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

September 10, 2009

Airworthiness Group Chairman's Factual Report

NTSB ID No.: LAX08PA259

A. ACCIDENT:

Location:	Weaverville, California
Date:	August 05, 2008
Time:	About 7:41 PM Pacific Daylight Time (PDT)
Aircraft:	Sikorsky S-61N Helicopter

B. AIRWORTHINESS GROUP:

Group Chairman: Mike Hauf
Aircraft Systems Engineer
National Transportation Safety Board

Member: Levi Phillips
Carson Helicopter Services, Inc
Grants Pass, OR

Member: Chris Lowenstein
Sikorsky Aircraft Corporation
Stratford, CT

Member: Jim Morrison
United States Department of Agriculture
Forest Service
Ogden, UT

Member: David Gridley
GE - Aviation
Lynn, MA

C. SUMMARY

On August 5, 2008, about 1941 Pacific daylight time,[1] a Sikorsky S-61N helicopter, N612AZ, impacted trees and terrain during the initial climb after takeoff from Helispot 44, located at an elevation of about 6,000 feet in mountainous terrain near Weaverville, California. The airline transport pilot, the safety crewmember and seven firefighters were killed; the commercial copilot and three firefighters were seriously injured.[2] Impact forces and a post crash fire destroyed the helicopter. The helicopter was being operated by the United States Forest Service (USFS) as a public flight to transport the firefighters from Helispot 44 to another location. The helicopter was registered to Carson Helicopters, Inc. (CHI) of Grants Pass, Oregon, and leased to Carson Helicopter Services, Inc. (CHSI) of Grants Pass. The USFS had contracted with CHI for the services of the helicopter.[3] Visual meteorological conditions prevailed at the time of the accident, and a company visual flight rules flight plan had been filed.

The Airworthiness Group convened at the accident site, from August 7 - 11, 2008, to commence examination and documentation of the helicopter wreckage. Participants in the Airworthiness Group consisted of representatives from the Federal Aviation Administration (FAA), General Electric Aviation Engines (GE), Sikorsky Aircraft, United States Forest Service (USFS), and CHSI.

D. DETAILS OF THE INVESTIGATION:

D.1 S-61N Helicopter - Description:

The S-61N helicopter was of all metal, semi-monocoque construction. At the time of the accident, N612AZ was equipped with a 5-blade main rotor system (with a rotor disc diameter of 62 feet), a 5-blade tail rotor system, S-61L style landing gear, and two 1500-horsepower General Electric CT58-140 turboshaft engines. The overall length of the aircraft was 72 feet 10 inches and its height was 18 feet 10 inches.

Helicopter N612AZ was equipped with an onboard SkyConnect Tracker Automated Flight Following (AFF) system, which is an aircraft tracking system that transmits aircraft position, speed, and direction to select ground stations every two minutes. The NTSB obtained data from the SkyConnect (AFF) system showing information for August 5, 2008 starting at 14:39:08 (Table 1).

As of August 4, 2008, N612AZ had accumulated 35,396.4 flight hours. Based on data from the Cockpit Voice Recorder (CVR) and SkyConnect, N612AZ accumulated about an additional 3.5 hours of operating time from August 5, 2008 up to the accident time for a total of about 35,399.9 hours.

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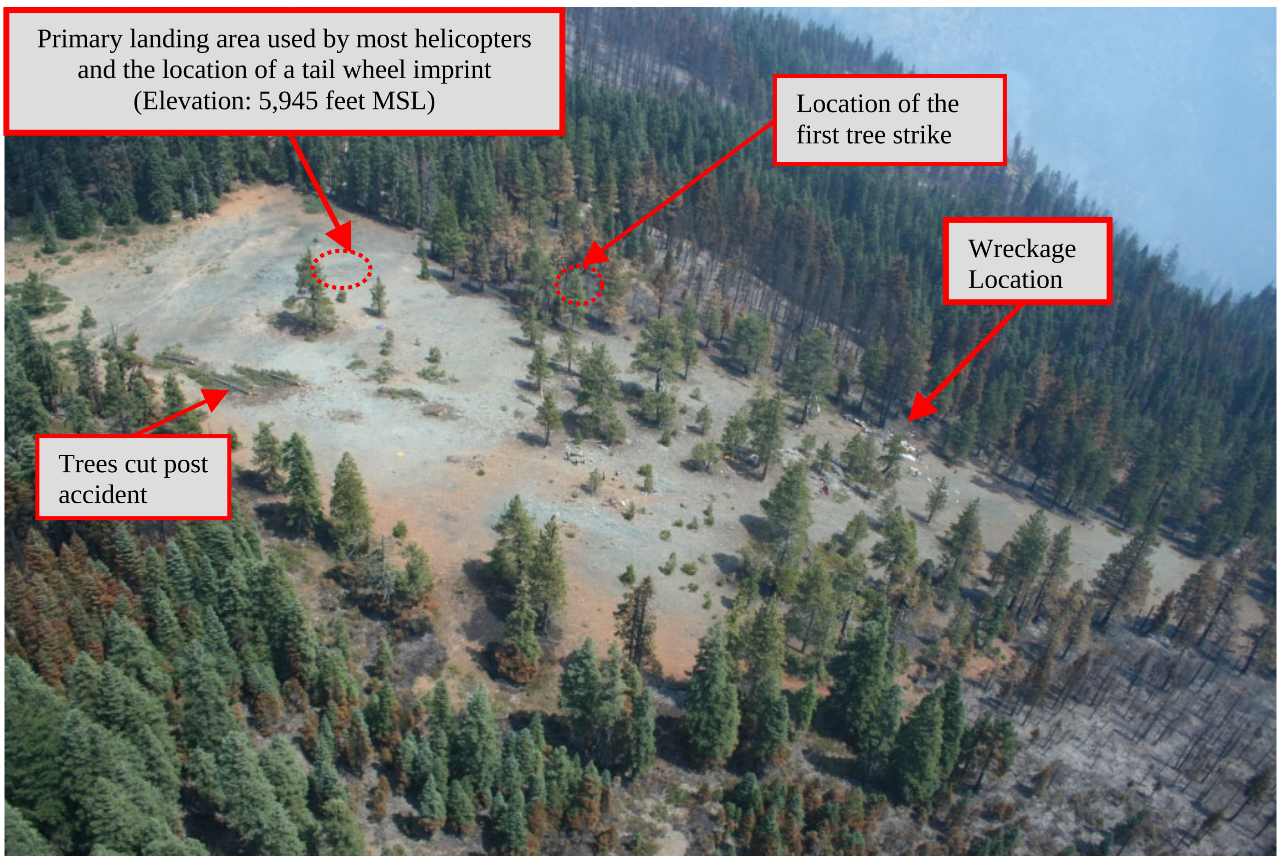
1 All times in this report are expressed in terms of a 24-hour clock and Pacific daylight time unless otherwise noted.
2 The safety crewmember was a USFS Inspector Pilot.
3 Initially, the NTSB was informed that the contract was between the USFS and CHSI. For further information refer to the Operations Factual Report.

D.2 Helispot 44 (H-44) Observations:

H-44 is located in a natural opening along a ridge in a wide saddle that is aligned west-southwest to east-northeast (Figure 1); it is located about 20 miles east of Willow Creek Helibase, 25 miles west northwest of Trinity Helibase, and 52 miles west northwest of Redding, California. The elevation at the reported landing pad at H-44 was surveyed and found to be 5,945 feet above mean sea level (MSL)[4]. The overall open area on the ridge at the time of the accident was approximately 400 ft. wide and 250 ft. long with numerous isolated trees dotting portions of the ridge. Because of those trees, the usable landing and hover area was limited to an area approximately 200 ft. wide by 250 ft. long. The surface is generally rocky and uneven thus allowing for limited of options for landing within the clear area. The natural opening slopes off gradually and steepens to the south-southeast. Due to the lack of natural vegetation and exposed soil, dusty conditions prevailed especially at the landing area adjacent to the foot trail.

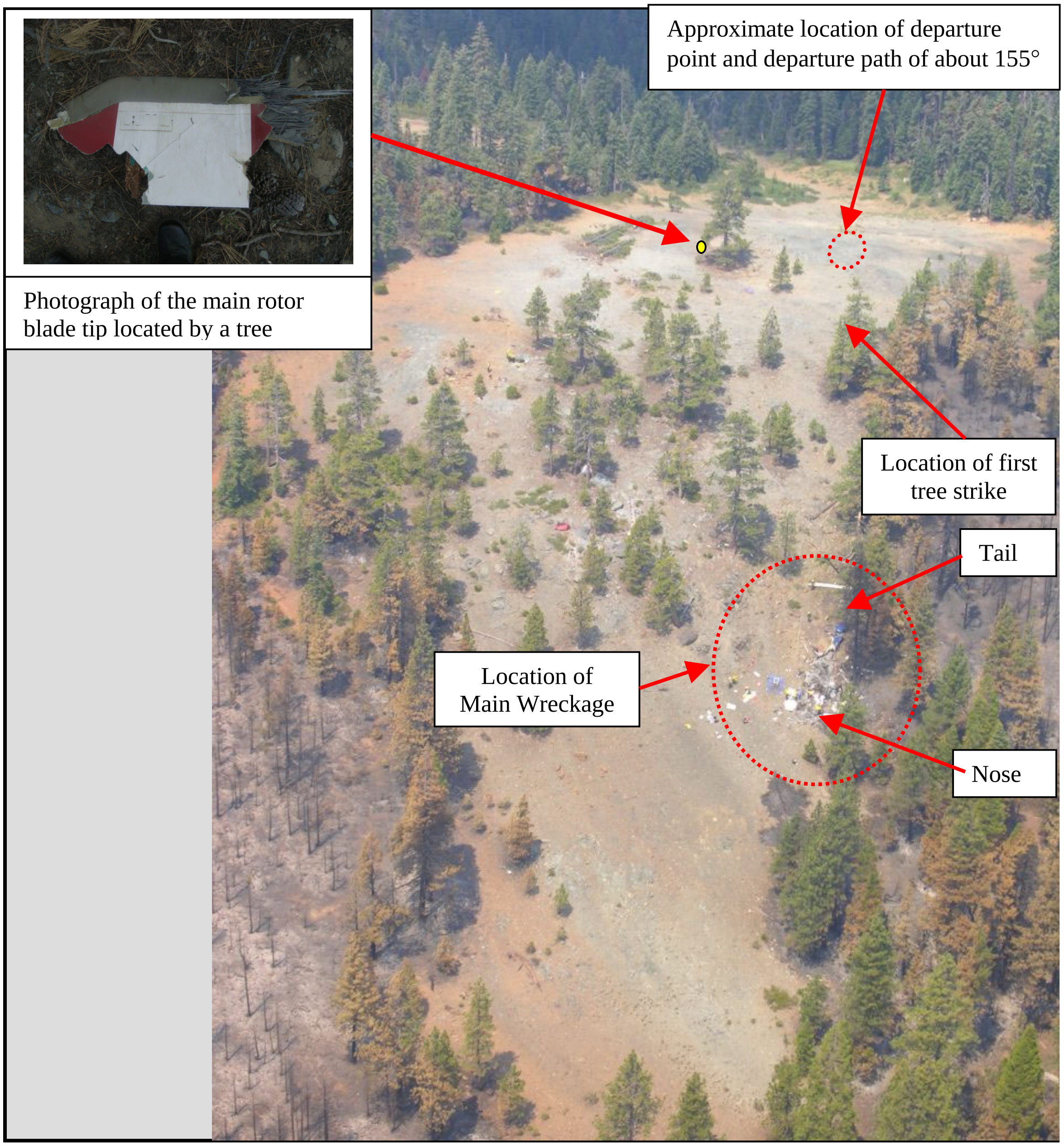
Figure 1View of H-44 showing the normal landing area for helicopters



[4] On Friday August 8 2008, a survey of the accident site was conducted by SHARRA DUNLAP SAWER, Inc.; their office is located in Redding CA, 96002. The survey data includes GPS positions and elevations of the accident site. The survey data was provided to the investigation team on Monday August 11 2008.

Information from the SkyConnect Tracker AFF system indicates that on August 5' 2008 at 19:39:17 local (about two minutes prior to the accident) the helicopter was located at a GPS position of Longitude 40.9142, Latitude 123.2522 traveling at a ground speed of zero knots on a heading of 138 degrees at an altitude of 5,955 feet mean sea level (MSL). It is not known if the helicopter was fully positioned on the ground when the SkyConnect Tracker AFF system transmitted the data. Investigation of this location revealed marks in the rocky soil that were consistent in size to the helicopter's tail wheel and also consistent with a witness statement of the helicopter's position just prior to takeoff. The ground elevation at this location was determined by the surveyor to be about 5,945 feet (Figure 2).

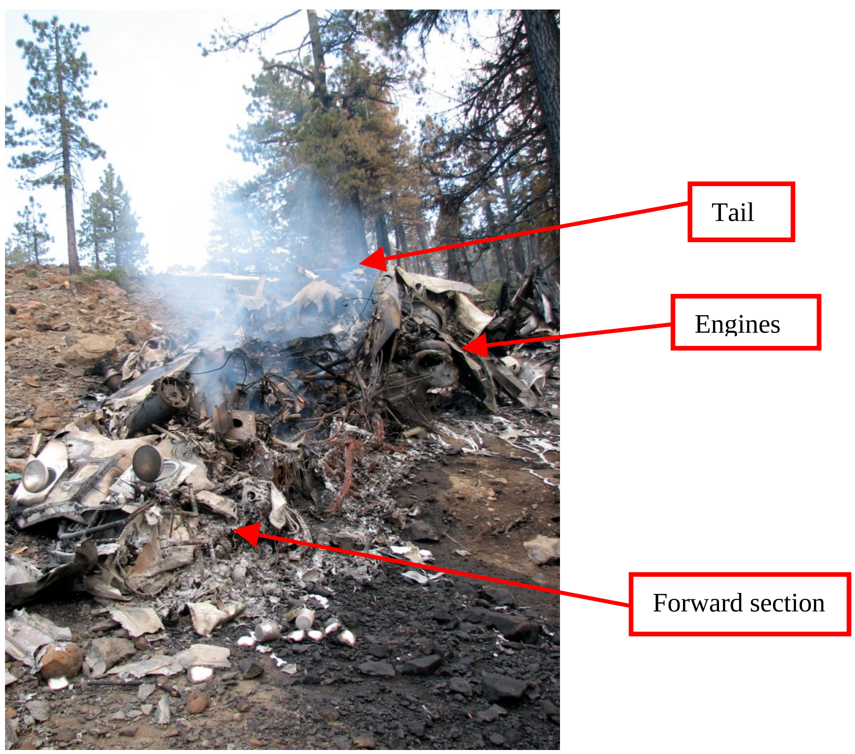
Figure 2 Photograph of Helispot 44 showing departure location and the main wreckage



The location of the main wreckage was surveyed[5] and found to be located about 150 yards south-southeast of the helicopter's reported departure point at an approximate GPS position of Longitude 40.91302, Latitude 123.25163. The last data transmitted by the SkyConnect Tracker AFF system installed on N612AZ occurred on August 5, 2008 at 19:41:25 local and located the helicopter at a GPS position of Longitude 40.913, Latitude 123.2517, which is consistent with the location of the main wreckage as determined by the surveyor.

The helicopter was found resting on its left side with its nose lower than its tail boom. A post-crash fire had consumed most of the helicopter's airframe and structure forward of station 640. Resolidified molten metal was noted in several areas of the wreckage and in the impact area. The magnetic heading of the wreckage was about 155°. The helicopter came to rest in an area that was estimated to have about a 15 to 20 degree slope. Sections of the helicopter's lower fuselage structure including all fuel tank cells and the cabin flooring were found consumed by the post-crash fire. Most of the forward fuselage including the cockpit and the electronics compartment were found inverted and consumed by fire (Figure 3).

Figure 3 Photograph showing the forward fuselage section of the helicopter.



[5] On Friday August 8, 2008, a survey of the accident site was conducted by SHARRA DUNLAP SAWER, inc.; their office is located in Redding CA, 96002. The survey data includes GPS positions and elevations of the accident site. The survey data was provided to the investigation team on Monday August 11, 2008.

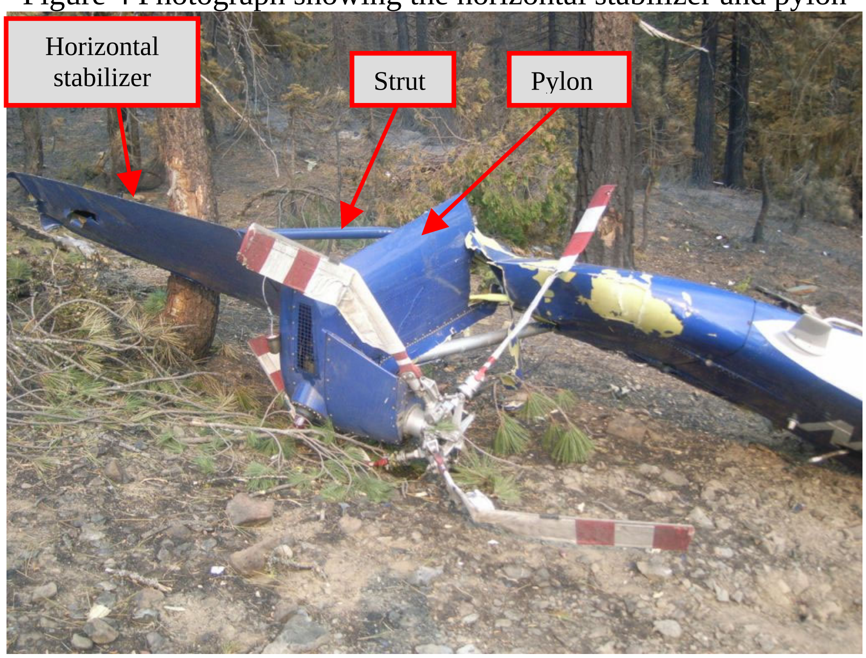
An assessment of the surrounding trees revealed evidence of a main rotor blade (MRB) contacting (striking) a pine tree that was located approximately 65 yards south-southeast (GPS position Longitude 40.91362, Latitude 123.25198) from the helicopter's reported departure area. To inspect the blade contact point on the tree, the Airworthiness Group had a feller from the Forest Service cut down the pine tree. Using a tape measure, the diameter of the tree, at the blade contact point, was measured and found to be about 3.5 inches. Prior to cutting the tree down, the height from the base of the tree to the blade contact point was measured by the surveyor and found to be about 49 feet, 5 inches. The ground elevation at the base of the tree was measured by the surveyor and found to be 5,931.6 feet MSL. An assessment of the blade contact section of the tree provided evidence that a tree strike occurred from the retreating side[6] (left side of aircraft) tipcap of an MRB. This was confirmed by evidence of red (tip end) paint transfer and green composite material (an internal blade component) found embedded in the tree.

