



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

**November 5, 2018**

**AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT**

**NTSB No: ERA17FA274**

**A. ACCIDENT**

Operator: Virginia State Police  
Aircraft: Bell 407, Registration N31VA  
Location: Charlottesville, Virginia  
Date: August 12, 2017  
Time: 1649 eastern daylight time

**B. GROUP**

|                 |                                                                                          |
|-----------------|------------------------------------------------------------------------------------------|
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**LIST OF ACRONYMS**

|      |                                      |
|------|--------------------------------------|
| AGB  | accessory gearbox                    |
| ALF  | aft looking forward                  |
| ATT  | aircraft total time                  |
| ECU  | engine control unit                  |
| EDT  | eastern daylight time                |
| FAA  | Federal Aviation Administration      |
| FCC  | flight control computer              |
| FTR  | force trim release                   |
| KIAS | knots indicated airspeed             |
| Nr   | main rotor speed                     |
| NTSB | National Transportation Safety Board |
| OGE  | out-of-ground effect                 |
| PCL  | pitch change link                    |
| PRCS | pedal restrictor control system      |
| PRCU | pedal restrictor control unit        |
| RSTA | rotor station                        |
| S/N  | serial number                        |
| SAS  | stability augmentation system        |
| STC  | supplemental type certificate        |
| TCDS | type certificate data sheet          |
| TRDS | tail rotor drive shaft               |
| VA   | Virginia                             |
| VSP  | Virginia State Police                |

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**C. SUMMARY**

On August 12, 2017, about 1649 eastern daylight time (EDT), a Bell 407 helicopter, N31VA, operated by Virginia State Police (VSP), was destroyed after impacting trees and terrain in Charlottesville, Virginia (VA). The airline transport-rated pilot and private pilot-rated observer were fatally injured. Visual meteorological conditions prevailed and no flight plan was filed for the public use aerial observation flight that departed Charlottesville-Albemarle Airport, Charlottesville, VA about 1554 EDT.

According to the VSP, the mission of the helicopter flight crew was to provide continuous video downlink to the VSP command center of the public demonstrations that were occurring in Charlottesville, Virginia. After being refueled, the helicopter departed CHO about 1554 EDT, with the airline transport pilot flying the helicopter, and observer operating the helicopter's camera.

The helicopter arrived over the city of Charlottesville at 1604 EDT and remained over the city until 1642 EDT when they were re-tasked to provide over-watch for the Governor of Virginia's motorcade. At 1643 EDT, the helicopter crew advised the VSP command center that they were heading directly to the motorcade and were about 30 seconds away. About 1649 EDT, another helicopter advised the VSP command center that the accident helicopter had crashed.

The main wreckage site contained the main fuselage (cockpit and cabin), the forward section of the tail boom, horizontal stabilizer, main rotor, and engine. The main wreckage came to rest at a magnetic heading of 333 degrees. A postcrash fire damaged a significant portion of the main wreckage. The aft section of the tail boom, including the vertical fin and tail rotor, was found within a tree canopy about 100–150 feet south-southwest of the main wreckage site.

**D. DETAILS OF THE INVESTIGATION****1.0 HELICOPTER INFORMATION****1.1 HELICOPTER DESCRIPTION**

The Bell Helicopter 407 is type certificated under Federal Aviation Administration (FAA) type certificate data sheet (TCDS) No. H2SW. The Bell 407 has a four-bladed 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 effect and control helicopter yaw. The helicopter is equipped with one Rolls-Royce (formerly Allison) Model 250-C47B turboshaft engine, mounted behind the main gearbox, 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.

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. Additionally, clock positions are in the ALF frame of reference unless otherwise specified.

**1.2 HELICOPTER HISTORY**

The accident helicopter, serial number (S/N) 53465, was manufactured in 2000. According to helicopter records, the airframe had accumulated an aircraft total time (ATT) of

about 5,997.1 flight hours as of August 9, 2017. The engine installed on the accident helicopter, S/N CAE 847508, had accumulated about 5,606.3 hours as of August 9, 2017. The helicopter was estimated to have weighed about 4,633 pounds at the time of the accident.

## 2.0 WRECKAGE DOCUMENTATION AT THE ACCIDENT SITE AND AFTER RECOVERY

On August 13–14, 2017, members of the Airworthiness Group convened at the accident site to document the wreckage. The wreckage was recovered on August 14–15, 2017 to a facility in Clayton, Delaware. On August 16–17, 2017, members of the Airworthiness Group documented the recovered wreckage.

### 2.1 STRUCTURES

#### 2.1.1 OVERVIEW

The helicopter structure comprises the main fuselage, the tail boom and empennage, and a skid-type landing gear. The main fuselage contains the cockpit, cabin, bladder-type fuel tanks, and baggage compartment. The main fuselage supports the engine, main rotor gearbox, and the main rotor and its controls. The tail boom and empennage contain the tail rotor drive system and the tail rotor and its controls. The left and right horizontal stabilizers are mounted about mid-length of the tail boom. Vertical end plates are mounted to the outboard edge of each horizontal stabilizer. The upper and lower vertical fins are mounted near the aft-end of the tail boom and are co-located with the tail rotor gearbox.

