National Transportation Safety Board
Office of Aviation Safety
Washington, DC 20594

CEN23FA071

AIRWORTHINESS

Group Chair's Factual Report

October 25, 2023

A. ACCIDENT

Location: Galliano, Louisiana
Date: December 29, 2022
Time: 0834 central standard time
Helicopter: Bell 407, registration N595RL

B. AIRWORTHINESS GROUP

Group Chair	Chihoon Shin National Transportation Safety Board Washington, District of Columbia
Group Member	Gary Howe Bell Fort Worth, Texas
Group Member	Dave Riser Rolls-Royce Indianapolis, Indiana
Group Member	Curtis Hudson Rotorcraft Leasing Company Broussard, Louisiana
Group Member	Robert Hardwick Federal Aviation Administration Baton Rouge, Louisiana

C. SUMMARY

On December 29, 2022, about 0834 central standard time, a Bell 407 helicopter, N595RL, operated by Rotorcraft Leasing Company (RLC), was destroyed when it crashed during takeoff from an offshore oil platform[1] in the Gulf of Mexico. The pilot and three passengers were fatally injured. The helicopter was operated as a Title 14 *Code of Federal Regulations* Part 135 flight.

The helicopter was submerged in the water near the offshore oil platform for about 4 days prior to its recovery. On January 2-3, 2023, the helicopter wreckage was located submerged in the water near the offshore platform, recovered, and transported to Southern Aircraft Recovery in Baton Rouge, Louisiana. On January 4-5,

[1] The offshore oil platform was operated by Walter Oil and Gas Corporation.

2023, members of the Airworthiness Group convened at Southern Aircraft Recovery to examine the recovered wreckage. On February 15, 2023, onboard data from the accident engine control unit (ECU) was downloaded at Triumph Engine Control Systems in West Hartford, Connecticut. From March to April 2023, the lateral servo actuator was scanned using computed tomography (CT) to assess the internal condition of the actuator. On May 5, 2023, the lateral servo actuator was examined and bench tested at Woodward, Inc. in Santa Clarita, California. On July 25, 2023, the Airworthiness group re-examined the flight control system and landing gear at Southern Aircraft Recovery in Baton Rouge, Louisiana.

D. DETAILS OF THE INVESTIGATION

1.0 Helicopter Information

1.1 Helicopter Description

The Bell 407 is type certificated under FAA type certificate data sheet (TCDS) No. H2SW. The Bell 407 has a four-bladed, fully-articulated main rotor system that provides helicopter lift and thrust, and a two-bladed teetering tail rotor system that provides thrust to counteract main rotor torque and for directional control of the helicopter. The helicopter is equipped with one Rolls-Royce Model 250-C47B turboshaft engine, mounted behind the main transmission, which has a maximum continuous power rating of 630 shaft horsepower per TCDS No. H2SW. The helicopter was equipped with a skid-type landing gear.

1.2 Accident Helicopter History

The accident helicopter was serial number (S/N) 53595 and was manufactured in 2004. The engine installed on the accident helicopter was S/N CAE-847406. According to helicopter records, the helicopter had an aircraft total time (ATT) of 16,722.0 flight hours when it landed on the offshore oil platform on December 29, 2022, immediately before the accident.

2.0 Wreckage Examination[2]

2.1 Structures

2.1.1 Overview

The Bell 407 helicopter fuselage is composed of three primary structural assemblies: the forward fuselage, the intermediate fuselage, and the tail boom. The forward fuselage begins at the nose of the helicopter and extends to the rear of the

[2] All left, right, up and down orientations as well as clock positions referenced in this report are in the aft-looking-forward frame of reference unless otherwise specified.

passenger compartment. The intermediate fuselage begins at the rear of the passenger compartment and ends at the front of the tail boom. The tail boom begins at the aft fuselage and ends at the vertical stabilizer and tail skid.

The forward fuselage is composed of an aluminum roof beam, supporting the transmission and the forward section of the engine, honeycomb panels for the floor and roof, and forward and aft bulkhead assemblies and a center post to connect the floors and the roof. The forward fuselage is the primary structure for crew and passenger seating, the fuel cell enclosure, the landing gear, and the transmission support. The intermediate fuselage is a semi-monocoque construction composed primarily of aluminum and composite skins. The intermediate fuselage provides support for the engine compartment, engine oil tank, oil cooler and its blower, and the steel tail rotor drive shaft. The tail boom is a monocoque shell composed primarily of aluminum. The tail boom supports the tail rotor drivetrain, the tail rotor, and the horizontal and vertical stabilizers.

2.1.2 Main Fuselage

The main fuselage was recovered as a single piece except for the landing gear, which had separated from the main fuselage (**Figure 1**). The main beam was present and attached to the control column (broom closet), but the broom closet was fractured and partially separated near its upper end. The tail boom had fractured and separated near its forward end, about a foot aft of the tail boom attachment point. The left cockpit door remained installed, but its frame was partially detached from the airframe. The right cockpit door was not present. None of the windows in the cockpit were present.

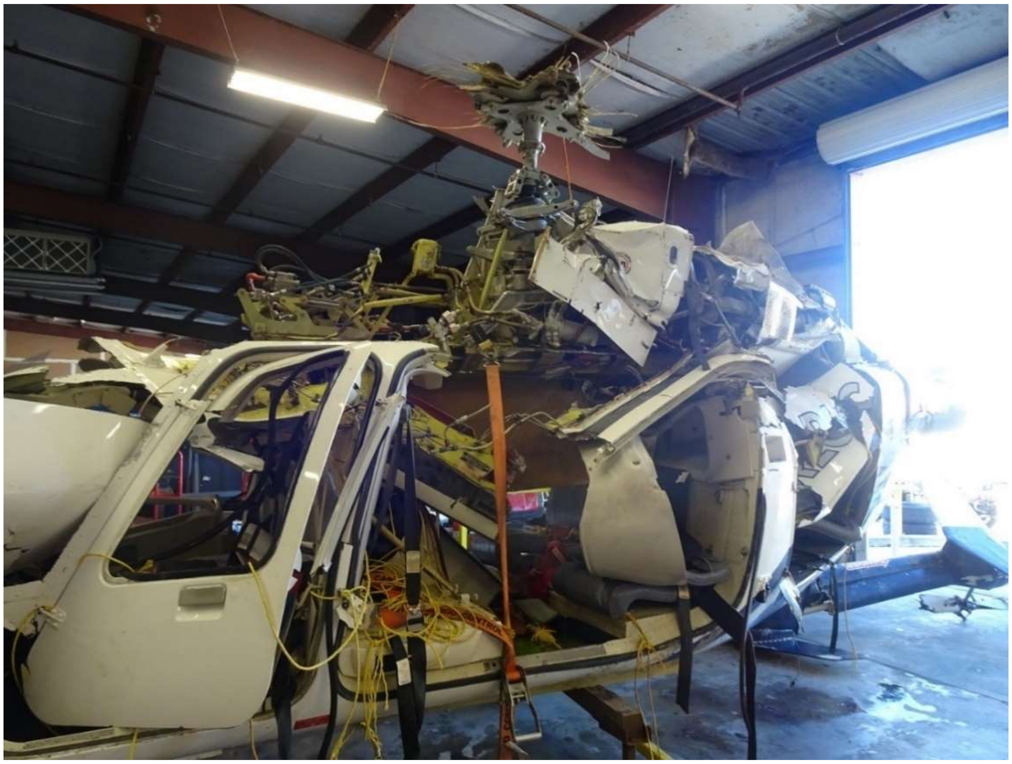

Figure 1. The recovered main fuselage.

The forward upper cowling, covering the main rotor servo actuators, had fractured into multiple pieces and was retained with the airframe via wire bundles. Significant areas of red paint transfer markings were observed on the forward upper cowling (**Figure 2**). Red paint transfer was also observed on the upper-right side of the airframe near the forward-upper corner of the right cabin door. The roof shell assembly was present but crushed and partially separated at the upper portion of the left-side cabin door frame and the forward and aft upper-corners of the right-side cabin door frame. In the intermediate fuselage, the deck between the engine pan and the baggage compartment had deformed downward (into the baggage compartment). The engine cowlings were present but exhibited deformation.

