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

June 2, 2020

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: ERA19FA210

A. ACCIDENT

Operator:	Challenger Management LLC
Aircraft:	Leonardo AW139, Registration N32CC
Location:	Big Grand Cay, Bahamas
Date:	July 4, 2019
Time:	0153 eastern daylight time

B. GROUP

Group Chairman:	Chihoon Shin National Transportation Safety Board Washington, District of Columbia
Member:	Patrick Lusch Federal Aviation Administration Washington, District of Columbia
Member:	Bill Gill Honeywell Aerospace Olathe, Kansas
Member:	Allie Engel Honeywell Aerospace Phoenix, Arizona
Member:	Mikael Amura Agenzia Nazionale per la Sicurezza del Volo Rome, Italy
Member:	Giorgio Dossena Leonardo Helicopter Division Cascina Costa, Italy
Member:	Marc Hamilton Transportation Safety Board of Canada Ottawa, Canada

Member: Merryn Spielman
Pratt and Whitney Canada
Montreal, Canada

LIST OF ACRONYMS

AAID	Air Accident Investigation Department [of the Bahamas]
ACTT	aircraft total time
AD	airworthiness directive
AEO	all engines operating
AFCS	automatic flight control system
AGL	above ground level
AIOP	actuator input/output with processor
ALT	altitude hold
ALTA	altitude acquire
AMM	aircraft maintenance manual
AP	autopilot
APP	lateral/vertical approach
ASB	alert service bulletin
CAMP	continuous airworthiness maintenance program
CAS	crew alerting system
CMC	central maintenance computer
DCU	data collection unit
ECL	engine control lever
EEC	electronic engine control
EEPROM	electronically-erasable programmable read-only memory
EFIS	electronic flight instrument system
EGPWS	enhanced ground proximity warning system
FAA	Federal Aviation Administration
FD	flight director
FMM	fuel management module
fpm	feet per minute

FTR	force trim release
HOV	hover/velocity hold
IAS	indicated airspeed
KIAS	knots indicated airspeed
LHD	Leonardo Helicopter Division
MAU	modular avionics unit
MFD	multifunction display
mm	millimeters
Nf	power turbine speed
NIC/PROC	network interface controller and processor
Nr	rotor speed
NTSB	National Transportation Safety Board
OEI	one engine inoperative
PCL	pitch change link
PCM	power control module
PFD	primary flight display
PI	power index
P/N	part number
psi	pounds per square inch
PWC	Pratt & Whitney Canada
RFM	rotorcraft flight manual
SAS	stability augmentation system
SB	service bulletin
S/N	serial number
STA	frame station
TAWS	terrain avoidance warning system
TSB	Transportation Safety Board [of Canada]
UTC	coordinated universal time
Vne	never exceed speed
VS	vertical speed

C. SUMMARY

On July 4, 2019, about 0153 eastern daylight time, a Leonardo AW139, N32CC, owned and operated by Challenger Management LLC, impacted the Atlantic Ocean near Big Grand Cay, Abaco, Bahamas. The commercial pilot, airline transport rated co-pilot, and five passengers were fatally injured. The helicopter was substantially damaged. The helicopter was being operated under the provisions of title 14 *Code of Federal Regulations* Part 91 as a personal flight. Dark night visual meteorological conditions prevailed at the time and an instrument flight rules flight plan was filed for a flight from Walker's Cay Airport in Walker's Cay, Bahamas to Fort Lauderdale/Hollywood International Airport in Fort Lauderdale, Florida. The flight originated about 0152 from a concrete pad located at Big Grand Cay, Abaco, Bahamas.

The accident investigation was initially under the jurisdiction of the Air Accident Investigation Department (AAID) of the Bahamas. On July 6, 2019, in accordance with Annex 13 to the Convention on International Civil Aviation, the AAID requested delegation of the accident investigation to the NTSB, which the NTSB accepted on July 8, 2019.

On July 6, 2019, the helicopter was recovered from the Atlantic Ocean and subsequently transported to Florida Air Recovery in Jacksonville, Florida. From July 8 to 12, 2019, members of the Airworthiness Group[1] examined the recovered wreckage at Florida Air Recovery. From August 26 to 29, 2019, the engines were examined at PWC under oversight from the Transportation Safety Board (TSB) of Canada.[2] From August 27 to 28, 2019, members of the Airworthiness Group[3] convened at Honeywell facilities in Phoenix, Arizona to examine components from the automatic flight controls system (AFCS).

D. DETAILS OF THE INVESTIGATION

1.0 HELICOPTER INFORMATION

1.1 HELICOPTER DESCRIPTION

The Leonardo[4] AW139 is type certificated under Federal Aviation Administration (FAA) type certificate data sheet (TCDS) No. R00002RD. The AW139 has a five-bladed fully articulated main rotor system that provides helicopter lift and thrust, and a four-bladed fully articulated tail rotor system that provides thrust for directional control. The cockpit flight controls are hydraulically-assisted via a dual hydraulic system. Additionally, the AW139 is equipped with the Honeywell Primus Epic integrated avionics system which includes an AFCS. The AFCS incorporates a four-axis autopilot (AP) that can control lateral cyclic, longitudinal cyclic, collective, and pedals. The helicopter is equipped with two Pratt & Whitney Canada (PWC) PT6C-67C turboshaft engines, mounted behind the main gearbox. The helicopter has a retractable wheeled landing gear in a tricycle configuration.

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. All locations and

[1] During this activity, members from NTSB, FAA, LHD, and PWC were present.
[2] During this activity, members from PWC, AAID of the Bahamas, and TSB of Canada were present.
[3] During this activity, members from NTSB, LHD, and Honeywell were present.
[4] In 2000, Agusta S.p.A. and Westland Helicopters merged to form AgustaWestland. AgustaWestland was renamed to Finmeccanica Helicopters in January 2016, which was subsequently renamed to Leonardo Helicopter Division (LHD) in January 2017.

directions will be viewed from aft-looking-forward unless otherwise specified. Additionally, clock positions are in the aft-looking-forward frame of reference unless otherwise specified.

1.3 HELICOPTER HISTORY

The accident helicopter, serial number (S/N) 31112, was manufactured in July 2007. Engine S/N KB0224 was installed in the No. 1 (left) position and S/N KB0239 was installed in the No. 2 (right) position. At the time of the accident, the airframe had accumulated an aircraft total time (ACTT) of about 2,158.1 hours, the No.1 engine had accumulated a total time of about 2,158.1 hours, and the No. 2 engine had accumulated a total time of about 2,158.1 hours.

2.0 POST-RECOVERY WRECKAGE EXAMINATION

2.1 STRUCTURES

2.1.1 OVERVIEW

The helicopter structure consists of the nose, center fuselage, aft fuselage, and tail section. The center fuselage consists of the cockpit and cabin. The nose landing gear resides under the cockpit. The cockpit contains doors on both left and right sides that are hinged at their forward end. The cabin contains two sliding doors, one on each side. The aft fuselage consists of the baggage compartment and supports the sponsons for the main landing gear. Two interconnected bladder-type fuel tanks reside within the aft fuselage. The center and aft fuselage support the main transmission and engine installations. The tail section consists of the tail boom, horizontal stabilizer and vertical fin, and also contains the tail rotor drive system and tail rotor control system.

2.1.2 NOSE AND CENTER FUSELAGE OBSERVATIONS

The cockpit remained attached to the center fuselage. The left windshield was not present and the right windshield was partially separated and fractured from impact. The left side cockpit skin panel and roof structure was missing (**Figure 1**). The floor was crushed and buckled along an approximate 45-degree line between the outboard lower longeron at frame station (STA) 3120[5] and the left nose landing gear support longeron. Some of the buckled skin panels appeared to be hydroformed. Some remaining structure in the area of the forward cabin floor was bent up, aft, and inboard. A part of the door track in this area was also bent up and aft. The co-pilot floor was pushed up by this buckling and the left side of the center console was crushed or buckled inboard. The cockpit skin main beam at butt line (BL) 0[6] was severed 25 inches from the STA 1500 rivet line. The bulkhead frame at STA 1500 was severed approximately 6 inches to the left of BL 0. Buckling and a fracture of the bulkhead frame was observed on the upper right side about 10 inches forward of the right pitot-static tube. The nose cowling was missing but some internal structure and components remained. The left side of the nose landing gear wheel bay had a bend 18 inches aft of the forward edge that turned the forward edge to the left. The right side cockpit upper window frame is separated at the outboard side and the lower door frame

[5] All STA values are in millimeters (mm). STA 0 starts about 560mm, or 22.04 inches, in front of the nose of the helicopter.
[6] All BL values are in mm. BL 0 is the axisymmetric longitudinal centerline of the airframe.

was separated at the forward side. The resulting large section of structure was twisted in a counter-clockwise direction.

Figure 1. The left side of the cockpit.

The upper right side of the cockpit roof was separated along the STA 3120 rivet line and bent downward (**Figure 2**). The right cockpit and cabin steps remained attached to the fuselage. The right main landing gear segments were relatively perpendicular to the fuselage. The majority of the right side skin paneling did not exhibit significant damage. The right side skin panel between the ceiling and the upper deck was bent inwards between STAs 3120 to 3900 and a shaded impact mark was present. A crack was observed on the lower door frame at STA 3900. The distance from the upper door frame to the lower door frame at STA 5700 and at STA 3900 measured 52.5 inches. Some skin panel buckling was observed at STA 6700 below the floor line.

Figure 2. The right side of the cockpit and center fuselage.

The fuselage belly skin panels generally did not exhibit deformation. Antennae remained attached to the belly of the main fuselage. The left engine inlet barrier was inspected. Both filters were present but were damaged and partially separated from the frame. No significant debris was identified in the filters. Both inlet scoops were crushed but the deformed screens appeared clean. The generator cooling screen was clean. The filter bypass door was missing. The right engine inlet barrier was inspected. Both filters were present but were damaged. The outboard filter was partially separated from the frame. No significant debris was identified in the filters. Both inlet scoop screens appeared clean. The generator cooling screen was clean. The filter bypass door was present and could easily be opened.

The nose landing gear remained attached to the airframe. The nose landing gear tires were deflated. The gear was in the down and locked position. The left and right main landing gear sponsons were cut from the main fuselage to facilitate transportation of the recovered wreckage. The right sponson aft fairing was found with the position/strobe light attached. The left position/strobe light was not located. The left and right main landing gear actuators were found extended consistent with being down and locked. The left tire pressure was above 160 pounds per square inch (psi). When a second gauge was used with a higher pressure capability, the pressure reading was 150 psi. The left brake stackup measured 0.875 inches, the wear pin extension measured 0.188 inches, and all brake pistons were flush to the pressure plate. The right tire pressure was 0 psi. The right brake stackup measured 0.875 inches, the wear pin extension was 0.188 inches, and all brake pistons were flush to the pressure plate. Both wheels rotated freely.