#### 2.1.2 FINDINGS

The main fuselage came to rest at a magnetic heading of 333 degrees ([Figure 1](#)). The cockpit and main rotor system came to rest upright. The engine deck, containing the engine, and the forward section of the tail boom came to rest on its right side. A postcrash fire damaged a significant portion of the main wreckage. The tail boom section containing the horizontal stabilizer was found adjacent to the nose of the helicopter. The left horizontal stabilizer exhibited an angled cut consistent with main rotor blade contact. The outboard forward slat of the left horizontal stabilizer was found separate from the horizontal stabilizer. The upper and lower vertical end plates were separated from the left horizontal stabilizer and found loose in the debris field. The right horizontal stabilizer remained mostly intact, with partial fractures of its upper and lower vertical end plates.

The left skid tube was found adjacent to the cabin. The aft crosstube remained attached to the left skid. The left skid was fractured at the forward crosstube, above the tube saddle. The right skid was fractured at the forward and aft crosstube above the crossmember tube saddle. The right-side high step, spanning the length of the right skid tube, was found loose in the wreckage and was fractured at its forward and aft crosstube attachment points. The aft-left step, normally mounted on the aft crosstube, had fractured at its attachment point and was not recovered. The forward-left step remained mounted on the forward crosstube. The left side of the crosstubes generally had a more flattened appearance compared to the right side of the crosstubes.

The aft section of the tail boom, containing a portion of the tail rotor drive system, tail rotor, and vertical stabilizer, was found within a tree canopy about 100-150 feet south-southwest of the main wreckage site (**Figure 2**). The left side of the tail boom aft section exhibited an impact mark consistent with tail rotor blade contact.



**Figure 1. The main wreckage site containing the main fuselage.**



**Figure 2. The aft section of the tail boom found in a tree canopy.**

## 2.2 MAIN ROTOR SYSTEM

### 2.2.1 SYSTEM OVERVIEW

Power from the engine reduction gearbox is transferred through a sprag clutch<sup>1</sup> to the main rotor gearbox via the input drive shaft. Kaflex couplings on both the forward

<sup>1</sup> A sprag clutch is a type of freewheeling unit.

and aft ends of the input drive shaft allow for minor misalignment. 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 plate. Each main rotor blade is attached to the yoke via a grip and elastomeric thrust bearing, the latter of which also accommodates blade pitch and lead-lag motions. The four main rotor blades are identified by color and the shape<sup>2</sup> 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. A Frahm-type damper is mounted to the top of the main rotor hub for vibration attenuation.

### 2.2.2 MAIN ROTOR FINDINGS

All four main rotor blades were found at the main wreckage site. The main rotor hub remained attached to the main rotor mast and was thermally damaged. The yoke exhibited separation of its composite layers. The yoke connection to the 'orange' main rotor blade was completely fractured. The main rotor fairing remained installed and was thermally damaged. The fairing was removed, revealing a Frahm-type damper and its springs in their normally installed positions.

The whole span of the 'blue' main rotor blade was found at the main wreckage site. The blade generally appeared intact. The outboard section of the blade was fractured but remained attached to the blade. The inboard section of the blade exhibited thermal damage. The blade remained attached to its grip, which remained attached to the hub. The pitch horn was attached to the grip and the pitch change link (PCL) upper rod end remained attached to the pitch horn, but the PCL body was fractured with evidence of thermal damage.

The majority of the spar of the 'orange' main rotor blade was found at the main wreckage site, with most of its afterbody missing. The spar was fractured near the outboard end of the blade. A blade tip end, about 24 inches in length, was found in the debris field and was matched to the 'orange' main rotor blade. The blade was separated from its grip, but was found adjacent to the hub. The hub still contained the grip for the 'orange' blade, but the hub was fractured at this location. The grip exhibited fractures and thermal damage. The pitch horn was attached to the grip and the PCL upper rod end remained attached to the pitch horn, but the PCL body was fractured with evidence of thermal damage.

The inboard section of the 'red' main rotor blade remained attached to its grip. The pitch horn remained attached to the grip, but the pitch horn lug normally connected

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<sup>2</sup> Stickers of a certain shape and color are normally placed on the Bell 407 main rotor blades to identify them. The stickers are typically found on the blade grip and the inboard end of the blade.

to the PCL upper rod end was fractured. The pitch horn lug fracture surface exhibited signatures consistent with overload and thermal damage. Neither the fractured pitch horn lug or the PCL upper rod end were recovered in the wreckage. The blade was fractured about 66 inches outboard of the bolts retaining the blade to the grip, about rotor station (RSTA) 88.<sup>3</sup> A 78-inch section of the spar, missing its afterbody and leading edge abrasion strip, remained attached at the RSTA 88 fracture. The leading edge abrasion strip was found loose in the main wreckage. A 103-inch blade piece, later matched to the red main rotor blade, was found underneath a pile of large tree branches about 20 feet south-southwest of the main wreckage. An impact mark was observed on the leading edge around RSTA 128. The tip end of the blade, about 16 inches in length, and the outboard adjustable weight package were found loose in the debris field.

The 'green' main rotor blade remained attached to its grip. The entire span of the blade was found at the main wreckage site. At the blade tip, the lower skin was peeled and partially separated, but still attached to the blade. The pitch horn remained attached to the grip. The PCL upper rod end remained attached to the pitch horn, but the PCL body was fractured and thermally damaged.

### 2.2.3 MAIN ROTOR DRIVE SYSTEM FINDINGS

The main rotor gearbox remained attached to the airframe. The gearbox housing exhibited soot deposits and was thermally damaged. The Kaflex coupling on the forward end of the input driveshaft<sup>4</sup> was intact. The Kaflex coupling on the aft end of the input driveshaft was fractured; a portion of the Kaflex coupling was found loose in the main wreckage and exhibited thermal damage. Rotation of the input driveshaft resulted in a corresponding rotation of the main rotor head.