Figure 2. The forward upper cowling exhibiting red paint transfer.

All occupant seats were present except for the pilot seat (right cockpit seat). None of the occupant seat pans exhibited deformation except for the cockpit-left seat pan, which exhibited downward deformation.

2.1.3 Tail Boom

Based on photographs taken by persons on the offshore platform shortly after the accident occurred, the tail boom was floating adjacent to and had separated from the fuselage. However, it could not be determined based on these photographs if the tail boom was partially or fully separated from the fuselage. The tail boom was recovered fully separated from the fuselage. The vertical fin remained attached to the aft end of the tail boom and exhibited impact damage on the bottom side of the lower fin. The tail skid assembly (stinger) had separated from the lower vertical fin but was not recovered. The right side of the horizontal stabilizer was fractured chordwise but remained partially attached via its upper skin. The right finlet remained installed. The left side of the horizontal stabilizer had fractured chordwise at its inboard end and was not part of the recovered wreckage.

2.1.4 Landing Gear

The high-skid landing gear was recovered as a single assembly with the emergency floats still attached. Photographs taken by persons on the offshore platform shortly after the accident occurred showed the landing gear separated from the airframe, both floating near each other and inverted. In these photographs, the emergency floats at all six positions, three on the left skid (forward, mid, and aft) and three on the right skid, were inflated. At the time the landing gear was examined by the Airworthiness Group, the emergency floats at all six positions had deflated. The emergency flotation system’s gas reservoir remained installed on the belly of the fuselage.

The high-skid landing gear had separated from the main fuselage due to fractures on all four saddle mounts (**Figure 2**). The forward-left mount plate had fractured with three of the four bolts still installed with remnants of the mount plate attached. The fourth bolt had fractured with remnants of the bolt still within the bolt hole. The forward-right mount plate was present, and its four bolts remained installed, but the plate had fractured where it mounts to the landing gear. A portion of the aft-left mount plate remained installed with four bolts, but the aft portion of this plate had fractured and separated along with its two aft bolts. The aft-right mount remained installed on the fuselage but had fractured at the mount.

Figure 2. The recovered landing gear and floats.

The forward and aft crosstubes did not exhibit significant deformation. The two forward mounts remained attached to the forward crosstube. The two aft mounts were not present on the aft crosstube, but the pivot plate, located at the center of the aft crosstube, remained installed. The left skid was generally [longitudinally] straight but exhibited impact deformation where the left skid is attached to the aft crosstube. The right skid toe (forward) section was laterally bent outboard while the right skid heel (aft) section was laterally bent inboard. The conical end cap of the right skid had impact damage (**Figure 3**). No other significant impact damage was observed on the remainder of the left and right landing skids.

Figure 3. Damage to the conical end cap on the aft end of the right landing skid.

Two distinct groups of impact marks were present on the accident helipad. One group of impact marks exhibited an arc-like appearance (**Figure 4**). The second group of impact marks exhibited a linear appearance (**Figure 5**). The spacing of these arc and linear impact marks were consistent with the spacing of the bolt heads on the right skid that attach the skid shoe to the skid starting at the forward crosstube (**Figure 6**). The white-colored paint on these skid shoe bolt heads was not present, but white-colored paint was present on adjacent bolt heads.

Figure 4. A group of impact marks on the helipad with an arc-like appearance.

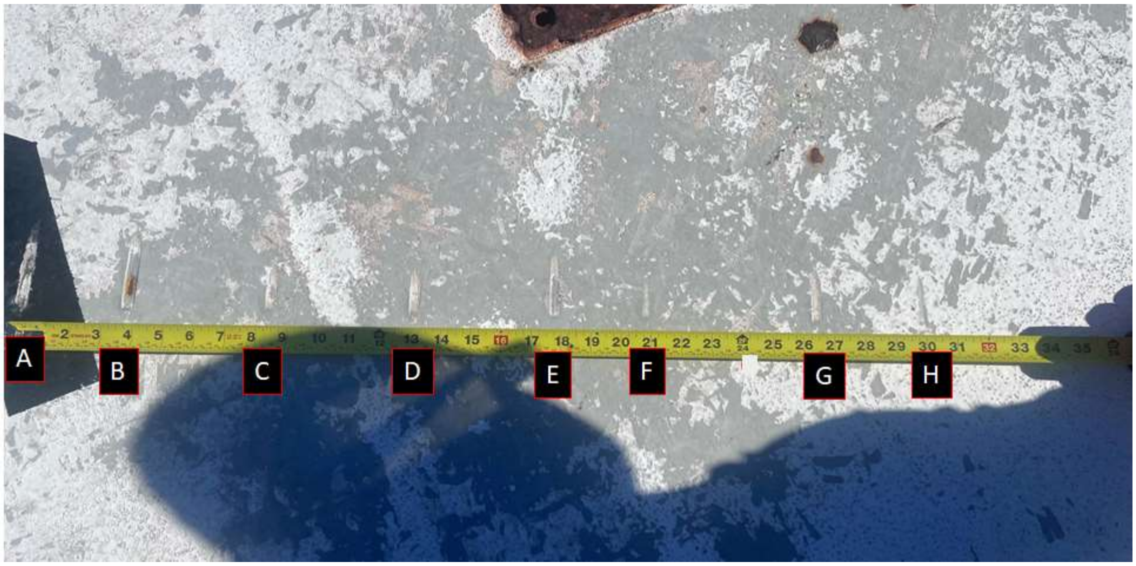

Figure 5. A group of impact marks on the helipad with a linear appearance.

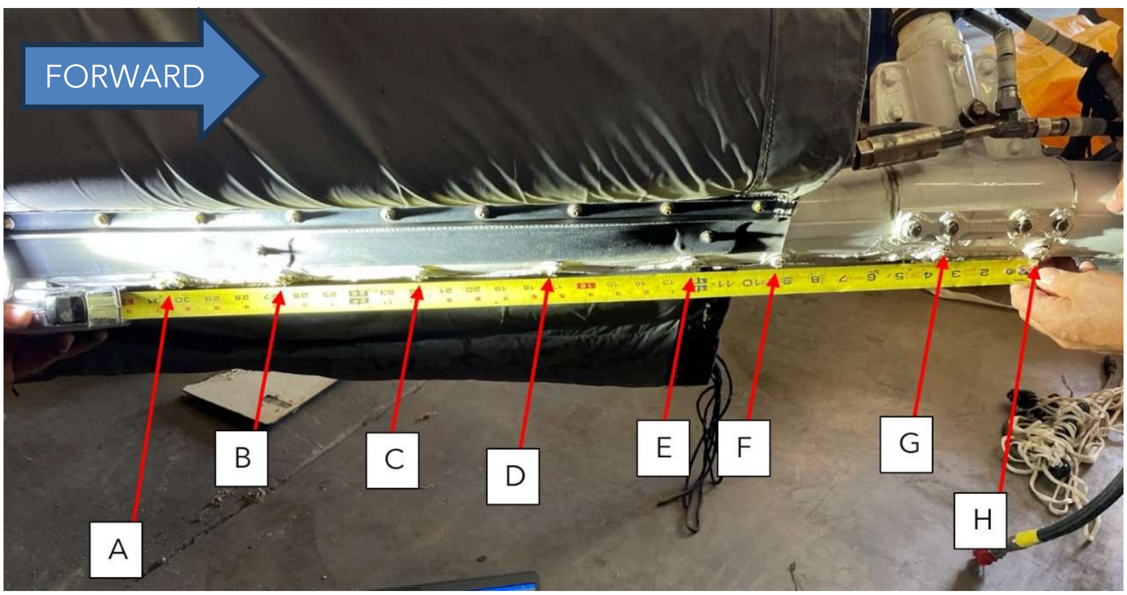

Figure 6. Damage seen on the bolt heads on the right skid.