2.1.3 AFT FUSELAGE OBSERVATIONS

The aft fuselage remained attached to the main fuselage but the tail boom was separated (**Figure 3**). The aft fuselage's aft section exhibited deformation to the right on the upper deck and left upper longeron, but the floor structure appeared to be relatively straight. A section of the right side skin panel was missing from STA 8700 to STA 8150. A portion of the roof panel was missing from the aft tail boom fracture, following an approximate 45-degree angle, to right side skin fracture at STA 8150. The remaining roof panel was deformed and had a long dent or buckle that started approximately 15 inches forward of STA 8700 and was perpendicular to the missing roof panel fracture. The upper left and lower right tail boom attachment points were present on the aft fuselage; within these attachment points were remnant pieces of the attachment bolt. The lower left, middle left, upper right, and middle right tail boom attachment points were fractured and separated from the aft fuselage. Portions of the aft fuselage containing the upper-right, middle-right, lower-left, and middle-left attachment points remained connected to the tail boom. The lower right attachment bolt was bent to the right and had a large lip on the left side.

The left cargo interior wall panel was bulged inboard. It was removed to gain access to the fuel valves. The fuel system 1, fuel system 2, and crossfeed valves were in the open position. The exposed fuel tank bladder was inspected with no anomalies noted. The majority of the right cargo interior wall panel covering the right fuel tank was separated from the structure in one large piece and was deformed outboard.

Figure 3. A view of the aft fuselage of the helicopter.

2.1.4 TAIL SECTION OBSERVATIONS

The tail boom was separated from the aft fuselage and was recovered in multiple pieces. Three large pieces containing the portion of the forward end of the tail boom, which attaches to the aft fuselage, were recovered. The three large pieces consisted of: 1) the lower surface and a portion of the left tail boom wall (wiring was present in the interior surfaces); 2) the lower portion of the right tail boom wall (no wiring was present); and 3) the upper portion of the right tail boom wall and the upper surface (no wiring was present). The aft portion of the three large pieces exhibited deformation to the right. A four foot section of the left tail boom wall immediately aft of piece 1 described above was recovered. It contained a long, narrow dent, and deformation consistent with a blade strike approximately 3.5 inches aft of STA 8700. A 1-foot section of the tail drive system cowling was found with black marks on its exterior. The tail boom strake was found separated from the tail boom. The strake outer surfaces had a black-colored contact mark.

The aft end of the tail boom, where it attaches to the vertical fin, was fractured and separated from the tail boom. The aft end of the tail boom was deformed to the right and exhibited an impact mark with yellow-colored paint transfer about 24 inches forward of the front vertical fin attachment point. The vertical fin remained attached to the aft end of the tail boom (**Figure 4**). The aft fairing of the vertical fin was separated. The forward fairing remained installed but was fractured near its base. A 30-inch section of the tail gearbox fairing with the strobe light and aft white position light attached was recovered. Most of the left and right sides of the horizontal stabilizers were recovered. The horizontal stabilizer was in multiple pieces.

Figure 4. The vertical fin attached to the aft end of the tail boom.

2.1.5 SEATS AND RESTRAINTS

The pilot seat was separated from its floor-mounting structure but remained connected via cable (routing to the vertical seat structure). The floor-mounting structure remained attached to the cockpit floor. A seat back cushion was present but the bottom cushion was missing. The copilot seat was partially attached to the cockpit floor. The right side of the seat was fractured and separated from its floor-mounting structure. Left side of the seat remained attached to its floor mounting structure. The copilot seat back and bottom cushions were present. A cushioned seat with armests was found partially inverted in the cockpit. A second cushioned seat with armrests was found in a bag containing recovered pieces of wreckage.

The cabin contained three rows of seats (**Figure 5**). In the first row, a single rear-facing seat remained attached to the floor and was located behind the cockpit center console. The copilot seat was resting on the left side of the seat back of the rear-facing seat. The seat bottom and back cushions and the headrest were present on rear-facing seat frame. In the second row, a single forward-facing seat remained attached to the floor in the center of the cabin and was located [laterally] in line with the rear-facing seat. The bottom and back cushions were not present on the second row seat, but a headrest was present. In the third row, three forward-facing seats remained attached to the floor. Of the four available installation positions, the three seats present were: outboard-right, inboard-right, and inboard-left. A seat was not present on the outboard-left seat installation position. All three seats remained attached to the floor. The outboard-right seat was missing a bottom cushion, but the back cushion and headrest were present. The inboard-right seat was missing both its bottom and back cushions, but its headrest was present. The inboard-left seat was missing its bottom cushion, but the back cushion and headrest were present. The inboard-right seat frame was partially collapsed downward on its right side.

Figure 5. A view of the cabin showing the passenger seats that remained installed.

2.2 MAIN ROTOR SYSTEM

2.2.1 SYSTEM OVERVIEW

Power from the output shaft from each engine is transferred to their respective input shaft module. Each input shaft module contains a sprag clutch[7] which connects to a cross shaft that transmits drive to a collector gear in the main gearbox. Additionally, each input shaft module powers a hydraulic pump and an alternator. The main gearbox contains a single-stage planetary gear train, driven by a sun gear on the collector gear, that drives the main rotor shaft (also known as the main rotor mast). The main rotor shaft transmits power to the main rotor hub via a splined connection. The collector gear also drives the tail takeoff pinion, two oil pumps, a hydraulic pump, and an environmental control system compressor. The tail takeoff pinion drives the oil cooler fan and is connected to a flange which provides tail rotor drive. A rotor brake is installed at the tail takeoff flange. The main gearbox is attached to the airframe via an anti-torque beam, at the bottom of the main gearbox, and four rods on the main gearbox upper housing.

The main rotor head is composed of a titanium hub, five composite tension link assemblies, and five dampers. The scissor attachment flange, to which the rotating scissors attach, is installed on the bottom of the main rotor head along with a sliding ring to which the droop stop plate contacts. Each tension link assembly is attached to the hub via an elastomeric thrust bearing that allows for pitch, flap/droop, and lead/lag motions. A flap limiter support is connected to the hub at each tension link assembly. A droop stop bracket and plate are attached to the underside of the tension link. A pitch control lever and main rotor blade are connected to the outboard end of the tension link via two blade bolts. Each damper, which moderates their respective blade's lead/lag motion, is connected to the hub

[7] A sprag clutch is a type of freewheeling unit. During autorotation, the clutches disengage the engines from the main gearbox.

on one end and the pitch control lever on its other end. Each pitch change link (PCL) is connected to its respective pitch control lever. A spherical "beanie" fairing is installed on the top of the main rotor head.

The five main rotor blades are identified by color, presented in the order of advancing rotation (when seated in the pilot seat and observing the blades pass from right to left): 'red', 'black', white', 'yellow', and 'blue'. The main rotor rotates counterclockwise when viewed from above. The main rotor blades are composite in construction, with a fiberglass spar, graphite upper and lower skins which enclose a honeycomb core, and tip cap that is bonded to the parent blade. The leading edge of each main rotor blade has an erosion shield constructed of stainless steel, while each blade tip cap has one that is constructed of nickel.

The swashplate assembly is installed below the main rotor head and is composed of the stationary swashplate, rotating swashplate, and both stationary and rotating scissor assemblies. Main rotor blade pitch control is achieved via PCLs, connected between each blade's pitch control lever and the rotating swashplate. Three main rotor hydraulic actuators transmit pitch, roll, and collective commands to the stationary swashplate.

2.2.2 MAIN ROTOR OBSERVATIONS

All five main rotor blades remained attached to their respective pitch control levers, but all five were separated from the main rotor hub. For all five blades, the outboard ends of the upper and lower tension links remained attached to the pitch control lever and the inboard end of the main rotor blades (**Figure 6**). All five tension links were fractured inboard of the droop stop attachment bolts. Except for the 'white' blade, the remaining tension link C-channel were recovered separated from their respective blade assemblies. The 'white' blade was partially separated but remained attached to the upper tension link. All five tension links had the inboard housing for the elastomeric thrust bearing still attached. The droop stop remained attached to the 'yellow', 'red', and 'black' tension links. The 'blue' blade droop stop was fractured and separated from its tension link. The 'white' blade droop stop was not recovered.

The 'red' blade exhibited separation of portions of its afterbody about 68 inches to 134 inches, and then from 169 inches to its outboard end. The leading edge abrasion strip exhibited deformation at 67 inches, 88 inches, and 211 inches, and a crack at 79 inches, all measured from the blade retention bolts. The tip cap was separated and not recovered. The four holes at the tip cap attachment remained filled.

The 'black' blade exhibited separation of portions of its afterbody from 129 inches to its outboard end. The leading edge abrasion strip exhibited cracks about 74 and 140 inches from the blade retention bolts. The leading edge abrasion strip was separated from 200 inches to its outboard end. The tip cap was separated but was recovered. A standoff bolt, consistent with those found on the tail boom interior for the hydraulic lines, was found lodged inside the afterbody on the underside of the 'black' bolt, about 181-182 inches from the blade retention bolts and about 14.5 inches aft of the leading edge.

The 'white' blade exhibited separation of its afterbody from about 68 inches to 135 inches, and then from 172 inches to its outboard end (measured from the blade retention bolts). The leading edge strip was deformed about 76-77 inches from the blade retention bolts. The tip cap was separated and not recovered. The four holes at the tip cap attachment remained filled. The underside of the 'white' blade exhibited three distinct areas of chord-wise scoring about 167 to 183 inches from the blade retention bolts. The inboard area of scoring had spacing about 2.5 inches apart. The middle-area of scoring had spacing about 2 inches apart. The outboard-area of scoring had spacing about 2.25 inches apart.

Figure 6. An overview of the five main rotor blades. From left-to-right, the blades are the following colors: 'black', 'red', 'white', 'blue', 'yellow'.

The 'yellow' blade exhibited separation of portions of its afterbody from 75 to 224 inches from the blade retention bolts. The leading edge abrasion strip was cracked about 123 inches from the blade retention bolts.

The 'blue' blade exhibited separation of portions of its afterbody from 58 inches to its outboard end (measured from the blade retention bolts). The leading edge and spar exhibited a fracture about 76 inches from the blade retention bolts. The tip cap was separated and not recovered. The four holes at the tip cap attachment were empty.

The main rotor hub remained attached to the main rotor shaft. The hub exhibited nicks and gouges but no evidence of fractures. All five dampers remained attached to the hub and all five dampers' piston were fractured near their spherical bearing end. All fractures exhibited signatures consistent with overload. The 'red' blade flap limiter and the outboard end of its elastomeric thrust bearing remained attached to the hub. The 'black' blade flap limiter was not present but the outboard end of its elastomeric thrust bearing remained attached to the hub. The 'blue' blade flap limiter was not present but the outboard end of its elastomeric thrust bearing remained attached to the hub. The 'yellow' blade flap limiter was present and the elastomeric thrust bearing, absent its inboard end housing, remained attached to the hub. The 'white' blade flap limiter and elastomeric bearing outboard end were not present. The center hub of the beanie remained attached to the main rotor hub but the beanie fairing was separated. A portion of the beanie fairing was recovered.

