The steel tail rotor drive shaft was recovered, but its aft flange, composed of aluminum, was consumed by the postcrash fire. Resolidified molten aluminum surrounded the forward flange of the steel tail rotor drive shaft, but hardware attaching the flexible coupling to the flange was observed along with remnant, splayed flexible coupling laminates. Attaching hardware remained connected to the aft flange of the power transmission shaft along with remnant, splayed flexible coupling laminates.

2.4 FLIGHT CONTROL SYSTEM

2.4.1 SYSTEM DESCRIPTION

The cyclic and collective control inputs are transmitted to the stationary swashplate through a series of push-pull tubes and bellcranks. The main rotor cyclic and collective controls are hydraulically assisted via three dual-cylinder main rotor servo controls: fore/aft, right-roll, and left-roll. Each main rotor servo control comprises two cylinders that are stacked in tandem. The main rotor servo controls are mounted to the transmission upper housing and the stationary swashplate. The pedal control inputs are transmitted to the single-cylinder tail rotor servo control through a series of control linkages, bellcranks, and a flexible ball control cable. A yaw load compensator, with an associated accumulator, is connected to the tail rotor servo control output piston via the compensator connecting link, which actuates a push-pull tube connected to the pitch change bellcrank mounted to the tail gearbox.

The AS350 B3e is equipped with a dual hydraulic system, comprising two independent hydraulic circuits, provides hydraulic assistance to the main rotor servo controls (**Figure 1**). The hydraulic circuits are identified as the "upper" and "lower" hydraulic circuit. Each hydraulic circuit contains its own pump, filter, regulator valve, and reservoir. A gear-driven hydraulic pump, mounted to the forward housing of the main transmission, provides hydraulic pressure to the upper hydraulic circuit. A belt-driven hydraulic pump, mounted adjacent to the main transmission input pulley flange, provides hydraulic pressure to the lower hydraulic circuit. Each hydraulic circuit powers its respective cylinder for each of the main rotor servo controls, i.e. for the fore/aft servo control, the upper hydraulic circuit powers the upper cylinder while the lower hydraulic circuit powers the lower cylinder. The dual hydraulic setup provides redundancy in hydraulic assistance to the main rotor servo controls in the event one of the hydraulic circuits fail.

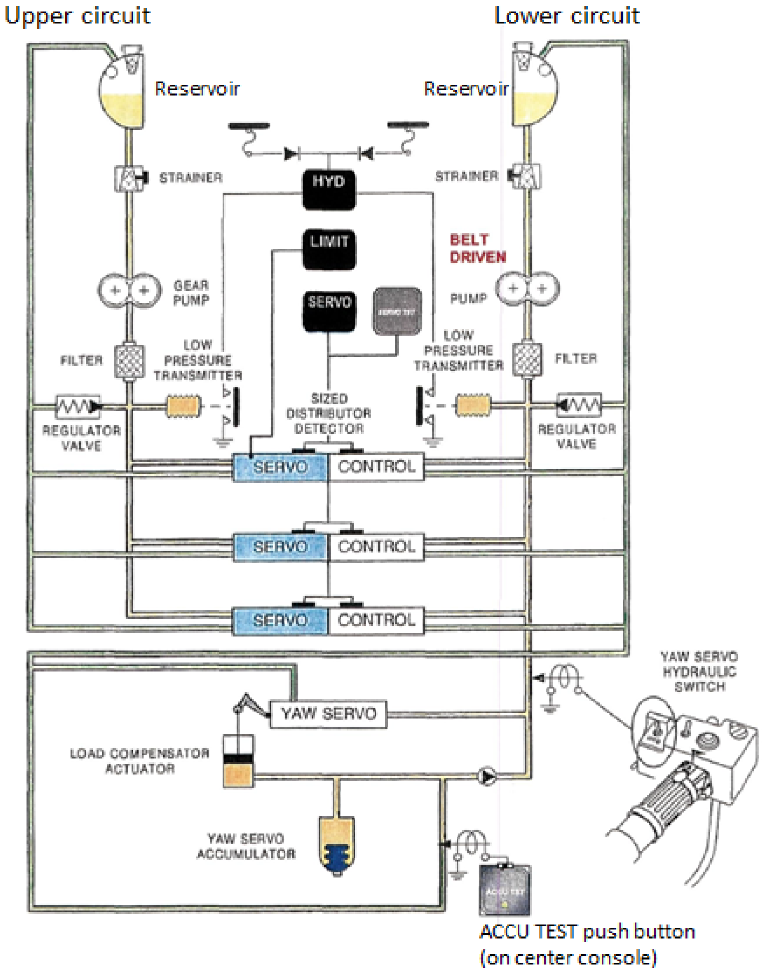

Figure 1. Diagram of the AS350 B3e dual hydraulic system. **(Image courtesy of AHI)**

The tail rotor servo control, yaw load compensator, and accumulator are powered only by the lower hydraulic circuit. The yaw load compensator and its accumulator provide a continuous hydraulic power reserve to assist the pilot in the event of a depressurization of the lower hydraulic circuit, i.e. the yaw load compensator accumulator does not deplete with pedal inputs. The accumulator contains a rubber bladder that is charged with nitrogen gas to about 218 pounds per square inch (psi). The hydraulic isolation electrovalve, installed on the transmission deck, provides a means to cut off hydraulic pressure to the tail hydraulic circuit in the event of a tail rotor servo control malfunction. Actuation of the hydraulic isolation electrovalve is controlled by a guarded yaw servo hydraulic switch[2], located on a control block installed at the forward end of the collective lever. During normal operation, the hydraulic isolation electrovalve is in the open position. The yaw load compensator assembly contains an accumulator test, or "ACCU TEST" electrovalve that allows hydraulic pressure to discharge from the yaw load compensator and accumulator when the guarded ACCU TEST push button, located on the cockpit center console, is depressed. The ACCU TEST function is used to check the functionality of the yaw load compensator accumulator.[3] Additionally, the ACCU TEST function is used in the emergency procedures for a tail rotor control failure. During normal operation, the ACCU TEST electrovalve is normally in the closed position.