The helipad was generally square in shape, painted white with the border perimeter, target circle, and logo painted in black (**Figure 7**). A chain-link fence perimeter surrounded the helipad. A red-painted landing with stairs was located on the western edge of the helipad. The helipad surface, excluding the fence perimeter, was about 24 feet by 24 feet. At the perimeter of the helipad, orange-colored lights were installed at the four corners of the helipad and blue-colored lights were installed about midpoint between each corner.

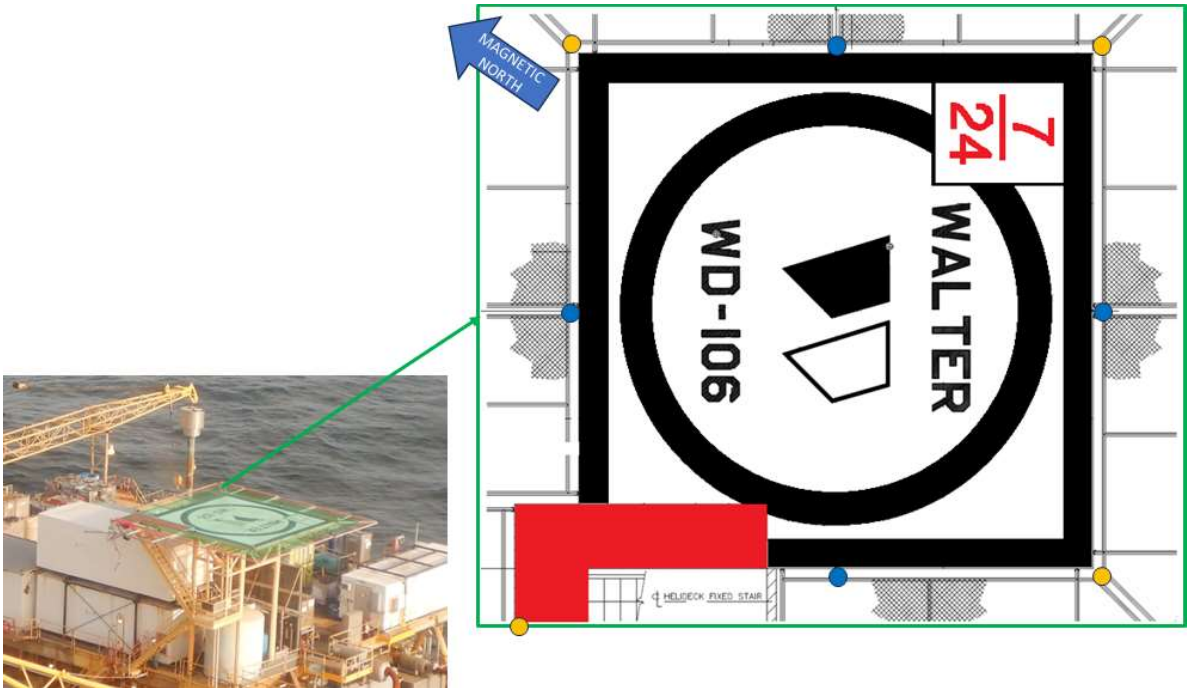

Figure 7. A diagram of the helipad with the location of the four orange-colored lights and the four blue-colored lights. (Right image courtesy of Walter Oil and edited by the NTSB)

The distance from the aft end of the right skid to the skid shoe bolt head that matched impact mark "A" (from Figure 6) was about 104.5 inches, or about 8.5 feet. Based on this distance and the platform dimensions provided by the platform operator, the aft end of the right skid (the conical end cap) would overlap with the blue-colored light near the northwestern edge of the helipad and whose light housing exhibited impact deformation and its blue-colored glass globe was shattered (**Figures 8, 9, and 10**).

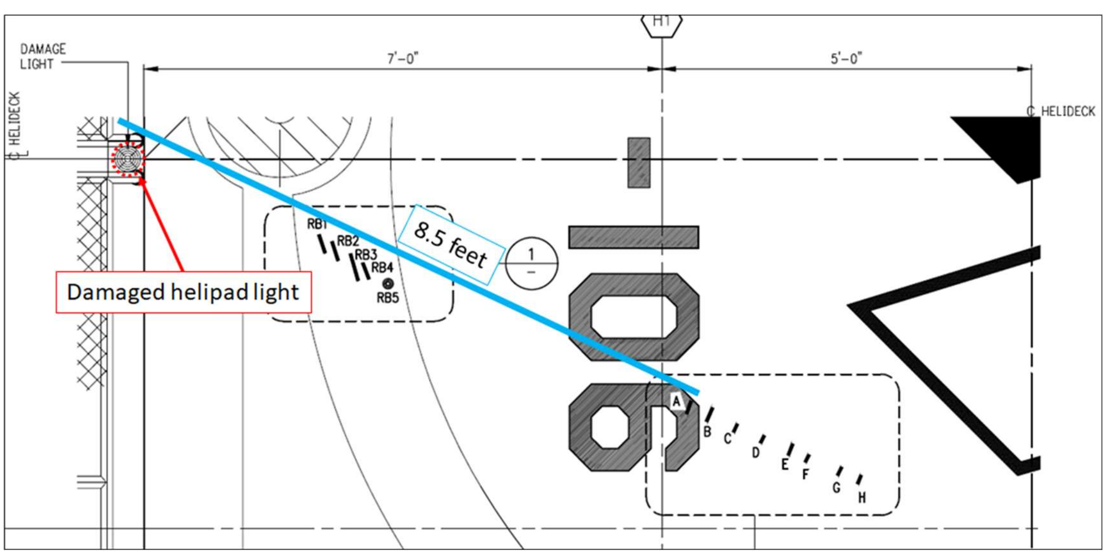

Figure 8. A diagram of the distance between impact mark "A" and the damaged blue-colored light on the helipad. (Image courtesy of Walter Oil and edited by the NTSB)

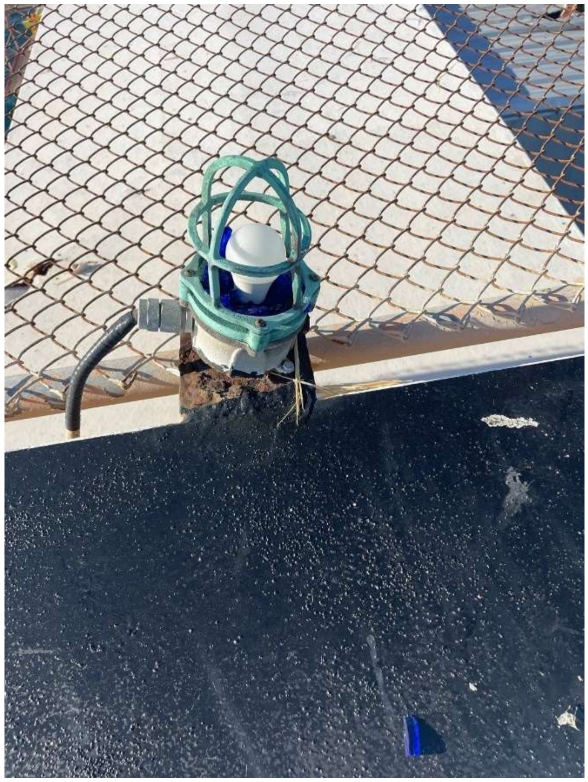

Figure 9. The damaged blue-colored light found at the helipad. The platform operator stated they had cut the light housing's installation bolts to facilitate its removal from the helipad. (Image courtesy of Walter Oil)

Figure 10. The damaged blue-colored light.

2.2 Main Rotor System

2.2.1 System Overview

Power from the engine accessory gearbox is transferred through a freewheeling unit to the main rotor gearbox via the engine-to-transmission (input) drive shaft. KAflex couplings on both the forward and aft ends of the input drive shaft allow for minor misalignment. The accident helicopter was equipped with a rotor brake which was installed between the engine accessory gearbox output flange and the aft KAflex coupling of the input drive shaft. The main rotor gearbox contains a single-stage sun and planetary gear system that turns the main rotor shaft (also known as the main rotor mast). The main rotor hub is splined onto the main rotor shaft. The main rotor gearbox is attached to the airframe via pylon assemblies containing elastomeric mounts for vibration dampening.