D.3 Helicopter Investigation:

D.3.1 Horizontal Stabilizer:

An on-scene assessment revealed that the horizontal stabilizer and its strut remained attached to the vertical pylon via their attachment hardware. The aft section of the pylon was found mostly separated from its mating section of the pylon just forward of the strut attachment (Figure 4).

Figure 4 Photograph showing the horizontal stabilizer and pylon



[6] The main rotor blade rotates counterclockwise as looking down at the rotor system.

D.3.2 Rotary Wing and Tail Rotor Blades Flight Control System:

Investigation of the flight control system was conducted at the accident site from August 7 through 11 2008. The following sections describe the flight control system and the results of the investigations findings.

The flight control system provides a means of controlling rotary wing and tail rotor blades for horizontal, vertical, and directional flight, and for taxiing during ground operations. The main rotor flight controls consist of a collective pitch lever and a cyclic pitch stick for each pilot. The collective pitch lever simultaneously changes the pitch on all rotor blades. The cyclic stick, used for longitudinal and lateral control, varies the pitch individually as a blade rotates. Control motion from the collective pitch lever and cyclic stick are combined in a mixing unit, located in the compartment aft of the pilot's seat, and are transmitted to the rotor head assembly by mechanical linkage through a swashplate. Control action is power boosted by two hydraulically operated flight control servo systems. Normally, control is achieved by use of both servo systems, though either system can control the helicopter separately.

Movements of the flight controls are transmitted mechanically through control rods and bellcranks to the auxiliary servo cylinders, which serves as a hydraulic boost, mechanically transferring the control movements to the mixing unit. The mixing unit consists of bellcranks and connecting links, which proportion and transmit the control movements individually or simultaneously by means of push-pull rods to the three primary servo cylinders and by push-pull rods and cables to the tail rotor. The primary servo cylinders, on the main gearbox, are hydraulically powered cylinders that transfer control movements to the stationary swashplate. Movements of the stationary swashplate are transferred into movements of the rotating swashplate, which alters the pitch on the rotor blades. The tail rotor control movements affect the helicopter’s movement about the yaw axis through changes in the pitch of the tail rotor blades.

Remnants of the flight control system components (pedals, quadrants, linkages collective sticks, and cyclic sticks, control rods, avionics equipment, and throttle quadrant) were located within the main wreckage in the area of the nose section and cockpit. Examination revealed that the integrity of the system had been compromised because the post-crash fire had consumed the aluminum portions of the controls. Therefore, flight control system continuity could not be established. Examination revealed that all attachment hardware (bolts, nuts, cotter pins) of the remaining flight control components appeared in place. Both tail rotor pedal adjusting assemblies were recovered and appeared to be intact. Both cyclic stick mounts were found as well as both tail rotor pedals.

D.3.3 Main Rotor:

Investigation of the main rotor was conducted at the accident site from August 7 through 11 2008. The main rotor was re-inspected by the Airworthiness Group at Plain Parts, Inc. located in Pleasant Grove, California on October 28, 2008. The main rotor consists of the rotary wing head, main rotor blades (MRB), and main gearbox.

The rotor head assembly, mounted directly above the main gearbox, consists of a hub assembly and a control swashplate assembly. The MRBs are attached to the sleeve-spindle assemblies and can individually move vertically [flap-droop], horizontally [lead-lag], or rotate [feather] on their span-wise axis. Droop stops limit the downward position of the blades and are in operation when the blades are stopped or turning at low speed. The droop stops release at approximately 75 percent main rotor speed. Hydraulic dampers dampen hunting [lead-lag] movement of the blades about the vertical hinges. The aircraft was equipped with a bifilar vibration absorber installed on the rotor head, which reduces fuselage vibration and rotor induced airframe structural loads. The rotary wing revolves and creates an airflow past the MRBs, which is used to obtain lift. When collective pitch is applied, the pitch of each blade is increased simultaneously. When lift exceeds the force of gravity, the helicopter will rise vertically. As cyclic pitch is applied, individual blade pitch increases and decreases in each revolution of rotary wing but the average pitch remains unchanged. The increasing and decreasing pitch on individual blades causes the rotor disc to tilt in the direction of stick movement. The highest flap angle will be 180 degrees opposite the stick movement, with the lowest being in the direction of stick movement. The helicopter moves in the direction the cyclic stick is moved as the result of a lower flap angle in that direction. The main gearbox reduces engine rpm, drives the main gearbox accessories, and conveys torque from the engines to the rotary wing and rotary rudder.

Each MRB has the following information stenciled on the bottom side of the blade: blade serial number, date of manufacture, and blade drawing number. Each installed blade has its cuff color-coded red, blue, yellow, white, and black to match the color-coding on the sleeves of the rotary wing hub. The installed blades are also identified numerically to correspond to the numbering of the sleeves. The Number 1 sleeve, color-coded red, is at the rotating scissors. The other sleeves and blades are identified by consecutive numbering in the direction of rotation (counter-clockwise).

D.3.3.1 Main Rotor Head (MRH):

Examination of the MRH revealed that the hub assembly, with its sleeve spindles, was intact and remained connected to the rotor shaft. The appearance of the MRBs and MRH are consistent with a significantly reduced N_R. All MRH components had evidence consistent with the exposure to fire and heat.

(a) Spindles:
All five spindles remained attached to the MRH and were found thermally damaged. Except for the Black spindle, the spindles could not be moved by hand. The Black spindle could only be moved in the flapping direction.

(b) Dampers:
All five dampers were located within the main wreckage. The Red damper was found separated from its sleeve & spindle assembly and the horizontal hinge pin. The Blue,

Yellow, and Black dampers were found separated from the horizontal hinge pin. The White damper had fractured at the piston rod itself.

(c) Pitch Change Rods (PCR):
The Yellow PCR was intact but found separated from the rotorhead. The Yellow PCR pitch change horn attachment fittings were found fractured from the horn, and both lower lugs were found fractured from the rotating swashplate. The White PCR was intact and remained attached to the rotating swashplate. The Red PCR had fractured from the pitch change horn fitting and was not located. The Blue PCR was found in the unburned initial ground impact area. The Black PCR was intact.

(d) Pitch Change Horns (PCH):
The Red, Blue, and Black PCH were found intact. The Yellow PCH was completely consumed by the post-crash fire. The White PCH was found separated from the White sleeve and spindle, but remained attached to the PCR, which remained attached to the rotating swashplate.

(e) Droop/Flap Stops:
The Red, White, and Black droop stops appeared to be in the retracted (ground) position. The Blue droop stop was missing, and the Yellow droop stop itself had separated; it was located on the sleeve in a clump of resolidified aluminum. The Red, Blue, and Yellow flap stops appeared to be in the retracted (ground) position. The White and Black flap stops appeared to be in the extended (flight) position. There was damage to several of the flap stop leading edge arms, and the lead-side flap stop return springs were missing from Red, White, and Black.

(f) Bifilars:
One bifilar weight (from between the Yellow and White arms) was found separated from the head and had impacted the ground in the central initial ground impact area. Its arm was found damaged in a downward direction and opposite main rotor rotation. The bifilar arm between Blue and Yellow was also bent downward, but did not separate. Several bifilar fairings were located just below the final wreckage area.

(g) Main Rotor Servos:
All three main rotor servos were present and remained in their respective positions. All three lower connections remained connected to the MGB magnesium housing; their mounting hardware was intact, however, the MGB housing had been completely consumed by fire. Two of the upper connections to the stationary swashplate remained connected. The ‘right lateral’ servo remained connected and intact with all hardware attached. The ‘fore and aft’ servo remained connected and intact with all hardware attached. The ‘left lateral’ servo was found disconnected from the swashplate, because its mounting lug on the stationary swashplate had melted away.

D.3.3.2 Main Rotor Blades (MRB):

At the time of the accident, N612AZ was equipped with Carson Helicopter Inc-designed, all composite (glass and carbon fiber-reinforced plastic) construction (except for a nickel-alloy leading edge abrasion strip and steel cuff) MRBs. The MRBs were manufactured by DuCommun AeroStructures Inc. Carson states on their website (http://www.carsonhelicopters.com) that these blades produce a 2000 lbs increase in lift. The blade length is 28 feet and 9 inches. The installation of the composite blades on to S-61N helicopters was approved via Supplementary Type Certificate (STC) number SR01585NY.

Fragmented sections of all five MRBs were located and recovered from the immediate area around the main wreckage of the helicopter. A visual assessment of fractured sections of the main rotor blades revealed that the resultant damage was consistent with multiple leading edge impacts with trees.

Two MRBs (Black and White) were found separated from their steel cuff[7] bolted connections. One three foot tip end section of a main rotor blade was located on the ground next to a tree about 100' southwest of the initial takeoff point. The following paragraphs provide a description of the blade tip and two main rotor blades:

1. Section of Blade found next to a Tree:
   The "as found" three foot tip end section of the blade was missing the outboard ~six inches of its outboard tipcap. At its inboard portion, the spar was found fractured in a broomstraw fashion. Two pieces of blade skin and two fragments of leading edge were positively matched to this section. All four pieces matched blade sections that were found in the immediate area of the first observed tree strike.

2. Black Blade:

   The Black blade, S/N CHI-0140, and was found fractured near its mid-span and draped over a tree branch approximately 75 feet from the wreckage. The outboard section (approximately 6 ft) was heavily damaged, with only spar remnants remaining. The blade was found separated from its cuff assembly; nine of the blade's ten retention bolts (on both the upper and lower airfoil surfaces) were found sheared. The outboard lag-side bolt and nut remained intact with a remnant of the cuff under the bolt head. The leading edge sheath was missing from blade station[8] (BS) 73 (start of sheath) to the fracture at BS 102. A six foot section of blade spar only was matched to the Black blade, and was located about 100' west north west of the aircraft. This section was found entangled in a purple mylar-type film that was tentatively identified as a mold-release film from the central spar cavity. It is reported by the blade manufacturer, Ducommun, that this film is used during the spar build process and is not an interlaminar film, but is used to prevent the internal spar inflatable bladder from

[7] The cuff permits attaching the blades to the rotary wing head sleeve-spindles.
[8] Blade Station (BS) is a measurement in inches from the cuff of the blade towards the tip.

adhering to the spar during the autoclaving process. A fragment of leading edge was positively matched to this blade. The matched piece was located just below the tree in which the blade came to rest.

3. White Blade:
   The White blade (from cuff to tipcap), S/N CHI-0133 was found lying on the ground just behind and uphill of the helicopter wreckage. The blade was found separated from its cuff assembly; eight of the blade’s ten retention bolts (on both the upper and lower airfoil surfaces) were found sheared. The outboard lead bolt head was intact, but the bolt was sheared at the lower airfoil surface. The White blade also exhibited leading edge impact at BS 256-270. The leading edge abrasion strip was found separated in this region; the strip was found nearby and matched. There was pocket damage and a section of skin missing in this section. The trailing edge in this section was found ‘popped’ (a term used to describe the trailing edge bond separation that occurs with rapid deceleration, as occurs when striking a fixed object). The outboard six inches of tipcap were not recovered, however, a section of the most outboard tipcap leading edge abrasion strip was found and matched.

   Blue paint transfer, (consistent with paint on the horizontal stabilizer), was noted between BS 118 and 142. Surface contamination (black ash or soot) inboard of about BS 90 was noted on both the upper and lower airfoil surfaces, more heavily along the leading edge (Figure 5). The contamination was sandpaper-rough in appearance, and there was evidence of some streaming in a combined chordwise and radial outward direction. The contamination could be removed by fingertip pressure and was not apparent on any of the other blades. For example, the Black blade inboard section had a dirty film appearance, but was not raised or rough in appearance as was the contamination on the White blade. The Black blade ‘dirty film’ could also be removed by fingertip pressure. During recovery, the White blade was saw cut at BS 66, 118, 190, and 264.

Figure 5
Photograph of the White main rotor blade showing evidence of contamination.

4. <u>Remaining Blades:</u>

Several sections of root end blades were located immediately adjacent to the wreckage.

- A 16-foot section of the Blue blade, S/N CHI-0018, was located just east of the tail cone. Its outboard end was found thermally damaged, and the inboard end was found heavily damaged (only spar remaining).
- A 9.5-foot section of an unknown blade was located just east of the tail cone. The blade was found fractured at the CG stripe, and its other end was found thermally damaged.
- A 14-foot section of an unknown blade was located next to the helicopter. The blade leading edge was found intact, but the remaining section of the blade was found thermally damaged.
- A 16-foot section of an unknown blade at the MGB was found with its leading edge intact, but the remaining section of the blade was found thermally damaged.

D.3.3.3 Main Gearbox:

The main gearbox (MGB), mounted above the cabin aft of the engines, changes the angle of drive from the engines to the main rotor, reduces engine rpm, and provides the means for driving the tail rotor and gearbox accessories. The MGB consists of a magnesium alloy housing that contains gears and shafts made from steel. The MGB is built so that the main rotor shaft is inclined forward 4°. The MGB supports and drives the main rotor and provides power to the tail take-off to drive the tail rotor and the gearbox accessories. The following accessories are mounted to the rear of the MGB: primary and secondary hydraulic pumps, N_R tach generator, torque-meter and MGB oil pumps, two AC generators, a DC motor-generator. An accessory drive through-shaft is installed, which turns only when the Number 1 engine is operating and the rotors are turning. During normal operation the through-shaft merely provides a redundant power path for driving the accessories, which are normally driven by the tail take-off through a free-wheel unit. Two (left and right) input freewheel units (IFWU) are also installed on the MGB.

A review of the maintenance records provided by CHSI revealed that on July 21, 2008, the main gearbox part number S6135-20600-046 having serial number 1101, was removed from the helicopter and replaced with main gearbox part number S6135-20600-046 having serial number A14-B-22-77-1057 per the Sikorsky Aircraft S-61N maintenance manual (Reference NTSB Maintenance Record's Group Chairman's Factual Report in the Safety Board's public docket for accident, LAX08PA259).

An assessment of the MGB at the accident site revealed that the post-crash fire had consumed the magnesium section of the main gearbox; only the steel components, such as gears, shafts and bearings remained. There was no evidence of any loss of drive.

D.3.4 Input Freewheel Units (IFWU):

The MGB in the Sikorsky S-61N incorporates a left and right IFWU on the input section to permit disengagement of the rotary wing when the transmission rpm exceeds the engine input rpm during autorotation or engine start-up/shutdown, and during single engine operation. The IFWU is basically a one-way clutch mechanism that allows the engine to drive the rotor, but prevents the rotor from driving the engine. The IFWU comprises three notable elements: a gear housing, a camshaft, and 12 rollers. In the event that an IFWU slips, the associated engine would be rapidly unloaded and may encounter an overspeed condition and automatically shut down.

The Carson IFWU Supplemental Type Certificate (STC) number SR02057NY, dated February 10, 2005 utilizes the HH-3E/F (23000 series) IFWU, which was a part of a military MGB developed by Sikorsky under contract to the United States Air Force (USAF). The significant difference in the original IFWU and the one introduced in the STC is the tapered roller bearings that replace the ball type bearings, which increased the "footprint" of the bearings supporting the gear housing from 0.75" to 1.5" (approximately). This provides for increased support of the gear housing, which reduces wear on the camshafts, rollers, and gear housing roller bore and roller retainer. Adding a second oil jet to each IFWU increased oil flow to the associated bearings and rollers.

According to the NTSB Maintenance Record's Group Chairman's Factual Report, the accident IFWUs were new when they were installed in the MGB having serial number A14-B-22-77-1057. IFWU having S/N G042-01023 was installed on the left side of the gearbox and IFWU having S/N B042-00031 was installed on the right side of the gearbox. At the time of the accident each of the IFWUs had accumulated approximately 26 hours total time in service[9].

The left and right IFWUs were located within the main wreckage and remained attached to the MGB. Visual inspection revealed that the units were coated with a white material and showed surface oxidation consistent with exposure to fire and heat. Both IFWUs were recovered from the helicopter wreckage, placed in shipping containers and shipped, along with the engines, to the Columbia Helicopter Inc., facility located in Aurora, Oregon. As directed by the NTSB, representatives from Columbia re-packaged

[9] Reference NTSB Maintenance Record's Group Chairman's Factual Report in the Safety Board's public docket for accident, LAX08PA259.

both IFWUs and then shipped them to the Helicopter Support, Inc facility located in Trumbull, Connecticut for examination.

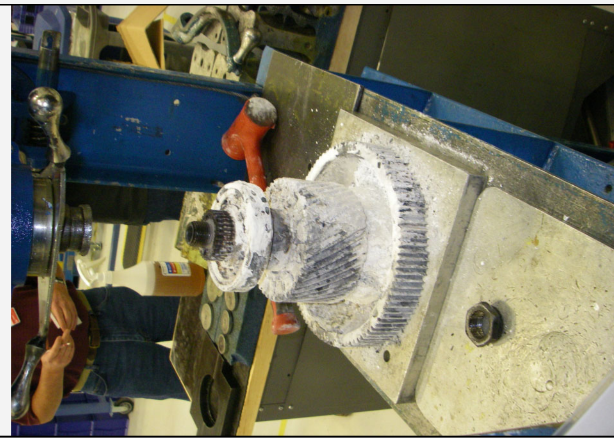
Investigation of the left and right IFWUs was conducted at the Helicopter Support, Inc. (HSI) facility located in Trumbull, Connecticut and at the Sikorsky Aircraft Corporation located in Stratford, Connecticut on September 3 and 4, 2008. The investigation of the IFWUs was conducted under the supervision of the NTSB and witnessed by representatives from General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service, and CHSI.