2.4.2 ON SCENE OBSERVATIONS

At the time of the accident, only the right-side set of pilot flight controls were installed in the helicopter.[4] The cyclic and collective sticks were found in their installed positions. The collective stick was observed to be near its lowest position. The cyclic stick was observed to be slightly bent to the left. The push-pull tubes and bellcranks for the cyclic and collective were connected at the forward end and exhibited evidence of heat distress. The mixing unit controls were found loose within the wreckage and exhibited evidence of heat distress. The both pedals were found in their installed position, with the left pedal near its forward stop. The push-pull tube for the right pedal was fractured about 4 inches aft of its rod end connection to the rocker arm and exhibited evidence of heat distress (**Photo 5**). The remaining pedal controls were continuous back to the flexible ball cable. Movement of the intermediate bellcrank resulted in an associated movement of the flexible ball cable.

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[2] The yaw servo hydraulic switch is also known as the yaw hydraulic isolation switch, the hydraulic cut-off switch, and the hydraulic pressure switch.

[3] Prior to and on the day of the accident, the flight manual procedures required the use of the ACCU TEST button and the yaw servo hydraulic switch during the hydraulics checks conducted as part of the preflight run up checks. At a later date, after the accident occurred, a change in the flight manual procedures by Airbus Helicopters and the FAA moved the hydraulics checks to the postflight checks. See Section 7.0 for additional details to this change.

[4] As part of approved helicopter air ambulance (HAA) modifications, the left-side flight controls were removed.

Photo 5. Fractured and heat distressed push-pull tube for the right pedal.

All three main rotor servo controls remained attached at both ends to the main transmission housing and the stationary swashplate with their respective attaching hardware. The input rod ends remained attached on all three main rotor servo controls with their respective attaching hardware, but the input rods were fractured and exhibited evidence of heat distress. The tail rotor hydraulic isolation solenoid and valve, normally mounted on the transmission deck to the aft and right of the main transmission, were not observed in the wreckage.

The tail rotor servo control and yaw load compensator were found loose in the main wreckage (**Photo 6**). The tail rotor servo control rod end was not attached to the tailboom structure, but the attaching hardware and resolidified molten aluminum was found within the rod end. Resolidified molten aluminum was found around the servo control body, while the aft end of the piston was exposed. The aft end of the input rod remained connected to the servo control with its associated attaching hardware. Attaching hardware remained installed on the forward end of the input rod, and the input rod tube exhibited a slight curve. The yaw load compensator, yaw load compensator accumulator, ACCU TEST electrovalve, and compensator lever were found as a single assembly, with resolidified molten aluminum surrounding the body of the assembly. Hydraulic lines remained attached to the yaw load compensator assembly but exhibited thermal damage. Attaching hardware was observed where the assembly body and the compensator lever are mounted to the tailboom structure. The compensator lever remained attached to the yaw load compensator piston with its respective attaching hardware. Additionally, the attaching hardware for connecting the compensator lever to the compensator connecting link remained within the compensator lever, but the compensator connecting link was consumed by the postcrash fire.

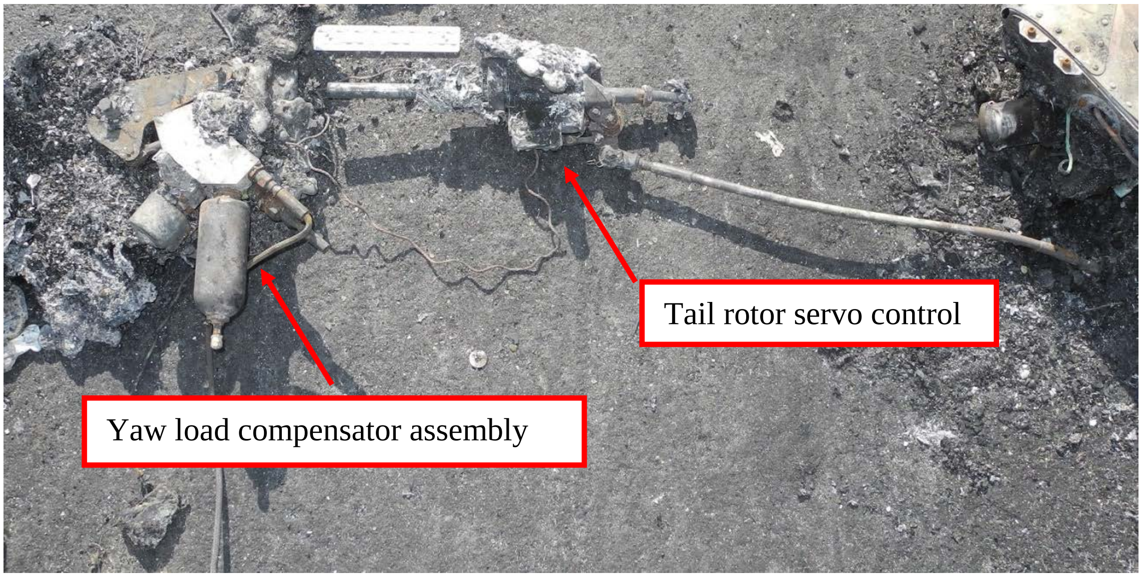

Photo 6. The yaw load compensator assembly and the tail rotor servo control recovered from the wreckage.

Both hydraulic reservoirs were found immediately aft of the main transmission and exhibited evidence of heat distress. The filler caps of both reservoirs remained attached. The gear-driven hydraulic pump remained installed on the transmission housing. Removal of the gear-driven hydraulic pump cover revealed the drive coupling and the six-lobed coupling, both of which were sooted but remained intact. The drive coupling splines did not exhibit evidence of abnormal wear. The belt-driven hydraulic pump was found adjacent to the input pulley flange and exhibited evidence of heat distress. The hydraulic pump belt was consumed by the postcrash fire. Removal of the belt-driven hydraulic pump cover revealed sooted material on the internal surfaces of the pump. The pump gear teeth exhibited no evidence of abnormal wear, but could not be manually rotated. The four-lobed coupling was mostly consumed by the postcrash fire, but remnants were observed on the coupling adapter. The coupling adapter drive key was removed and did not exhibit abnormal wear.

The rotor brake was found in the forward position, consistent with disengagement of the rotor brake as required for normal flight. The fuel shutoff control lever was found in the forward position, consistent with the fuel flow valve in the open position. Cockpit instruments were observed to be heat distressed and thus could not be reliably read (**Photo 7**). The control block at the forward end of the collective stick was partially consumed by the postcrash fire (**Photo 8**). The yaw servo hydraulic switch was observed to be in the forward position. The ACCU TEST push button was partially consumed by the postcrash fire. The engine control twist grip, located on the collective stick, was heat distressed and observed to be in the flight position.[5]

The tail rotor servo control and yaw load compensator assembly were retained for computed tomography (CT) scans; see Section 4.0 of this report for additional information. The remnant collective stick control block, and the ACCU TEST push button were retained for x-ray examination; see Section 5.0 of this report for additional information.