The main rotor hub is composed of a yoke which is mounted between an upper and lower [hub] plate. Each main rotor blade is connected to the yoke via a grip assembly. An elastomeric thrust bearing that accommodates blade pitch changes, a pitch horn, and an elastomeric shear (pivot) bearing are attached to the blade grip. An elastomeric lead-lag damper bearing, installed inboard of the pitch horn, is connected to the pivot bearing as well as both the upper and lower hub plates. The four main rotor blades are identified by color and the shape of identification stickers adhered to each rotor blade, presented in the order of advancing rotation: 'blue' (diamond), 'orange' (square), 'red' (triangle), and 'green' (circle). The main rotor blades are composite in construction, with a fiberglass spar, fiberglass skin, and Nomex honeycomb core. Blade pitch control is achieved via pitch change links (PCL) connected between each blade's pitch horn and the rotating swashplate.

2.2.2 Main Rotor

The upper and lower plates, to which the main rotor yoke is attached, remained installed on the main rotor mast. The main rotor yoke exhibited a broomstrawed appearance. Photos from the offshore platform showed one large segment of the 'green' main rotor blade (composed of a portion of the yoke, spindle, and the inboard segment of the blade) was present on the platform as well as many smaller fragmented pieces from the main rotor blades. The large segment of the 'green' main rotor blade was recovered. The main rotor cover and main rotor Frahm damper assembly were removed by the operator prior to the accident flight; these items are optional.

2.2.3 Main Rotor Drive System

The main transmission remained attached to the airframe. The main transmission housing exhibited corrosion due to saltwater immersion. The upper chip detector was removed and contained no magnetic chips. The lower chip detector could not be removed from the housing. The main transmission input shaft (KAflex driveshaft) was present and remained attached to the main transmission input quill but exhibited multiple fractures at the aft KAflex coupling. Rotational scoring was observed on the KAflex driveshaft. The airframe-mounted oil filter was present, and its bypass indicator had not extended. The airframe-mounted fuel filter bypass indicator had partially extended.

2.3 Tail Rotor System

2.3.1 System Overview

Power from the freewheeling unit is transferred to the tail rotor via the tail rotor drive system. The tail rotor drive system contains a total of 6 tail rotor drive shafts: 1) the forward short shaft, composed of steel, mounted between the freewheeling unit and the oil cooler blower; 2) the aft short shaft, composed of aluminum, attached to the aft side of the oil cooler blower; and 3) four additional tail rotor drive shafts, also composed of aluminum, that leads up to the tail rotor gearbox. A flexible coupling disc pack is present at the connecting flanges of each tail rotor drive shaft to allow for axial and angular misalignments. Starting with the aft short shaft, the aluminum tail rotor drive shafts are numbered 1 through 5. The tail rotor gearbox, mounted near the aft end of the tail boom, changes the direction of drive and also reduces the output speed of rotation at the tail rotor. The tail rotor gearbox output shaft connects to the tail rotor yoke.

The two-bladed tail rotor is a semi-rigid (teetering) rotor system. Each tail rotor blade, composite in construction, is attached to the tail rotor yoke via two spherical bearings. The two spherical bearings accommodate blade pitch changes. The tail rotor blades are identified by color as 'white' and 'red'.

2.3.2 Tail Rotor Drive System

The tail rotor drive system was present in the wreckage. The forward short shaft (steel) had separated from the forward end of the oil cooler blower shaft and remained connected to the freewheeling unit. The oil cooler blower shaft and its blower fan (commonly referred to as a "squirrel cage") remained installed on the airframe but could not be rotated by hand due to damage. The aft short shaft (aluminum, No. 1) had separated from the aft splines of the oil cooler blower shaft but was retained with the airframe via its aft hangar bearing. The aft short shaft had bent about mid-length of the shaft. The No. 2 aluminum drive shaft segment had separated from its riveted adapter and was retained within the tail boom wreckage. The five aluminum drive shaft were contained within the tail boom. The No. 5 aluminum drive shaft was connected to the tail gearbox. All hangar bearings remained installed on the tail boom. The tail rotor drive shaft could not be rotated by hand due to seizure of the tail gearbox.

The tail gearbox remained installed on the aft end of the tail boom but its housing exhibited significant corrosion due to saltwater immersion. The tail rotor could not be rotated due to seizure within the gearbox, likely due to the extensive corrosion of the tail gearbox. The tail gearbox chip detector was removed, and its magnetic end contained magnesium oxide deposits, but no evidence of magnetic chips.

2.3.3 Tail Rotor

Both tail rotor blades was fractured just outboard of the tail rotor yoke (**Figure 11**). The inboard ends of both tail rotor blades remained retained within the yoke. Both tail rotor blades exhibited fractures of the composite material with a broomstrawed appearance. Fragmented portions of the outboard sections of the tail rotor blades were observed at the helipad.

Figure 11. The tail rotor and pitch change links.

2.4 Flight Control System

2.4.1 System Overview

The cockpit flight control system is composed of cyclic, collective, and directional (pedal) controls. The mechanical linkages for the pilot (right seat) flight controls are routed below the pilot and passenger seats, aft to the center of the helicopter, and vertically, in a compartment known as the "broom closet", to the cabin roof. On the cabin roof, the cyclic and collective flight control linkages connect to three [hydraulic] servo actuators that are installed on the hydraulic rack in front of the main rotor gearbox. The three servo actuators are identified as lateral cyclic (left), collective (center), and longitudinal cyclic (right). The lateral and longitudinal cyclic main rotor servo actuators provide hydraulic assistance for the cyclic control of the main rotor while the collective main rotor servo actuator provides hydraulic assistance for the collective control of the main rotor.

The pedal control linkages are routed from the cabin roof back to the tail rotor servo-actuator, located in the intermediate fuselage (between the main fuselage and tail boom), and then to the tail rotor. The tail rotor servo actuator provides hydraulic assistance for the pedal controls.

2.4.2 Fixed Flight Controls

The right-seat cockpit controls were installed. The left-seat cyclic and collective controls were not installed. The left pedals were installed with a pedal lockout kit. The [right] collective control was fractured at its base and the collective control's splined connection was separated, but the collective control exhibited no significant deformation. The [right] cyclic control remained installed with no significant deformation. The emergency flotation system activation handle remained installed on the cyclic control. Continuity of the interconnection between the left and right cyclic and collective bases and the pedals were confirmed. The collective twist grip throttle was found adjacent to, but not precisely at, the "FLY" position.

The access panels to the broom closet were removed. When the belly access panel was removed, an aluminum ring (about 1.25 inches in diameter) and a rubber support were on the access panel. The lower throttle cable support bracket located on the right-side broom closet wall was partially separated, deformed, and without its rubber support. The throttle cable support bracket above the aforementioned lower throttle cable support bracket was intact with its rubber support present between the throttle cable and support bracket. The mixer bellcrank assembly remained installed. The lateral cyclic and longitudinal cyclic control rods remained connected at their rod ends to their respective mixer bellcranks. The collective control rod was present but its aft rod end bearing separated from its outer race housing in overload. The collective control rod's aft rod end outer race housing remaining connected to the collective control rod while its bearing and attachment hardware remained installed on the collective mixer bellcrank.

The broom closet contained four push-pull tubes. The yaw control push-pull tube was separated from its lower rod end threaded connection and the tube was pulled outside the broom closet and partially within the cabin. The remaining three push-pull tubes (for lateral cyclic, longitudinal cyclic, and collective control) were mostly contained within the broom closet. Of the three push-pull tubes, the lateral cyclic tube contained its clevis end and had a remnant piece of the servo actuator support connection attached, the longitudinal cyclic was fractured at the threaded clevis end, and the collective tube was fractured at the upper end of the tube. Corresponding fractures were observed on the servo support. All fractures observed were in overload.

The cockpit cyclic control could not be moved manually. The lateral cyclic control rod end connection to its mixer bellcrank was disconnected by the investigation team, and subsequent manual inputs to the cockpit cyclic control in the lateral axis resulted in a corresponding movement of the lateral cyclic control rod.