## 2.3 TAIL ROTOR SYSTEM

### 2.3.1 SYSTEM OVERVIEW

Power from the engine reduction gearbox sprag clutch 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 (TRDS): 1 steel TRDS<sup>5</sup> is mounted between the sprag clutch and the oil cooler blower followed by 5 aluminum TRDS<sup>6</sup> between the oil cooler blower and the tail rotor gearbox. The 5 aluminum TRDS are supported along the tail boom structure by 4 hanger bearings. In this report, the aluminum TRDS are numbered 1 through 5 from the forward to aft position, respectively. Each aluminum TRDS is connected to each other via flexible coupling disc packs. A flexible coupling disc pack is also found between the No. 5 TRDS and the tail rotor gearbox input flange. 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.

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<sup>3</sup> On the Bell 407, RSTA 0 for the main rotor is located at its axis of rotation. The main rotor blade tip end is located at RSTA 121.0. All RSTA values are in inches.

<sup>4</sup> The input driveshaft is also known as the engine-to-transmission driveshaft.

<sup>5</sup> The steel TRDS is also known as the forward short shaft.

<sup>6</sup> The forward-most aluminum TRDS is also known as the aft short shaft.

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 FINDINGS

Both tail rotor blades remained attached to its yoke (**Figure 3**). The 'white' tail rotor blade exhibited damage on its tip end. The leading edge of the 'white' tail rotor blade exhibited impact damage about 15.5 inches inboard of the tip end; the impact damage was about 3 inches wide. The 'red' tail rotor blade did not exhibit anomalous damage.



**Figure 3. The recovered aft tail boom section and tail rotor.**

### 2.3.3 TAIL ROTOR DRIVE SYSTEM FINDINGS

The steel TRDS was found whole and thermally damaged. The forward flexible coupling was fractured but its attaching hardware was present. The splines at the aft end of the steel TRDS did not exhibit evidence of smearing or missing spline teeth. The oil cooler blower remained attached to the airframe. The forward and aft splines of the oil cooler blower were thermally damaged but did not exhibit evidence of smearing or missing spline teeth. Rotation of the forward splines resulted in a corresponding rotation of the blower and the aft splines.

The No. 1 TRDS was fractured about 1/3 of its length from its aft end, which was in line with an impact mark on the tail boom consistent with main rotor blade contact. The splines on the forward end of the No. 1 TRDS did not exhibit evidence of smearing or fractures. The aft end of the forward portion of the No. 1 TRDS exhibited thermal damage and remained attached to the forward flange of the No. 2 TRDS via aft coupling disc pack. The aft coupling disc pack exhibited a serpentine (wavy) appearance about its circumference as well as opening of the disc pack laminates.

The No. 2 TRDS was fractured about its longitudinal midpoint, which was in line with an impact mark on the tail boom consistent with main rotor blade contact. The shaft tube was separated from its riveted connection to the forward end cap and flange. The forward end cap and flange remained attached to the hanger bearing, which remained attached to the support bracket and a remnant piece of tail boom. The aft coupling disc pack exhibited a serpentine appearance about its circumference.

The majority of the No. 3 TRDS was not recovered and only the forward riveted end cap and flange for the No. 3 TRDS was found; its rivets exhibited signatures consistent with shear.

The No. 4 TRDS was fractured about 1/3 of its length from its aft end. Only the aft portion of the No. 4 TRDS was recovered. The aft coupling disc pack exhibited opening of the disc pack laminates. The fractured attachment flange lobes from the No. 5 TRDS, along with its attaching hardware, remained connected to the aft portion of the No. 4 TRDS. The aft portion of the No. 4 TRDS remained attached remnant tail boom structure.

The No. 5 TRDS remained intact except for the fracture of its forward attachment flange lobes. The No. 5 TRDS remained attached to the tail rotor gearbox input flange via its aft flexible coupling. The aft flexible coupling remained attached to the No. 5 TRDS and was intact.

The tail rotor gearbox remained attached to the tail boom structure and did not exhibit evidence of looseness. Manual rotation of the tail rotor gearbox input flange resulted in a corresponding rotation of the tail rotor. The oil sight gauge revealed presence of oil within the gearbox. The chip detector was removed and revealed no evidence of chips or debris.



**Figure 4. A reconstruction of the recovered tail rotor drive train.**

## 2.4 FLIGHT CONTROL SYSTEM

### 2.4.1 SYSTEM OVERVIEW

The flight control system comprises cyclic, collective, and directional (pedal) controls which are hydraulically assisted. 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 to the cabin roof. The Bell 407 is equipped with an airspeed-actuated pedal restrictor control system (PRCS). The PRCS restricts tail rotor authority at higher airspeeds by automatically adjusting the left pedal [forward] stop. When the helicopter accelerates above 55 knots indicated airspeed (KIAS), the PRCS solenoid energizes, engaging a cam which limits forward travel of the left pedal by 25%. When the helicopter decelerates below 50 KIAS, the PRCS solenoid de-energizes, disengaging the cam, allowing full forward travel of the left pedal. Pulling the emergency release cable for the PRCS allows for manual disengagement of the PRCS pedal stop. The emergency release cable has a copper witness wire to prevent inadvertent disengagement of the PRCS pedal stop.