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[5] The twist grip contains the label “vol”, a French word which translates to “flight”.

Photo 7. Cockpit instruments exhibiting heat distress.

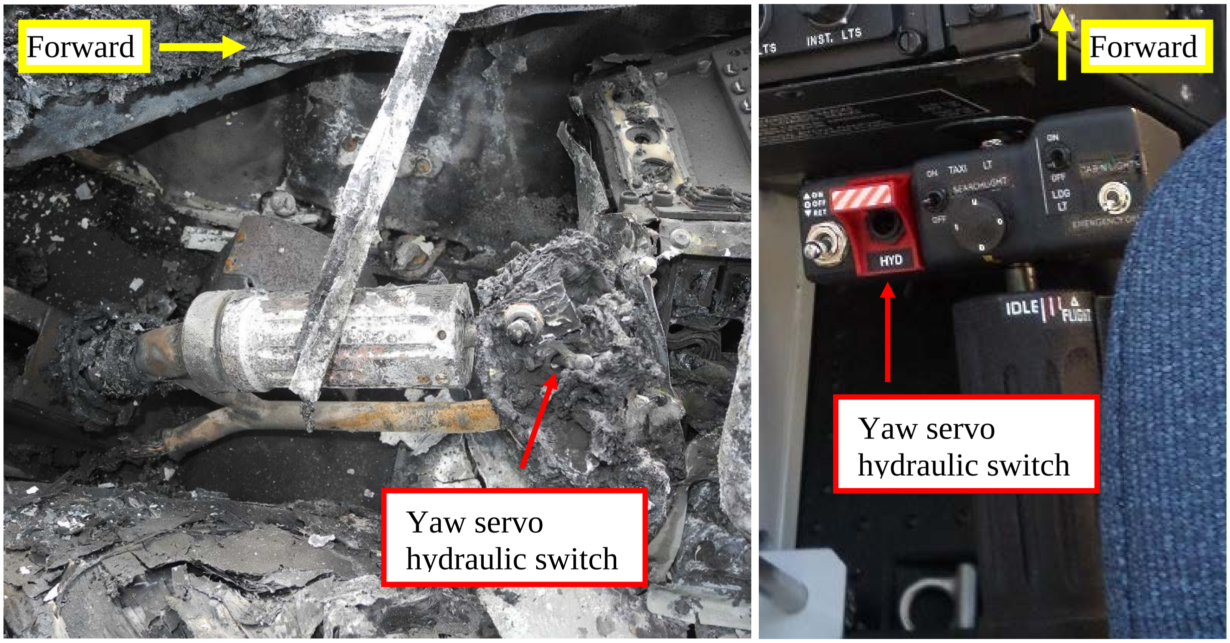

Photo 8. Collective stick with the control block from the accident helicopter at the accident site (left) and from a photograph prior to the accident (right, courtesy of AMC).

2.4 ENGINE

2.4.1 ENGINE DESCRIPTION

The Turbomeca Arriel 2D turboshaft engine features a single-stage axial flow compressor and a single-stage centrifugal flow compressor, an annular combustor, a single-stage turbine rotor that drives the compressor, and a free turbine[6] rotor. A

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[6] The free turbine is also known as the power turbine.

reduction gearbox is driven via a splined coupling (also known as a “muff coupling”) connected to the free turbine rotor. The reduction gearbox provides final drive to the power transmission shaft. An accessory gearbox is driven off of a gear coaxial to the shaft between the axial and centrifugal flow compressors.

2.4.2 ON SCENE OBSERVATIONS

The engine, still attached to the engine deck via the engine rear mounts, was found at the main wreckage site laying on its right side. The overall external surfaces of the engine exhibited heat distress. The engine air inlet duct was consumed by the postcrash fire, but remnants remained attached to the flange adjacent to the axial compressor. The air inlet filter frame and the filter were found aft of the main transmission and exhibited evidence of heat distress. The engine exhaust duct was deformed upward (**Photo 9**). The flanged connection between the free turbine case and the gas generator case was observed to be partially separated (**Photo 10**).

The leading edges of the axial flow compressor blades exhibited evidence of hard body impact damage (**Photo 11**).[7] The gas generator[8] could not be manually rotated. All free turbine blade roots exhibited overload fractures consistent with blade shedding (**Photo 12**).[9] The reduction gearbox was removed and inspection of the input pinion slippage mark[10] was offset in the tightening (torque) direction by about 0.04 inches (**Photo 13**). The forward ends of the splines of the reduction gearbox splined nut exhibited abnormal wear on both the drive and non-drive contact surfaces. The outboard ends of the high pressure (compressor) turbine blades exhibited a rough appearance and material loss of about one quarter of their span length. The free turbine nozzle guide vanes exhibited heat distress and exhibited deformation on its left side.

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[7] Hard body impact damage is characterized by a serrated appearance and deep cuts or tears to the blade leading and trailing edges. Hard body impact damage can result from impact with metal parts, concrete, asphalt, and rocks. Soft body impact damage is characterized by a large radius of curvature or curling deformation to the blade and can cause curling deformation of the blades in the direction opposite of normal rotation. Soft body impact damage can result from impacts with pliable objects such as birds, ice, tire rubber, and plastic objects.

[8] The gas generator comprises the axial flow compressor, centrifugal flow compressor, and single-stage turbine rotor.

[9] According to the engine manufacturer, the free turbine blades are designed to fracture, or shed, when the free turbine rotor speed exceeds 150% free turbine rotations per minute (N_F). The manufacturer stated the free turbine blade shedding is a safety measure designed to keep the free turbine rotor from becoming unbalanced and coming apart during an engine overspeed.

[10] Within the reduction gearbox is a splined nut that secures the input gear and also transfers drive from the muff coupling to the input gear. The nut is indexed by a final torque (index) mark, also referenced as a “slippage” mark. The amount of “slippage” (or amount of overtorque) can be referenced by the quantity and direction the index mark on the splined nut is offset from the index mark on the input gear.

Photo 9. The right side of the engine after extraction from the main wreckage. The exhaust duct was observed to be deformed upward. The red arrow points to the partially separated flanged connection between the free turbine case and the gas generator case. (Photo courtesy of Turbomeca)

Photo 10. The left side of the engine after extraction from the main wreckage. The red arrow points to the partially separated flanged connection between the free turbine case and the gas generator case. (Photo courtesy of AMC)

Photo 11. Axial compressor blade leading edges exhibiting evidence of hard body impact damage.