The servo support assembly remained partially installed and exhibited multiple fractures (**Figure 12**). The lateral cyclic and collective servo actuators remained installed. The longitudinal cyclic servo actuator was present near its normally installed location but separated due to fracturing of the servo support structure. The pressure lines to the right cyclic and collective servo actuators were fractured at their respective pressure ports, but all remaining hydraulic lines remained installed on the three servo actuators. The left servo actuator appeared to be fully retracted and the right servo actuator appeared to be fully extended. The collective servo actuator appeared to be mid-extension. The input valve for all three actuators exhibited play when manipulated by hand. Red paint transfer was observed on the forward attachment bellcranks of the collective and right cyclic servo actuators. All observed fractures in the servo support assembly area exhibited signatures consistent with overload.

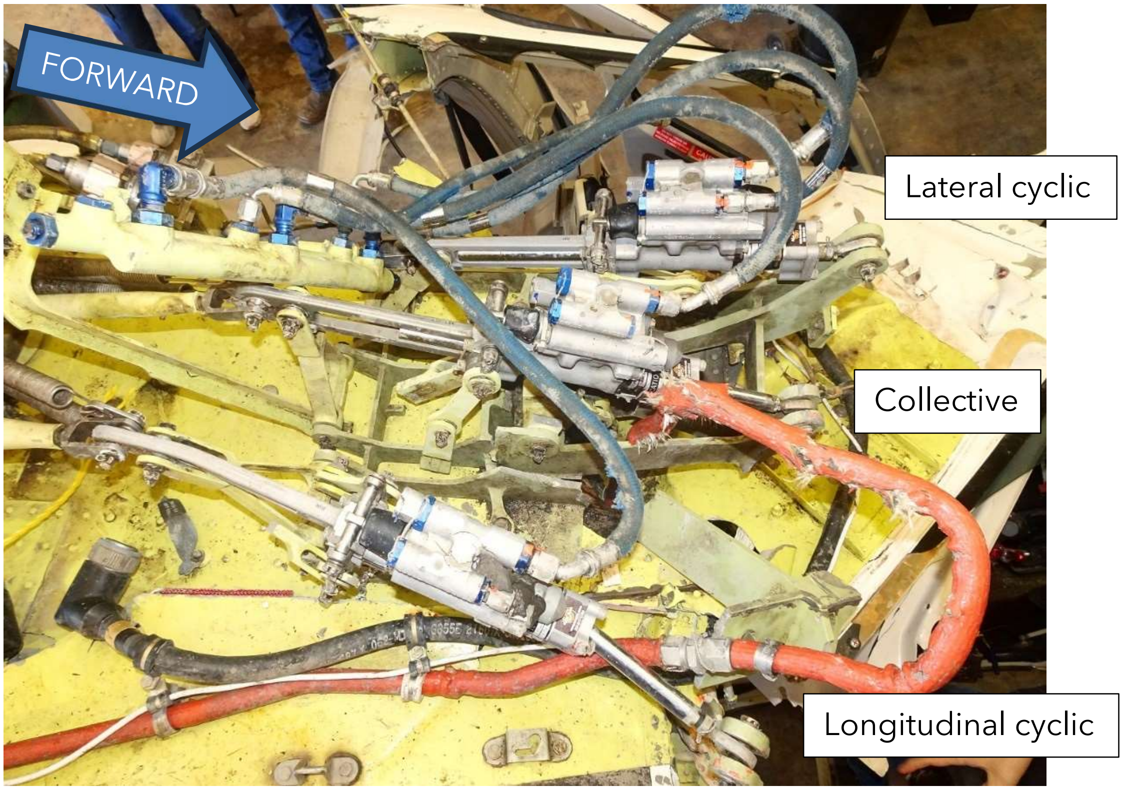

Figure 12. The servo support assembly and the three main rotor servo actuators.

The control linkages between the servo actuators to the bellcrank support (attached to the forward end of the main transmission housing) remained installed. The left portion of the bellcrank support was fractured with signatures of overload, but there was no evidence of disconnection. The left cyclic, right cyclic, collective, and tail rotor servo actuators were removed from the airframe and retained for further investigation.

2.4.3 Main Rotor Rotating Controls

The control rods remained attached to the stationary swashplate and the collective lever. The collective sleeve was separated from the collective lever. Three of the four pitch links were not present but their lower rod ends were present on the rotating swashplate. The fourth pitch link remained connected to the stationary swashplate and was fractured about mid-length of the rod. Both drive links were present but had fractured in overload. The anti-drive link remained installed and was not fractured.

2.4.4 Tail Rotor Controls

The longitudinal tail rotor control tube was continuous from the forward bellcrank (at the top of the broom closet) to the aft bellcrank (at the aft end of the cabin). The control tube between the aft bellcrank and the tail rotor servo actuator was fractured about mid-length. The tail rotor servo actuator remained installed within the intermediate fuselage and appeared to be in the fully retracted position.

The tail rotor push-pull tube connected between the tail rotor servo actuator and the tail rotor gearbox bellcrank was present and installed at both ends but had fractured near the interface between the tail boom and aft fuselage (also where the tail boom had fractured and separated from the aft fuselage). There was a small cylindrical piece of material within the forward portion of the [fractured] tail rotor push-pull tube that was later determined to be excess bonding material (for the rod end). Manual pitch changes to the tail rotor blades resulted in a corresponding movement of the tail rotor push-pull tube within the tail boom.

The tail rotor pitch change links remained installed. All associated hardware was present. One pitch change link exhibited deformation. The outboard ends of the tail rotor pitch change links had evidence of contact damage. The pitch change mechanism remained installed within the tail rotor output shaft.

2.4.5 Hydraulic System

The hydraulic reservoir remained installed. The reservoir was opened and its screen and the reservoir contained mud and a mixture of water and hydraulic fluid-colored liquid. The manifold supply to the main rotor servo actuators was present and there were no disconnections of the hydraulic lines. The hydraulic pump remained attached to the main gearbox housing.

2.5 Engine

2.5.1 Engine Overview

The Rolls-Royce Model 250-C47B is a turboshaft engine that comprises a single-stage centrifugal compressor, a single can-type combustor, and a turbine assembly that incorporates a two-stage gas producer turbine and a two-stage power (free) turbine. The gas producer turbine and the power turbine are not mechanically connected and rotate independently of each other. The gas producer turbine drives the compressor as well as most of the engine-mounted accessories. The power turbine delivers power to the helicopter rotor system via the engine accessory gearbox. The accessory gearbox drive splines deliver power to the main transmission.

The engine incorporates a full authority digital engine control (FADEC) system. The FADEC system is composed of an airframe-mounted engine control unit (ECU) and an engine-mounted hydromechanical unit (HMU). The ECU, manufactured by Triumph Engine Control Systems, monitors engine parameters and performance, as well as the helicopter's power demands, and delivers the necessary control commands to the HMU. As a result, the HMU delivers a metered fuel flow to the engine in accordance with ECU control commands. The ECU continuously monitors the FADEC system for faults and will alert the pilot of specific faults that could significantly impact engine performance. Faults that are considered minor will not be alerted to the pilot during flight, but will be recorded and displayed during engine shutdown. The recorded fault history can be retrieved by maintenance personnel using a computer with the requisite software. Additionally, whenever certain engine parameters are exceeded, the ECU will record the exceedance in a separate incident recorder dataset; this dataset is only accessible by the engine and ECU manufacturers.

2.5.2 Engine Observations

The engine remained installed on the airframe but portions of the engine mounts partially fractured. All engine components were present and the engine remained whole as a single assembly. All pneumatic, oil, and fuel lines to the engine remained installed and exhibited no evidence of looseness. The combined engine filter assembly (CEFA) oil filter bypass indicator had extended (the fuel filter does not have a bypass indicator). Removal and examination of the CEFA revealed water within the filter and housing, but no evidence of debris. The compressor impeller blades, when viewed through the engine inlet, exhibited significant deformation on several blades (**Figure 13**). The turbine module exhibited no damage. The fourth stage turbine blades, when viewed through the exhaust, showed no evidence of damage and were all present. Water was present within the engine. The gearbox housing was partially consumed due to corrosion due to saltwater immersion. Neither gas producer (N1) nor power turbine (N2) drivetrains could be manually rotated. The engine throttle was at about the 90% position.