The accident helicopter was equipped with an optional quick disconnect dual control modification which allowed for copilot (left seat) operations.<sup>7</sup> Additionally, a copilot pedal “lockout” kit was installed.<sup>8</sup> According to the operator, the copilot cyclic and collective sticks were removed for the accident flight.

The accident helicopter was equipped with a Genesys Aerosystems HeliSAS stability augmentation system (SAS) and two-axis autopilot. The HeliSAS system was installed under supplemental type certificate (STC) No. SR02344LA. The HeliSAS system is designed to provide attitude stabilization and force feedback to the cyclic stick using electromechanical servo actuators connected in parallel to the existing cyclic flight controls. The HeliSAS is designed to accept attitude data from the attitude gyro and sensors within the flight control computer (FCC). The FCC commands the servo actuators to apply small corrections in the roll and pitch axes to maintain attitude. HeliSAS functions are controlled using the front panel-mounted control panel, a cyclic-mounted force trim release (FTR) switch, and a cyclic-mounted autopilot disconnect switch. When the helicopter is powered on, the HeliSAS is in standby mode following a successful self-test. The SAS function is activated by pressing the button labeled “SAS” on the control panel.

### 2.4.2 ROTOR FLIGHT CONTROL FINDINGS

All the three main rotor hydraulic actuators were found loose in the main wreckage near their normally installed location. For one of the main rotor hydraulic actuators, of which the installation position could not be determined, the actuator body was consumed by the postcrash fire and only the piston remained.

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<sup>7</sup> The dual control quick disconnect was installed via Bell Helicopter installation instruction No. BHT-407-II-9.

<sup>8</sup> The helicopter was equipped with a pedal lockout kit via STC No. SR00513AT. The copilot pedals are “locked out” when the copilot pedal linkages are physically disconnected from the copilot pedal bell crank. The pedal lockout kit allow for a quick disconnect of the copilot pedals from the flight control system when changing from single to dual pilot configuration.

The rotating swashplate arms connecting to the 'red' and 'orange' main rotor blade PCL lower rod end were fractured consistent with overload and thermally damaged. The rotating swashplate arm connecting to the 'orange' PCL was found loose in the recovered main wreckage debris. Two PCL lower rod ends, with attaching hardware still installed within the rod end, were found loose in the wreckage debris. The swashplate drive lever between the 'green' and 'red' main rotor blades remained connected to both the main rotor shaft and swashplate, but exhibited fractures and thermal damage. The lower link of the swashplate drive lever between the 'blue' and the 'orange' main rotor blades was fractured and separated from the swashplate; this swashplate drive lever was also thermally damaged. The non-rotating swashplate anti-drive upper lever was melted but the lower lever was present.

The tail rotor control tube was continuous from the tail rotor to where the aft section of the tail boom was fractured. The two PCLs for the tail rotor blades remained installed and exhibited slight bending near its rod ends. Witness marks were observed near the upper rod end of both tail rotor PCLs. Contact with the outboard washer weights on the tail rotor blades would produce witness marks on the upper rod end of the tail rotor PCL.

#### 2.4.3 COCKPIT FLIGHT CONTROL FINDINGS

The pilot cyclic and collective sticks were found in the cockpit. The cyclic grip was melted and the stick remained connected to its base near the floor of the cockpit. The left and right cyclic attachment points remained connected to the lateral jack shaft. The lateral push-pull tube was thermally damaged and exhibited a small fracture on a portion of its tube, but the fracture was not through the tube cross-section which allowed for motion to be transmitted between the two cyclic attachment points. The lower rod end of the left cyclic vertical push-pull tube remained connected to the mixing unit, but the push-pull tube was fractured at its lower end and thermally damaged; the remainder of the tube was not observed. The lower portion of the right cyclic vertical push-pull tube was fractured above the rod end threads and the tube exhibited evidence of impact damage. The remainder of the right cyclic vertical push-pull tube was not observed.

The collective stick was fractured near its base in the aft-center of the cockpit. The left and right collective stick attachment points remained connected to the collective jack shaft. The collective jack shaft remained connected to the mixing unit. The lower portion of the collective vertical push-pull tube was fractured at the threaded end for the rod end threads. The remainder of the collective vertical push-pull tube was not observed.

The copilot cyclic and collective sticks were found loose in the rear baggage compartment of the helicopter. According to the operator, the copilot cyclic and collective sticks were normally removed and stored in the rear baggage compartment for single-pilot flights.

The HeliSAS roll and pitch servo actuators, as well as the FCC, remained installed underneath the copilot seat (**Figure 5**). The linkage between the roll servo actuator and the cyclic lateral jackshaft was bent. The linkage between the pitch servo actuator and the cyclic lateral jackshaft bracket was intact but the bracket had separated

from the jackshaft. For information on the examination of the HeliSAS, see section 4.0 of this report.