2.2.3 MAIN ROTOR CONTROL SYSTEM OBSERVATIONS

The swashplate assembly remained installed on the main rotor shaft centering plate (**Figure 7**). The upper boot for the rotating swashplate was attached at its lower end but separated at its upper end. The lower boot for the stationary swashplate was attached but partially torn. The 'blue' pitch change link remained attached to the rotating swashplate and was continuous up to the nut securing the upper rod end to the link body. The 'blue' pitch change link upper rod end was fractured from the link; the upper rod end remained attached to the 'blue' pitch change lever. The 'blue' pitch change link exhibited deformation inboard and in the direction opposite of advancing rotation. The remaining four pitch change links were fractured at their lower rod ends; the lower rod ends remained attached to the rotating swashplate. The remaining four pitch change links remained attached to their respective pitch change levers. The 'yellow' pitch change link did not exhibit deformation between the link body and the upper rod end. The 'white' pitch change link body did not exhibit deformation but the upper rod end thread was bent outboard. The 'red' pitch change link body exhibited slight deformation but its upper rod end connection did not appear bent. The 'black' pitch change link body did not exhibit deformation but its upper rod end threaded connection was bent inboard. All observed fractures exhibited signatures consistent with overload.

Figure 7. The main gearbox, swashplate assembly, and main rotor hub.

Both rotating scissors remained attached to the rotating swashplate and the scissor drive plate. For both rotating scissors, the bolt attaching the lower scissor link to the rotating swashplate was bent in the direction of advancing rotation. The scissor drive plate was separated from the hub; remnant bolt material was seen on the hub. The attaching hardware between the scissor drive plate and the hub was not present. The droop stop floating ring was resting on the scissor drive plate.

The three main rotor hydraulic actuators remained connected to the stationary swashplate and their respective actuator support (on the main gearbox housing). The right and forward hydraulic actuators did not exhibit anomalous damage. The left hydraulic actuator’s feedback link was displaced from its upper bearings. The strap had separated from the inner surface of the left hydraulic actuator’s forward feedback link. For all three main rotor hydraulic actuators, their respective input linkages remained connected to their input levers.

2.2.4 MAIN ROTOR DRIVE SYSTEM OBSERVATIONS

The main gearbox remained attached to the main fuselage via its four support struts and the anti-rotation plate. All four support struts did not exhibit fractures and no visible deformation was seen. Scuff marks through the paint were observed on portions of the forward-left and aft-left support struts. No fractures were observed on the main gearbox housing.

A fluid level was not visible in the main gearbox oil sight gauge. Removal of the three main gearbox chip detectors revealed no evidence of ferrous debris. The main rotor shaft was continuous up to the main rotor head. Manual rotation of the main rotor head was attempted but could not be rotated.

The rotor brake disc remained installed on the tail rotor takeoff flange. The rotor brake remained in the stowed position and was connected to the rotor brake actuation rod. The forward end of the No. 1 tail rotor drive shaft remained attached to the tail rotor takeoff flange via flexible coupling. The oil cooler fan and heat exchanger remained installed on the aft side of the main gearbox housing.

The No. 1 input shaft module was attached to the main gearbox housing. A piece of fairing was pressed onto the outboard face of the No. 1 input shaft module. The gimbal remained installed between the No. 1 input shaft module and its support tube (mounted to the forward end of the engine gearbox), but the engine gearbox was fractured near its forward mounting surface. The No. 1 engine was displaced upward and inward relative to the No. 1 gimbal support tube. The input drive shaft, normally between the No. 1 engine and the diaphragm coupling of the No. 1 input shaft module, was separated from the engine gearbox but the shaft tube was continuous to the diaphragm coupling. The diaphragm coupling was fractured at its forward hub but the thin walls remained intact.

The No. 2 input shaft module was attached to the main gearbox housing. No anomalous damage was observed on the No. 2 input shaft gearbox. The gimbal remained installed between the No. 2 input shaft module and its support tube (mounted to the forward end of the engine gearbox). The input drive shaft, between the No. 2 engine and the diaphragm coupling on the No. 2 input shaft module, was continuous. There was no evidence of fractures on the No. 2 diaphragm coupling.

After engine removal the No. 1 and No. 2 input shaft module freewheel units were turned. They both turned counter-clockwise with expected force and could not be turned in the clockwise direction.

2.3 TAIL ROTOR SYSTEM

2.3.1 SYSTEM OVERVIEW

Power from the main gearbox tail takeoff pinion is transmitted to the tail rotor gearbox via three tail rotor drive shafts (TRDS) and an intermediate gearbox. Flexible couplings are used at the forward and aft connection points between the TRDS and the main, intermediate, and tail gearbox flanges. The stainless steel No. 1 TRDS is located between the two engines and has a damper assembly supporting the shaft. The aluminum No. 2 TRDS is located in the tail section and has both a damper assembly and an anti-flail assembly supporting the shaft. A hanger bearing support assembly is located near the connection between the Nos. 1 and 2 TRDS. The aluminum No. 3 TRDS is located between the intermediate and tail gearboxes. The intermediate gearbox changes the angle of tail rotor drive from down the tail section to up the vertical fin. The tail gearbox changes the angle of tail rotor drive to the outboard-right direction for a tractor (“puller”) tail rotor configuration. The intermediate and tail gearboxes are mounted directly to the airframe.

The tail rotor head is composed of a titanium hub that retains the four tail rotor blades via elastomeric thrust bearings. The elastomeric thrust bearing allows for blade pitch, flap/droop, and lead/lag motions. Four elastomeric dampers are affixed between the hub and each blade. The tail rotor head is driven by the tail rotor output shaft from the tail gearbox.

The tail rotor blades are composite in construction. The leading edge of each tail rotor blade has an erosion shield constructed of nickel. The four tail rotor blades are identified by color, presented in the order of advancing rotation (when viewed from the right side of the helicopter and observing the blades rotating counterclockwise): ‘red’, ‘white’, ‘blue’, and ‘yellow’.

A PCL is connected between each tail rotor blade and a pitch change spider. Movement of the pitch change spider via the tail rotor hydraulic actuator pitch control rod results in tail rotor blade collective pitch changes. Two rotating scissors attach the pitch change spider to the tail rotor hub.

2.3.2 TAIL ROTOR OBSERVATIONS

The ‘red’ tail rotor blade was fractured about 9 inches from its root end. The rest of the blade was not found. The root end of the blade remained attached to the hub via the elastomeric thrust bearing. The lightning strip remained installed on the root end and was bent outboard. The ‘red’ damper bracket was fractured on the blade side and the damper was separated from the ‘red’ tail rotor blade. The damper’s hub-side rod end was bent at the threads. The ‘red’ PCL attachment was fractured on the blade side and the link was bent near the inboard rod end bearing, but the ‘red’ PCL remained connected to its attachment points on both ends. Delamination was present at the base of the ‘red’ PCL lugs.

The ‘white’ tail rotor blade was fractured about 9.25 inches from its root end and the remaining blade airfoil was found separated from the tail rotor. The root end of the blade remained attached to the hub via the elastomeric thrust bearing. The span of the blade exhibited slight deformation in the direction of the blade tip bending inboard. Adjacent to the root end fracture, about 7 inches of the trailing edge of the blade was split open. Delamination of the inboard side of the was present from the root end fracture to about 23 inches outboard. About 22 inches outboard from the root end of the erosion shield, four small dents were observed on outboard side of the erosion shield. About 17 inches from the root end of the erosion shield, chordwise scratches were observed on the inboard side of the erosion shield. The lightning strip remained installed on the root end and was bent outboard. The ‘white’ damper bracket was fractured on the blade side and the damper was separated from the ‘white’ tail rotor blade. The damper’s hub-side rod end exhibited a slight bend of its threads. Both ends of the ‘white’ PCL remained connected but the inboard side rod end was bent.

The ‘yellow’ tail rotor blade was fractured about 9 inches from its root end and the remaining blade airfoil was found separated from the tail rotor. The root end of the blade remained attached to the hub via the elastomeric thrust bearing. The outboard side of the erosion shield exhibited chordwise scratches about 16 inches from the erosion shield root end as well as a small dent about 31.5 inches from the erosion shield root end. The inboard side of the blade exhibited delamination from the root end fracture to about 9 inches outboard. Additionally, delamination was observed on the inboard side of the blade near its tip end. Adjacent to the root end fracture, about 11 inches of the trailing edge of the blade was split open. The outboard side of the blade tip is missing some yellow-colored paint at its trailing edge. The lightning strip remained installed on the root end and was bent outboard. The ‘yellow’ damper bracket was fractured on the blade-side and separated from the ‘yellow’ tail rotor blade. The damper’s hub-side rod end was slightly bent at the threads. Delamination was present at the base of the ‘yellow’ PCL lugs. The ‘yellow’ PCL remained connected on both ends.

The ‘blue’ tail rotor blade was fractured about 13 inches from its root end and the remaining blade airfoil was found separated from the tail rotor. The root end of the blade remained attached to the hub via the elastomeric thrust bearing. The span of the blade exhibited deformation in the direction of the blade tip bending inboard. The blade exhibited delamination impact damage along multiple locations of the blade span. The outboard side of the erosion shield exhibited scratches starting from about 10 inches from the erosion shield root end to about 24 inches from the erosion shield root end. The inboard side of the erosion shield exhibited scratches about 16 inches from the erosion shield root end. Additionally, a thin, blue-colored contact mark was observed on the inboard side of the erosion shield; it started about 10 inches from the erosion shield root end and extended about 10 inches outboard. The blade afterbody exhibited fractures and was bent inboard from the root end fracture to about 13 inches outboard. Chordwise scratches were observed on the inboard side of the erosion shield about 16 inches from the erosion shield root end. Delamination was present on the inboard side from the root end fracture to about 14 inches outboard from that fracture. The lightning strip remained installed on the root end and was bent outboard. The 'blue' damper bracket was fractured on the blade-side and the damper was separated from the blue rotor blade. The 'blue' damper housing exhibited a puncture. The damper's hub-side rod end was bent at the threads. Delamination was present at the base of the 'blue' PCL lugs. The 'blue' PCL remained connected on both ends but its inboard rod end was bent.

2.3.3 TAIL ROTOR CONTROL SYSTEM OBSERVATIONS

The two rotating scissors remained attached to the hub and the spider. The distance between the spider slider and hub measured 4.12 inches. The PCLs were disconnected and the end cap was removed by the group in order to move the spider along its sliding axis; it could be moved freely with no evidence of restriction. The rotating scissors were disconnected by the group in order to manually rotate the spider, which rotated with no evidence of binding and no evidence of excessive play in its bearings. There was no evidence of leakage from the output shaft seal.