Figure 13. The compressor impeller blades viewed through the engine inlet. (Image courtesy of Rolls-Royce)

The electronic engine control unit (ECU) was installed on the transmission deck and was removed by investigators. The ECU was drained of saltwater and placed in a container with distilled water for preservation. The ECU was retained for further investigation.

2.6 Appareo Vision 1000

The operator had previously installed an Appareo Vision 1000 cockpit image recorder on the accident helicopter.[3] The Vision 1000 was normally installed on the right side of the broom closet structure such that the recorder was behind the [right-seated] pilot's left shoulder (**Figure 14**). On the accident helicopter wreckage, the structure to which the Vision 1000 mounted was partially fractured. The Vision 1000 mount was present, but the image recorder itself was not present on the mount and was not found in the recovered wreckage (**Figure 15**).

Around January 27-28, 2023, divers searching the area adjacent to the accident offshore oil platform found additional helicopter debris, including the Appareo Vision 1000. Data from the Vision 1000 was recovered by the NTSB Recorders Laboratory in Washington, District of Columbia (DC). The recovered parametric data showed that during takeoff, the helicopter suddenly rolled to the right. Additional details of the recovered data from the Vision 1000 can be found in the Onboard Image, Audio, and Data Recorder Group Chair's Factual Report in the docket for this investigation.

[3] The FAA Form 337 for the installation of the Appareo Vision 1000 onto N595RL was dated August 6, 2020 and stated the installation was similar in design to FAA Supplemental Type Certificate (STC) No. SR02825CH. This STC is for the installation of the Appareo Vision 1000 for Bell 206B, 206L-1, 206L-3, and 206L-4 helicopters. These models are also found in TCDS No. H2SW.

Figure 14. An Appareo Vision 1000 installation on an exemplar Bell 407. The viewpoint of this image is forward-looking-aft.

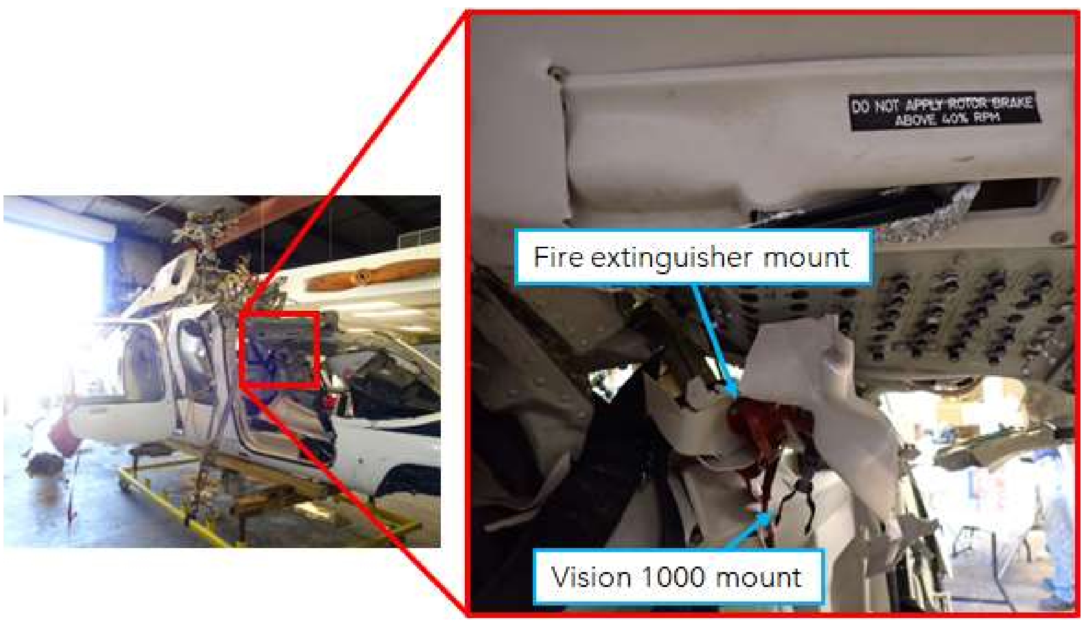

Figure 15. The Vision 1000 mount as seen on the accident helicopter.

3.0 Engine ECU Data Recovery

The ECU for the Rolls-Royce 250-C47B contains nonvolatile memory (NVM) that records engine history and engine faults in the [onboard] maintenance terminal and engine exceedances in the [onboard] incident recorder. The incident recorder captures data from multiple parameters at a 1.2 second interval. (Each interval containing recorded parameters is referred to as a 'line' in this section.) When a fault or an exceedance[4] occurs, the ECU will record the last 10 lines prior to the exceedance and 40 lines after the fault or exceedance.

The accident ECU was transported to the NTSB Recorders Laboratory in Washington, DC, to dry and clean the internal circuit card assemblies (CCA). Subsequently, the ECU CCAs were shipped to Triumph Engine Controls to recover the recorded data. On February 13, 2023, the representatives from the NTSB, Rolls-Royce, RLC, and Triumph Engine Controls convened at Triumph facilities in West Hartford, Connecticut to recover the recorded data from the accident helicopter's ECU. Inspection of the CCA identified white-colored particles and debris on some of its components. In order to mitigate the risk of damage by applying power to the CCA, the U40 chip[5] was removed from the CCA and installed onto a test CCA. The NVM was successfully recovered from the U40 chip, which included engine fault history, engine history, and engine incident recording data. The data recovered from the ECU containing helicopter rotor speed (Nr), gas generator speed (Ng[6]), power turbine speed (Np[7]), collective position (CLP) and measured gas temperature (MGT) can be found in **Appendix A** of this report.

4.0 Lateral Servo Actuator Detailed Examination

4.1 CT Scan of Actuator

From March to April 2023, the lateral servo actuator was scanned using CT to document the internal condition of the actuator prior to disassembly. The CT imaging showed no evidence of cracks, fractures, or metallic debris within the hydraulic passageways. Additional details of the CT imaging can be found in the Computed Tomography Specialist Report in the docket for this investigation.

4.2 Actuator Examination at Woodward

On May 5, 2023, representatives from the NTSB, FAA, RLC, Bell, and Woodward convened at Woodward's facilities in Santa Clarita, CA to examine the lateral actuator from the accident helicopter. The scope of the examination included visual evaluation, bench testing, and a limited disassembly of the actuator.

4.2.1 Bench Testing

Prior to installation on the test bench, the two idler links were removed from the actuator. Aside from light corrosion, the attachment hardware for the idler links

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[4] A fault is an error that may preclude normal operation of the ECU. An exceedance is when a parameter, such as torque or gas generator speed, operates outside of a predefined limit.
[5] The U40 chip was the NVM to which faults and exceedances are written.
[6] Ng is identical to N1.
[7] Np is identical to N2.

did not exhibit anomalous damage. Silt-like debris was present underneath the rubber boot of the input linkage.

The actuator was installed on a test bench.[8] A separate return line was attached to collect fluid within the actuator; the collected fluid was filtered. Minimal pressure, about 400 pounds per square inch (psi), was applied to the actuator to open the bypass springs to measure the force needed to move the input lever, about 5 pounds (lbs) for the extend direction and 3 lbs for retract direction. A second measurement of the force needed to move the input lever resulted in about 3 lbs for the extend direction. The residual fluid within the actuator that came out of its return port had a milky-red appearance. The presence of mud and debris on the input linkage was possibly a factor in the increased breakout force observed for the first measurement in the extend direction (**Figure 16**).