**Figure 5. The cyclic and collective jackshafts and the HeliSAS actuators.**

The pilot pedal set was found loose in the wreckage. The lateral push-pull tube was fractured. The copilot pedal set remained installed in the cockpit but its lateral push-pull tube was fractured. The left pedals were consistent with being “locked out.” The tail rotor fore-aft push-pull tube, routed under the right collective stick, remained connected at the forward bell crank and was continuous to the bell crank underneath the mixing unit. The bell crank underneath the mixing unit remained attached to the airframe. The vertical push-pull tube was fractured at the lower rod end threads, but the lower rod end remained attached to the bell crank underneath the mixing unit. The vertical push-pull tube exhibited thermal damage. The lateral push-pull tubes, between the left and right pedals, remained attached to the forward bell crank. The PRCS remained installed but its solenoid exhibited impact damage ([Figure 6](#)). The PRCS cam was found in the engaged position ([Figure 7](#)). The PRCS emergency release cable was found in the cockpit and was thermally damaged, but the copper witness wire for the emergency release cable was found unbroken. The plastic pull knob for the emergency release cable was not recovered, and the adjacent cable housing exhibited thermal damage. The PRCS was removed from the helicopter for further examination. For information on the examination of the PRCS, see section 5.0 of this report.



**Figure 6. The pedal bell crank and PRCS solenoid.**



**Figure 7. The PRCS cam (red arrow) found in the engaged position.**

## 2.7 COCKPIT INSTRUMENTS

The majority of the cockpit instruments were damaged by the postcrash fire and were unreadable. Cockpit instruments that were not thermally damaged but were unreadable were: the fuel quantity indicator; the fuel pressure gauge; the battery amperage gauge; the main gearbox oil gauge; and the engine oil and temperatures gauges. The caution and warning panel was found thermally damaged and unreadable. The Hobbs meter was found and read 1291.8 hours.

## 2.8 POWERPLANT

### 2.8.1 POWERPLANT DESCRIPTION

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 front and rear drive splines deliver power to the main rotor and tail rotor drive trains, respectively.

The engine incorporates a full authority digital engine control (FADEC) system. The FADEC system is composed of an airframe-mounted electronic 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.

### 2.8.2 POWERPLANT FINDINGS

The engine was found laying on its right side near its normally installed location on the airframe. The engine mounts had fractured from the airframe. The engine assembly was bent about 30 degrees laterally, at the junction of the turbine and gearbox modules, resulting in a disengagement of the compressor-to-turbine shaft as well as the power turbine shaft. The engine exterior exhibited damage consistent with exposure to the postcrash fire. The first stage compressor blades exhibited evidence of hard body foreign object debris (FOD) damage ([Figure 8](#)).<sup>9</sup> Manual rotation of the compressor resulted in a corresponding rotation of the N1 gear train. The splined coupling to the compressor turbine shaft also rotated. Due to the decoupling of the compressor turbine shaft, rotation of the gas generator turbine from the compressor was not possible. Rotation of the transmission input shaft (the engine power output shaft) resulted in a corresponding rotation of the N2 gear train. The power turbine had decoupled from the

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<sup>9</sup> 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.

gearbox, resulting in no corresponding rotation of the N2 gear train when the power turbine was rotated.



**Figure 8. The engine inlet vanes and the first stage compressor blades.**

The fuel spray nozzle was removed and the gas generator turbine was examined via borescope. The gas generator turbine nozzle vanes and blades appeared intact and did not exhibit anomalous damage. Small silver-colored globules were observed on the turbine nozzle vanes, turbine blades, and in the combustion chamber. The fourth stage turbine exhibited no evidence of anomalous damage.

The oil and pneumatic lines were checked by hand for evidence of looseness, none of which was found. The combined engine oil and fuel filter was recovered separated from its normally installed location. The filter elements were present but the aluminum filter bowls were consumed by the postcrash fire. Residual oil was observed within the engine accessory gearbox had a tar-like appearance but oil from the top of the gearbox appeared bright brown in color. The gearbox chip detectors were removed and the upper and lower chip detectors were missing their magnets.

The ECU was found loose in the wreckage with one of its electrical connectors attached. The ECU exhibited thermal damage. The ECU was retained for further examination (see section 3.0 of this report).

### 3.0 ECU DOWNLOAD

On September 12, 2017, members of the Airworthiness Group convened at Triumph Engine Control Systems in West Hartford, Connecticut to attempt to retrieve any information that may have been stored within the ECU. The ECU was opened and exhibited thermal damage to its internal circuit boards. The U40 chip, the non-volatile memory of interest, remained installed on its circuit card assembly (CCA) but exhibited evidence of thermal exposure. The U40 chip was removed from the CCA and examined under a microscope. The U40 chip appeared in good condition except for a slight surface

crack to the substrate of the component package and thermal damage near the short edges of the chip. The U40 chip was installed on an exemplar CCA and ECU and the stored data was downloaded successfully. The U40 had recorded engine fault history, engine history, and engine incident recording data. Attachment 1 contains the data downloaded from the U40 chip. Three faults were recorded in the fault history: fault 1 was engine overtorque recorded at timestamp 6572:01:04.824<sup>10</sup>; fault 2 was main rotor speed (Nr) overspeed recorded at timestamp 6572:01:08.616; and fault 3 was main rotor droop recorded at timestamp 6572:01:09.912. The incident history records several parameters at a 1.2 second interval. (Each interval containing recorded parameters is referred to as a 'line' in this section.) When a fault occurs, the incident history records and retains parameters from the last 10 intervals prior to the fault and the next 40 intervals after the fault. A total of 150 lines were downloaded from the incident history. For Line 1, the recorded timestamp was 6572:00:53.640. Line 18 contained the end of useable data and had timestamp 6572:01:14.040. Line 19 contained some parameters but no timestamp. Lines 20 through 150 were empty of data. For additional information on the use of the downloaded ECU data, see the Aircraft Performance Study in the docket for this investigation.