The tail rotor hydraulic actuator remained installed on the tail rotor gearbox (**Figure 8**). The input control rod and the piston remained attached to the control lever. The input lever remained connected between the control lever and the hydraulic actuator. The piston extension, measured in line with the piston from the control lever attachment bolt to the actuator housing, was approximately 2.5 inches. The pressure transducers and hydraulic fittings remained installed.

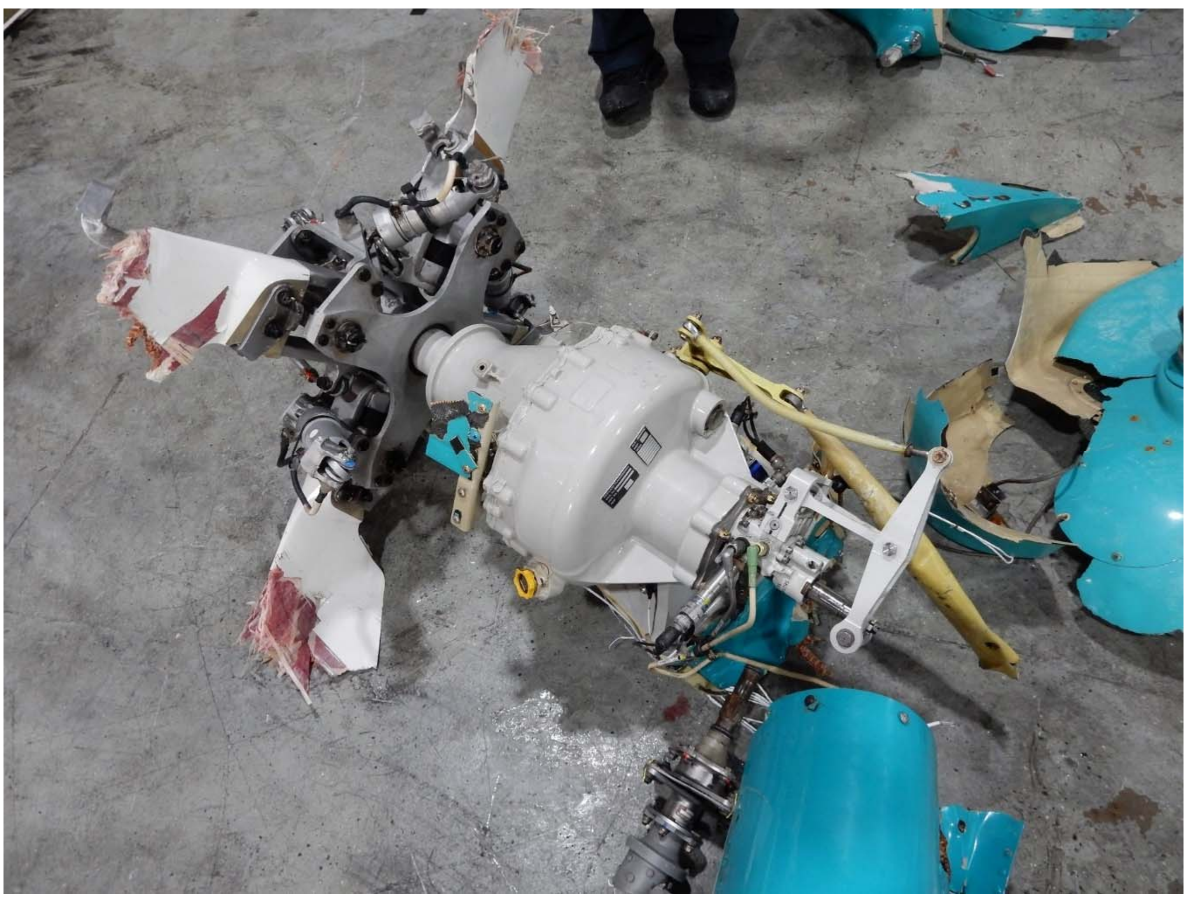

Figure 8. The tail rotor gearbox and its hydraulic actuator.

2.3.4 TAIL ROTOR DRIVE SYSTEM OBSERVATIONS

The No. 1 TRDS remained attached to the tail rotor takeoff flange. The No. 1 TRDS was fractured about 30 inches aft of its forward attachment flange. The aft portion of the No. 1 TRDS, about 33 inches in length, was also found but the center section was not recovered. The damper support for the No. 1 TRDS was attached to the fuselage, but the damper was not present. The airframe attachment bracket for the bearing support assembly between the Nos. 1 and 2 TRDS remained installed at the base but was fractured in two parts, one remaining attached to the bracket and one remaining attached to the drive shaft. The hanger bearing and flexible coupling section was recovered. The hanger bearing rotated freely but was detached from the housing.

The No. 2 TRDS section, about 99 inches in length, was fractured from its forward attachment flange and twisted in multiple locations. The forward 10 inches of the No. 2 TRDS remained attached to the flexible coupling. The No. 2 TRDS had a second fracture about 36 inches from its aft attachment flange. The aft attachment flange for the No. 2 TRDS remained connected to its flexible coupling. Fracture surfaces were matched between the middle (99 inch) segment and aft (36 inch) segment of the No. 2 TRDS. The intermediate gearbox input drive splines and flange remained attached to the aft attachment flange and flexible coupling for the No. 2 TRDS. The sleeves for the anti-flail support and No. 2 damper remained pressed against the No. 2 TRDS tube. A portion of the anti-flail support remained attached to the aft end of the tail boom. The damper for the No. 2 TRDS was not recovered.

The intermediate gearbox remained installed on the aft end of the tail boom. Continuity of the drive was established through the intermediate gearbox by inserting the splined shaft to the intermediate gearbox output and seeing rotation of its input. The intermediate gearbox chip detector was not present.

The No. 3 TRDS lower attachment flange remained connected to the intermediate gearbox output drive splines via flexible coupling. The No. 3 TRDS was fractured about 34 inches from its lower/forward attachment flange. The tail gearbox input with all splines present and a small section of attached drive shaft was recovered. The fracture surface of the drive shaft attached to the gearbox input did not match the fracture surface on the forward No. 3 TRDS section.

The tail rotor gearbox drive splines appeared to be intact. The temperature sensor was attached. The low oil level sensor was attached but the connector’s back shell was broken. The chip detector was inspected and was clean but water was found in the chamber. The drive shaft was inserted and the tail rotor rotated freely.

2.4 FLIGHT CONTROL SYSTEM

2.4.1 OVERVIEW

The cockpit flight control system is composed of cyclic, collective, and directional (pedal) controls at each crew seat. Additionally, the engine control levers (ECL) and rotor brake handle are centrally located at the forward portion of the cockpit ceiling-mounted control panel. Pitch, roll, and yaw trim actuators as well as a collective actuator allow the

AFCS to physically move the cyclic, collective, and directional controls. Two sets of pitch, roll, and yaw linear actuators provide automatic stabilization inputs that do not result in cockpit control feedback. The trim, collective, and linear actuators are electrically driven. Additional information on the AFCS can be found in Section 4.0 of this report.

The cyclic and collective inputs are transmitted, via a system of control tubes and bell cranks, and combined within a mixing unit and to the three main rotor hydraulic actuators. Pedal inputs are transmitted, via a series of push-pull tubes and bell cranks, to the tail rotor hydraulic actuator. The brakes for the landing gear are mounted on the pedals.

The main hydraulic power system is composed of two independent hydraulic circuits. Each system contains two hydraulic pumps that provide pressurized hydraulic fluid to a power control module (PCM), the latter of which also contains the hydraulic reservoir for their respective hydraulic circuit. The No. 1 hydraulic circuit has a mechanically-driven and an electrically-driven hydraulic pump, while the No. 2 hydraulic circuit has two mechanically-driven hydraulic pumps. The No. 1 hydraulic pump is the mechanically-driven pump in the No. 1 hydraulic circuit and is mounted on the left input shaft module. The No. 3 hydraulic pump is the electrically-driven pump in the No. 1 hydraulic circuit and is located on the upper deck in front of the main gearbox. The Nos. 2 and 4 hydraulic pumps are the mechanically-driven pumps in the No. 2 hydraulic circuit. The No. 2 hydraulic pump is mounted on the right input shaft module and the No. 4 hydraulic pump is mounted on the main gearbox front housing. The No. 1 PCM is located in front of the left input shaft module while the No. 2 PCM is located in front of the right input shaft module. The PCMs provide hydraulic power to the three main rotor hydraulic actuators and the tail rotor hydraulic actuator. Additionally, the No. 2 PCM provides normal hydraulic power to the landing gear while the No. 1 PCM provides emergency hydraulic power to the landing gear.

2.4.2 COCKPIT FLIGHT CONTROL OBSERVATIONS

The right-seat pedals remained installed. Of the right-seat pedals, the left pedal stem was deformed and the right pedal stem was intact. The two brake pistons remained installed on the pedals. There was no evidence of disconnection on the rod linkages from the right-seat pedals to the control tube going into the cockpit floor.

The right-seat cyclic control remained installed (**Figure 9**). The cyclic grip was present but the cyclic head was fractured but attached via wiring. The pilot-side door post interior covers were removed to facilitate examination of the control tubes. Movement of the cyclic in the fore-aft axis resulted in movement of the vertical control tube (in the pilot-side door post) but the freedom of movement was extremely limited. Movement of the cyclic in the fore-aft axis resulted in movement of the transmission deck-mounted bell cranks (connected to the main rotor servo input rods). The cyclic control could not be moved in the lateral axis.

The right-seat collective control was present but the collective head was fractured from the control but remained attached via wiring. A passenger chair was resting on top of the right collective. After the chair was removed, the collective control was seen to be partially attached at its base. Movement of the collective control resulted in movement of its base and the composite jackshaft (torque tube) connected to it. The jackshaft was fractured about 5 inches inboard of its connection to the left collective control base.