D.3.4.1 Number 1 (Left) IFWU Examination:

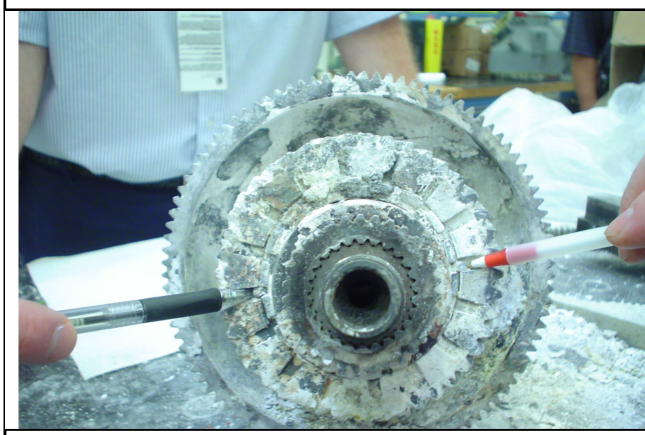
Prior to disassembly, visual examination revealed that the IFWU was heavily coated in a material consistent with magnesium oxide (Figure 6). A representative from HSI removed the shaft from the input spline and found the gear housing nut-locking tab in place and secured. After removing the camshaft aft lock nut, the camshaft aft roller bearing was examined and found to rotate smoothly, with no visible scoring. An attempt to remove the gear housing from the camshaft was unsuccessful. The Airworthiness Group determined that to inspect the internal mechanisms of the IFWU, the unit should be cut in half. The IFWU was re-packaged and hand carried to the Sikorsky Aircraft Corporation located in Stratford, Connecticut. Utilizing a band saw at the Sikorsky Aircraft Corporation fatigue lab machine shop, the IFWU was sectioned into two pieces along its axial centerline.

Figure 6
Photographs of the Number 1 IFWU

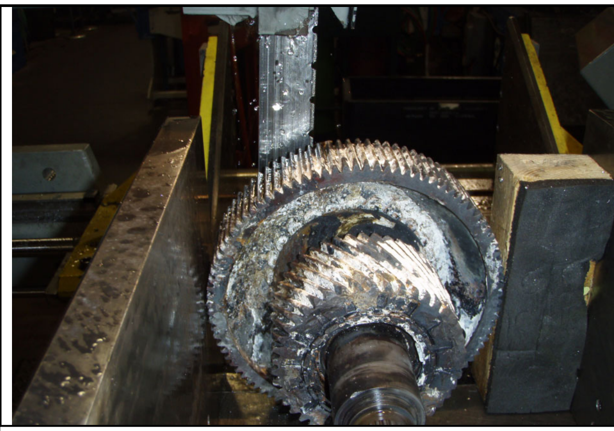
Photograph of Number 1 IFWU as received at HSI



Photograph of Number 1 IFWU in test fixture with shaft removed



Photograph of Number 1 IFWU showing the gear housing nut locking tab in place securing the gear-



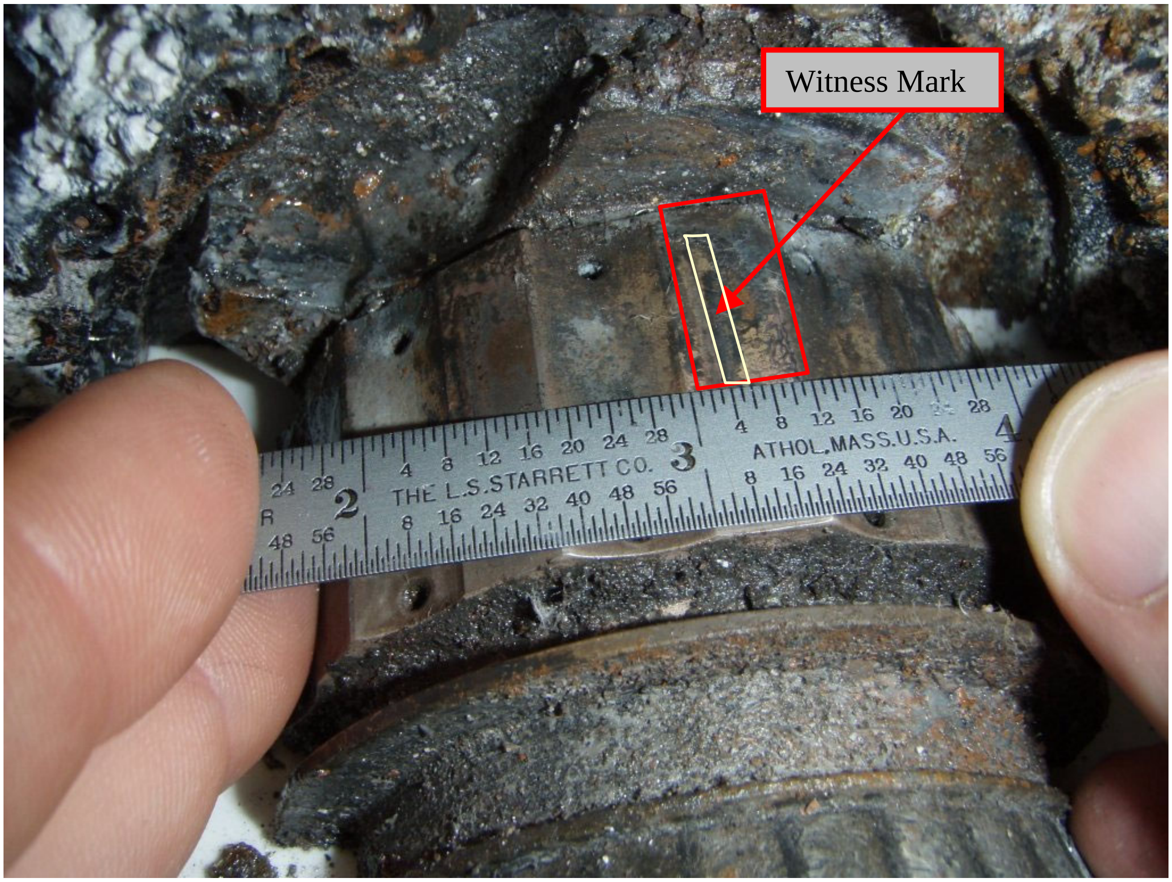
Photograph of Number 1 IFWU showing it installed in the band saw

Photograph of Number 1 IFWU after being cut in half

Photograph of Number 1 IFWU roller bearings

The group inspected the unit's camshaft flats for witness marks and noted marks approximately 0.050" wide at approximately 0.125" to 0.187" up from the beginning of the ramp (Figure 7). An inspection of the cam flats indicated no visible wear depressions, evidence of skidding, impact marks, contamination or any other indications of improper operation on the cam flat surface or on the gear housing. Visual examination of each of the 12 bearing rollers found no excessive wear, flat spots, contamination or deformations. The bronze Oilite bushings (two each) were found thermally destroyed; however, remnants of the bushings were visible in the exposed channel.

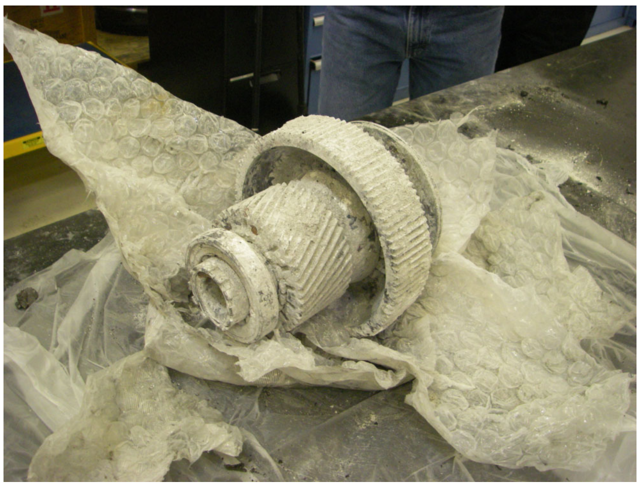
Figure 7 Photograph of the Number 1 IFWU showing the approximate 0.050"witness mark positioned at approximately 0.125" to 0.187" up from beginning of the ramp



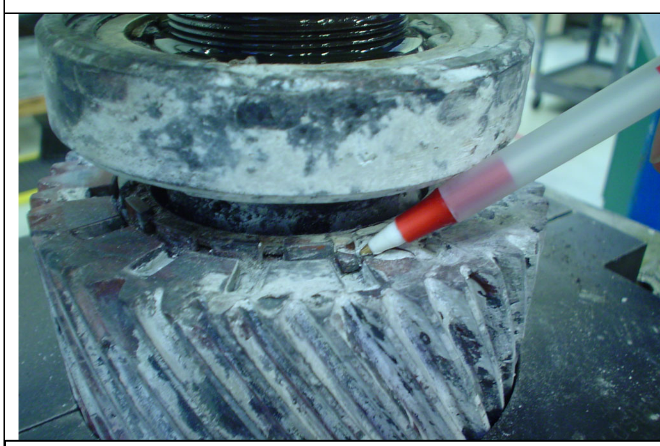
D.3.4.2 Number 2 (Right) IFWU Examination:

Prior to disassembly, visual examination revealed that the IFWU was heavily coated in a material consistent with magnesium oxide (Figure 8). A representative from HSI removed the camshaft aft locknut and found the gear housing nut-locking tab in place and secured. After removing the camshaft lock nut, the camshaft aft roller bearing was examined and found to rotate smoothly, with no visible scoring. The gear housing was separated from the camshaft successfully after about 16 tons of pressure was applied to the shaft. Upon inspection, three sequential rollers were found heat-welded to the camshaft, preventing removal of the roller retainer. The Airworthiness Group determined that to inspect the internal mechanisms of the roller retainer; the retainer should be cut in half. The IFWU was re-packaged and hand carried to the Sikorsky Aircraft Corporation. Utilizing a band saw at the Sikorsky Aircraft Corporation fatigue lab machine shop, the roller retainer was cut into two pieces with two diametrically opposed cuts through the retainer. The less heat-damaged section of the roller retainer was tapped off the camshaft using a brass drift, and the cam flats and rollers were evaluated.

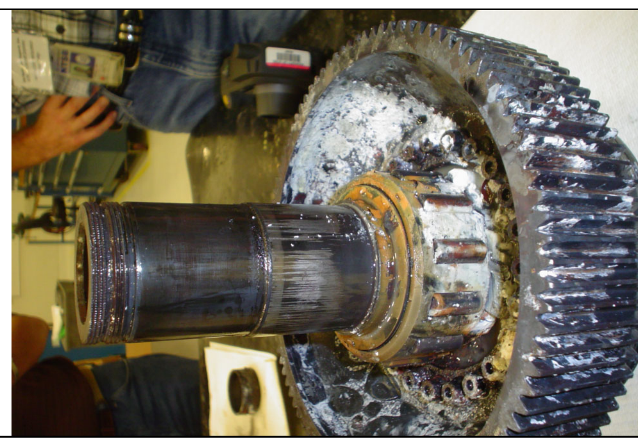
Figure 8
Photographs of the Number 2 IFWU



Photograph of the Number 2 IFWU as received at HSI



Photograph of Number 2 IFWU showing the gear housing nut locking tab in place securing the gear-housing nut.



Photograph of Number 2 IFWU showing the gear housing removed.

The group inspected the cam flats for witness marks and noted marks approximately 0.050” wide at approximately 0.078” to 0.140” up from the beginning of the ramp (Figure 9). An inspection of the cam flats indicated no visible wear depressions, evidence of skidding, impact marks, contamination or any other indications of improper operation on the cam flat surface or on the gear housing. Visual examination of each of the 12 bearing rollers found no excessive wear, flat spots, contamination or deformations. The bronze Oilite bushings (two each) were found thermally destroyed; however remnants of the bushings were visible in the exposed channel.

Figure 9 Photograph of Number 2 IFWU showing the 0.050”witness mark positioned at approximately 0.078 to 0.140” up from beginning of ramp

D.3.5 Tail Rotor System:

Investigation of the tail rotor system was conducted at the accident site from August 7 through 11 2008. The following sections describe the tail rotor system and the results of the investigations findings.

The tail rotor system consists of the tail rotor hub (TRH) assembly and five tail rotor blades (TRB); each TRB is color-coded (Red, Blue, Yellow, White, and Black). The TRH is supported and driven by the horizontal output shaft of the tail gearbox (TGB). The five TRBs are attached to the hub by hinges and spindles. The tail rotor is semi-articulated with a feathering hinge and a flapping hinge.

The TRBs are constructed of a one-piece all-metal wrap-around skin, bonded to a solid leading edge spar and a honeycomb core. In addition to the conventional pedal control the tail rotor system is mechanically coupled to the collective control system through a mixing unit. Thus, the TRB pitch angle is automatically and proportionately varied with changes in the collective pitch without having to move the pedals. This coupling serves to help maintain aircraft heading by varying the tail rotor thrust in proportion to main rotor torque.

1. <u>Tail Rotor Blades:</u>
   Examination of the TRBs revealed that four of the five rotor blades remained attached to the rotary rotor head. A section of White TRB also remained attached to the rotary rotor head, but was found severed at about blade station 9. According to the Sikorsky representative, all TRB blade damage appears to have occurred with minimal (low) rotational energy as evidenced by minor rotational scarring evidence, only flat wise bending, and no damage to the leading edges or the leading edge end cap portions of the tipcap fairings.

   a. <u>Red Blade</u>:
      Inspection of the Red Blade revealed that it was bent flat-wise on its outboard side at blade station[10] (BS) 22. Punctures were found in the blade on its inboard side at BS 35 and 39. The pitch change rod-ends were found bent outboard 30º and inboard 15º. The grease zerk fitting was found sheared from hub.

   b. <u>Blue Blade</u>:
      Inspection of the Blue Blade revealed that it was bent flat-wise on its outboard side at BS 18. A gash was found in the blade on its outboard side at BS 35. The blade’s tipcap was found damaged on its inboard and outboard sides and there was no blade leading edge damage. The pitch horn inboard flange was found detached. The nut and cotter pin were missing from the inboard pitch change attachment bolt. The outboard pitch change link rod-end was found bent approximately 25º. The grease zerk fitting was found sheared from the hub.

   c. <u>Yellow Blade</u>:
      Inspection of the Yellow Blade revealed that it was bent flat-wise at BS 22. The pitch change link and the pitch horn had separated from the hub. The blade tip remained attached to the blade and was found intact.

   d. <u>White Blade</u>:
      The remainder of the White Blade, outboard of BS 9, was located approximately 10 feet aft of the stabilizer, except the tip cap, which was found (with no tip end or leading edge damage) about 25 feet behind the helicopter wreckage. Inspection

[10] The blade station is measured in inches from the tail rotor cuff.

revealed that it was bent flat-wise at BS 22. The aft 6 inches of the trailing edge had separated from the blade tip. The blade exhibited a longitudinal pocket tear from about BS 22 to 35 aft of the spar. The pitch horn attachment flange was found separated from the horn.

e. Unmarked Blade (should be Black):
Inspection of this blade revealed that its blade pocket was deformed the entire length on both sides of the blade. The blade did not have flat-wise bending as noted on the other four blades. A puncture through the blade was found at BS 48.

2. Tail Gear Box:
Inspection of the TGB revealed no external damage and that it remained mounted to the vertical pylon.

3. Tail Rotor Control System:
The tail rotor cable system was visually examined to evaluate control continuity from the tail rotor control forward quadrant to the tail rotor control aft quadrant. Control continuity could not be verified because both forward cable end swaged fittings had separated from the melted forward quadrant. The forward cable end swaged fittings were intact. Both cables remained intact from the forward quadrant up to the forward section of the forward turnbuckles. The forward turnbuckles were located in the wreckage in an area of high thermal damage. Portions of the middle link were missing from each turnbuckle. Cable integrity from the aft section of the forward turnbuckles to the aft quadrant area was confirmed. Both aft turnbuckles were observed connected and safetied.

The tail rotor pitch change shaft remained connected to the gearbox bellcrank and all hardware was found intact. The push tube from the gearbox bellcrank to the negative force gradient (NFG) spring bellcrank, NFG spring and bellcrank hardware and the upper pylon push tube to center idler were found intact and in good condition. The lower pylon push tube was found broken at the clevis shank; the shank was found sheared off at the tube. The lower pylon push tube had several small dents with blue paint about mid length. The push tube from the bottom pylon to aft quadrant remained connected at both ends. The left and right control cables remained connected to the aft quadrant. Continuity was confirmed with a slight pull on cables at the tail cone attach point moving the push tube at the pylon end indicating integrity through the tail cone.

4. Tail Rotor Drive System:
The tail rotor drive system was visually examined to evaluate drive continuity from the MGB tail takeoff to the TGB input. Overall, the examination did not find any pre-impact malfunctions with the tail rotor drive system hardware that would result in the loss or reduction of tail rotor rpm or thrust. The following paragraphs provide the results of the examination.

All five sections of the tail shafting were found within the main wreckage. Most of the hanger bearings were found thermally damaged, but remained attached to the airframe.

The Number 1 Tail Rotor Drive Shaft (TRDS) remained connected to the Main Gearbox (MGB) tail takeoff. The tail takeoff connection was found intact, and the tail takeoff gear was distorted out of plane by heat.

The Number 2 and 3 TRDS remained connected with all hardware intact. Their flexible couplings were distorted in bending because the airframe in this area had been consumed by fire, and the hanger bearing mounts were missing.

The Number 4 TRDS had pulled out of its forward flange. There were no signs of rubbing or distortion of the shaft or the flange. The forward bearing had been pulled out of the hanger bearing mount, and was found heavily burned. The hanger bearing mounting assembly was distorted and cracked; its flange alignment marks remained aligned. One of the hanger bearing bolts was missing where the mount had fractured.

The intermediate gearbox (IGB) was intact, with no signs of leakage; no visible damage, and safety wire remained installed. The wire to the chip detector was found separated. The input flange was intact, with all hardware installed. The flexible coupling was found slightly deformed.

The Number 5 TRDS lower flange remained connected to the IGB output flange, however, due to an airframe fracture in the area, there was about a 45° angle between the IGB output and the centerline of the Number 5 TRDS. One of the three TRDS lobes was found broken. The flexible coupling was heavily distorted in bending. The mounting hardware was intact. The Number 5 TRDS remained connected to the TGB input flange. The mounting hardware was also intact. The chip detector was installed and did not appear to have any indication of oil leaks.