Photo 12. Free turbine rotor exhibiting evidence of blade shedding (module was removed from engine). (Photo courtesy of Turbomeca)

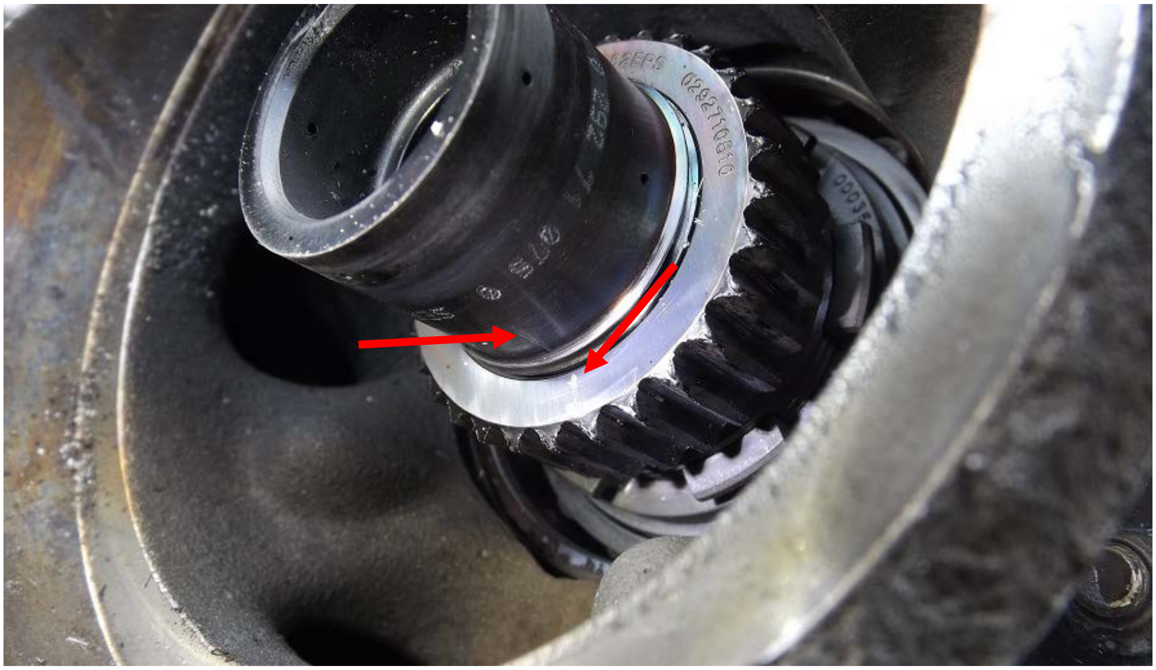

Photo 13. Red arrows point to the two index marks on the reduction gearbox splined nut and input gear. **(Photo courtesy of Turbomeca)**

The digital engine control unit (DECU) was located on the bottom side of the engine deck and exhibited heat distress. The engine data recorder (EDR) was consumed by the postcrash fire. The emergency backup control ancillary unit (EBCAU) was removed from the fuel control and its keyway was found to be in the 12 o'clock position. In normal governing mode (non-emergency mode), the keyway is at the 12 o'clock position. The vehicle engine monitoring display (VEMD) was located in the cockpit but exhibited evidence of heat distress.

The DECU were retained for further examination by the NTSB Recorders Laboratory. However, due to the severe thermal damage sustained by the DECU, no useful data could be recovered from these units. For additional details on the VEMD and DECU examination, see the Digital Engine Control Unit Specialist's Factual Report in the docket for this investigation.

3.0 WRECKAGE EXAMINATION AT BEEGLES AIRCRAFT SERVICES

On August 20, 2015, members of the Airworthiness Group convened at Beegles Aircraft Services in Greeley, CO, to search for the tail rotor hydraulic isolation valve and solenoid and to further examine the tail rotor head and directional control push-pull tubes.

The tail rotor hydraulic isolation valve and solenoid were not found in the recovered wreckage. The tail rotor head was disassembled from the tail gearbox output shaft. The output shaft key (also known as the "woodruff key") was found intact (**Photo 14**).

Photo 14. Output shaft key (red arrow) on the tail gearbox output shaft.

3.1 DIRECTIONAL CONTROL LATERAL PUSH-PULL TUBE

The AS350 B3e helicopter has two pedal mounts, one for the front-left seat and one for the front-right seat. Depending on the desired control configuration (e.g. single pilot controls versus dual pilot controls), a pedal set comprising two pedals (left and right) is installed at the desired pedal mount location. The directional control lateral push-pull tube is connected to both pedal mount bellcranks and provides coordinated movement between both pedal mounts. For example, pushing the left pedal forward on the left pedal mount will result in the same left pedal forward movement at the right pedal mount. The directional control lateral push-pull tube is located underneath the cockpit floorboard and is routed through a cutout in the sheet metal that is part of the floor structure.

The push-pull tube was found in its normally installed location in the wreckage. Continuity of control was confirmed between the left and right pedal mounts. The push-pull tube was found to have a dent in the upward direction, about mid-length of the push-pull tube. The dent was observed to be near this cutout. When the left pedal was manually moved to its full forward position, the dent on the push-pull tube moved laterally to the right side of the helicopter (**Photo 15**). When the right pedal was manually moved to its full forward position, the dent on the push-pull tube moved laterally to the left side of the helicopter. Furthermore, near the right pedal full forward position, the dent on the push-pull tube lined up with the sheet metal cutout within the floor structure (**Photo 16**).