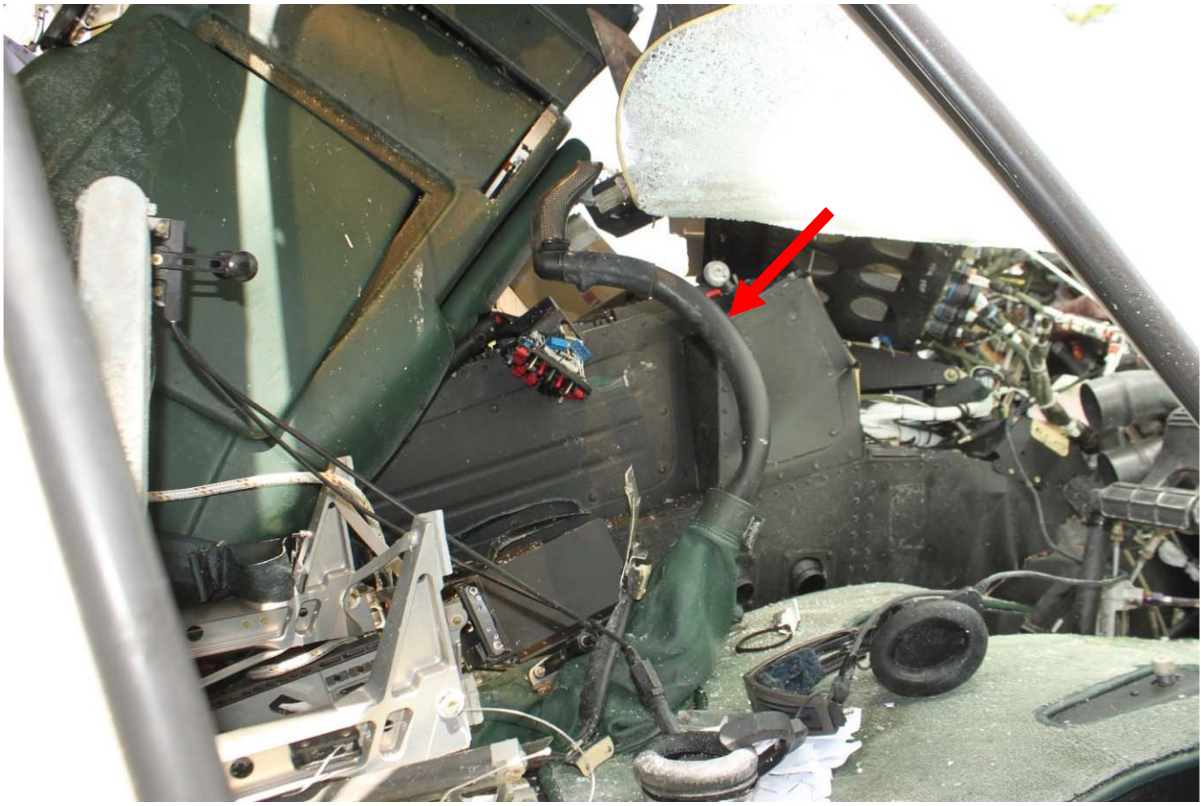

Figure 9. The red arrow points to the right-seat cyclic control. The head of the collective control and a portion of the pedals are also visible.

The left-seat cyclic control was installed to its base but was rotated to the right. The cyclic grip was fractured from the cyclic control but remained attached via wiring.

The left-seat pedals were not present but the pedal mounts and base remained installed on the cockpit. The brake pistons remained attached but it's linkages were deformed. The linkages routing to the cockpit floor were deformed but exhibited no evidence of disconnection.

The left collective control was fractured and separated from its base. The collective control was connected to the base via wiring. The grip and the head were both fractured from the collective control but remained attached via wiring.

Both ECLs were in the OFF position. An occupant seat that was present in the cockpit was underneath both ECLs. The rotor brake handle was in the OFF position.

2.4.3 FLIGHT CONTROL SYSTEM OBSERVATIONS

The cabin roof was removed to facilitate examination of the flight controls. The control sticks were coupled, with limited movement, in pitch and roll. Moving the pitch and roll vertical control rods in the forward cabin resulted in limited movement at the sticks and movement of the main rotor servo input rod. The collective torque tube between the left and right controls in the cockpit was severed. The vertical control rod in the forward cabin was bent aft in a "C" shape. The torque tube in the upper left of the forward cabin was severed with a 45-degree break. Turning the torque tube inboard of this fracture resulted in movement of the main rotor servo input rod. The input linkages to the main rotor hydraulic actuators remained attached to the transmission deck-mounted bell cranks.

The engine control flex cables were detached from the engines to facilitate engine recovery. They were attached correctly to their respective engine control. Movement of both left and right power levers resulted in movement of the internal cables at the terminal ends.

The tail rotor control tube, between the left pedals and the right pedals, remained connected at the pedals as well as its combining bell crank. The control tube running aft from the combining bell crank was separated at this point. The aft portion of this control tube was continuous through the center fuselage but exhibited a fracture of its rod end in the aft fuselage at STA 8700, the location where the tail boom attaches to the aft fuselage. The tail rotor control tubes that normally reside in the tail boom and vertical stabilizer exhibited fractures in multiple locations consistent with overstress failure. A segment of the tail rotor control tube that resides between STA 8700 and the yaw linear actuator bell crank was not recovered. All fracture surfaces observed exhibited evidence of overstress failure and no evidence of disconnection was found at the rod end connection points.

2.4.4 AFCS ACTUATORS

The pitch and roll linear actuators were found intact and both measured about 20 inches from its inboard rod end to closest outboard rod end connection (i.e. not inclusive of the short links connected to the pitch and roll bellcranks located near the right cockpit door frame). The yaw linear actuator was found separated in the recovered wreckage, but remained connected to the control rod bell crank at its forward end and the support bracket on its aft end. The front portion of the forward yaw linear actuator was fractured at its housing, but remained connected to the assembly via wiring.

The pitch, roll, and yaw trim actuators and the collective actuator were located in the crushed structure in the forward left side of the cockpit. All four actuators remained attached to the airframe and their housings did not exhibit significant deformation. All connections between the actuators and the cyclic, collective, and pedal control system remained intact.

2.4.5 HYDRAULIC SYSTEM

The No. 1 and No. 2 hydraulic pumps remained installed on their respective input shaft modules. The No. 3 hydraulic pump remained installed on the upper deck. The No. 4 hydraulic pump was partially separated from its main gearbox mounting flange. All lines remained connected and secured to all four hydraulic pumps. The left and right power control modules (containing their respective hydraulic reservoirs) remained installed on the transmission deck. The left reservoir exhibited an inward deformation while the right reservoir did not exhibit visible deformation. A fluid level was not visible on both the left and right hydraulic reservoir gauge. All hydraulic lines attached to the left and right power control modules remained secured. For both left and right power control modules, the pressure and return filter clogged indicators did not pop out.

The tail rotor shutoff valve remained installed on the airframe near the No. 2 PCM. The hydraulic and electric lines remained connected to the tail rotor shutoff valve. The overboard drain line from the No. 1 PCM was fractured outboard of the T-junction. The electric pump return line to the No. 1 PCM was fractured near the attachment to the No. 1 PCM port.

2.5 COCKPIT INSTRUMENTS AND AVIONICS

2.5.1 OVERVIEW

The AW139 helicopter is equipped with the Honeywell Primus Epic integrated avionics system. The instrument panel contains four flat-panel electronic flight instrument system (EFIS) displays. A set of two EFIS display panels, comprising a primary flight display (PFD) and multifunction display (MFD) installed side-by-side, is present for each crew seat. All four EFIS display panels have a portrait orientation.[8] The PFDs are located on both the left and right outboard sides of the instrument panel for the copilot and pilot, respectively. The PFD displays shows indicators for attitude, heading, indicated airspeed (IAS), vertical speed (VS), power index (PI), and triple tachometer[9], as well as barometric and radio altimeters, outside air temperature and wind vector, and radio frequencies. The MFD contains multiple selectable tabs that change the information presented. From the systems perspective, the MFD can present engine parameters (such as gas generator speed and interturbine temperature), engine and drive system indications, hydraulic and electrical system indications, as well as the crew alerting system (CAS) messages window. A standby attitude indicator is also installed on the instrument panel. The helicopter is equipped with two pitot-static probes and two alternate static-source valves used for air data.

The instrument panel has two display controllers: the pilots display controller is located to the left of the pilot MFD, while the copilot display controller is located to the right of the copilot MFD. The AFCS guidance controller, which controls the flight director (FD), and AP control panel are mounted on the center console.[10] A miscellaneous control panel is installed on the center console and contains on/off switches for force trim and collective/yaw trim. Two modular avionics units (MAU), identified as the No. 1 MAU (MAU1) and the No. 2 MAU (MAU2), are installed in an avionics cabinet located in the aft fuselage. MAU1 is installed in the left avionics cabinet and MAU 2 is installed in the right avionics cabinet. The accident helicopter was equipped with an enhanced ground proximity warning system (EGPWS), which incorporated a terrain awareness warning system (TAWS) function, that was installed in the aft fuselage.

2.5.2 OBSERVATIONS

The left pitot-static probe was pulled out from the surrounding structure and was hanging from the pitot hose and electrical wire. The pitot and static inlet holes did not

[8] A display with a portrait orientation has an aspect ratio where the display's vertical dimension is larger than its horizontal dimension.
[9] The triple tachometer shows rotor speed (Nr) as well as power turbine speed (Nf) values from both engines.
[10] More information about the AFCS can be found in Section 4.0 of this report.

appear to be blocked. The static 1 connection to the pitot-static probe was secure but the hose was not connected on its inboard terminal. The static 2 connection to the pitot-static probe was secure but the connecter was bent and the hose was missing. The pitot hose connection was secure. The electrical cannon plug was secure. The right pitot-static probe was pulled out from the surrounding structure and was hanging from the hoses and electrical wire. The pitot and static inlet holes did not appear to be blocked. The two static hoses were connected and secure. The pitot hose connection was secure but the hose was bent. The electrical cannon plug was secure.

For both the pilot and copilot seats, both PFDs and MFDs were retained in the instrument panel. The copilot MFD screen exhibited substantial impact damage while the copilot PFD and the pilot PFD and MFD exhibited minor damage. The pilot display controller was present on the instrument panel. The copilot display controller was separated from its mount on the instrument panel.

The guidance controller and AP control panel remained retained in the center console but the control cover exhibited slight deformation due to impact (**Figure 10**). On the guidance controller, the lateral/vertical approach (APP), altitude (ALT), altitude acquire (ALTA), and hover/velocity hold (HOV) buttons were found depressed (down position); the remaining buttons on the guidance controls were found not depressed (up position). The AP switch positions could not be determined. On the miscellaneous control panel, both the force trim and the collective/yaw trim switches were in the "on" (forward) position.