D.3.6 Cockpit:

Investigation of the components located in the cockpit was conducted at the accident site from August 7 through 11 2008. All of the components were found within the main wreckage in the area of the cockpit buried under ash and debris.

D.3.6.1 Circuit Breaker Panel:

Both the overhead switch panel and the circuit breaker panel were found thermally damaged and all switches and all circuit breakers were found compromised.

D.3.6.2 Throttle Quadrant:

Both throttles were found in about the full forward speed position. Emergency throttles were found mismatched with the Number 2 advanced about halfway and the Number 1 was shut-off, however, this position may not be representative as they are friction-detented only.

D.3.6.3 Avionics Components:

All of the avionic system components were found heavily damaged due to the high temperatures experienced during the post-crash fire. Due to the damage, the avionic system components were not recovered for testing.

D.3.6.4 Airspeed Indicator:

One airspeed indicator was located with the wreckage, it remained mostly intact and its airspeed pointer indicated about 50 knots (Figure 10).

D.3.6.5 N_G Indicators:

Both N_G indicators were located; one indicator indicated zero N_G and the pointer was missing from the other indicator. The N_G indicators are DC powered instruments and will indicate zero as soon as power is removed.

D.3.6.6 Main Transmission Oil and Hydraulic Pressure Gauge:

The main transmission oil pressure gauge indicated about 168 psi (35 - 90 psi normal) and the primary hydraulic pressure gauge indicated about 1,100 psi (1,300 to 1,600 psi normal).

D.3.6.7 Engine Torque Indicators:

The engine torque indicating system provides a method of gauging engine loads by measuring the amount of torque applied at each engine input and displaying these amounts in the cockpit. The system consists of an oil pump, a torque-sensing mechanism associated with each main gearbox input freewheel unit, two pressure transmitters, and two dual-pointer indicators (torque meters).

At the time of the accident, helicopter N612AZ was equipped with a total of three dual-engine torque indicators; one externally[11] mounted indicator, and two internally mounted indicators. All three dual-pointer engine torque indicators were recovered from the helicopter wreckage and visually examined at the accident site. Examination revealed each indicator was thermally damaged and that each indicator's dual-pointers remained attached and were indicating the following engine torque values:

1. Cockpit Torque Indicator: (Figure 10)
   a. The Number 1 needle indicated a torque of approximately 62%.
   b. The Number 2 needle indicated a torque of approximately 58%.

2. Cockpit Torque Indicator: (Figure 11)
   a. The Number 1 needle indicated a torque of approximately 62%.
   b. The Number 2 needle indicated a torque of approximately 50%.

[11] The external torquemeter was installed onto the helicopter on August 10, 2007

3. Externally-Mounted Torque Indicator: (Figure 12)
   a. The Number 1 needle indicated a torque of approximately 70%.
   b. The Number 2 needle indicated a torque of approximately 36%.

Electrical power for the engine torque indicating system is normally supplied by the Autotransformers at 26 Volts AC which in turn are supplied by the 115 V AC ESSENTIAL BUS. The Number 1 AC generator powers the AC ESSENTIAL BUS. In the event that the Number 1 AC generator fails, the Number 2 AC generator will automatically supply the AC ESSENTIAL bus, and will shed the non-essential (Number 2 bus) load. A DC to AC inverter, which can be powered by the DC generator or battery, will provide AC power to the engine instruments during engine start and in emergency operations. The DC generator will produce power at rotor speeds above about 32% and is rated to provide 200 amps at 40% N_R (limited to 15 minutes) and will produce 300 amps at >50% N_R. In the event that power is lost, the dual-pointers on the engine torque indicators will remain positioned at the indication displayed when power was lost.

Figure 10 Photograph of an airspeed indicator and a cockpit engine torque indicator at the wreckage site

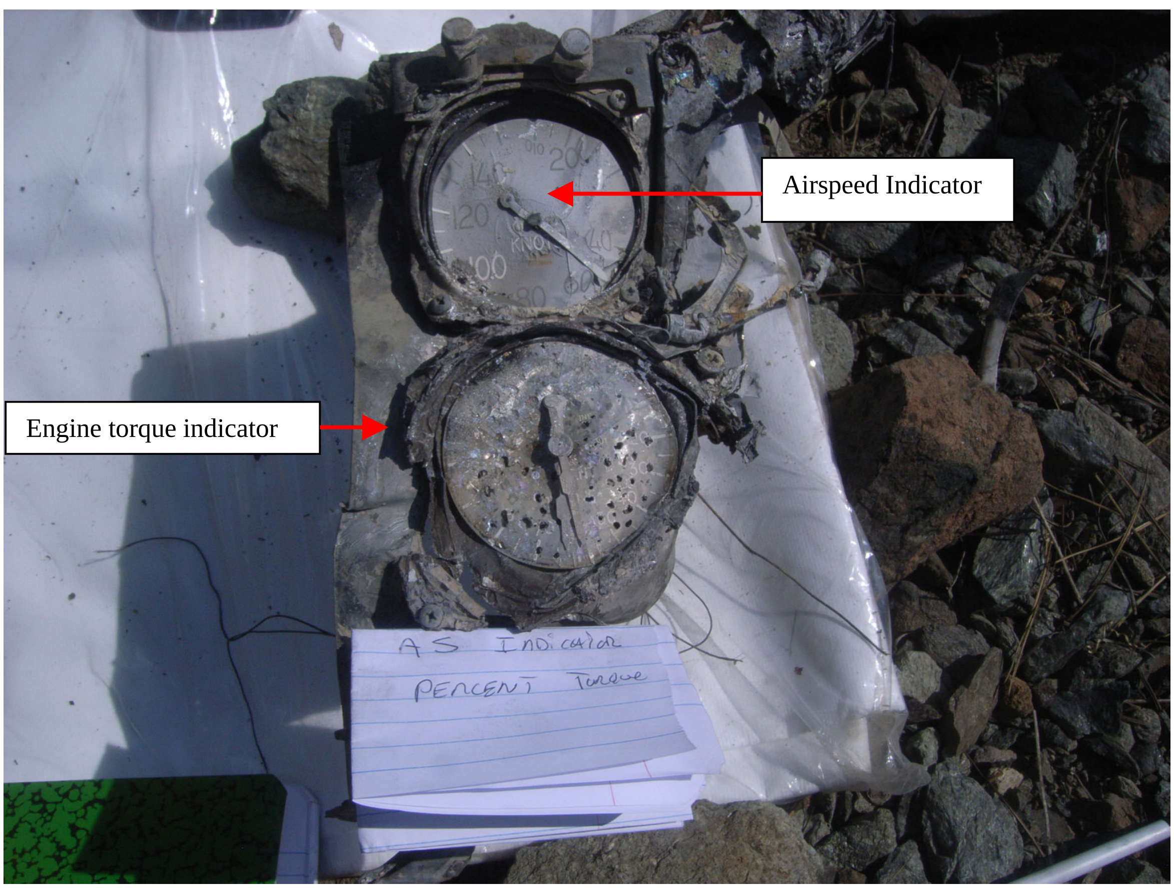


Figure 11 Photograph of a cockpit engine torque indicator at the wreckage site

Figure 12 Photograph of the exterior mounted engine torque indicator at the wreckage site

D.3.6.8 Digital Readout T_5 Indicator (Howell):

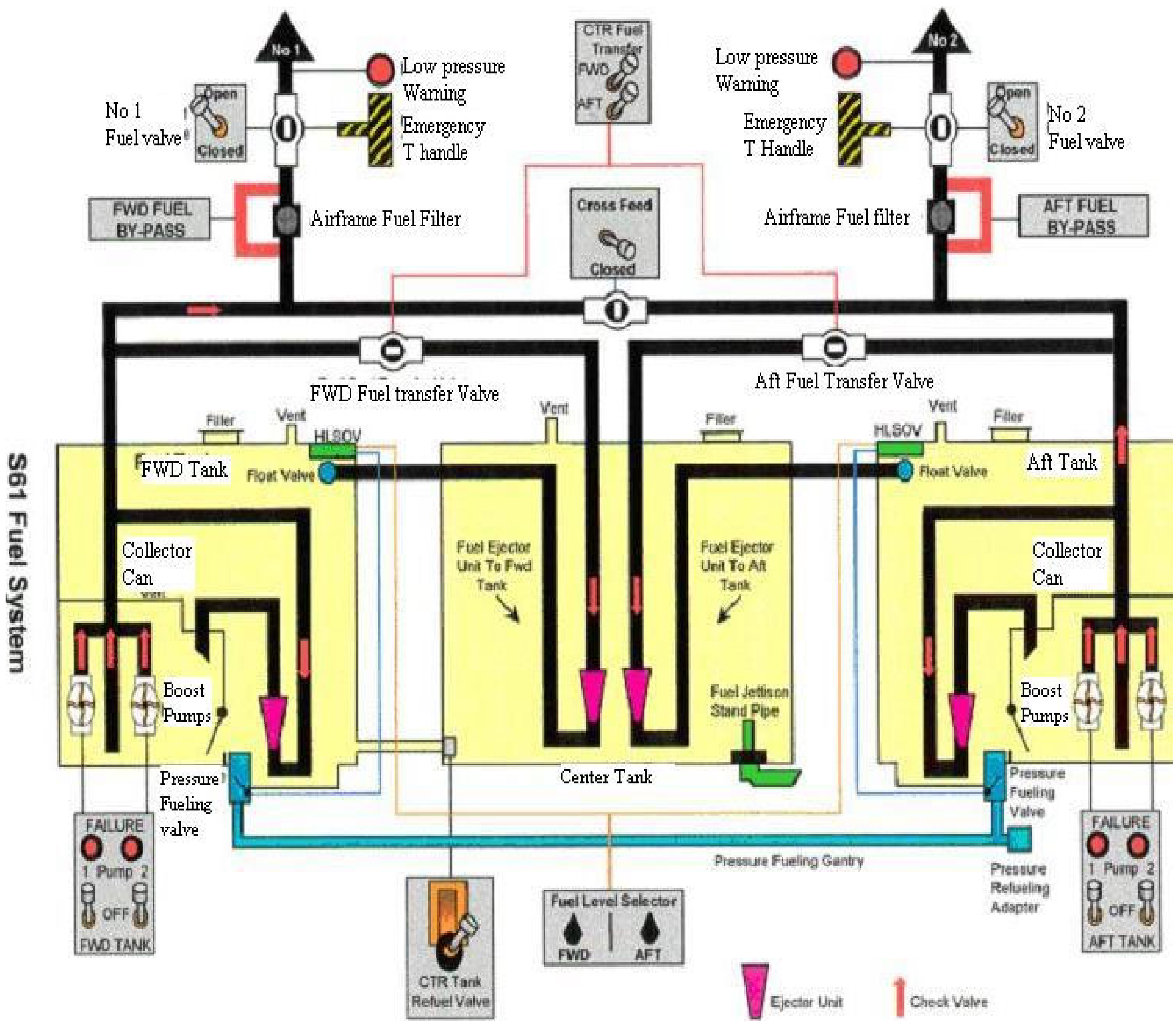
At the time of the accident, helicopter N612AZ was equipped with two digital readout T_5 indicators (Howell) that gave both a bar display and a digital readout. There was one indicator per engine and they were located in the center of the instrument panel. Each power turbine inlet temperature indicator contains a scale and pointer, and a digital read-out; both of which indicate temperature in degrees Celsius. An amber warning light and an OFF flag are also contained on the indicator. The warning light should illuminate when the gage reaches 745°C (CT58-140 engines).

Neither T_5 indicator was located within the helicopter wreckage.

D.3.7 Airframe Fuel System:

The airframe fuel system provides an independent fuel supply to the left and right engines (Figure 13). Helicopter N612AZ was equipped with three fuel tanks: a forward tank, a center tank (auxiliary) and an aft tank. Normally, the forward tank supplies fuel for the Number 1 (left) engine and the aft tank supplies fuel for the Number 2 (right) engine. The dividing line between the airframe and the engine fuel system is the firewall shut-off valve situated immediately beneath the engine bay.

Figure 13 Schematic showing the Airframe Fuel System



All three fuel tanks were located in the hull of the helicopter beneath the passenger cabin floor in watertight compartments comprised of three aluminum-ribbed cavities with bulkheads between them. Each fuel tank comprises a rubberized (flexible) fabric cell (constructed of BTC-99[12]), which is installed within a fiberglass liner that is attached to structure inside the fuel tank compartment. Lightweight bladder constructions such as BTC-99 are not designed to have the puncture resistance or the tear resistance required to survive a crash impact. As such, BTC-99 and other lightweight bladder constructions can be penetrated or ruptured in a crash sequence.

Pressurized fuel is supplied from each fuel tank via two electrically operated fuel booster pumps through a mixing chamber, fuel manifold, airframe fuel filter, and then to the engine fuel supply system. The forward and the aft fuel tanks each contain a fiberglass collector can, which houses the two fuel booster pumps. Within a tank, the fuel booster pumps draw fuel in through a mesh strainer and then discharges the fuel (between 26-50 psi) from their outlet port through a hose to the mixing chamber. After the mixing chamber, the fuel flows through the airframe fuel manifold into a 40-micron airframe fuel filter and then on to its respective engine. The aircraft contains two independent airframe fuel filters (one for each engine), which are secured to the lower fuselage fuel controls in the compartment immediately aft of the forward fuel tank.

Each airframe fuel filter assembly contains a built-in bypass valve, which allows fuel to bypass a partially clogged filter, and an electrical switch. Helicopter N612AZ was equipped with a 40-micron airframe fuel filter assembly having P/N S6130-63072-1 containing a metal cleanable filter element having P/N 52-0505-1. During the course of this investigation, Sikorsky advised the NTSB that they were working approval of a 10-micron filter as an alternative to the 40-micron filter for the S-61L, S-61N, S-61R, and S-61NM having general electric model CT58-140-1 or CT58-140-2 engines installed.

With the S6130-63072 type fuel filter, the electrical switch is actuated by pressure differential acting upon a diaphragm in the poppet assembly of the bypass valve. When activated, the switch illuminates a "FWD FUEL BYPASS" or "AFT FUEL BYPASS" caution light, as appropriate, on the cockpit Caution/Advisory Panel to indicate a clogged airframe fuel filter. Spacing between the corrugations of the filter element collects any contaminant in the fuel. In the S6130-63072 type fuel filter, the electrical switch actuates at a pressure differential of 1.5 plus 0.2, minus 0.4 psi to illuminate the caution light and the bypass valve opens at a pressure differential not greater than 3.5 psi; therefore, the caution light illuminates to indicate a partially clogged filter before the bypass valve opens. Access to the fuel filters is through the access cover or the flooring over the fuel control compartment.

A fuel control panel, located on the instrument panel contains fuel quantity indicator[13]; pump switches, valve switches, and a crossfeed switch. The pump switches control the application of 30V AC to the fuel boost pumps. A pressure line from each

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[12] BTC-99 is an FAA TSO-C80 approved bladder construction that has been in service for many years. It is used in most fixed wing lightweight utility class aircraft and rotorcraft that pre-dated the FAA commercial rotorcraft crash resistant requirements.

[13] The indicator receives 115 volt, AC electrical power through the FUEL QTY circuit breakers, on the AC circuit breaker panel.

pump to a pressure switch will operate a red light on the fuel control panel at or below 16 psi (for each pump) with the pump switched ON.

According to Civil Air Regulations (CAR), Part 7 "Rotorcraft Airworthiness transport Categories, dated August 1, 1956", section 7.42 (b), Fuel tanks and their installation shall be designed or protected so as to retain the fuel supply without leakage when the rotorcraft is subjected to the emergency landing conditions specified under CAR 7.260.

CAR 7.260 part (a) states the following:

The structure shall be designed to give every reasonable probability that all of the occupants, if they make use of the seats, belts, and other provisions made in the design (see 7.355) will escape serious injury in the event of a minor crash landing in which the occupants experience the following ultimate inertia forces relative to the surrounding structure:

a. *Upward 1.5 g (downward 4.0 g)*
b. *Forward 4.0 g*
c. *Sideward 2.0 g.*

CAR 7.260 part (b) states the following:

The use of a lesser value of the downward inertia force specified in paragraph (a) of this section shall be acceptable if it is shown that the rotorcraft structure can absorb the landing loads corresponding with the design maximum weight and an ultimate descent velocity of five feet per second without exceeding the value chosen.

D.3.7.1 Investigation Findings:

N612AZ landed at the Trinity Helibase at about 7:06 PM PDT local and remained on the ground for about 17 minutes for re-fueling[14]. Once refueled, N612AZ had approximately 2,500 pounds of fuel on board; 1,200 pounds in the forward tank, 100 pounds in the center tank, and 1,200 pounds in the aft tank. N612AZ departed Trinity Helibase at about 7:23 PM PDT, flew for about twelve minutes and then landed at H-44 for its third load of passengers at about 7:35 PM PDT.

An assessment of the helicopter wreckage revealed that the post-crash fire thermally compromised the hull of the helicopter. Each fuel tank including their associated fabric cells (fuel bladders), fuel lines, fuel boost pumps, airframe fuel filters and crossfeed valves were not identified within the wreckage.

One fuel quantity indicator having serial number 754B was located and recovered from the helicopter wreckage. The indicator was found within in the area of the cockpit buried under ash and debris. According to a representative from CHSI, the indicator was determined to be either the forward or the aft fuel quantity indicator. The pointer on the recovered indicator pointed to a fuel quantity of approximately 900 pounds.

[14] According to data obtained from the SkyConnect Tracker AFF system and fueling records.

The CHSI and Sikorsky Airframe Fuel Filter maintenance requirements are the same. CHSI's inspection program defines the interval for inspecting, cleaning, and functional check of the airframe fuel filters (2) to be performed during a Phase 3 Inspection. The last filter inspection and cleaning was performed on July 11, 2008 at an aircraft Total Time of 35,355.2.