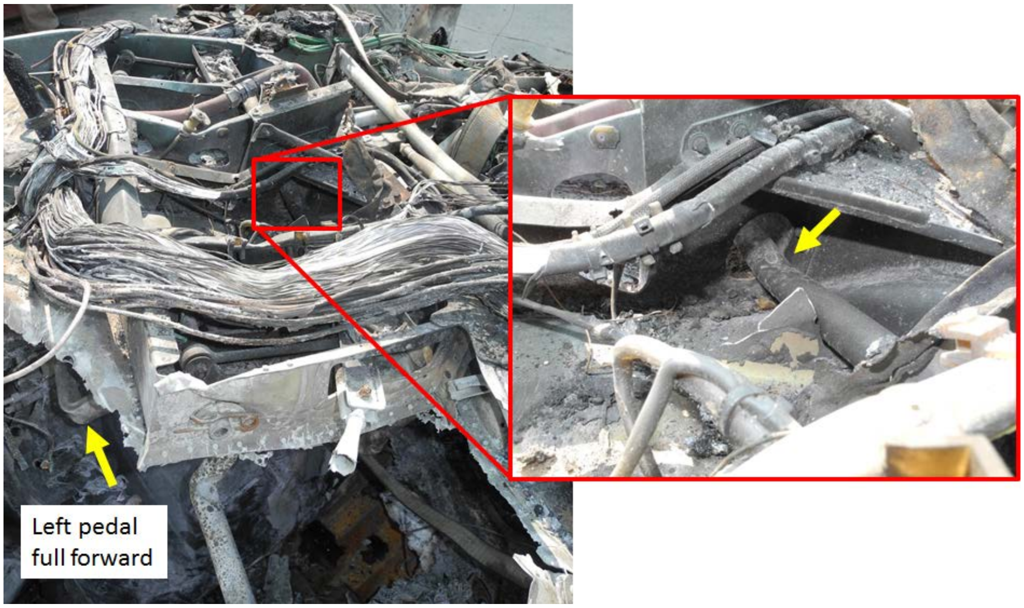

Photo 15. Dent on the lateral push-pull tube (yellow arrow in the inset photo) when the left pedal was placed fully forward.

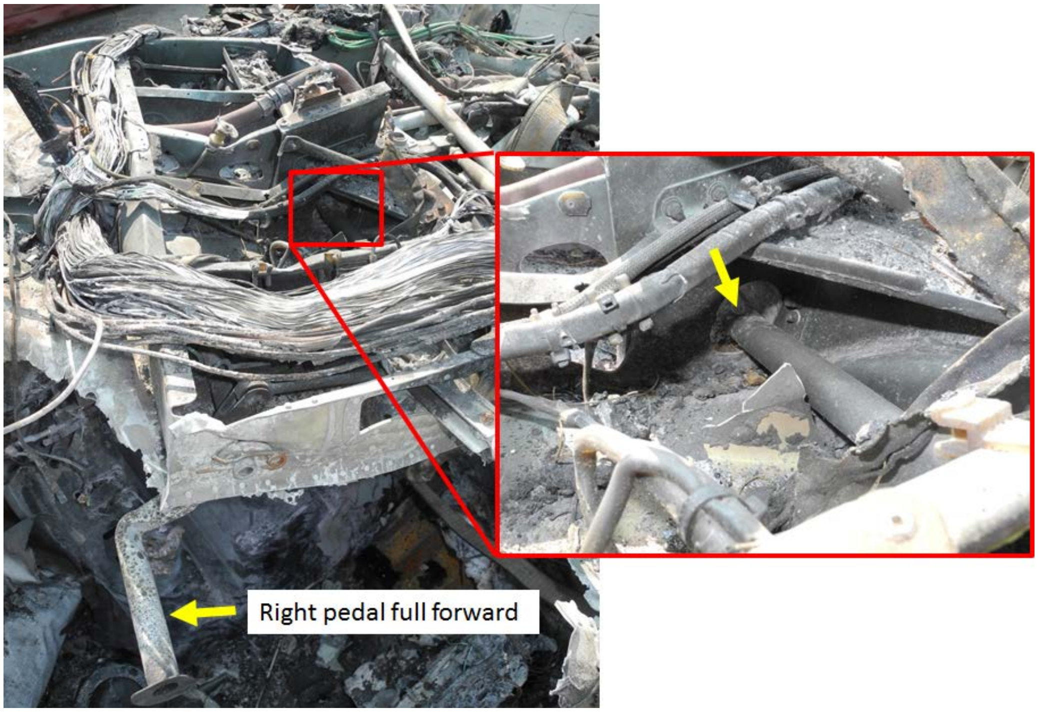

Photo 16. Dent on the lateral push-pull tube (yellow arrow in the inset photo) when the right pedal was placed fully forward.

4.0 TAIL ROTOR SERVO CONTROL AND YAW LOAD COMPENSATOR

On September 15-16, 2015, the tail rotor servo control and yaw load compensator were CT scanned at Varian Medical Systems in Chicago, Illinois. The scans revealed the tail rotor servo control manual lock pin (also known as the bypass pin) was engaged. The spring-loaded manual lock pin engages when hydraulic pressure is removed from the servo control. The edges of the control valve sleeve ports did not exhibit damage. Additionally, no foreign material was noted within the actuator filter. The yaw load compensator accumulator exhibited pitting on the inner walls of the accumulator.

Debris consistent with thermally decomposed rubber bladder was observed within the accumulator. The compensator piston appeared normally seated within the compensator body.

For additional details, see the Computed Tomography Specialist's Factual Report in the docket for this investigation.

5.0 COLLECTIVE STICK CONTROL BLOCK SWITCHES AND ACCU TEST PUSH BUTTON

The collective stick control block and ACCU TEST push button were submitted to the NTSB Materials Laboratory for x-ray scans. The position of the switches on the collective stick control block and of the ACCU TEST push button could not be conclusively determined due to the severity of damage sustained by those components. For additional details, see the Materials Laboratory Factual Report No. 15-127 in the docket for this investigation.

6.0 RECENT MAINTENANCE PERFORMED ON N390LG

According to Air Methods, N390LG was maintained under an FAA approved aircraft inspection program (AAIP). The maintenance history is presented from the most recent entries first.

According to the helicopter logbook records, on July 2, 2015, at ATT 483.5 flight hours, the following inspections were performed in accordance with Air Methods' AS350-series AAIP: a 15-hour/7 day inspection and a 25-hour inspection of the engine and VEMD, a 180-day inspection of various equipment installed on the helicopter, a 500-hour inspection and lubrication of the tail rotor drive shaft hanger bearings, and a 500-hour/12 month inspection of various installed equipment on the helicopter. Additionally, FAA Airworthiness Directive (AD) No. 2007-12-22 was accomplished; AD No. 2007-12-22 requires replacing the hydraulic fluid at a certain interval during cold weather operations.

On July 1, 2015, at ATT 481.0 flight hours, the following inspections were performed in accordance with Air Methods' AS350-series AAIP: a 30-hour inspection of the engine freewheel assembly, a 30-hour inspection of the tail rotor blades, and a 300-hour/180-day inspection of changing the engine oil and filter.