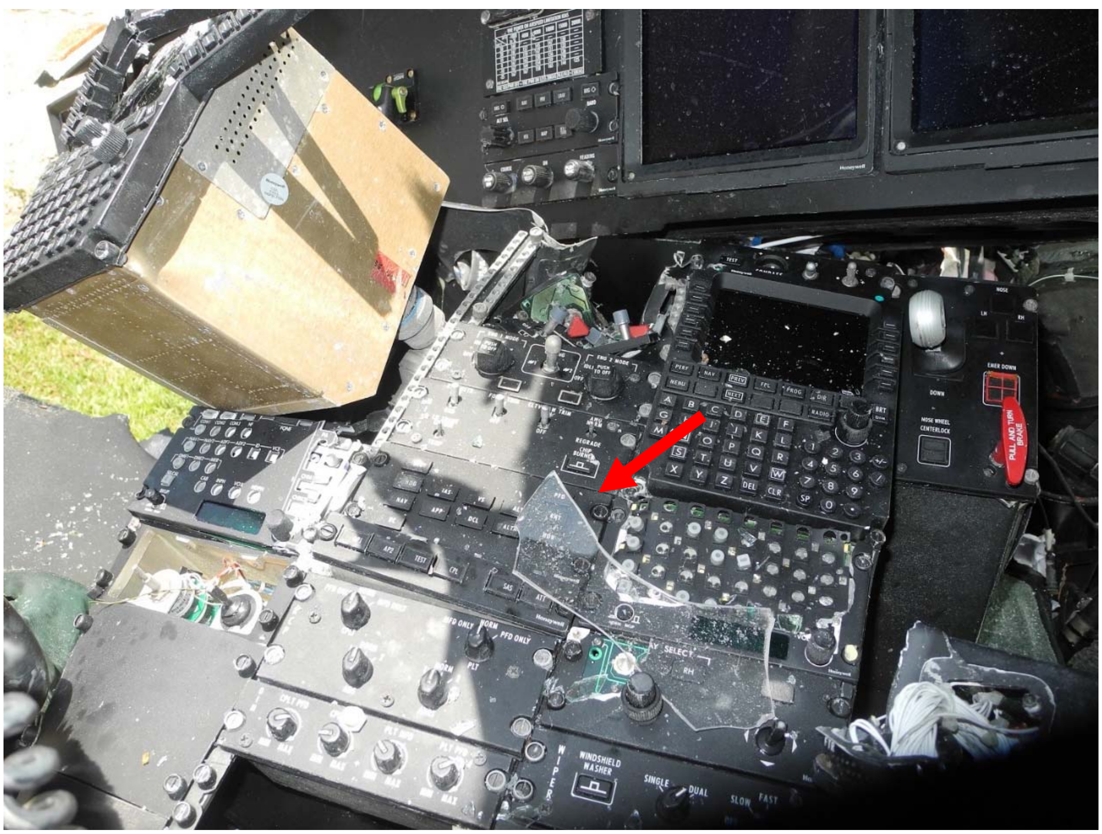

Figure 10. The center console of the cockpit. The red arrow points to the guidance controller and AP control panel.

The MAU from the right avionics cabinet was removed by the AAID after recovery and was shipped to Jacksonville immersed in fresh water. The MAU from the right avionics cabinet contained two white-colored labels that stated the following (a "?" signifies unreadable text due to damage to the label):

P/N 3G4600A00133
DESC: M.A.U. N.1
ESP: B STC: 531-1
O.L: 60485552

MODULAR AVIONICS UNIT
TSO AUTHORIZ???ON FOR CA?INET ONLY
PART NO. 70??629-1950
SERIAL NO. ??030314
W.T. 10.74 ??? TSO ?153

The left avionics cabinet still contained its MAU. When the MAU from the left avionics cabinet was removed, two white-colored labels were present. Those labels stated the following:

P/N 3G4600A00233A1R
DESC: M.A.U. N.2 + GPS
ESP: C STC: 548-3
O.L: 60450984

MODULAR AVIONICS NIT
TSO AUTHORIZATION FOR CABINET ONLY
PART NO. 7031629-1950
SERIAL NO. 07020305
WT. 10.74 LBS TSO C153

The investigation group used a black-colored marker to write "MAU1" on the MAU from the left avionics rack and "MAU2" on the MAU from the right avionics rack. MAU1 and MAU2 were retained for further examination.[11]

The EGPWS was found in its normally installed location and was removed by the group for further examination. A Hobbs meter was found on the right avionics rack access panel. The Hobbs meter read "2611.1" hours and a sticker label below it read "add 1897.0".

2.6 ENGINES

2.6.1 ENGINE DESCRIPTION

The PWC PT6C-67C is composed of a four-stage axial and single stage centrifugal compressor, a reverse-flow annular combustion chamber, a single stage compressor turbine, and a two-stage power (free) turbine. The free turbine powers the output shaft

[11] According to Leonardo, at the time of the helicopter's manufacture, the MAU installed in the left avionics cabinet (MAU1) was P/N 7031629-1950 and S/N 07030314 and the MAU installed in the right avionics cabinet (MAU2) was P/N 7031629-1950 and S/N 07020305.

through a direct drive section. Exhaust gases from the turbine are ejected out to the atmosphere through an exhaust collector duct routed aft of the engine. All engine-operated accessories are installed on the accessory gearbox at the rear of the engine. The engine is equipped with a single channel electronic engine control (EEC) and an electro-hydro-mechanical fuel management module (FMM), driven by the engine accessory gearbox, to modulate fuel flow.

2.6.2 NO. 1 ENGINE OBSERVATIONS

The No.1 engine mounts remained attached to the engine but were separated from the airframe, resulting in an aft and inboard displacement of the engine. The engine data collection unit (DCU), S/N DP06-447, remained attached, was removed by the group for further examination and subsequently rinsed and submerged in fresh water.[12]

The air inlet screen was deformed but the mesh was intact. Several first stage axial compressor blades were bent opposite the direction of rotation. The power turbine rotor was seized and the 1st stage compressor could be rotated with heavy rubbing. The single magnetic chip detector was clean. The No. 5 bearing housing support case was fractured circumferentially into two pieces (**Figure 11**). One of the two power turbine speed and torque sensors was intact and remained with the front portion of the No. 5 bearing housing support case; the second sensor remained with the aft portion of the No. 5 bearing housing support case and was partially sheared off.[13] The oil scavenge lines were intact. The firewall deformation was consistent with impact damage, and the exhaust case was deformed. The interturbine temperature wiring harness was intact, and the terminal box was impact damaged. The igniter wires and igniter were intact. The oil breather tube was crushed. The oil dipstick was intact and the oil appeared normal. About 2 gallons of fluid was drained from the oil reservoir, during which water mixed with the oil was apparent. The oil bypass had not activated. The fuel lines and their connections remained intact. A fuel nozzle was removed close to the 12 o'clock and 6 o'clock positions, and both had fuel in them. This was tested with a water detecting dye; small traces of water at the bottom nozzle but very little water at the top.

The FMM indices indicated power level just above "MIN". An attempt was made to change the ECL to the maximum detent, and although the ECL achieved this detent the FMM indicator did not move. The oil and fuel filters were removed and both were clean. The No. 1 engine EEC, S/N 06110077, was removed by the group for further examination, rinsed, and submerged in fresh water for preservation.

The EEC and DCU were subsequently shipped to the TSB of Canada for further examination. The engine was removed from the airframe on July 9th and subsequently shipped to PWC for further examination.

[12] As a standard procedure, electronics that were submerged in water, and are of interest to an investigation, are rinsed and submerged in fresh water to reduce the risk of corrosion that could preclude successful download of its nonvolatile memory.
[13] Each engine is equipped with two sensors, each of which are capable of measuring both power turbine speed and torque.

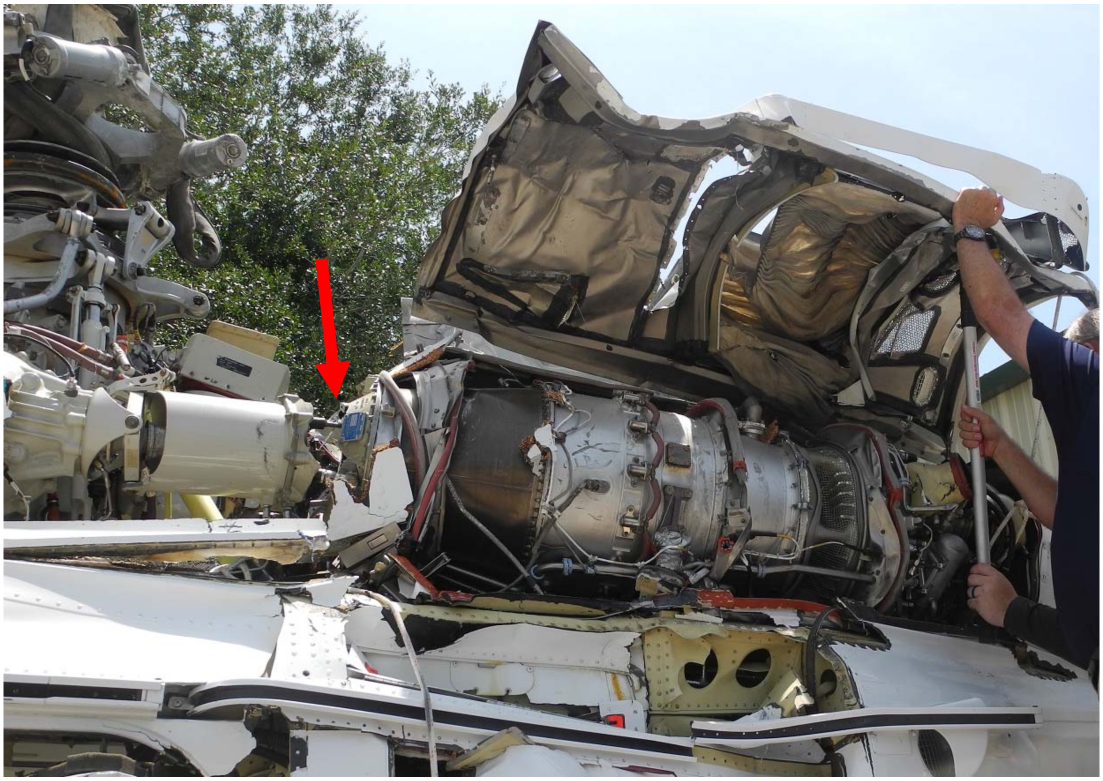

Figure 11. The No. 1 engine installed on the accident helicopter. The red arrow points to the fractured No. 5 bearing support case.

2.6.3 NO. 2 ENGINE OBSERVATIONS

Examination of the No. 2 engine, PT6C-67C, S/N KB0239, revealed no visible damage was noted to the engine exterior (**Figure 12**). The outboard mount was slightly deformed upward. The DCU, S/N 18061916, remained attached, was removed by the group for further examination, and subsequently rinsed and submerged in fresh water.

The air inlet screen was deformed but the mesh was intact. The No. 5 bearing housing support case was intact and both power turbine speed and torque sensors remained installed. The oil bypass had not activated, and the single chip detector was clean. All second stage power turbine blades were present and could be easily rotated by hand both by using the power turbine blades as well as the output shaft. The 1st stage compressor rotor could be rotated by hand. Three 1st stage compressor blades were bent in the direction opposite of normal rotation.

The accessories, fuel and air lines showed no deformation and were all correctly installed. The fuel filter was clean. Fuel was present in the fuel filters, and some fuel was collected and inspected and found to be without metallic particles. Two fuel nozzles were removed for inspection and fuel was present in both. The oil filter was removed and oil was present. No debris was found on the oil filter, but there was a slight trace of water. Oil was visible in the sight glasses and the level was above the minimum mark, or about 1.75 gallons of oil, mixed with water, was drained.

Figure 12. The No. 2 engine installed on the accident helicopter.