#### 4.0 HELISAS SYSTEM EXAMINATION

According to the Bell 407 maintenance manual No. BHT-407-MM-8, chapter 67, paragraph 67-57 provides instructions on adjusting the cyclic longitudinal balance spring when operating with single or dual cyclic controls.<sup>11</sup> With a single-pilot cyclic installation, the nominal longitudinal balance spring length should be 6.61 inches (168 mm); with a dual-pilot cyclic installation, the nominal longitudinal balance spring length should be 7.08 inches (180 mm). A note within the cyclic longitudinal balance spring adjustment instructions state that adjustment beyond these nominal dimensions, or replacement of the spring, may be required to balance the cyclic. According to the operator, the cyclic longitudinal balance spring<sup>12</sup> was typically adjusted to about 5.71 inches (145 mm) in length, with a single cyclic installation, in order to best meet the HeliSAS cyclic centering force criteria of 2.5–3.5 pounds<sup>13</sup>, when SAS mode was engaged, and the cyclic minimum friction requirement of 0.5–1.5 pounds. Additionally, according to the operator the cyclic longitudinal balance spring was not adjusted when the copilot cyclic stick was installed, or removed, as needed.

##### 4.1 CYCLIC LONGITUDINAL BALANCE SPRING TESTING AT VSP FACILITIES

On November 30, 2017, members of the Airworthiness group convened at a VSP hangar at Chesterfield County Airport in Chesterfield, VA to examine an exemplar Bell 407 operated by VSP. The cyclic longitudinal balance spring was measured to be about 5.71 inches. A handheld force measurement gauge was used to measure the cyclic centering force with the SAS mode engaged. With a dual cyclic installation, the centering force was 1.7–3.2 pounds. With the cyclic sticks centered, disengaging SAS mode resulted in no movement of the cyclic sticks. The copilot cyclic was removed but the cyclic longitudinal balance spring was not adjusted. With a single cyclic installation, the centering force was 2.5–3.3 pounds. With the cyclic stick centered, disengaging SAS mode resulted in the cyclic stick traveling aft.

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<sup>10</sup> The timestamps are listed in HH:MM:SS.mmm, interpreted as hours:minutes:seconds.milliseconds.

<sup>11</sup> A note within the cyclic longitudinal balance spring adjustment instructions, "the balance spring is used to keep to a minimum the cyclic control stick mass imbalance forces in the longitudinal control system."

<sup>12</sup> The cyclic longitudinal balance spring is used to keep to a minimum the cyclic control stick mass imbalance forces in the longitudinal control system.

<sup>13</sup> The Genesys Aerosystems HeliSAS instructions for continued airworthiness, manual No. AA-09109, contains requirements for the cyclic centering force.

The cyclic longitudinal balance spring was adjusted to 6.61 inches and the copilot cyclic was removed; the resultant centering force was between 2.3–4.5 pounds. With the cyclic stick centered, disengaging SAS mode resulted in the cyclic stick traveling aft. The aft travel of the cyclic stick appeared quicker than the previous test with the cyclic longitudinal balance spring set at 5.71 inches.

The cyclic longitudinal balance spring was adjusted to 7.08 inches and the copilot cyclic was reinstalled; the resultant centering force was between 1.1–3.7 pounds. With the cyclic sticks centered, disengaging SAS mode resulted in no movement of the cyclic sticks.

#### 4.2 CYCLIC LONGITUDINAL BALANCE SPRING TESTING AT BELL PINEY FLATS

On February 21, 2018, members of the Airworthiness group convened at Bell Helicopter facilities in Piney Flats, Tennessee to examine an exemplar Bell 407 using a test procedure similar to that of the examination of an exemplar Bell 407 at VSP facilities. The exemplar Bell 407 at Piney Flats was equipped with the Genesys Aerosystems HeliSAS system and the longitudinal balance spring was set at about 5.04 inches (137 mm) with only the pilot cyclic installed. The cyclic stick was balanced in this configuration. The cyclic centering force was about 1.5–2.5 pounds.

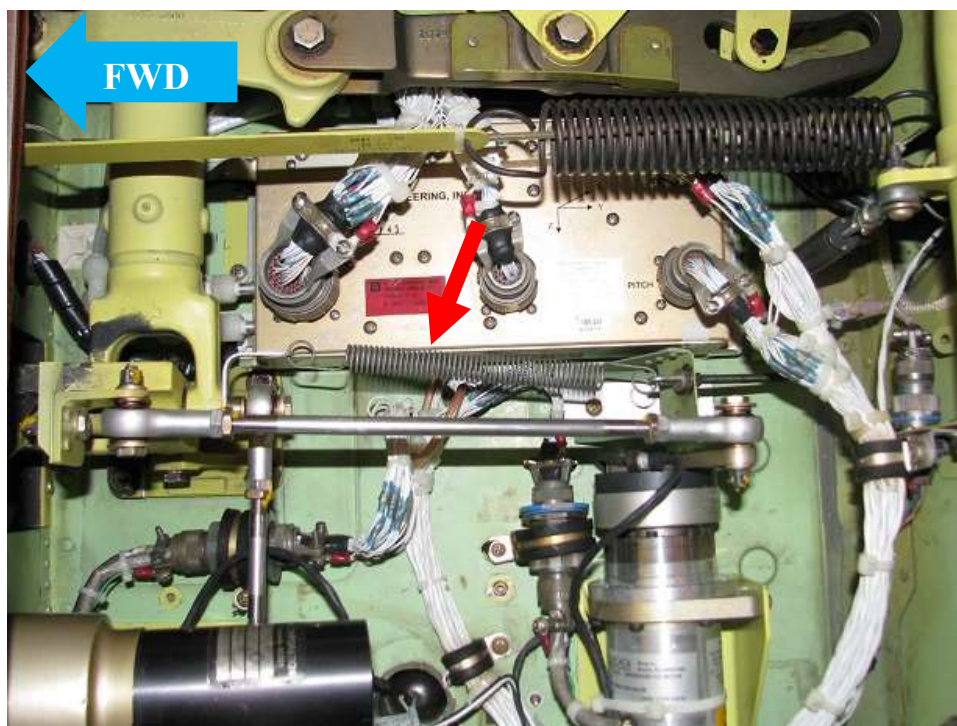
With the cyclic longitudinal balance spring adjusted to 7.08 inches and a dual-cyclic installation, the cyclic centering force was about 1.5–3.0 pounds. With the cyclic stick centered, disengaging SAS mode resulted in no movement of the cyclic stick. SAS mode was re-engaged and the cyclic was trimmed slightly forward of center, illuminating the “cyclic centering” light on the caution and warning panel. SAS mode was subsequently disengaged using the control panel button; no movement of the cyclic was observed.