On June 30, 2015, at ATT 476.3 flight hours, a 200-hour/90-day inspection and a 12-month inspection[11] of various equipment installed on the helicopter was performed in accordance with Air Methods' AS350-series AAIP.

From June 3-8, 2015, at ATT 416.5 flight hours, a multi-day maintenance event occurred which included the removal and reinstallation of the hydraulic [pump] pulley assembly (to facilitate maintenance and inspections) and the replacement of a cracked hydraulic pump spring coupling.

7.0 AS350 B3E FLIGHT CONTROL HYDRAULIC SYSTEM HISTORY

The AS350 B3, approved under TCDS No. H9EU on May 7, 1998, was originally equipped with the Arriel 2B engine and a single[-cylinder] hydraulic system. Subsequently, the AS350 B3 was

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[11] The mechanic noted the 12-month inspection was not applicable since the equipment listed for inspection was not installed on the helicopter, thus the 12-month inspection entry was primarily for recordkeeping purposes.

provided the option of a dual hydraulic flight control system.[12] The AS350 B3 equipped with the Arriel 2D engine received type design approval on August 24, 2011, but was marketed by AHI, formerly American Eurocopter, as the AS350 B3e.

For the United States (US) market, AHI made the decision for AS350 B3e helicopters delivered in the US to be equipped with a dual hydraulic system as a standard option. Originally, the AS350 B3e dual hydraulic system was not equipped with a yaw load compensator, i.e. pedal hydraulic boost was primarily through the [single cylinder] tail rotor servo control. Design changes in the AS350 B3e tail rotor, primarily via modified chin weights incorporated through modification (MOD) No. 07-5601, reduced pedal control loads to acceptable levels when hydraulic assistance from the tail rotor servo control was lost. The design changes eliminated the need for the yaw load compensator normally found on the AS350 B3 (without MOD No. 07-5601 incorporated). Supplement 23 (SUP.23) of the AS350 B3e rotorcraft flight manual (RFM) contains normal and emergency procedures for the dual hydraulic system. SUP.23 included additional instructions for the preflight run up checks specific to the dual hydraulic system. These instructions included a tail rotor servo hydraulic check, requiring actuation of the collective-mounted yaw servo hydraulic switch to the “off” position[13], which closed the hydraulic isolation electrovalve and isolated the tail rotor hydraulic circuit from hydraulic pressure, and a check of the pedals for increased pedal control loads. The switch is subsequently reset to its normal position (to open the hydraulic isolation electrovalve) and pedals are checked for no increased load, i.e. normal hydraulic boost to the pedals.

Due to premature wear of the tail rotor laminated half-bearings caused by MOD No. 07-5601, MOD No. 07-5606 was later introduced by Airbus Helicopters, which removed the modified chin weights and added the yaw load compensator to the dual hydraulic system to keep pedal control loads at an acceptable level in the event of a loss of hydraulic assistance from the tail rotor servo control. For AS350 B3 helicopters with MOD No. 07-5601 incorporated, FAA AD No. 2014-05-10 imposed a never exceed velocity limitation of 100 knots indicated airspeed until MOD No. 07-5606 was incorporated. The yaw load compensator was required to be functionally checked during the preflight run up checks within SUP.23, which were modified to the following:

- First, the collective-mounted yaw servo hydraulic switch was set to the “off” position, closing the hydraulic isolation electrovalve and isolated the tail rotor hydraulic circuit from hydraulic pressure from the lower hydraulic circuit. Pedal control loads were checked to ensure they remained low due to the partial boost provided by the yaw load compensator, which remained hydraulically boosted.
- Second, the ACCU TEST push button, located on the center console, was activated, resulting in the ACCU TEST electrovalve opening and depressurizing the yaw load compensator accumulator. The pedals were checked for increased pedal control loads due to the depletion of the yaw load compensator accumulator.
- Third, the ACCU TEST push button was reset in order to close the ACCU TEST electrovalve.
- Fourth, the yaw servo hydraulic switch was set to the “on” position, opening the hydraulic isolation electrovalve, restoring hydraulic pressure to the tail rotor servo and the yaw load

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[12] Both the single and dual hydraulic systems for the AS350 B3 helicopter, equipped with the Arriel 2B or 2B1 engine, have a single cylinder tail rotor servo control and a yaw load compensator.

[13] The yaw servo hydraulic switch is in the “on” position during normal operation. When the yaw servo hydraulic switch is in the “on” position, the switch is pointed in the forward direction. When the yaw servo hydraulic switch is in the “off” position, the switch is pointed in the aft direction.

compensator. This step essentially resets the tail rotor hydraulic circuit back to its normal configuration.

Depressurization of the entire lower hydraulic circuit will result in the illumination of the "HYD" light on the cockpit caution and warning panel (CWP). However, in both the pre- or post-MOD No. 07-5606 configuration, and prior to the introduction of MOD No. 07-4622 (discussed later in this section), the actuation of the yaw servo hydraulic switch to the "off" position does not illuminate any lights on the CWP or generate any aural warning.

On August 21, 2014, after two events involving dual-hydraulic AS350-series helicopters taking off with an immediate left yaw and reported stuck pedals[14], Airbus Helicopters released Safety Information Notice (SIN) No. 2776-S-29. The SIN warned pilots that during the preflight run up hydraulic checks, if the step to restore hydraulic pressure to the tail rotor hydraulic circuit was omitted, the pilot could encounter difficulty in moving the pedals, i.e. the perception of locked, jammed, or stuck pedals. The SIN goes into detail explaining what each step of the preflight tail rotor hydraulic check accomplishes and how it affects the tail rotor hydraulic circuit. Attachment 1 contains Airbus Helicopters SIN No. 2776-S-29.

On February 25, 2015, Airbus Helicopters released SB No. AS350-67.00.64 which introduced MOD No. 07-4622 to modify the lights associated with the dual hydraulic system on the CWP. The SB modifies the CWP so that actuation of the yaw servo hydraulic switch to the "off" position would result in a flashing "HYD2" light on the CWP. According to AMC, Airbus Helicopters SB No. AS350-67.00.64 had not been incorporated into the accident helicopter (N390LG) at the time of the accident. Since the modification did not add a mechanism to detect the hydraulic pressure solely on the tail rotor hydraulic circuit, the flashing "HYD2" light was primarily an indication of the position of the yaw servo hydraulic switch (versus an indication of insufficient hydraulic pressure to the tail rotor hydraulic circuit). Attachment 2 contains Airbus Helicopters SB No. AS350-67.00.64.