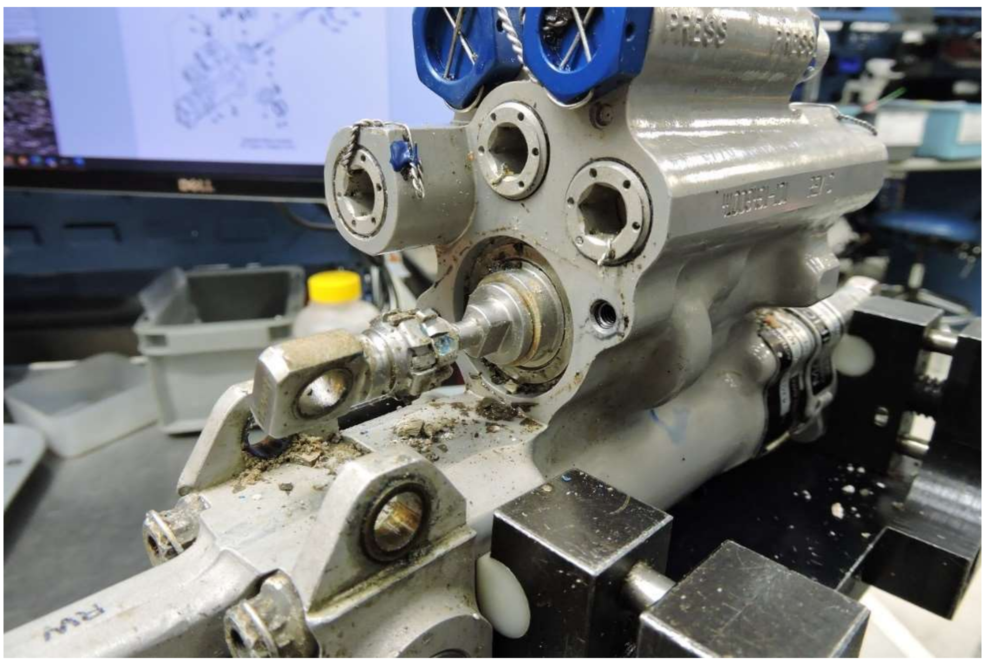

Figure 16. The input lever after removal of its rubber boot.

The actuator was manually actuated for several full-stroke cycles on the test bench at a supply pressure of about 1,000 psi (the same as system pressure when installed on the helicopter) to flush it. The return fluid was collected and subsequently filtered using a 0.8 micron filter to collect debris within the return fluid. At the end of the flushing process, the return fluid had a clear-red appearance.

Using the Woodward acceptance test procedures (ATP) for the servo actuator as guidance, bench testing was conducted on the actuator.[9] The ATP showed no

[8] The test bench utilized MIL-PRF-5606 hydraulic fluid in its supply.
[9] Proof pressure, low pressure leaking, and dynamic leakage tests were not conducted as part of this examination.

anomalous behaviors except for the bypass valve test. The bypass valve's secondary spool showed normal displacement in the retract direction, but did not show displacement in the extend direction; however, bypass functionality was achieved in both directions. The spring cover was removed, revealing the springs for the spool which contained a white-colored material that had the appearance of salt or aluminum oxide (**Figure 17**).[10]

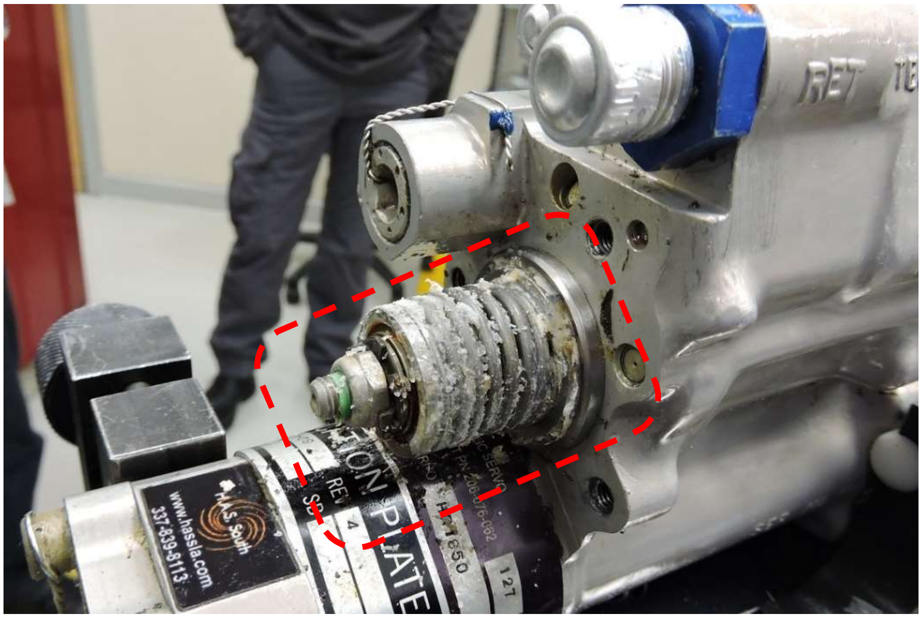

Figure 17. The white-colored material visible after removal of the spring cover.

4.2.2 Disassembly

The actuator was removed from the test bench for limited disassembly. The actuator's main control valve assembly was removed to inspect its condition. The spool and sleeve exhibited no anomalous damage. All O-rings and backup rings were intact. Three specks of debris were present near the center land of the spool (**Figure 18**). The debris was clear (no color) and had the appearance of glass under magnification. The clear debris from the spool, the debris collected from filtration during flushing of the actuator, and a hydraulic fluid sample from the actuator were retained for further analysis. Analysis of the hydraulic fluid sample found it to be consistent with MIL-H-5606 hydraulic fluid in addition to high sodium content, likely from saltwater contamination.[11]

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[10] The spring cover is vented to atmosphere via a slot at its 6 o'clock position.
[11] At the time of this report, analysis of the debris was in progress at the NTSB Materials Laboratory.

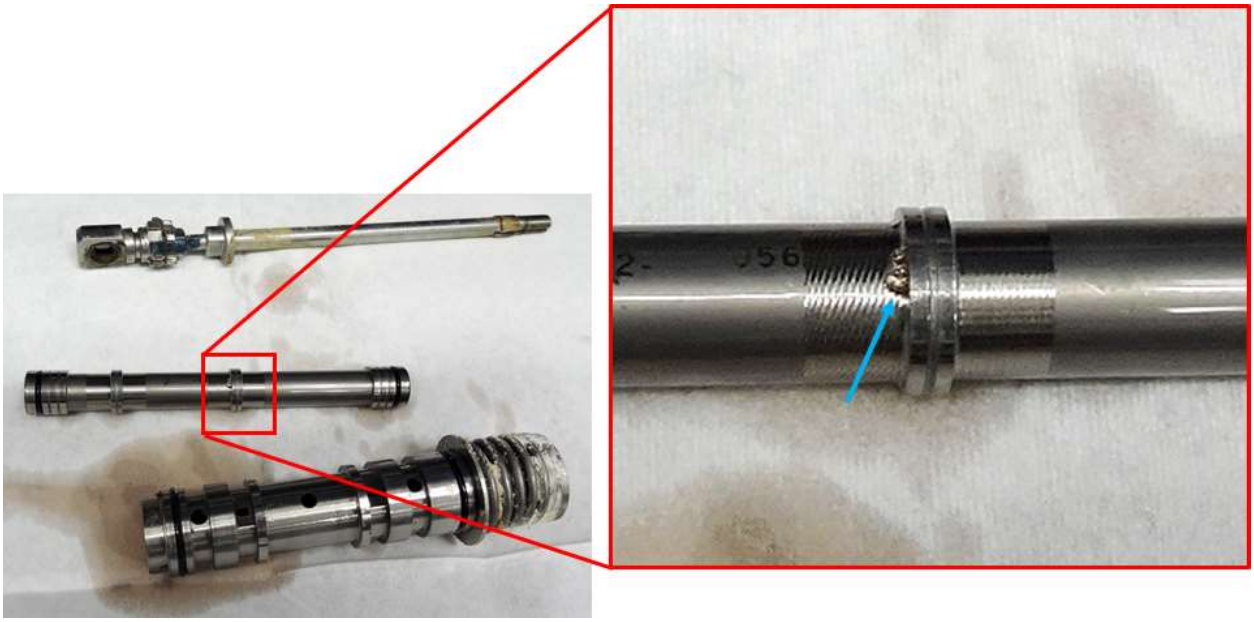

Figure 18. Clear debris (blue arrow) near the center land of the spool from the lateral servo actuator.