With the cyclic longitudinal balance spring adjusted to 6.61 inches and a single-cyclic installation, the cyclic centering force was about 1.5–3.0 pounds. With the cyclic stick centered, disengaging SAS mode, using the control panel button, resulted in no movement of the cyclic stick. SAS mode was re-engaged and the cyclic was trimmed slightly forward of center, illuminating the “cyclic centering” light on the caution and warning panel. SAS mode was subsequently disengaged using the control panel button, resulting in the cyclic stick traveling aft.

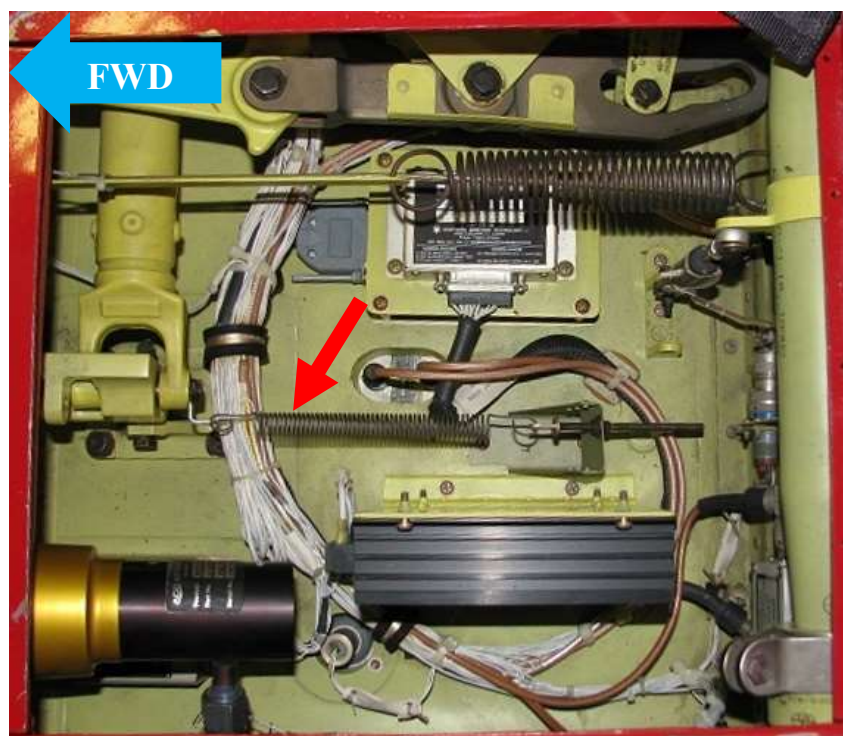
The cyclic stick was placed in the full aft position and the lateral distance between the cyclic longitudinal balance spring brackets and adjacent structural web was measured. The distance between the forward bracket and web was about 7.38 inches. The distance between the aft bracket and web was about 7.75 inches. With the cyclic longitudinal balance spring measured to be about 5.44 inches, the angular offset between the spring and the airframe longitudinal axis was calculated to be about 4 degrees ([Figure 9](#)).<sup>14</sup> On an exemplar Bell 407 without the HeliSAS installation ([Figure 10](#)), measurements between the cyclic longitudinal balance spring brackets and adjacent structural web were taken. The measured distances for the forward and aft brackets to the web were nearly identical, indicating there was no angular offset between the spring and the airframe longitudinal axis.

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<sup>14</sup> The standard installation for the cyclic longitudinal balance spring has the forward bracket pointing to the left. According to the installation instructions for the HeliSAS system, the longitudinal balance spring forward bracket is rotated such that the bracket points to the right in order to accommodate the adjustable rod end for the roll servo actuator link.



**Figure 9. The HeliSAS installation and the adjusted cyclic longitudinal balance spring (red arrow).**



**Figure 10. An exemplar cyclic longitudinal balance spring installation without the HeliSAS.**

#### 4.3 HELISAS COMPONENT EXAMINATION

On April 26, 2018, members of the Airworthiness group convened at Bell Helicopter facilities in Hurst, Texas to examine the HeliSAS pitch and roll servo actuators and the FCC

recovered from N31VA. The FCC exterior exhibited impact damage in multiple locations. The FCC cover was opened, revealing the circuit board. The circuit board exhibited impact damage, but the majority of the chips appeared to be in good condition. Application of power to the circuit board resulted in no illumination of its indicator lights. A Genesys laptop was hooked up to the circuit board, but a download of the onboard data was unsuccessful.