On August 26, 2015, Airbus Helicopters released SB No. AS350-67.00.66 to modify the hydraulic checks of the yaw load compensator in the RFM for the AS350 B3 helicopter, including the AS350 B3e. SB No. AS350-67.00.66 affected AS350 B3 helicopters equipped with either a single or a dual hydraulic system. SIN No. 2944-S-29 was issued in conjunction with the SB to explain the flight manual changes introduced by the SB. For dual hydraulics AS350 B3 and AS350 B3e helicopters, the preflight run up hydraulic checks within SUP.23 were modified to remove the tail rotor hydraulic check in its entirety. Procedures to check the yaw load compensator were added to the engine and rotor shutdown (postflight) procedures. On October 22, 2015, Revision 1 of SB No. AS350-67.00.66 was released; the SB stated the reasons for the revision were to provide procedures for inserting appendices into the RFM and make improvements in the appended pages. Attachment 3 contains Airbus Helicopters SB No. AS350-67.00.66 Revision 1.

Subsequent to the release of Revision 1 of Airbus Helicopters SB No. AS350-67.00.66, on October 28, 2015 the FAA released emergency AD No. 2015-22-52 to prohibit performing the yaw load

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[14] The first event was an accident which occurred on April 9, 2014 in Albuquerque, New Mexico with an AS350 B3e helicopter, N395P, which impacted a hospital roof top following departure from the roof top helipad and sustained substantial damage (NTSB case number CEN14FA193). The second event was an incident which occurred on June 26, 2014 in Temple, Texas, with an AS350 B3 helicopter, N808LF, which successfully performed a hovering autorotation shortly after takeoff and did not sustain substantial damage (NTSB case number CEN14IA329). Both events were nonfatal to persons on board the helicopters. More information about these investigations available on the NTSB website at http://www.ntsb.gov/_layouts/ntsb.aviation/index.aspx

compensator check during the preflight procedures and to instead perform the check during the postflight procedures. Movement of the procedures to the postflight procedures would eliminate the risk of a pilot taking off with the yaw servo hydraulic switch inadvertently in the “off” position, which would result in a loss of hydraulic boost to the pedal controls. The AD affected all AS350 B3 helicopters, including the B3e variant. A subsequent emergency AD, No. 2015-22-53, released on October 30, 2015, superseded emergency AD No. 2015-22-52; this superseding emergency AD corrected an error in terminology and a defect in recording compliance with emergency AD No. 2015-22-52. FAA AD No. 2015-22-53 was subsequently entered into the Federal Register on December 1, 2015. Attachment 4 contains FAA AD No. 2015-22-53.

8.0 N390LG PERFORMANCE STUDY AND SURVEILLANCE VIDEO ANALYSIS

8.1 N390LG CALCULATED PEDAL CONTROL LOADS DURING HOVER

14 CFR 27.395 and 27.397 discusses requirements for control system design, including limit forces and torques for the control system. For foot controls, a limit of 130 pounds is prescribed. The Airworthiness Group Chairman requested Airbus Helicopters calculate the pedal control loads required for an AS350 B3e to maintain a stationary heading when hover in ground effect (HIGE) using ambient conditions representative of the day of the accident. A helicopter weight of 4,440 pounds, an ambient temperature of 73.4 degrees Fahrenheit, and an altitude of 9,170 feet (mean sea level) were utilized for the calculations. Two wind conditions were analyzed: a 0 knot “no wind” condition and a 15 knot wind originating from the helicopter’s left side (when in the pilot’s seat), the latter being representative of the estimated steady state wind conditions at the time of the accident.[15] Three different tail rotor hydraulic circuit configurations affecting pedal control loads were considered. The first configuration represented normally boosted pedals, with the yaw servo hydraulic switch in the “on” position and the ACCU TEST button deactivated. The second configuration represented partially boosted pedals, with the yaw servo hydraulic switch in the “off” position and the ACCU TEST button deactivated. The third configuration represented non-boosted pedals, with the yaw servo hydraulic switch in the “off” position and the ACCU TEST button activated. **Table 1** shows the calculated right pedal control load provided by Airbus Helicopters for the aforementioned conditions. Given a pedal position of 0% for full left pedal and 100% for full right pedal, in a “no wind” condition, a pedal position of 78% (right pedal) was calculated to be required to maintain a steady heading in HIGE using the aforementioned conditions; in a 15 knot left wind condition, a pedal position of 83.5% (right pedal) was required to maintain a steady heading in HIGE.

Table 1. Calculated right pedal control load (italicized) for three tail rotor hydraulic circuit configurations to maintain a steady heading when HIGE using ambient conditions from the day of the accident.

Tail rotor servo control condition	Yaw load compensator accumulator condition	Calculated pedal control loads with no wind	Calculated pedal control loads with 15 knot wind from left
Boosted	Charged	*~ 3-5 pounds*	*~3-5 pounds*
No boost	Charged	*~29 pounds*	*~45 pounds*
No boost	Depleted	*~142 pounds*	*~161 pounds*

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[15] For additional meteorology information, see the Meteorology Group Chairman’s Factual Report in the docket for this investigation.

8.2 PEDAL CONTROL LOADS DURING THE PREFLIGHT HYDRAULICS CHECK

The pedal control loads experienced by a pilot during the preflight hydraulics checks, to verify proper functionality of the yaw load compensator, were expected to be different than that of the pedal control loads at takeoff and in flight. On May 6, 2016, NTSB investigators and representatives from the FAA, AHI and AMC convened at AHI facilities in Grand Prairie, TX, to measure the pedal control loads on an exemplar [dual hydraulics] AS350 B3e helicopter after performing the preflight hydraulic system check steps in various combinations. The measurements were taken with the helicopter on the ground with the engine twist grip in the "idle" position. The winds were about 5 knots originating from a magnetic heading of 170°; the helicopter's magnetic heading was 180°. The exemplar helicopter was configured to a weight of 4,720 pounds. A handheld force gauge was used to measure the pedal control load on both the left and right pedals at the end of each step combination. Furthermore, three measurements were taken for both the left and right pedals at the end of each step combination in order to provide consistency in pedal control load measurements. A total of 11 step combinations were performed, resulting in four unique configurations for the condition of both the tail rotor servo control and yaw load compensator accumulator. **Table 2** provides the average measured pedal control loads for the four unique tail rotor hydraulic circuit configurations. For additional details of this activity, see the Human Performance Specialist's Factual Report in the docket for this investigation.