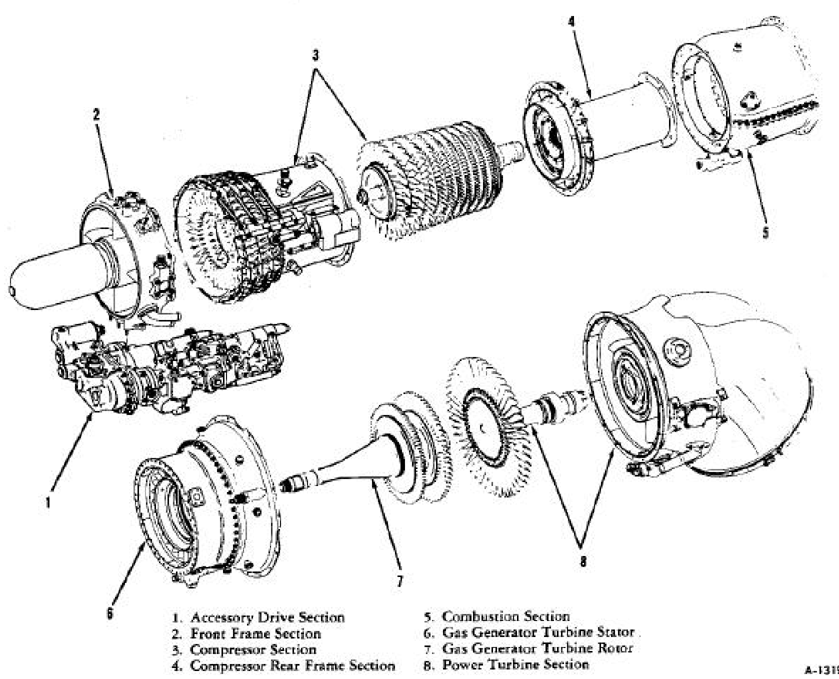
D.4 Powerplant:

D.4.1 Engine Description:

Helicopter N612AZ was equipped with two 1500-horsepower General Electric CT58-140 turboshaft engines. The CT58-140 engine is an axial-flow turboshaft engine incorporating the independent free turbine principle (Figure 14). A single-stage free power turbine (PT) extracts output power, which is mechanically independent of the gas generator rotor system, and provides output power of 1500 shaft horsepower (SHP) at the 2½-minute rating and 1,250 SHP at the maximum continuous rating. The gas generator consists of a ten-stage compressor, annular combustor and a two-stage gas generator turbine (GGT). The inlet guide vanes (IGV) and stator vanes (in Stages 1, 2 and 3) are variable. The turbine section includes a gas turbine section, which consists of two turbine rotors that provide rotational power to the compressor. Various instruments monitor engine performance; these instruments include the torquemeter indicator (two needles labeled 1 and 2 for the Number 1 and Number 2 engine torque in percent (%), respectively), and the triple tachometer (three needles labeled 1, 2 and R for Number 1 power turbine speed, Number 2 power turbine speed, and main rotor percent rated speed (%), respectively).

The CT58-140 engine is 59 inches long and weighs 340 lbs without the starter and tachometer generators. 100% gas generator speed (N_G) is 26,300 rpm and 100% power turbine speed (N_F) is 18,966 rpm. All of the engine flow path components, such as the discs, spool, blades and vanes are GEAM355 (stainless steel).

Figure 14 Schematic showing the major sections of the CT58-140 engine



D.4.1.1 Compressor Section:

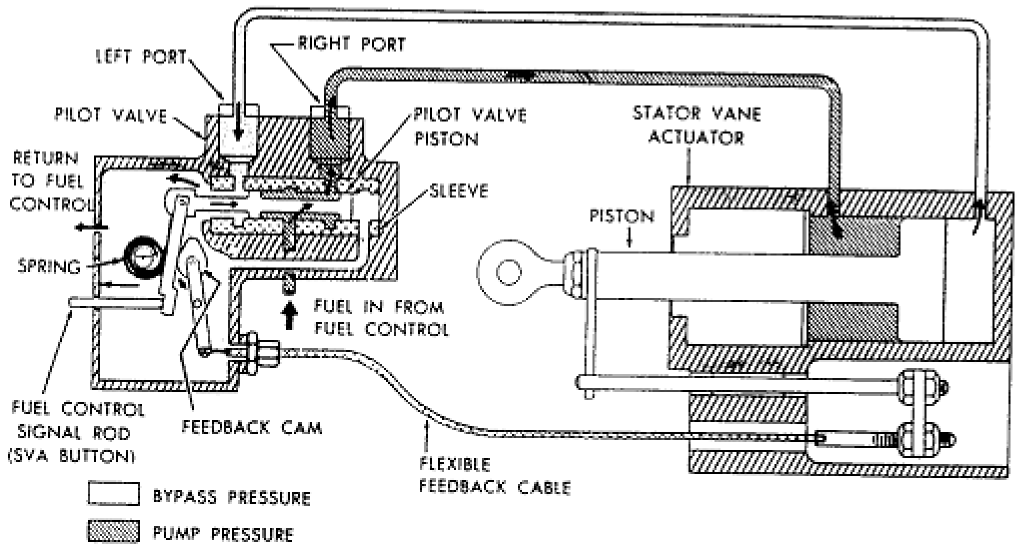
The CT58-140 engine employs an axial flow compressor, which consists of the compressor stator assembly and the compressor rotor assembly. Air taken from the atmosphere passes through the front frame to the inlet of the compressor section. Mounted on the inner surface of the compressor stator are ten stages of stator vanes, one stage of inlet guide vanes, and one stage of exit guide vanes. The compressor rotor assembly is a ten stage steel unit. The first two stages consist of a disk and a front shaft with blades dovetailed axially into the rim of each. The remaining eight stages consist of blades dovetailed circumferentially into a tapered, one-piece spool.

The stator vane system controls the flow of compressor intake air by regulating the angle of the inlet guide vanes and the vanes of the first four compressor stator stages. The air increases in pressure as it flows between the blades to the progressively smaller cross sectional area. Compression in the final stage is approximately eight times greater than atmospheric pressure. Engine idle speed is about 56% gas generator speed (N_G). At this speed, the variable stator vanes are fully closed. Variable stator vanes are required to produce efficient, stall-free operation throughout the entire speed range. The variable stator vanes are at their closed position during engine starting and open as the engine speed increases. They start to open at about 65% N_G and are fully open at about 95% N_G, on a standard day. As the N_G decreases through 64% during a normal engine shutdown, reduced fuel pressure causes the actuator piston to fully retract and the vanes rotate to fully closed and remain there during coast down.

The stator vane actuating system comprises a stator vane actuator (SVA), a pilot valve mounted on the forward face of the fuel control unit, a signal actuating mechanism in the fuel control, a bellcrank assembly, four actuating rings, and multiple actuating levers. The stator vane schedule is provided by a contour on the 3D cam. A cam follower positions the stator vane actuator button, which protrudes from the fuel control into the pilot valve. The button activates the pilot valve piston to port high-pressure fuel (from the fuel pump) to the actuator. Motion of the SVA piston is fed back to the pilot valve to provide nulling of the system when the desired position is reached.

During an engine start, the 3D cam is rotated by the T_2 sensor to provide the proper stator vane schedule contours. As N_G increases, the 3D cam translates allowing the stator vane cam follower to trace the scheduling contour. Assume the stator vane actuator button moves to the left as indicated in Figure 15. This movement results in the pilot valve piston moving to the right, uncovering ports in the sleeve. High-pressure fuel exits through the right port and is directed to the actuator via an external line. Fuel enters the chamber at the left side of the actuator piston, moving it to the right closing the variable stator vanes. Fuel in the chamber at the right end of the piston is forced back to the pilot valve left port, where it is directed to the fuel control case through an external port. As the actuator piston moves to the right, the feedback cable moves to the right also. A change in N_G or T_2 input would cause a vane angle change to occur within the region of regulation.

Figure 15 - Schematic of the pilot valve and stator vane actuator



D.4.1.2 Combustor Section:

The combustor section consists basically of the combustion casings, the annular combustion liner, the compressor rear frame assembly and the first stage turbine nozzle. Fuel enters the combustor through two fuel manifolds. One manifold is active at low gas generator speeds and both are active at higher speeds ensuring a proper spray pattern throughout the operating range.

D.4.1.3 Gas Generator Turbine Description:

The gas generator turbine section consists of first and second stage turbine casings, second and third stage turbine nozzles and the turbine rotor. Gases from the combustion chamber drive the gas generator turbine rotor, which drives the compressor rotor by means of the turbine forward shaft.

The gas generator turbine rotor assembly consists of a turbine forward shaft, the first stage turbine wheel, a coupling shaft, the second stage turbine wheel, the turbine rear shaft, and a roller bearing. Cooling air plates are mounted on the front and aft faces of both turbine wheels and are held in place by both forward and rear turbine shafts, and by a coupling shaft. The front face of each turbine wheel and each blade base is cooled directly by compressor air bled internally from the ninth stage through the hollow turbine shaft bolt. The air passes into an annular space within the turbine forward shaft and the turbine-coupling shaft. It then passes through holes in the curvic couplings and through holes in the base of the forward cooling air plates on the first and second stage turbine wheels. This cooling air then circulates aft through the shanks of the blades and passes through holes in the rear cooling air plates.

Gas generator speed (N_G) is primarily dependent upon fuel flow and is monitored by the engine fuel control unit. The principle purpose of monitoring gas generator speed is to control acceleration and deceleration characteristics, prevent overspeed and establish a minimum idle setting. Gas generator speed controls mass airflow pumped through the engine and, consequently, the power available to the engine

D.4.1.4 Power Turbine Section Description:

The power turbine assembly is flanged and bolted to the second stage turbine casing; it consists of the exhaust casing assembly, the power turbine rotor assembly, and the power turbine accessory drive system. A single stage power turbine, which is mechanically independent of the gas generator, provides the output power of the engine. The power turbine rotor derives its power from the gasses which are directed to it by the gas generator turbine nozzles.

The drive for the power turbine governor is via a flex drive shaft, which is driven by the free turbine power gearbox. The shaft provides overspeed protection for the engine through the free turbine governor by operating the overspeed shut-off valve, which if operated, shuts the engine down. In the event of a flex drive shaft failure from the free turbine gearbox to the FCU (N_F governor), there is no overspeed protection of the power turbine. However, if using the speed select lever the engine will not accelerate beyond topping (maximum contingency) setting of 102% N_G; 721°C T_5 (power turbine inlet temperature). Topping speed cannot be exceeded when using the emergency (manual) throttle, however, T_5 limits can be exceeded[15].

The power turbine speed (N_F) is dependent upon engine control input shaft position and main rotor load. The principle purpose of monitoring N_F is to regulate fuel flow to maintain an essentially constant power turbine speed for a given engine control input shaft position. To prevent power turbine overspeed, in the event of a loss of power turbine load, a governor within the fuel control senses N_F and shuts off fuel flow to the engine at a power turbine speed of about 120% (N_F).

D.4.1.5 Engine Accessory Drive Description:

The engine accessory drive system is composed of a series of gear systems used to extract power from the gas generator to drive the engine accessories. Power is extracted from the gas generator by means of a front frame accessory drive (FFAD), which is housed in the hub of the front frame and connected to the compressor rotor front shaft by a splined shaft.

During engine operation, pressurized fuel from the airframe fuel system is supplied to the engine driven dynamic fuel filter. This dynamic fuel filter purifies fuel

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15 The emergency rpm control affects engine operation more directly by allowing manual control of the fuel to the engine. The primary function of the emergency rpm control is to override the automatic features of the fuel control. This control must be used with extreme caution as it has a positive influence on fuel flow and misuse can cause engine overspeed or over temperature.

(>80% efficiency) by a centrifugal action rotating filter turning at 4,200 rpm @ 100 % N_G. After leaving the dynamic fuel filter, the fuel enters the engine driven fuel pump inlet. When the engine is operating at 100% N_G the fuel pump discharge pressure and flow rate are 850 psig and 2,333 lb/hour (min). After being boosted by the pump, the fuel is supplied to the Fuel Control Unit (FCU) and to the stator vane actuator. Metered fuel is passed from the FCU through the oil cooler (fuel/oil heat exchanger), through the static filter (40 microns), and then to the flow divider. At the flow divider fuel flow is directed through two manifolds, the primary and the secondary, and on to the fuel nozzles for ignition. Fuel at pump discharge pressure is used to operate the stator vane actuator.

D.4.2 Engine History:

D.4.2.1 Engine Number 1 (Left):

A review of the maintenance records provided by CHSI, revealed the left engine was identified as having part number CT58-140-1, serial number 295-120. However, the dataplate attached to the engine identified it as having part number CT58-140-2, serial number 295-120C[16]. Reference the maintenance records group chairman's factual report for the number of engine flight hours and cycles.

D.4.2.2 Engine Number 2 (Right):

A review of the maintenance records provided by CHSI revealed the right engine was identified as having part number GE CT58-140-1 and serial number 296-024D. However, the dataplate attached to the engine identified it as having part number CT58-140-2, serial number 296-024[17]. Reference the maintenance records group chairman's factual report for the number of engine flight hours and cycles.

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[16] According to the General Electric service bulletin (CT58 72-122 "CEB 200", dated April 9, 1969 "Conversion between CT58-140-1 and CT58-140-2), the serial number "295" denotes that the engine was configured as a "CT58-140-1" and a "C" at the end of the serial number denotes that the engine was converted to a CT58-140-2. Carson's maintenance records indicate that a re-conversion to a CT58-140-1 was completed at a light overhaul. According to the service bulletin, the data plate should have been updated to show the engine model as "CT58-140-1" and the serial number as "295-120."

[17] According to the General Electric service bulletin (CT58 72-122, dated), the addition of a "D" to the serial number denotes that the engine was converted from a "CT58-140-2" to "CT58-140-1". The service bulletin also indicates that the dataplate should have been updated to show the engine model as "CT58-140-1" and the serial number as "296-024D."

D.4.3 Turbine Engine Power Assurance Check:

According to USDA Forest Service Contract number AG-024B-C-08-9340, Section C: "*A power assurance check shall be accomplished on the first day of operation, and thereafter within each 10-hour interval of contracted flight operation unless prohibited by environmental conditions (i.e. weather, smoke). The power assurance check shall be accomplished by the contractor in accordance with the Rotorcraft Flight Manual (RFM) or approved company performance monitoring program. A current record of the power assurance checks will be maintained with the aircraft under this Contract and any renewal periods*".

A review of the maintenance records provided by CHSI revealed that Carson had performed seven "topping" checks on helicopter N612AZ from July 3, 2008 up to the date of the accident (Reference Table 1).

Table 1 – N612AZ Last Seven Topping Checks

	Hour Meter	Pressure Altitude (feet)	°C	N1 Compressor	Engine Gas Temperature (EGT)	Torque #1	Torque #2	Chart Reading	Δ #1	Δ #2
July 3, 2008	171.0	3200	21	102.1 / 102.1	721 / 717	105	107	105	0	+2%
July 5, 2008	172.0	4200	22	102.0 / 102.0	716 / 700	100	100	98	+2%	+2%
July 7, 2008	188.0	700	34	101.8 / 101.5	721 / 718	102	103	99	+3%	+4%
July 18, 2008	204.0	5000	22	102.0 / 102.1	710 / 700	99	98	94	+5%	+4%
July 25, 2008	213.9	2000	28	102.0 / 102.	712 / 702	105	105	102	+2%	+3%
July 29, 2008	223.2	2500	24	102.0 / 102.1	712 / 706	105	106	101	+4%	+5%
August 4, 2008	233.4	3160	30	101.8 / 101.5	720 / 714	97	100	94	+3%	+6%

Note: N612AZ accumulated an additional 3.5 hours after the power check on August 4, 2008. This is based on the fact that N612AZ did not fly on August 4, 2008 and the fact that data from SkyConnect and the CVR, indicates N612AZ accumulated an additional 3.5 hours on August 5, 2008.

Sikorsky only developed Power Assurance Check (PAC) charts for -500/0/+500/+500/+1000 foot Pressure Altitudes (PA) for the CT58-140 engines. As documented in Table 1, six of the last seven PACs were accomplished at a PA greater than 1000 feet. Because the PAC charts do not contain data at PA's greater than 1000 feet, CHSI could not perform a PAC as specified in the flight manual. Instead, they performed a "Topping Check" as specified in the Sikorsky Maintenance Manual (SA4045-80, chapter 73-20-0), recording ambient temperature and PA, and reading the cockpit N_1, T_5, and torque gauges (actual)[18].

Sikorsky requires topping checks every 450 flight hours; any time an engine, fuel control unit, engine fuel pump, or stator vane actuating system component is changed; and following adjustments to the fuel control or stator vane actuating system.

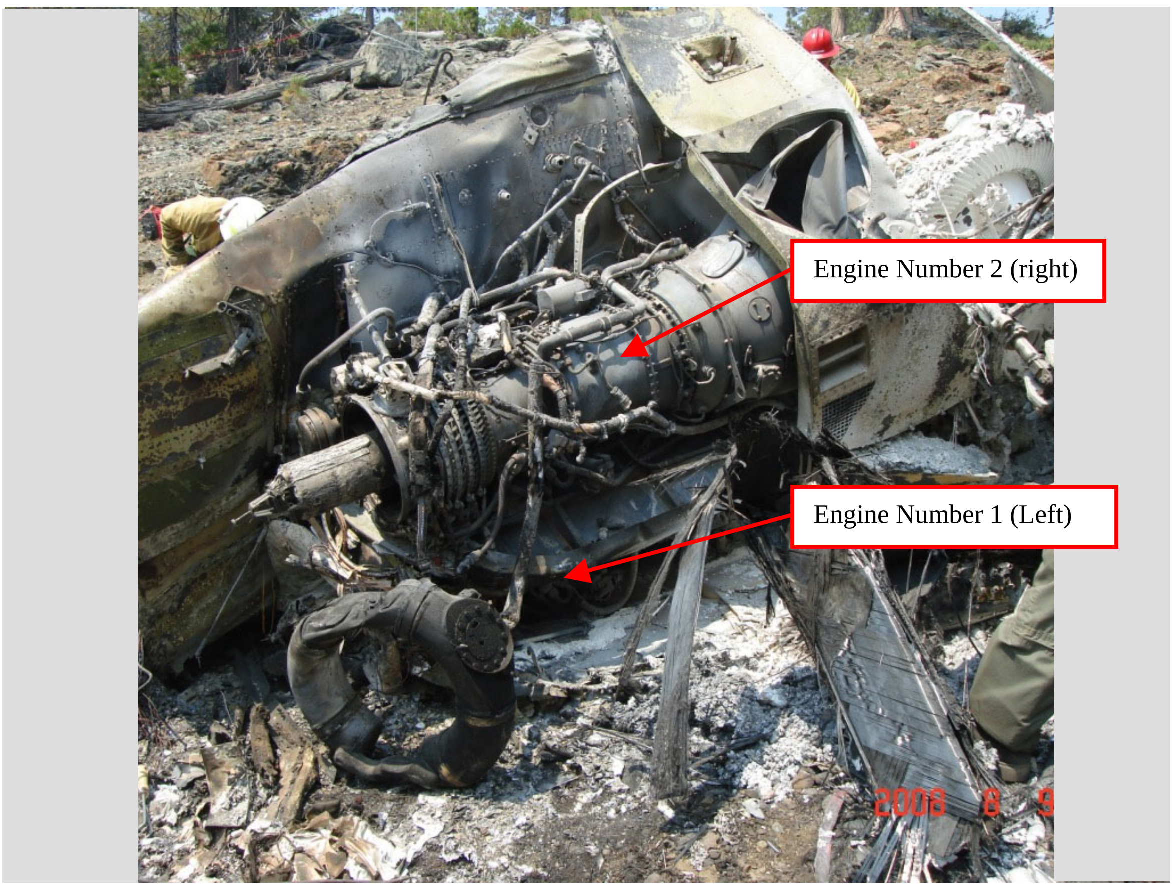
[18] For further information regarding the topping checks performed by Carson, see the Operations Factual Report.