The pressure inlet screen was examined and found to be intact. There was no evidence of debris or obstructions within the pressure inlet screen. The pressure inlet screen lock ring was removed but the screen would not come out. During the attempt to remove the screen, the screen mesh separated from its housing; the screen housing remained within the inlet and was not removed. The lateral actuator was not disassembled further.

5.0 Maintenance

The operator maintained the accident helicopter under an approved airworthiness inspection program (AAIP). Additionally, the helicopter was inspected under a progressive inspection program composed of six, 50-hour Events, with all six Event inspections required to be accomplished within 12 months. According to aircraft records, the helicopter had an ATT of 16,722.0 hours at the time of landing at the offshore platform on December 29, 2022 (immediately before the accident flight). According to the operator, the servo actuators are on-condition components but the operator had elected to have them to overhaul every 4,000 hours.

The last Event 1 inspection was performed on August 5, 2022 at an ATT of 16,283.2 hours. The last Event 2 inspection was performed on September 18, 2022 at an ATT of 16,413.9 hours. The last Event 3 inspection was performed on October 27, 2022 at an ATT of 16,549.4 hours. The last Event 4 inspection was performed on December 2, 2022 at an ATT of 16,661.7 hours. The last Event 5 inspection was performed on May 14, 2022 at an ATT of 16,018.1 hours. The last Event 6 inspection was performed on June 18, 2022 at an ATT of 16,150.4 hours. The last 1200-hr/24-month inspection was performed on January 7, 2022 at an ATT of 15,695.6 hours.

The longitudinal cyclic servo actuator, S/N HR1071, was installed on the accident helicopter on August 12, 2021 at an ATT of 15,336.5 hours and a component time since overhaul (CTSO) of 0.0 hours. The collective servo actuator, S/N HR045, was installed on the accident helicopter on December 6, 2020 at an ATT of 14,419.0 hours and a CTSO of 0.0 hours. The lateral cyclic servo actuator, S/N HR1850, was installed on the accident helicopter on August 12, 2021 at an ATT of 15,336.5 hours and a CTSCO of 0.0 hours. The tail rotor servo actuator, S/N HR339, was installed on the accident helicopter on March 19, 2022 at an ATT of 15,857.7 hours and a CTSO of 0.0 hours.

Submitted by:

Chihoon Shin
Aerospace Engineer – Helicopters

APPENDIX A

Table A-1. Excerpt of data from the ECU incident recorder.

Snapshot Record #	IR Data Record #	Time Stamp[12] 2136:*XX:XX.XXX*	Nr %	Ng %	Np %	Torque %	CLP %	MGT °F
	1	53:55.968	100	84	100	28	4	960
	2	53:57.168	100	84	100	28	4	980
	3	53:58.368	100	84	100	28	4	960
	4	53:59.568	101	86	101	36	16	1020
	5	54:00.768	102	84	102	24	26	940
	6	54:01.968	100	86	100	36	32	1000
	7	54:03.168	100	89	100	48	40	1060
	8	54:04.368	100	91	100	52	46	1080
	9	54:05.568	100	94	100	68	52	1160
	10	54:06.768	100	97	100	78	50	1240
1		54:07.200	96	96	86	112	4	1220
2		54:07.224	87	96	80	124	4	1220
3		54:07.536	49	95	39	120	32	1180
	11	54:07.968	30	98	31	130	98	1280
4		54:08.952	106	103	110	70	94	1560
5		54:09.024	113	103	121	44	72	1580
	12	54:09.168	124	97	123	34	50	1440
6		54:09.240	124	94	122	18	-16	1380
	13	54:10.368	115	81	113	6	76	1120

[12] The time stamps are listed in hours:minutes:seconds:milliseconds.

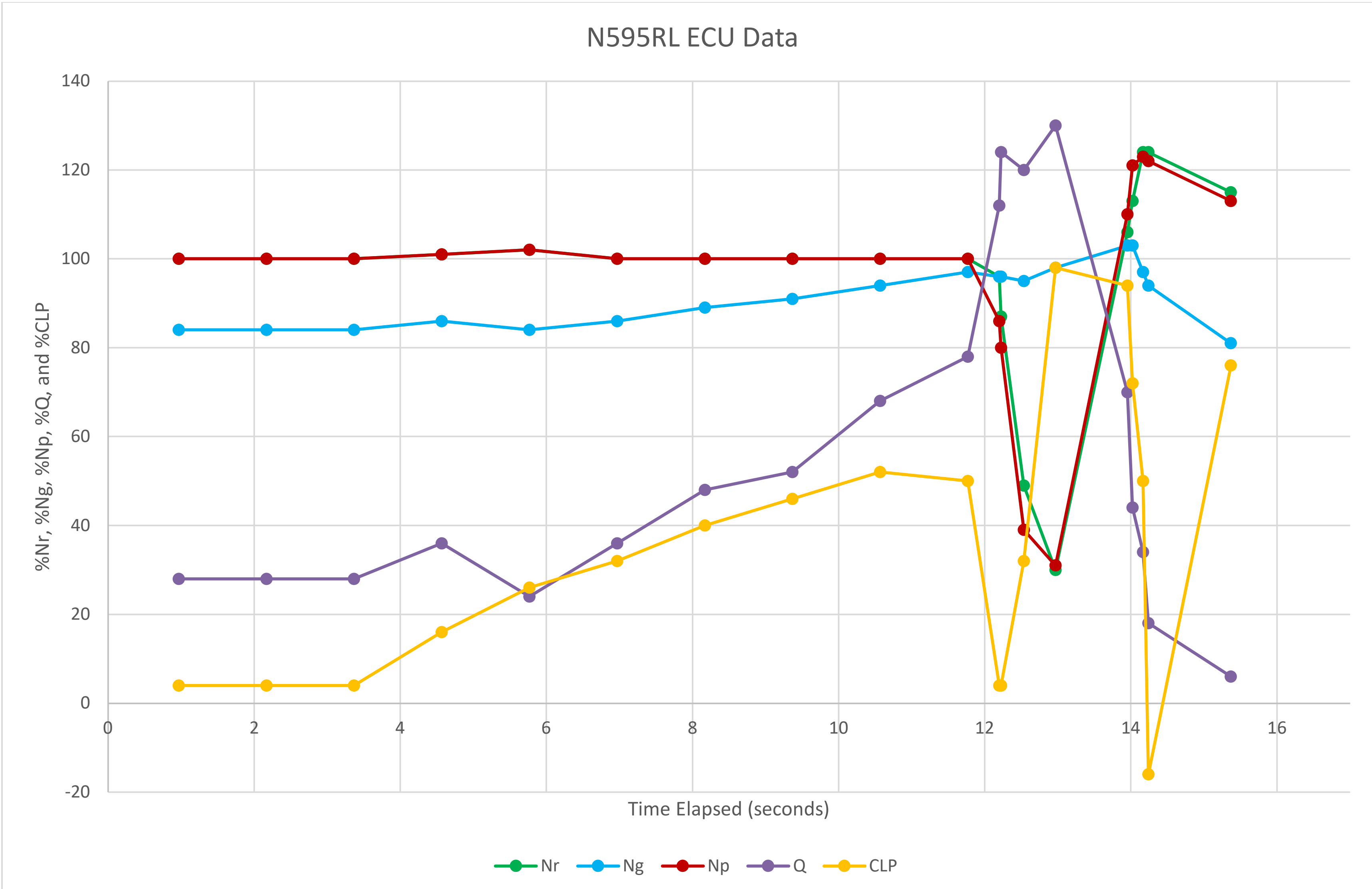

Figure A-1. A graph of select parameters from the recovered ECU data.