Table 2. Measured pedal control loads (italicized) for four different tail rotor hydraulic circuit configurations with the helicopter on the ground with rotors turning at the ground idle setting.

Tail rotor servo control condition	Yaw load compensator accumulator condition	Left pedal control load	Right pedal control load
Boosted	Charged	*~3 pounds*	*~2 pounds*
Boosted	Depleted	*~3 pounds*	*~2 pounds*
No boost	Charged	*~60 pounds*	*~34 pounds*
No boost	Depleted	*~28 pounds*	*~39 pounds*

8.3 SURVEILLANCE VIDEO ANALYSIS

The NTSB Vehicle Performance division analyzed the surveillance videos showing the helicopter taking off the helipad and the ground impact sequence. The analysis estimated the following:

- An average yaw rate of 30 degrees per second over the 12 seconds showing the helicopter takeoff.
- An average yaw rate of 22 degrees per second for the first 8 seconds of the helicopter takeoff.
- An average yaw rate of 45 degrees per second the following 4 seconds of the helicopter takeoff.
- An average vertical climb rate of 5.6 ±0.5 feet per second, shortly after takeoff.
- An estimated ground impact speed of 58±5 feet per second.

For additional details of the surveillance video study, see the Video Study report in the docket for this investigation.

8.4 N390LG PERFORMANCE STUDY

The Airworthiness Group Chairman requested Airbus Helicopters perform simulations of the estimated left yaw rate that would develop for a given reduction in right pedal input. The yaw rate simulation used the same ambient conditions as the pedal control loads simulation except that only a 15 knot left wind condition was used. Furthermore, the yaw rate simulation only considered a scenario with the tail rotor servo control boosted and the yaw load compensator accumulator charged. After 1 second in a steady heading HIGE, the right pedal input (initially at a pedal position of 83.5%[16]) was reduced by a range of 10% to 30%, in 5% increments. The simulation estimated that 4 seconds after a 10% right pedal reduction (to a pedal position of 73.5%), the yaw rate would be about 45 degrees per second.[17]

The NTSB Vehicle Performance division performed FlightLab simulations of an AS350 helicopter on the required pedal inputs with varying wind conditions (both in azimuth and in magnitude) when at a hover in ground effect (HIGE). The Airworthiness Group Chairman requested Airbus Helicopters perform simulations using the same criteria utilized in the FlightLab simulation. Both the FlightLab and Airbus Helicopters simulations concluded that adequate right pedal margin should be available given the estimated steady state wind conditions at the time of the accident.

For additional details on both simulations, see the Performance Study Specialist's Report in the docket for this investigation.

9.0 FUEL SYSTEM CRASH RESISTANCE STANDARDS

On October 3, 1994, the FAA improved the crash resistance standards for normal category helicopter fuel systems via amendment 27-30 to 14 CFR 27.952.[18] The intent of the improved standards was to minimize fuel spillage near ignition sources in order to improve the evacuation time needed for crew and passengers to escape a postcrash fire. However, the improved crash resistance standards were not retroactively applicable to newly manufactured helicopters whose certification basis and approval predated the effectivity of amendment 27-30.

All versions of the AS350-series helicopter hold type design approvals under TCDS No. H9EU. The first of the AS350-series helicopter approved in the US, the AS350 C, initially received FAA type certificate design approval on December 21, 1977. The certification basis for all AS350-series helicopters approved under TCDS No. H9EU consist of 14 CFR 21.29 and Part 27 effective February 1, 1965, plus the requirements contained in amendments 27-1 through 27-10. There are no exemptions or equivalent levels of safety findings listed for the AS350-series helicopter's fuel tank within TCDS No. H9EU. Thus, there are no regulatory requirements for the AS350-series helicopter to be equipped with a crash resistant fuel system (CRFS) meeting the standards of amendment 27-30 to 14 CFR 27.952.

The certification basis for the Airbus Helicopters EC130 B4, which received type certificate design approval under TCDS No. H9EU on December 21, 2000, consist of 14 CFR 21.29 and Part 27, including amendments 27-1 through 27-32, but 14 CFR 27.952 was not adopted. However, the

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[16] Full right pedal was defined as a pedal position of 100% and full left pedal was defined as a pedal position of 0%.

[17] According to Airbus Helicopters, the yaw simulations tended to overestimate the yaw rate as the simulation could not consider factors such as the aerodynamic effect of the tailboom and vertical fin and wind effects during rotation.

[18] 59 *Federal Register* 50380, October 3, 1994. The revised airworthiness standards became effective on November 2, 1994.

certification basis for the EC130 T2, which received type certificate design approval under TCDS No. H9EU on July 30, 2012, did include compliance with 14 CFR 27.952 (amendment 27-30).

On July 23, 2015, the NTSB made a safety recommendation to the FAA to require, for all newly manufactured helicopters regardless of the design's original certification date, that the fuel systems meet the improved fuel system crash resistance requirements of 14 CFR 27.952 or 29.952.[19] The basis of the safety recommendation was the analysis that from the time 14 CFR 27.952 and 29.952 was updated in 1994, only about 15% of newly manufactured helicopters were equipped with fuel systems that met the improved standards. On September 28, 2015, the FAA responded that they agreed with the recommendation and started the rulemaking process by sending a tasking statement to the aviation rulemaking advisory committee. The safety recommendation is currently classified as "open – acceptable response".

At the time of the accident, N390LG was not equipped with a CRFS, nor was it required to be equipped with a CRFS. However, according to AHI and AMC, prior to the accident there were no available retrofit options to equip an AS350 B3e helicopter with a CRFS which met the improved standards of 14 CFR 27.952. According to AHI, in March 2015, a CRFS was introduced into AHI's AS350 B3e final assembly line as a standard option; the first of those helicopters equipped with a CRFS was delivered in late 2015. On March 3, 2016, Airbus Helicopters released SB No. AS350-28.90-22 to provide the option to retrofit existing AS350 B3e helicopters with a CRFS.[20]

Chihoon Shin
Aerospace Engineer – Helicopters

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[19] More information about this safety recommendation, NTSB Safety Recommendation No. A-15-12, is available on http://www.ntsb.gov/safety/safety-recs/_layouts/ntsb.recsearch/RecTabs.aspx

[20] According to Airbus Helicopters, the AS350 B3e CRFS is based the design of the EC130 T2 CRFS, the latter of which was certified to the standards of 14 CFR 27.952. However, as of the date of this report, the CRFS for the AS350 B3e has not yet been certified to the standards of 14 CFR 27.952.