The motor cover of the roll servo actuator exhibited impact damage and light corrosion, but the motor exhibited no significant damage. The clutch could be rotated by hand but its movement was restricted by the clutch cover. The clutch cover exhibited impact damage. The arm, connected to the adjustable rod end, was slightly bent toward the motor. The motoring gearbox and wiring to the motor were intact.

The motoring gearbox for the pitch servo actuator was unable to be moved, but the clutch was free to rotate. The arm, connected to the adjustable rod end, did not exhibit damage. The motor cover exhibited impact damage and light corrosion. Gouges were observed on the exterior of the motor. The wire bundle to the motor was damaged and several wires in the bundle were severed.

## 5.0 PRCs EXAMINATION

On April 26, 2018, members of the Airworthiness group convened at Bell Helicopter facilities in Hurst, Texas to examine the PRCs components, including the pedal bell crank, pedal restrictor control unit (PRCU), solenoid, emergency release cable, pedal stop pin, and PRCs cam.

The pedal bell crank contained integral left and right pedal stops. A light coat of dirt covered the area of the left pedal stop. Cleaning the dirt from the left pedal stop revealed an area of wear consistent with contact with the pin. The contact area exhibited wear through the bell crank primer coating. The contact area furthest from the edge of the left stop had a triangular shape, and the contact area closest to edge of the left stop contained narrow strip of more severe wear within the area of worn primer. The right pedal stop had a light coat of dirt. Once the dirt was removed, a witness mark consistent with contact with the pin was observed. The witness mark did not appear to have worn through the bell crank primer coating.

A pitot-static test bench was used to functionally test the PRCU. When power was initially applied to the PRCU, it did not command the solenoid to activate. The PRCs engagement and solenoid activation lights illuminated and then extinguished within the requisite 5 seconds after power was applied. When the bit test switch was turned on, the caution light illuminated and stayed on. When the pitot-static pressure was increased to simulate an increase in airspeed, the PRCs engagement and solenoid activation lights illuminated at about 53.2 KIAS. When the pitot-static pressure was decreased, the PRCs engagement and solenoid activation lights extinguished at about 48.0 KIAS.

The solenoid coil spring, which returns the PRCs cam to the disengaged position, remained attached at both ends. The coil spring was continuous but deformed. The pin was inserted into the solenoid drive shaft to attempt to manually rotate the solenoid, but rotation was unsuccessful due to pushed material, due to impact damage on the solenoid housing, impinging on the magnetic rotor. The coil spring was removed and the pin was reinserted to attempt manual rotation of the magnetic rotor; a very slight rotation was observed but the magnetic rotor remained restricted. The pushed material was removed, allowing for full rotation of the magnetic rotor. The rotation of the magnet was smooth. Power was applied to the solenoid and the solenoid functioned normally, with the magnetic rotor rotating to the

engaged position when power was applied. Because the coil spring was removed, the magnetic rotor did not return to its starting position when power was removed from the solenoid. The solenoid drive shaft was bent slightly to the left. The drive shaft's flat surface exhibited a shiny wear mark on the forward-right side of its vertical surface. The corresponding drive shaft hole, on the pin, did not exhibit anomalous wear.

The tab on the emergency release cable housing, normally adjacent to the pin, was deformed consistent with impact forces. The deformation exhibited a leftward directionality (in the ALF frame of reference). Visual examination of the pin did not reveal evidence of fractures. The pin contained two bosses on its upper surface. The right/forward boss exhibited damage due to contact with the tab; the left/aft boss did not exhibit any damage. Visual examination of the PRCS cam did not reveal evidence of fractures. Anomalous wear was not observed on the cam contact surfaces. The index pin, normally residing between the pin and the cam, did not exhibit anomalous damage. The release cable was index-marked and subsequently pulled by hand, resulting in movement of the tab. The tab rotated about 90 degrees, confirming functionality of the emergency release cable. Resistance was felt when pulling the release cable.

## 6.0 TAIL ROTOR AUTHORITY STUDY

To determine the effect on tail rotor authority if the PRCS were to remain engaged below 50 KIAS, the Airworthiness group chair requested Bell Helicopter perform calculations to determine the left pedal control margin available in different airspeed conditions with the PRCS cam engaged. The calculations used a gross weight of 4,633 pounds, an ambient temperature of 86 degrees Fahrenheit (30 degrees Celsius), and a pressure altitude of 2,200 feet.

Bell Helicopter determined the critical condition would be an out-of-ground effect (OGE) hover, and that the left pedal margin would increase with increasing airspeeds due to efficiencies gained with the main rotor the power it requires decreases (thus resulting in a decrease of the needed main rotor anti-torque). Therefore, the left pedal margin calculations were performed only for the critical condition. The estimated tail rotor blade collective pitch angle needed to maintain heading in an OGE hover was between 10–11 degrees. If the PRCS were to remain engaged during an OGE hover, a tail rotor blade collective pitch angle of 18.5 degrees could still be achieved with the left pedal at full forward travel (contacting the PRCS cam). Without the PRCS engaged, a tail rotor blade collective pitch angle of about 26.5 degrees could be achieved with the left pedal at full forward position.

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