The FMM cable position was below the off position, however the entry cable was severely deformed during impact. An attempt was made to change the ECL position to see if the FMM position would move, but the ECL was jammed in position and would not move. The No. 2 engine EEC, S/N 07021053, was removed by the group for further examination, and was rinsed and submerged in fresh water for preservation.

The EEC and DCU were subsequently shipped to TSB of Canada for further examination. The engine was removed from the airframe on July 9, 2019 and subsequently shipped to PWC for further examination.

3.0 EXAMINATION OF THE ENGINES AT PWC

From August 26 to 29, 2019, the engines from the accident helicopter were examined at PWC facilities in Longueuil, Quebec, Canada. Representatives from AAID, TSB of Canada, and PWC were present for the examination. Evidence of salt deposits and corrosion, due to saltwater immersion of the engines, was seen throughout the disassembly of the engines. Thus, in this section, any damage observed due to corrosion will not be specifically mentioned unless warranted.

3.1 NO. 1 ENGINE EXAMINATION (S/N KB0224)

The No. 5 bearing housing support case was fractured into two pieces. The power turbine speed/torque sensor, which had remained installed in the aft portion of the No. 5 bearing housing support case, was bent. The sensor tip showed evidence of contact with the torque sensing teeth. Four of the torque sensing teeth had sheared off and a fifth tooth was bent in the direction opposite of normal rotation. The bolts holding the two shafts together had also sheared off. The other power turbine speed and torque sensor remained with the broken front portion of the No. 5 bearing housing support case and was undamaged.

The compressor discharge air lines remained attached. The compressor bleed valve was removed and the piston was moved by hand, resulting in water exiting the torque motor cavity. The torque motor on the bleed valve passed its functional test. The piston was found cracked and the bleed valve diaphragm was slightly cracked; according to PWC these cracks would not have affected normal engine operation when in auto mode. The chip detector and oil filter exhibited no evidence of ferrous debris.

All blades of the first, second, third, and fourth axial compressor stages were present. Several of the first stage compressor blades exhibited bending in the direction opposite of normal rotation, but the degree of bending differed between these blades. There was evidence of blade contact (rubbing) against the first stage shroud due to the blades' deformation. The second, third, and fourth stage compressor blades showed no evidence of rubbing or anomalous damage. The second and third stage shrouds showed no evidence of rubbing. None of the stators' airfoils exhibited anomalous damage. The centrifugal impeller exhibited evidence of rubbing at the exducer portion of the blades with corresponding contact marks observed on the impeller shroud. There was no other anomalous damage to the impeller as well as the fourth stage stator. The No. 1 and No. 2 bearings and their air seals exhibited substantial corrosion.

The combustion chamber outer and inner liner surfaces exhibited soot deposits, but these soot marks were in an evenly distributed pattern with no signs of streaking. No anomalous damage was observed within the combustion chamber liner.

The compressor turbine guide vane ring, the turbine shroud, and the turbine disk and blades showed no evidence of anomalous damage. All blades from the compressor turbine were present.

All interturbine temperature probes remained installed. The harness and busbar were present with no anomalous damage. Due to saltwater immersion, these components were not tested.

The power turbine housing did not exhibit any damage. The first stage power turbine stator showed no evidence of rubbing on its upstream side, but evidence of light tip rubbing was observed on its downstream side. The second stage power turbine stator showed no evidence of rubbing on its upstream side, but its downstream side showed evidence of rubbing with the second stage power turbine knife edge tips. The power turbine shroud showed evidence of light rubbing. All blades from the first and second stage power turbines were present. The first stage power turbine blades and disk showed evidence of light rotational scoring on its downstream side and the blade tips exhibited light rubbing on its upstream side. The second stage power turbine blade knife edges exhibited evidence of rubbing against the shroud, and the upstream tips of the shrouded blades had rubbed against the backside of the shroud. The upstream side of the second stage power turbine blades showed no other damage. The downstream side of the second stage power turbine blades exhibited evidence of rubbing against the No. 3 bearing cover. The power turbine shaft bearings were seized and had extensive corrosion. Shaft itself had black-colored deposits on its surface consistent with carbon deposits typical from in-service operation.

The accessory gearbox was opened and extensive corrosion was observed on the gears and bearings. All bearings were seized. The ignitor plugs were corroded but had no other anomalous damage. The fuel heater, fuel pump, fuel control unit, and fuel nozzles did not exhibit anomalous damage. The oil filler cap remained installed and its O-ring was in good condition.

3.2 NO. 2 ENGINE EXAMINATION (S/N KB0239)

The compressor discharge air lines remained intact. The bleed valve had no anomalous damage. The accessory gearbox chip detector and the oil filter were removed and had no ferrous debris. During removal of the fuel filter, about 3.4 fluid ounces of fuel came out with the filter. The fuel exhibited a clean appearance with no evidence of debris.

All blades from the first, second, third, and fourth axial compressor stages were present. Three of the first stage compressor blades were bent in the direction opposite of normal rotation. The first stage compressor blades had evidence of rubbing against their shroud, and circumferential rub marks were present on the shroud. There were minor nicks on the leading edges of several first stage compressor blades, consistent with typical wear seen from in-service operation, but no other anomalous damage was observed. The second, third, and fourth stage compressor blades showed no evidence of deformation. The second and third stage shrouds showed no evidence of rubbing from their respective compressor blades. The compressor's first stage, second stage, and third stage stator airfoils showed no evidence of anomalous damage. The exducer portion of the centrifugal impeller blades showed evidence of rubbing against its shroud, but the impeller blades showed no other anomalous damage. No anomalous damage was observed on the front stub shaft. The No. 1 and No. 2 bearings and their air seals did not exhibit anomalous damage.

The combustion chamber outer and inner liner surfaces exhibited soot deposits, but these soot marks were in an evenly distributed pattern with no signs of streaking. No anomalous damage was observed within the combustion chamber liner.

The compressor turbine guide vane ring and the compressor turbine shroud did not exhibit anomalous damage. All blades from the compressor turbine were present. No evidence of rubbing or anomalous damage was observed on the compressor turbine blades.

All interturbine temperature probes remained installed. The harness and busbar were present with no anomalous damage. Due to saltwater immersion, these components were not tested.

The first stage power turbine stator showed no evidence of rubbing or anomalous damage on its leading edges, but light rubbing was observed on the downstream side of the stator. The second stage power turbine stator exhibited light rubbing on the upstream side of the stator hub, but no other anomalous damage was observed. The first and second stage power turbine shrouds showed evidence of rubbing due to contact with the knife edges of the first and second stage power turbines, respectively. The rubbing on the second stage power turbine shroud was more prominent than that seen on the first stage power turbine shroud. All blades from the first and second stage power turbine were present. The downstream side of the first stage power turbine blade platforms showed evidence of rubbing with its respective stator inner shroud. The upstream side of the second stage power turbine blade platforms showed evidence of rubbing into its respective stator. The second stage power turbine blade tips also exhibited evidence of rubbing against its respective shroud. The power turbine shaft bearings were seized and had extensive corrosion. Shaft itself had black-colored deposits on its surface. Both power turbine speed/torque sensors remained installed. The left power turbine speed and torque sensor connector was partially dislodged while the right sensor connector remained intact. The probes from both left and right power turbine speed and torque sensors exhibited no anomalous damage.

The accessory gearbox was opened and extensive corrosion was observed on the gears and bearings. All bearings were seized. The ignitor plugs were corroded but had no other anomalous damage. The fuel heater, fuel pump, fuel control unit, and fuel nozzles did not exhibit anomalous damage. The oil filler cap remained installed and its O-ring was in good condition.

4.0 PRIMUS EPIC AFCS

4.1 SYSTEM DESCRIPTION

The AW139 AFCS is part of the Honeywell Primus Epic integrated avionics system. The AFCS has two APs, AP No. 1 (AP1) and AP No. 2 (AP2), as well as two FDs, flight director No. 1 (FD1) and flight director No. 2 (FD2). The APs normally operate together and each is capable of providing full functionality in case of a failure of the other AP. Each AP commands a set of three limited-authority, high-bandwidth linear actuators (pitch, roll, and yaw). The linear actuators are installed in series with the flight controls and do not result in cockpit control feedback when active. The priority AP controls the set of full-authority trim actuators (pitch, roll, yaw, and collective). The trim actuators are installed in parallel with the cockpit controls, such that movement of a trim actuator will result in movement of the corresponding cockpit control. The priority AP will move the trim actuators as necessary to allow the linear actuators to operate about their center positions. The FDs operate simultaneously with a priority/non-priority relationship, and the non-priority FD synchronizes with the priority FD. Both Aps receive commands from the priority FD during coupled operations. The crew can override the AFCS at any time by manually operating the flight controls and provides full authority over the flight controls regardless of whether the AP is engaged or disengaged. Two AFCS modules are contained in each MAU cabinet. The MAU1 contains two actuator input/output with processor (AIOP) modules for the No. 1 AFCS (AP1 and FD1): AIOP A1 and AIOP B1. The MAU2 contains two AIOP modules for the No. 2 AFCS (AP2 and FD2): AIOP A2 and AIOP B2. Additionally, each MAU has a network interface controller and processor (NIC/PROC) module. A central maintenance computer (CMC) module resides in MAU1.

When a FDs are in standby mode (no guidance modes are active) but the APs are engaged, the APs operate in the attitude stabilization (ATT) mode or the stability augmentation system (SAS), as chosen by the pilot. ATT mode provides long-term pitch and roll attitude retention by using the pitch and roll trim actuators and short-term attitude retention and stability via the pitch and roll linear actuators. SAS mode is a “hands on” flying mode that provides helicopter rate dampening as well as dampening of short-term helicopter disturbances, such as turbulence, in the pitch and roll axes using those respective linear actuators. Yaw axis functionality, such as body rate dampening and turn coordination, is provided irrespective of ATT or SAS mode via the yaw linear actuators and the yaw trim actuator. FD/AP coupled operation is possible only when the APs are operating in the ATT mode.

During flight, the crew can set a desired airspeed, using the IAS mode, and a desired altitude, using the ALTA mode, by activating those FD modes on the guidance controller. The AFCS will control the cyclic, collective, and pedals, via their respective trim actuators, to achieve the desired conditions. In the AW139 rotorcraft flight manual (RFM), Supplement 34, titled “4-axis enhanced flight director (EPIC Phase 4),” Section 1 (“Limitations”) contains the airspeed ranges and minimum use heights, in feet above ground level (AGL), for the FD modes, a selection of which is seen in **Table 1**. The desired IAS can be set by using the cyclic beep trim switch or by depressing the cyclic force trim release (FTR) switch while flying to the desired IAS and subsequently releasing the cyclic FTR switch. The desired altitude can be set by using the altitude select knob on the display controller. Once the helicopter reaches the desired altitude, the collective FD mode will automatically change to an altitude hold (ALT) mode. Additionally, a desired VS to achieve the desired altitude can be specified by the crew, but will default, at ALTA mode engagement, to 1,000 feet per minute (fpm) when below the desired altitude, unless changed by the crew. The desired VS can be changed by using the collective beep trim switch or by depressing the collective FTR switch while flying to the desired VS and subsequently releasing the collective FTR switch. The crew can manually override the trim actuators by two means: 1) depressing and holding the respective control's FTR switch or 2) pushing against the trim actuator's force gradient spring.