D.4.4 Powerplant Observations at the Accident Site:

From August 8 -11, 2008 the helicopter's two General Electric CT58-140 turboshaft series engines were visually examined at the accident site.

The Number 1 engine (S/N 295-120C) remained in position beside the Number 2 engine (S/N 296-024D) (Figure 16). Both engines remained attached to the engine deck/cabin roof by their mounting structure; the engines remained separated by the firewall. The engine deck was found resting on its left side, pointing downhill at an angle of about 25 – 30 degrees. The inlet of the left engine was resting on the ground, which consisted of light brown soil, rocks, ash, resolidified aluminum and other debris.

Figure 16 Photograph of the engines at wreckage site



Both engines exhibited widespread heat/fire damage on their external components and there were no visible "burn through" areas on any of the engine casings. For each engine, the engine speed control flex cable assembly remained intact, contained within its clamps attaching it to the engine, and connected to the fuel control unit. The left and right engine speed control flex cable assemblies remained attached to its respective fuel control unit input linkage. To facilitate the removal of the engines from the wreckage, the engine speed control flex cable assemblies were disconnected from the fuel control unit by removing the bolt that attaches the cable to the unit.

The Number 2 engines high-speed shaft remained attached to the spur gear and gimbal; however, the MGB housings had been completely consumed by post-crash fire. Drive shaft continuity from the power turbine to input spur gear checked positive for continuity. The engine exhaust duct was found buckled inwards, but remained attached to the engine. The engine oil tank had separated from the front of the engine.

On August 11, 2008, both engines were removed from the wreckage site by helicopter[19] and transported by road to Columbia Helicopters, Inc., located in Aurora, Oregon for further examination.

D.4.5 Engine Number 1 (Left) S/N 295-120C Examination:

On August 13, 2008, the Airworthiness Group convened at the Columbia Helicopters, Inc facility to commence the examination and teardown of the engine identified as having serial number (S/N) 295-120C[20] (Figure 17). The engine was examined under the supervision of the NTSB and witnessed by representatives from the Federal Aviation Administration, Columbia Helicopters, Inc, General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service and CHSI. After the examination, the disassembled engine was boxed by Columbia Helicopters, Inc[21] and shipped to Plain Parts, Inc. in Pleasant Grove, CA.

On October 28 and 29, 2008 the Airworthiness Group re-examined the engine components at Plain Parts under the supervision of the NTSB. Representatives from General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service and CHSI witnessed the re-examination.

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[19] The engines were removed from the accident site via long line operation, using a Lama aircraft on 11 August 2008.

[20] According to the dataplate attached to the external surface of the power section of the engine, the engine was a model number CT58-140-2 having serial number GEE295-120C.

[21] No investigative representation was present at the time that Columbia Helicopter's, Inc packaged and shipped the engine components.

Figure 17 Photograph of Engine Number 1 (S/N 295-120C) at Columbia Helicopter Inc.

D.4.5.1 Engine External Condition Observations:

There was no evidence of casing penetrations. The exterior portion of each engine had experienced exposure to high thermal temperatures, with most accessories and external components thermally damaged. Most of the external fuel, oil lines and electrical harnesses were found compromised due to the thermal exposure. P_3 air-lines were found thermally damaged however the remaining portions appeared to be properly attached to the respective locations.

D.4.5.2 Engine Compressor Section Observations:

Examination of the front frame assembly, Number 1 bearing and forward stub shaft was conducted. The front frame assembly had evidence of impact damage between the 3:00 strut and the 6:00 strut position on the front flange. The Front Frame Accessory Drive (FFAD) cavity was found coated in a light layer of soot. The oil jet manifold was in location and appeared to be undamaged. The Number 1 roller bearing was found retained in its bearing housing; it appeared to be dry but shiny. To check the integrity of the bearing, the compressor rotor was rotated by hand. During this operation, all of the roller elements could be seen rotating. The forward stub shaft bearing retainer nut was undamaged and in its proper location.

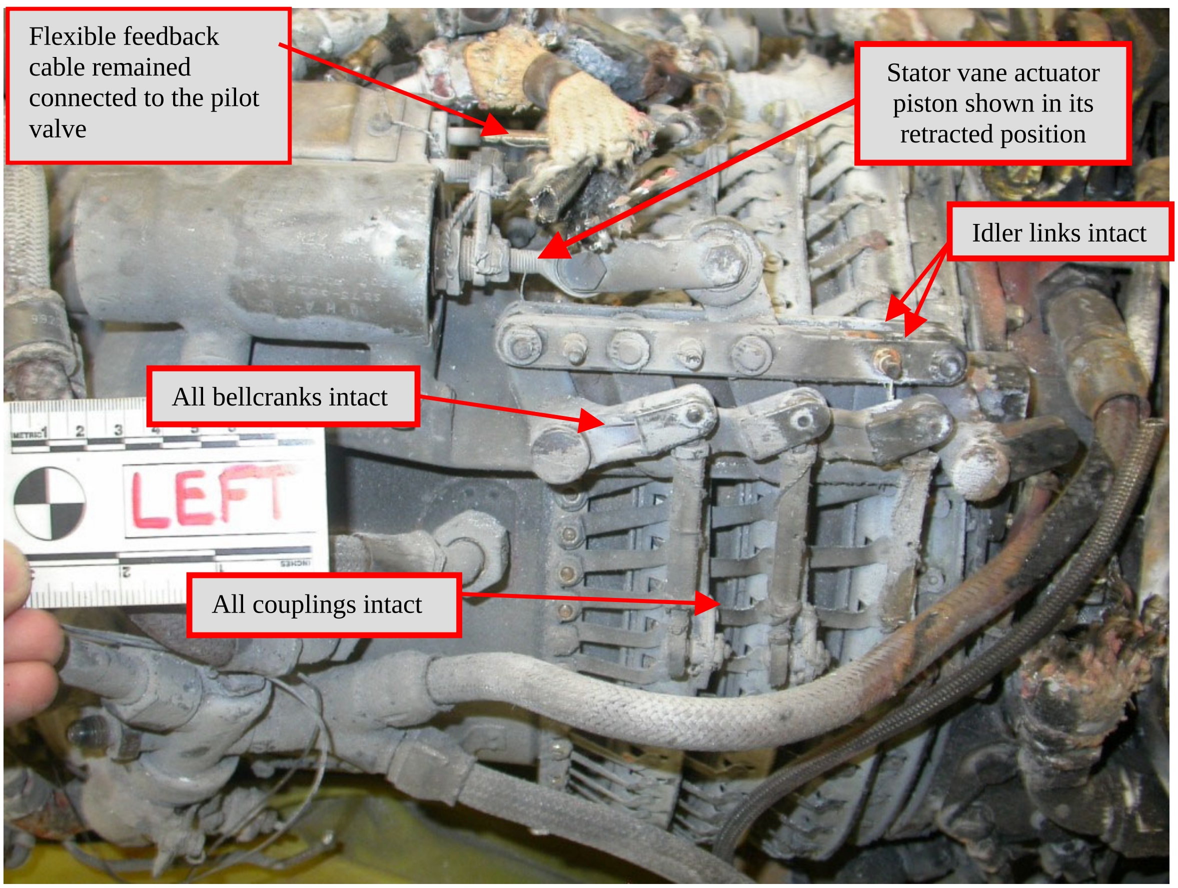
Inspection of the stator vane system was conducted to check the integrity of the inlet guide vanes (IGV), vanes of the first three compressor stator stages and the stator vane actuator (SVA). Prior to the engines being removed from the helicopter, the IGVs were noted to be at or near their closed position. All of the IGV lever arms were present, appeared to be undamaged, had a black flaky coating and were free to rotate although with a large clearance, consistent with loss of elastomeric bearing integrity due to exposure to heat.

The first, second and third stage variable stator vanes (VSVs) rotated freely when hand pressure was applied. All blades were found coated with a light brown colored powdery (very fine) material that could easily be removed by hand. Examination of the first stage VSVs revealed that the outer leading edges of five adjacent VSVs at approximately the 2:30 position were torn with associated material loss; the largest tear was about 0.10 inches. The material deformation was in the direction of rotation. One vane at approximately the 5:30 position had damage to the outer leading edge in the direction opposite of rotation. Three of the second stage VSVs had deformation on the leading edge outer tip. Six of the third stage VSVs blades had leading edge tearing on the outer tips.

Examination of the stator vane actuator (SVA) and its components (support linkage, actuator links, idler links, bellcranks, and couplings) revealed evidence of sooting and surface oxidation consistent with exposure to fire and heat. The SVA and its components remained connected, secure and intact (Figure 18).

Figure 18

Photograph of the stator vane actuator and support linkage from Engine Number 1 (S/N 295-120C)



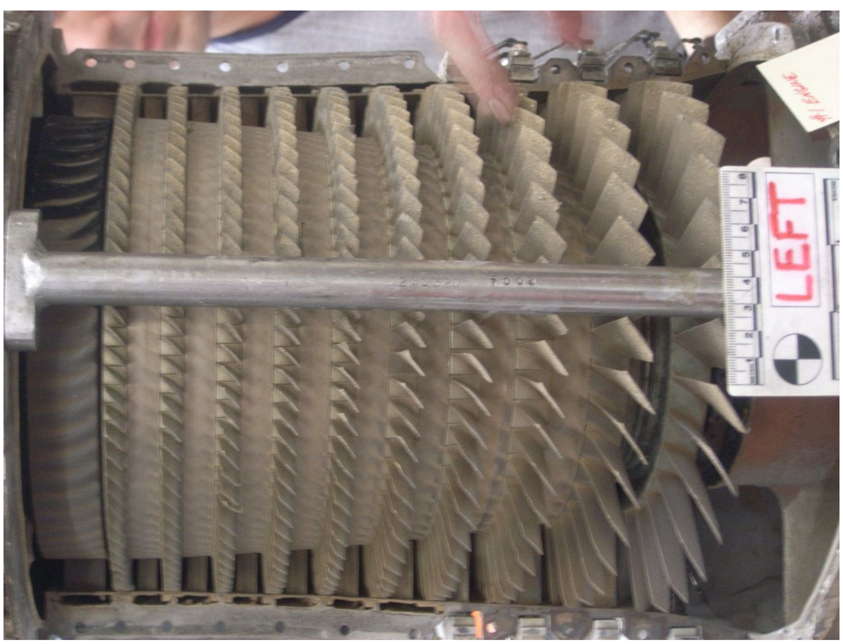
The SVA actuator remained attached to the compressor casing and its piston was found fully retracted. The SVAs flexible feedback cable remained connected to the pilot valve (feedback lever) and to the SVA assembly (overspeed control). The feedback cable was observed in position consistent with the SVA in a fully closed position. Under the supervision of the GE representative, a representative from Columbia Helicopter Inc. removed the piston from the SVA actuator and then sectioned the actuator housing into two pieces to allow for an internal visual inspection of the housing and the piston for witness marks. One possible witness mark was observed 1/2 inch from the end on the piston consistent with the piston being positioned at open (>95% N_G) during impact.

The fuel control unit's pilot valve linkage remained intact, the pilot valve piston could not be moved and there was no resistance on the fuel control unit pilot valve arm when applying hand pressure. The Airworthiness Group submitted the pilot valve to the National Transportation Safety Board Materials Laboratory for metallurgical examination[22]. Reference NTSB Materials Laboratory Factual Report number 09-019 in the Safety Board's public docket for accident number LAX08PA259.

Inspection of the compressor blades was conducted to check the integrity and condition of the blades. All of the compressor blades were present and remained connected to the compressor disk. Loose fine light brown dirt was observed throughout the entire compressor (Figure 19) and there was evidence (dirt, bent blades) of FOD ingestion. After the turbine section was disconnected, the compressor section was rotated (by hand) about ¼ turn and rubbing was observed.

Figure 19

Photograph of the compressor assembly from Engine Number 1 (S/N 295-120C) at Columbia Helicopter Inc.



[22] Reference NTSB Materials Laboratory Factual Report number 09-013 in the Safety Board's public docket for accident number LAX08PA259.

Examination of the first stage compressor wheel blades revealed that the blades exhibited leading edge impact damage. Eight of the blades exhibited tearing damage to their leading edge tips associated with loss of blade material (the tearing direction was opposite of rotation). The leading edges of nine blades were found dented. There was minor rotational scoring on the front half of the blade tip chord. The trailing edges of all the second stage compressor blades were found dented near the blade tip; the denting was in the direction opposite of rotation. Five of the second stage compressor blades exhibited tearing damage to their leading edge tips associated with loss of blade material. Examination of stages 3 – 10 of the compressor wheel revealed that about 5 % of the blades exhibited tearing on the leading and trailing edges and about 60% of the blades were dented and deformed on their leading and trailing edges (all deformation was in the direction of rotation). About 5 % of the blades exhibited leading edge deformation near the outer edges of the vanes. All blades were evenly coated with a light brown colored powdery (very fine) material that could easily be removed by finger rubbing.

The two-piece compressor case was intact, appeared to be undamaged and could be separated by hand following the removal of the securing fasteners. The outside of the case was coated with a white powder.

The inner race of the Number 2 bearing appeared to be undamaged; there were witness marks on the inner land indicating stationary balls. All balls were present and appeared to be discolored but intact. The Number 2 bearing air seal appeared to be undamaged.

D.4.6 Engine Number 1 Combustor Section Observations:

The combustor case (two piece) was intact and the inner surface was coated with the light brown colored powdery material. The boss weldment of the start bleed valve was buckled on the aft side. The start bleed valve was still attached to the combustor case. The valve was observed to be in the closed position. The solenoid housing of the start bleed valve was damaged due to heat; exposing the solenoid wires. T he overboard discharge tube was fractured. The fuel manifold pass-through boss appeared to be undamaged.

The entire combustor liner was coated with a light brown colored powdery (very fine) material that could easily be removed. All fuel nozzles were present and were evenly coated with the brown powder, but were otherwise unobstructed. The fuel manifolds appeared to be undamaged.

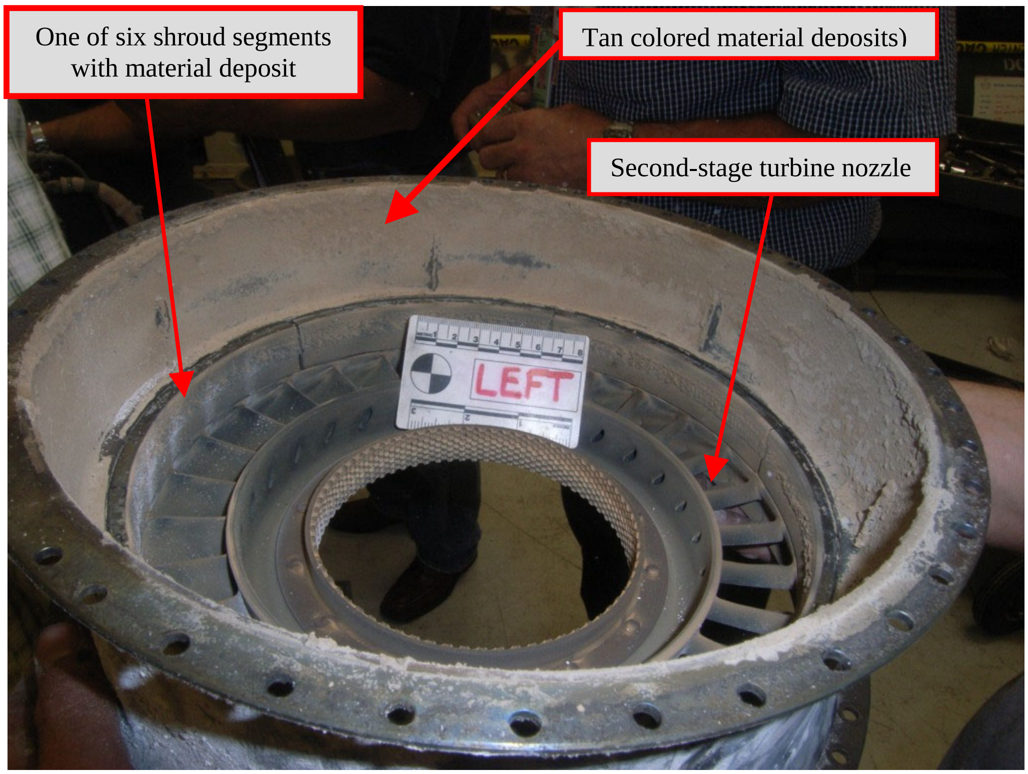
D.4.6.1 Engine Number 1 Gas Generator Turbine Observations:

The gas generator turbine rotor assembly consists of a turbine forward shaft, the first stage turbine wheel, casing, a coupling shaft, the second stage turbine wheel, the turbine rear shaft, and a roller bearing. Cooling air plates are mounted on the front and aft faces of both turbine wheels and are held in place by both forward and rear turbine shafts, and by a coupling shaft.

The first stage turbine nozzle was undamaged and all of the vanes remained intact. The convex side of all the vanes was coated with a light brown colored material. A light white powdery coating covered the light brown colored material on convex side of the vanes for approximately 180 degrees. The turbine casing appeared to be undamaged and intact. The exterior and interior surface was found covered with light brown material (Figure 20).

Figure 20

Photograph of the first stage turbine casing from Engine Number 1 (S/N 295-120C)



Examination of the turbine forward shaft indicated that its forward spline appeared to be undamaged. The knife-edges of the turbine air seal were rubbed and rolled. The curvic coupling appeared to be undamaged and all teeth were present. The stationary turbine air seal exhibited minor rub consistent with contact against with the turbine forward shaft seal and the stationary labyrinth seal for the first stage turbine forward cooling plate was intact and appeared to be undamaged.

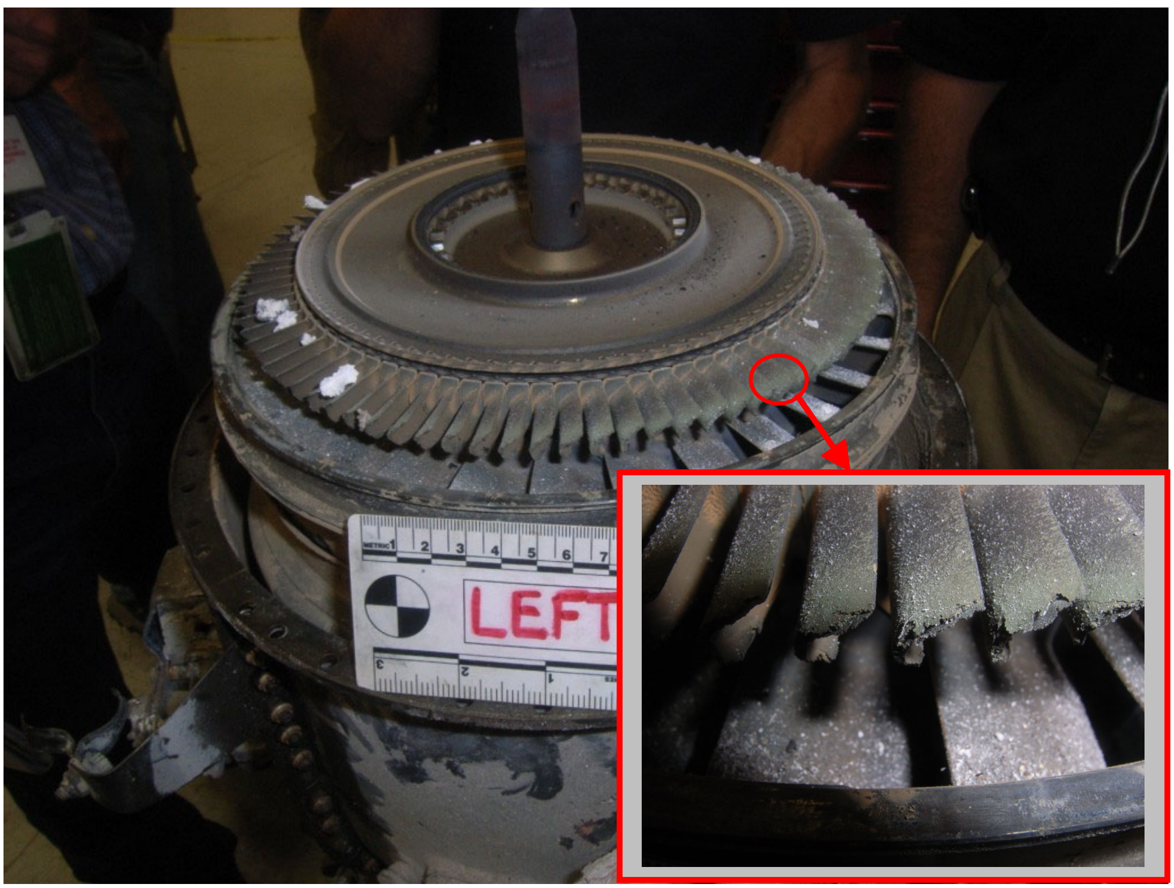
The first stage forward turbine cooling plate was intact and appeared to be undamaged. The knife-edge seals were undamaged. There was a pattern of light brown dust consistent with the pattern of the curvic coupling on the inner diameter.

The first stage aft turbine cooling plate was warped approximately 1/8 inch aft around the full circumference (360 degrees).

Examination of the first stage turbine wheel revealed that its wheel hub appeared to be undamaged (Figure 21). A spiral shaped cooling pattern was observed on the forward face of the hub and the forward curvic appeared to be undamaged. The aft face of the hub appeared to be undamaged. All of the blades exhibited a coating of light brown material on their concave surfaces, and a coating of the same material on about thirty percent of their convex surface. All blades tips were found fractured at approximately 2/3 their length. The wheel diameter measured about 9.2 inches vs. 9.7 inches nominal (as compared to the Number 2 engine wheel). The aft curvic coupling appeared to be undamaged. The Airworthiness Group submitted the first stage turbine wheel to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-013 in the Safety Board's public docket for accident number LAX08PA259).

Figure 21

Photograph of the first stage turbine wheel from Engine Number 1 (S/N 295-120C)



1. First Stage Turbine Shroud:
   All six of the first stage turbine shroud segments had a buildup of deposited material in-line with the first stage turbine wheel. The gaps between the turbine shroud segments were found uneven and bridged by the material deposits. The Airworthiness Group submitted the first stage turbine casing (containing all six shroud segments) to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-013 in the Safety Board's public docket for accident number LAX08PA259).

2. Second Stage Turbine Nozzle:
   The second stage turbine nozzle was intact but was seized in the turbine casing. The leading edges appeared to be undamaged. The trailing edges of several vanes were deformed, warped and exhibited fractures at the vane trailing edge tips. Several vanes had outer leading edges that were cracked at the outboard platform. A segment of 90 degrees of the vanes were coated with white powder. The honeycomb seal land appeared to be undamaged.

3. Second Stage Turbine Casing:
   The turbine casing appeared to be undamaged and intact. The exterior surface was covered with a light powder.

4. Second Stage Turbine Wheel:
   The second stage turbine hub remained intact and appeared to be undamaged. The front and aft curvic couplings appeared to be undamaged. The knife-edge seal appeared to be undamaged. All of the second stage turbine blades remained intact and all blade tips showed rub with evidence of mushrooming of the blade tips in the forward and aft direction. The Airworthiness Group submitted the second stage turbine wheel to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-013 in the Safety Board's public docket for accident number LAX08PA259).

5. Second Stage Turbine Shroud:
   The six second stage turbine shrouds were intact and in location. There was a rough metallic deposit throughout 360 degrees on the shroud. There was a heavy rub approximately at the 9:00 position.

6. Number 3 Bearing Assembly:
   The Number 3 bearing assembly appeared to be dry and distressed. The inner race and rollers could be rotated by hand.

7. Thermocouple Harness:

   The turbine inlet temperature (T_5) is of prime importance in operating this engine. The temperature is sampled by eight thermocouples, located in the second stage turbine casing just forward of the power turbine. The thermocouples are connected in parallel so the average of the temperature is transmitted to the cockpit indicator where the temperature signal is indicated in degrees Celsius.

   The thermocouple harness was intact and the probes appeared to be undamaged. The connector to the airframe harness was incomplete, missing the connector nut portion and the insulation around the probe was missing. The blank-off connector was present but fractured.

8. Turbine Shaft Bolt and Nut:
   The turbine shaft bolt appeared intact and the shaft bolt-breakaway torque was 180 ft-lbs.

D.4.6.2 Engine Number 1 Power Turbine Observations:

1. Exhaust Casing:
   The exhaust casing was found covered in white ash (consistent with magnesium oxide) and the casing was found thermally damaged. The power turbine assembly (exhaust casing, power turbine rotor, rear support, and main drive shaft) was removed from the engine. The rear support was intact but bent. The main drive shaft could not be rotated. The isolator was melted away, however the isolator bolts remained attached to the rear support and yoke on main gearbox input housing. Approximately 1/2 square foot of sheet metal material was found consumed at approximately the 1:00 to 3:00 position.

2. Power Turbine Nozzle:
   The power turbine nozzle was intact. There was a white powdery deposit between the 1:00 and 6:00 positions of the concave and convex side of the vanes. There was a rough metallic deposit on the outboard convex surface.

3. Power Turbine Wheel:
   The power turbine wheel was intact. White powdery debris was found stuck onto the blades between 3:00 and 5:00 position. The power turbine wheel could not be rotated due to resolidified metal stuck between the blades and casing assembly. The leading edges appear to be undamaged. The forward side of hub appeared to be undamaged. All the blades were present and in location.

4. PT Accessory Drive Assembly:
   Most of the housing of the accessory drive assembly had evidence consistent with exposure to fire and heat. The two bevel gears remained intact. The bevel gear bearings were present on the bevel gear shaft found but seized. The accessory drive assembly boss was found undamaged.

5. Flex Drive Shaft:
   The flex drive shaft was found intact and its internal cable could be rotated, the ends of the cable were square and not rounded.

6. High speed shaft:
   The high-speed shaft was in-place, attached to the spur gear and could not be rotated.

D.4.6.3 Engine Number 1 Accessory Drive observations:

1. Starter:
   The starter cover was intact, but fire damaged and melted along the forward third of the length. The starter P/N 20069-010, S/N 1109 was found intact, with no evidence of impact, however, it exhibited signs of heat distress. The outer surface of the starter housing was coated with a hard black material. There were small solidified metal droplets on the outside of the housing at the 10:00 “as installed” position. The starter dog drive teeth appeared to be undamaged and the Bendix operated. The starter could not be rotated with hand effort.

2. Front Frame Accessory Drive (FFAD):
   The front frame accessory drive bevel gears were found engaged and normal, no tooth wear was observed. Normal breakaway torque was noted on the locknut. The FFAD appeared to be undamaged but dry, both bevel gears appeared to be undamaged and both shafts rotated with hand effort but with roughness. The compressor rotor input spline appeared to be undamaged. There was a black sticky tar-like substance on the aft face (interior) of the conical section from 10:00 to 3:00.

3. Engine Accessory Drive Gearbox (EAGB):
   Using hand pressure, the Airworthiness Group could not rotate the EAGB and/or fuel control unit through the radial drive shaft when they remained attached to the fuel pump. Upon disassembly from the fuel pump, the EAGB x and fuel control unit rotated freely. No metal was noted on the EAGB magnetic plug. The fuel control unit’s spline appeared normal and its radial drive shaft remained intact and the spline appeared to be undamaged. The engine and EAGB was found to be completely dry of oil and fuel. When the EAGB input spline shaft was rotated by hand, all gears turned in unison

indicating continuity. Output splines (fuel pump, oil pump, and DCFF) appeared to be undamaged.

4. Dynamic (Centrifugal) Filter (DCFF):
Examination of the DCFF found that it had been exposed to high thermal temperatures. The unit's input spline shaft (which is attached to the rotating element via a bronze bearing) appeared to be intact and rotated freely when hand pressure was applied. Disassembly of the unit revealed that no fuel was present and that its sealing O-ring was brittle and broken. The DCFFs rotating element (centrifuge driving end assembly, fallout screen, inner shell and centrifuge driven end) could not be withdrawn from the housing by hand. Inspection found that the rotating element bearing was seized to the shaft. Using a punch and hammer, the rotating element was removed from the unit. No excessive wear was visually observed with the bearing. The filter element was removed from the assembly and no debris or contamination was found.

5. Engine Driven Pump:
All elements of the engine driven pump were intact including the gears, internal splines, and the pumping element teeth. The O-rings were not present and assumed to have been thermally consumed. Re-solidified globular-shaped metallic material consistent with babbitted bearing metal was found between the gears and bearings.

6. Scavenge System:
There was no metal noted on the auxiliary sump tee fitting magnetic plug. The Number 3 scavenge magnetic plug was found thermally destroyed as was the power turbine accessory drive (Number 4 scavenge) chip detector.

   The main oil tank remained attached to the engine; it had impact damage and was leaking oil. The tank was removed to gain access to the front frame of the engine. The input coupling assembly had no signs of rotational scoring.

   The engine oil filter (clean) was observed loose due to the gasket being thermally consumed, no oil was present in the oil tank, and the oil filter was found dry and has areas of heat damage. The Number 2 bearing seal was found normal and the Number 2 oil jet was found normal.

7. Flow Divider:

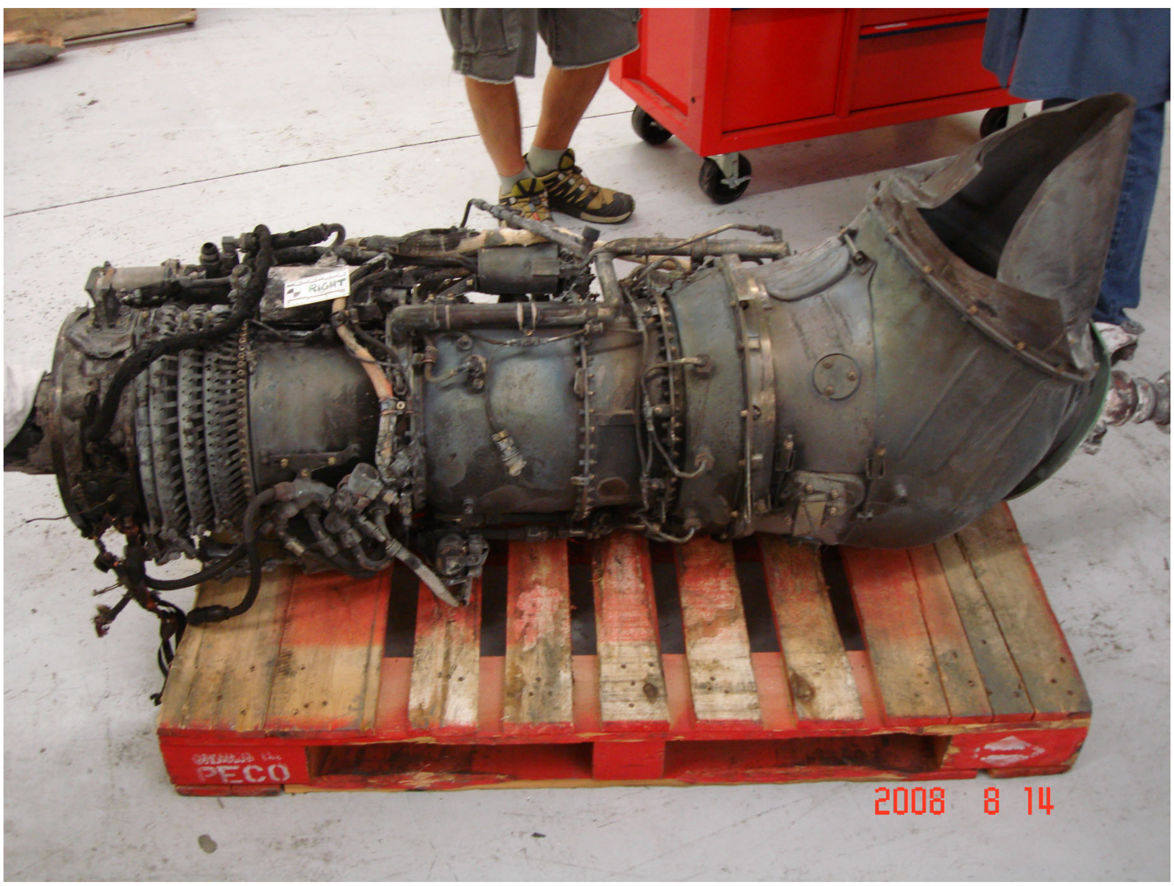
   The flow divider regulates the flow of metered fuel to the combustion chamber. An external visual inspection indicates no obvious damage (other than thermal) on the flow divider. The Airworthiness Group submitted the flow divider to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-019 in the Safety Board's public docket for accident, LAX08PA259).

D.4.7 Engine Number 2 (Right) S/N 296-024D Examination:

The Airworthiness Group convened at the Columbia Helicopters, Inc facility to commence the examination and teardown of the engine identified as having serial number (S/N) 296-024D [23] (Figure 22) on August 14, 2008. The engine was examined and disassembled under the supervision of the NTSB and witnessed by representatives from the Federal Aviation Administration, Columbia Helicopters, Inc, General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service and CHSI. The disassembled engines was then boxed by Columbia Helicopters, Inc[24] and shipped to Plain Parts, Inc. in Pleasant Grove, CA.

On October 28 and 29th, the Airworthiness Group re-examined the engine components at Plain Parts under the supervision of the NTSB. Representatives from General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service and CHSI witnessed the re-examination.

Figure 22 Photograph of Engine Number 2 (S/N 296-024D) at Columbia Helicopter Inc.



[23] According to the dataplate attached to the external surface of the power section of the engine, the engine was a model number CT58-140-2 having serial number GEE296024.

[24] No investigative representation was present at the time that Columbia Helicopters, Inc packaged and shipped the engine components.

D.4.7.1 Engine Number 2 External Condition Observations:

There was no evidence of casing penetrations. The exterior portion of each engine had experienced exposure to high thermal temperatures, with most accessories and external components thermally damaged. Most of the external fuel, oil lines and electrical harnesses were found compromised due to the thermal exposure. P_3 air-lines were found thermally damaged however the remaining portions appeared to be properly attached to the respective locations.