Table 1. Select FD mode engagement limits and minimum use height per the AW139 RFM.

FD Mode	*Airspeed Range*	*Minimum Use Height*
IAS	60 knots indicated airspeed (KIAS) to the never exceed speed (Vne) less 5 KIAS	150 feet AGL to 50 feet AGL during approach
ALT	0 KIAS to Vne	300 feet AGL (airspeed greater than 55 KIAS) 50 feet AGL in HOV or airspeed less than 55 KIAS
ALTA	60 KIAS to Vne	150 feet AGL
VS	60 KIAS to Vne within -1500 fpm and 2000 fpm	150 feet AGL

When a FD mode is controlling the collective axis, collective movement by the AFCS is limited, based on PI values, by a PI limiting function, also known as the "PI limiter".[14] According to the AW139 RFM Supplement 34, during all engines operating (AEO) flight, the maximum PI value is 106% for airspeeds less than 60 knots indicated airspeed (KIAS) and 97% for airspeeds above 60 KIAS. During a one engine inoperative (OEI) scenario, the maximum PI value is 140%. When the PI limiting function is active and is limiting maximum collective movement, an amber "LIM" caption is displayed adjacent to the collective cue on the PFD. Furthermore, the RFM states that if PI limiting is active with IAS engaged and the reference for the collective mode enabled, such as ALTA or VS, cannot be achieved, the FD will reduce airspeed automatically to achieve the collective mode reference. Lastly, the RFM states that if the collective mode reference cannot be maintained and the airspeed has reached a minimum of 80 KIAS, the FD will maintain that airspeed and reduce the collective mode reference and display that mode's caption on the PFD in amber.

4.2 EXAMINATION OF AFCS MODULES

On July 8, 2019, the recovered MAUs were rinsed with fresh water. On August 8, 2019, the MAUs were shipped to Honeywell in Phoenix, Arizona for preservation and further

[14] The PI value is a composite of three primary engine parameters: gas generator speed, interturbine temperature, and torque. Conceptually, the PI value is similar to a first limit indicator and the value displayed will be the parameter that is closest to reaching its operational limit. The PI scale is shown as a percentage value and is presented in three colored ranges: green, amber, and red. The green range is where the engine(s) may operate without a time restriction. The amber range is where the engine(s) may operate for a limited time. The red range is where engine operation is prohibited. During an AEO regime, the colored bands are defined as follows: 0 to 100.4% is green; 100.5% to 110.4% is amber; and 110.5% to 200.4% is red.

examination. According to Honeywell, the MAU from the left avionics cabinet conformed to a MAU1 configuration and the MAU from the right avionics cabinet conformed to a MAU2 configuration.

On August 27 to 28, 2019, members of the Airworthiness Group convened at Honeywell facilities in Phoenix, Arizona, to examine the NIC/PROC modules from MAU1 and MAU2, the AIOP modules from MAU1 and MAU2, and the CMC module from MAU1.

A Primus Epic emulator test bench was set up such that the modules would boot in safe mode to prevent erasure of stored data. An exemplar NIC/PROC module was initially connected to the test bench to demonstrate communication between the test bench and the module.

The NIC/PROC modules from MAU1 and MAU2 were installed on the test bench and powered. Neither module visually exhibited damage nor was a burnt smell present when the modules were powered. However, communication between the emulator and both modules was unsuccessful.

The microchip, called the "U13 chip", which contains the fault data from the electronically-erasable programmable read-only memory (EEPROM), was removed from each NIC/PROC module and installed onto a surrogate circuit board in order to download its data. The conformal coating was removed from the U13 chips prior to their installation onto a surrogate circuit board. The surrogate circuit board was installed onto a test bench and the EEPROM from each NIC/PROC module was successfully downloaded.

The fault data from the MAU1 NIC/PROC started on June 29, 2019 at 13:05:51 coordinated universal time (UTC), ended on July 3, 2019 at 22:38:10 UTC, and contained a total of 6 entries.[15] The fault data from the MAU2 NIC/PROC started on June 29, 2019 at 13:35:12 UTC, ended on July 3, 2019 at 22:38:13 UTC, and contained a total of 7 entries.[16]

The AIOP A1 (S/N 07050830) and AIOP B1 (S/N 07010764) modules were installed on the test bench and powered. The network interface controller communicated with the test bench, but neither module would communicate with the test bench.

The AIOP A2 (S/N 07040903) module was installed on the test bench and powered. Initially, both the network interface controller and the module did not communicate with the test bench. The AIOP A2 module was removed from the test bench. After the test bench was powered (without the AIOP A2 module installed), the network interface controller communicated with the test bench.

The AIOP B2 (S/N 07040913) module was installed on the test bench and, initially, both the network interface controller and the module did not communicate with the test bench. The AIOP B2 module was removed from the test bench. After the test bench was powered (without the AIOP B2 module installed), the network interface controller communicated with the test bench. The AIOP B2 module was reinstalled onto the bench and its EEPROM, boot flash, operations flash, and database flash memory were successfully downloaded.

[15] The first entry in the fault log for the MAU1 NIC/PROC was a fault queue initializer and is not considered a fault.
[16] The first entry in the fault log for the MAU2 NIC/PROC was a fault queue initializer and is not considered a fault.

The U13 chip from the AIOP A1, AIOP A2, and AIOP B1 modules was removed, their conformal coating was removed, and the chips were subsequently installed onto a surrogate circuit board. The surrogate circuit board was installed onto a test bench and the EEPROM from each of the aforementioned AIOP modules was successfully downloaded.

The fault data from the four AIOP modules showed a log of faults recorded at a timeframe consistent with the accident time and preceding the accident. The AIOP A1 fault data started on July 3, 2019 at 22:38:08 UTC, ended on July 4, 2019 at 5:54:32 UTC, and contained a total of 22 entries.[17] The AIOP A2 fault data started on May 21, 2019 at 19:34:48 UTC, ended on July 4, 2019 at 5:54:32 UTC, and contained a total of 179 entries. The AIOP B1 fault data started on June 20, 2019 at 22:04:31 UTC, ended on July 4, 2019 at 5:54:32 UTC, and contained a total of 94 entries. The AIOP B2 fault data started on May 13, 2019 at 13:52:20 UTC, ended on July 4, 2019 at 5:54:32 UTC, and contained a total of 179 entries.

A review of the fault data showed them in two categories: 1) faults that were logged at a time consistent with the accident helicopter's impact with water and 2) faults that were logged during the operational history of the helicopter prior to the accident. For the latter, the majority of the faults were those associated with powering up of the AFCS and other helicopter systems or were known 'nuisance' faults that are generated by the AFCS preflight test.

The CMC module was installed on the test bench and powered. The module successfully communicated with the bench, and the bench was able to access the base operating system on the module but was not able to load beyond the module boot disk due to a boot fault. Power was cycled on the test bench but the result was identical as the initial attempt. The "DiskOnChip 2000"[18] was removed from the CMC module and one of its pins was found to be partially broken; it was later determined the broken pin had no connection. The DiskOnChip 2000 was installed on an exemplar circuit board and subsequently installed on a test bench, but the result was identical to the prior two attempts. The DiskOnChip 2000 was removed, cleaned, and reinstalled onto the same exemplar circuit board. The exemplar circuit board was installed on the test bench and powered, but the results were identical to the prior three attempts. X-ray images were taken of the DiskOnChip 2000 and no evidence of cracks were found. Using the test bench, the boot disk on the DiskOnChip 2000 was circumvented manually, but the data on the DiskOnChip 2000 still could not be accessed. Inspection of the DiskOnChip 2000 under a microscope revealed damage to its surface that could have led to saltwater intrusion into the chip.

4.3 PI LIMITER SIMULATION STUDY

The Airworthiness Group Chairman requested Honeywell perform a simulation to study the collective PI limiter behavior. The simulation study provided a basis to compare the behavior of the PI limiter outputs, i.e. its effects on the collective servo commands, using the accident flight parameters as inputs to the simulation.[19] The study showed a strong correlation between the accident flight collective servo position and the simulation collective servo commands. During the final 8 seconds of the simulation (flight), the PI limiter decreased the rate of increasing collective servo commands in order to prevent the PI from reaching and exceeding the upper limit.

[17] The first entry in the fault log for AIOP A1 was a fault queue initializer and is not considered a fault.
[18] "DiskOnChip 2000" was the name on the chip installed on the module.
[19] For information on the accident flight data, see the Flight Data Recorder Factual Report in the docket for this investigation. For information on the helicopter performance during the accident flight, see the Aircraft Performance Study in the docket for this investigation.

5.0 MAINTENANCE

According to the helicopter logbook, the last maintenance action on the accident helicopter began in late May 2019 and was completed and signed off on June 28, 2019. At the time of the last maintenance action, the ACTT was about 2,154.1 hours. The last maintenance action was performed by Rotortech Services in West Palm Beach, Florida under work order No. 2320. According to the helicopter logbook, the following inspections were performed under work order No. 2320:

- 12-hour inspection per the AW139 aircraft maintenance manual (AMM)
- 25-hour inspection per the AW139 AMM
- 50-hour continuous airworthiness maintenance program (CAMP) cards per AW139 AMM
- 300-hour/1-year CAMP cards
- 600-hour CAMP cards
- Periodic Engine 1 Inspection/600-hour CAMP cards
- Periodic Engine 2 Inspection/600-hour CAMP cards
- 900-hour/1-year CAMP cards per AW139 AMM
- 12-month servicing per AW139 AMM
- 50-hour main and tail rotor track and balance

Additionally, work order No. 2320 listed several alert service bulletins (ASB), service bulletins (SB), and airworthiness directives (AD) with which were complied or checked for compliance. Lastly, additional maintenance actions not associated with the aforementioned inspections were performed.

Attachment 1 of this report contains the helicopter logbook entries for the last maintenance activity that was completed and signed on June 28, 2019.

The accident helicopter flight logs, provided to the NTSB by the operator, contained a record of flights, including flight time, engine cycles, landing, and a description of the flight. At the end of each calendar month, a summary that included ACTT, engine cycles, and landings was written on the bottom of the last flight log of that month. The flight log for May 2019 reflected, as of May 31, 2019, an ACTT of 2,154.1 hours. The flight log for June 2019 reflected, as of June 30, 2019, an ACTT of 2,155.9 hours. A flight log for July 2019 was not available.

Attachment 2 of this report contains the helicopter flight logs for May and June of 2019.

Chihoon Shin
Aerospace Engineer – Helicopters