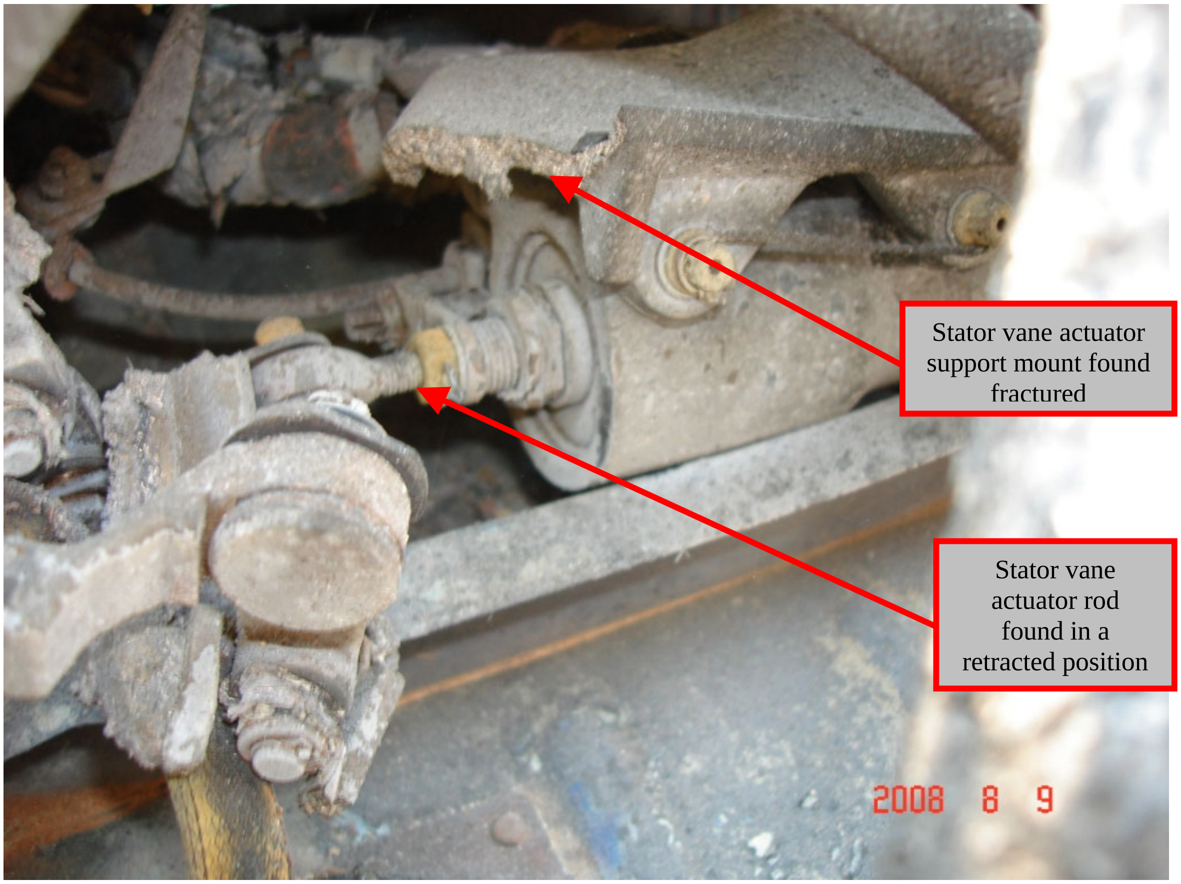
D.4.7.2 Engine Number 2 – Compressor Section Observations:

Examination of the front frame assembly, Number 1 bearing and forward stub shaft was conducted. There was no impact damage observed on the front flange assembly, it was found wetted with oil. The FFAD cavity was oil wetted with no presence of soot. Most of the inner airflow path was coated with a medium to heavy black soot. The FFAD appeared to be undamaged, both bevel gears appeared to be undamaged and both shafts rotated with hand effort and rotated smoothly. All the drive shafts and gears were oil wetted. The compressor rotor input spline appeared to be undamaged. The Number 1 roller bearing was still retained in its location and not separated from the assembly. The bearing could easily be rotated by hand by rotation of the compressor rotor. All of the rotor elements could be seen rotating during this operation. The bearing appeared to be oil wetted and shiny. The forward stub shaft bearing retainer nut was undamaged and in location.

Prior to the Number 2 engine being removed from the helicopter, the inlet guide vanes (IGV) were noted to be at or near their closed position. During the examination at Columbia Helicopter Inc, the variable IGVs were confirmed positioned at their closed position. All IGVs were present, in position and coated with soot. The vanes were free to move within the elastomeric bearings with moderate resistance, consistent with elastomeric bearing integrity. All the IGVs lever arms were present and most were bent. All the upper and lower IGV actuator rings were intact and appeared to be undamaged and had a white powdery coating. Five adjacent and two adjacent IGV trailing edges were torn in the direction of compressor rotation near the tip, consistent with contact against the leading edge of the compressor blades.

Prior to the Number 2 engine being removed from the helicopter, the stator vane actuator (SVA) piston was found fully retracted and all lever arms were found in the closed position. The SVA actuator bellcrank links to the Stage 2 and Stage 3 stator vanes were found broken (Figure 23). The aft section of the actuator support mount (aluminum) was found fractured and the idler link was found broken. Examination of the SVA and its components (support linkage, actuator links, idler links, bellcranks, and couplings) revealed evidence of sooting and surface oxidation consistent with exposure to fire and heat. Safety wire was noted on the stator vane actuator's top and forward mounting bolts.

Figure 23 Photograph of the stator vane actuator on Engine Number 2 at accident site



Examination found that the SVA rigging is on the C mark.[25]. The fuel control's pilot valve linkage remained intact, the pilot valve piston could not be moved and using hand pressure, there was no resistance on the fuel control unit pilot valve arm. The Airworthiness Group submitted the pilot valve to the National Transportation Safety Board Materials Laboratory for metallurgical examination[26]. Reference NTSB Materials Laboratory Factual Report number 09-019 in the Safety Board's public docket for accident number LAX08PA259.

The SVAs flexible feedback cable remained connected to the pilot valve (feedback lever) and to the SVA assembly (overspeed control). The feedback cable was observed in position consistent with the SVA in a fully closed position. Under the supervision of the GE representative, Columbia removed the piston from the SVA actuator and then sectioned the actuator housing into two pieces to allow for an internal inspection. The piston was examined for witness marks; none were observed.

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[25] The "C" mark is for rigging purposes, with the SVA rod fully retracted and the stator vanes fully closed, the chisel mark on the support ring must line up with the "C" mark on the third stage actuating ring.

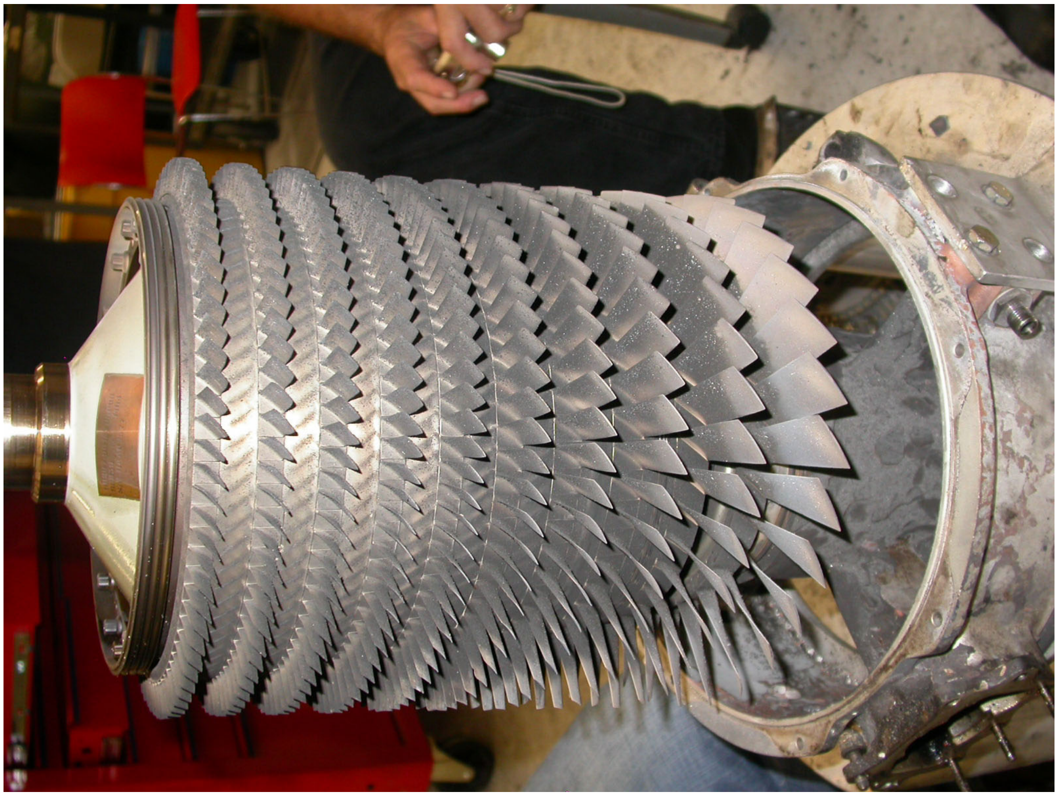
[26] Reference NTSB Materials Laboratory Factual Report number 09-013 in the Safety Board's public docket for accident number LAX08PA259.

Examination of the first, second and third stage variable stator vanes (VSVs) revealed that near the root of the vanes for all IGV and VSVs there is evidence of light brown powdery substance, that could easily be removed by finger pressure, adhered to the inner casing flow path surface. Examination of the first stage VSVs revealed that the inner span of the concave and convex side of the first stage blades were coated with black soot varying from approx 1/8 of a span to 7/8 of a span. All the blades were free to rotate. All of the second stage VSVs were present and appeared to be undamaged. Approximately 1/2 of the vanes were free to move within the elastomeric bearings with moderate resistance, consistent with elastomeric bearing integrity. The inner span of the concave and convex side was coated with black soot varying from approx 1/4 of a span to 3/4 of a span in an oval pattern. The leading edges of the inner vane tips exhibited scoring. Approximately 1/3 of the third stage VSVs were free to move within the elastomeric bearings with moderate resistance, consistent with elastomeric bearing integrity.

All of the compressor blades remained installed in the compressor disk. The compressor section was noted to have a coating a black material consistent with soot on its blades (Figure 24). After the turbine section was disconnected, the group rotated the compressor section (by hand) about ¼ turn; resistance was felt during the rotation.

Figure 24

Photograph of the compressor from Engine Number 2 (S/N 296-024D) at Columbia Helicopter Inc.



Examination of the first stage compressor wheel blades revealed that most blades appeared to be undamaged. The leading edges of five blade tips were found bent in the direction opposite of rotation. Two blades were found dented at the leading edge mid-span. The blades and platforms were found coated with a light powdery gray coating. Approximately ¾ of the blades have black soot up to the mid-span. There was very light rotational scoring on the blade tips. The leading edge tip of one second-stage compressor blade exhibited a small nick, and rollover and there was very light rotational scoring on the blade tips. The blades and platforms were found coated with a light powdery gray coating. Examination of stages 3 – 10 of the compressor wheel revealed that all the blades were coated with a layer of black soot along the span on the concave and convex side. The platform and drum surfaces are coated with a sooty pattern biased to one side. Approximately 5% of the blades were nicked or torn predominately on the trailing edge tips.

The two-piece compressor case was intact, appeared to be undamaged and could be separated by hand. The outside of the case was coated with a white powder.

Using a torque wrench, the Number 2 bearing lock nut broke free at 165 ft-lbs. The inner race of the Number 2 bearing appeared to be undamaged. All balls were present, shiny and appeared to be intact. The Number 2 bearing air seal was intact; all the knife seals were sharp with no evidence of rounding.

The inner race of the Number 2 bearing appeared to be undamaged. All balls were present, shiny and appeared to be intact. The Number 2 bearing air seal was intact; all the knife seals were sharp with no evidence of rounding.

The stationary compressor discharge seal was rotationally rubbed. The outer surface of the housing was clean and oil wetted. The inner and outer diffuser rings appeared to be undamaged. All six struts were clean and oil wetted. The 2:00 igniter boss weldment was buckled.

D.5 Engine Number 2 Combustor Observations:

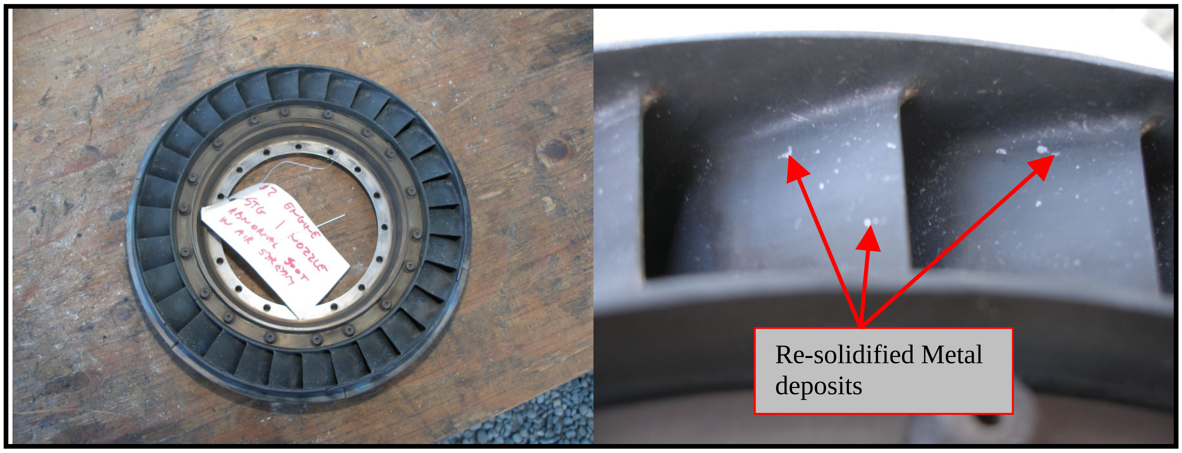
The two-piece combustor case was intact and its inner surface was coated with the light brown colored powdery material. The start bleed valve was still attached to the combustor case. The overboard discharge tube was fractured. The start bleed valve was observed to be in the closed position. The solenoid housing was damaged due to heat, exposing the solenoid wires. The fuel manifold boss appeared to be undamaged.

The entire combustor liner was found coated with a light layer of black soot. All the fuel nozzles were present and were generally coated with black soot, but were otherwise not obstructed. The fuel manifolds appeared to be undamaged.

D.5.1.1 Engine Number 2 Gas Generator Turbine Observations:

1. First Stage Turbine Nozzle:
   The first stage nozzle and all vanes were found intact, undamaged and coated with black soot (Figure 25). Multiple vanes were found with light metal deposits on their convex side. The Airworthiness Group submitted the first stage turbine nozzle to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-013 in the Safety Board's public docket for accident number LAX08PA259).

Figure 25
Photograph of the first stage nozzle and casing from Engine Number 2 (S/N 296-024D)



2. First-Stage Turbine Casing:
   The turbine casing appeared to be undamaged and intact. The exterior surface was found covered with a light brown material.

3. Turbine Forward Shaft:
   The forward spline appeared to be undamaged. The knife-edges of the turbine air seal were rubbed and rolled. The curvic coupling appeared to be undamaged, all teeth were present. The two last knife-edges were buckled at one location for the length of about ½ inch.

4. Stationary Turbine Air Seal:
   The stationary turbine air seal exhibited a uneven scoring pattern; two arcs each approximately one inch long exhibited light rub to none and the deeper score marks were about 180 degrees from the light rub marks. Rub marks are consistent with contact with the turbine forward shaft seal knife-edges. The original clocked orientation of the seal could not be established since it had been previously removed.

5. Stationary Labyrinth Seal:
   The stationary labyrinth seal for the first stage turbine forward cooling plate was intact and appeared to be undamaged. All safeties were found intact.

6. First Stage Forward and Aft Cooling Plate:
   The forward cooling plate knife-edge seals were found intact and undamaged. The forward cooling plate appeared to be flat. The aft cooling plate was intact and appeared to be undamaged. There was a light brown powdery material on the raised ridge radius.

7. First Stage Turbine Wheel:
   The first stage turbine wheel was intact and all blades were present and intact. There was a light metal material deposit on the concave side of three blades. All blades exhibited a light tip rub. The front and aft curvic couplings appeared to be undamaged.

8. First Stage Turbine Shroud:
   All six of the first stage turbine shroud segments were in place and intact. All the segments showed minor kissing, consistent with contact against the first stage turbine wheel blade tips.

9. Second Stage Turbine Nozzle:
   The second stage turbine nozzle was intact and no cracks were observed. There were light sooty deposits on the concave side of the blades and a light metal spray on convex side. The honeycomb seal is intact and undamaged.

10. Second Stage Front and Aft Cooling Plates:
    The second stage front and aft cooling plates appeared to be undamaged. The plates both appeared to be flat. The knife-edges are sharp and not rounded.

11. Second Stage Turbine Wheel Assembly:
    The second stage turbine wheel hub, curvic couplings, and attachment hardware appeared to be undamaged. Several turbine blades had dents on the leading edge tips. Five blades had metal spray on the concave side. The turbine wheel had loose material and debris on its aft end. The turbine lock nut cap retainer remained intact.

12. Turbine Shaft Tie Bolt:
    The turbine locknut pin was installed. A torque value of 80 ft-lbs was required to break the turbine lock nut loose for extraction, and the lock nut appeared normal. The turbine shaft and bolt appeared to be undamaged.

13. Curvic Coupling Shaft:
    The curvic coupling shaft appeared to be undamaged.

14. Number 3 Bearing Assembly:
    The Number 3 bearing assembly appeared to be undamaged. The roller elements were oil wetted. The knife-edges are clean, undamaged and sharp edged. The curvic coupling appeared undamaged.

15. T_5 Thermocouple Harness:
    The thermocouple harness was intact and the probes appeared to be undamaged. The connector to the airframe harness was incomplete, missing the connector nut portion and the insulation around the probe was missing. The blank-off connector was present but fractured.

D.5.1.2 Engine Number 2 Power Turbine Observations:

1. Exhaust Casing:
   The exhaust casing was found slightly damaged. The power turbine assembly (exhaust casing, power turbine rotor, rear support, and main drive shaft) was removed from the engine. The rear support was intact but bent. The isolator was thermally destroyed, however the isolator bolts remained attached to the rear support and yoke on main gearbox input housing.

2. Power Turbine Nozzle:
   The second stage turbine shroud segments were all present and in location and appeared to be undamaged. There was metal spray on the concave side on several blades.

3. Power Turbine Wheel:
   Initially, the power turbine wheel could not be rotated with hand pressure; there was no evidence of FOD damage, but resolidified molten metal was observed between the 10:00 and 11:00 position of power turbine rotor. Severe fire damage was observed to the power turbine assembly. After the Airworthiness Group removed the re-solidified molten metal, the power turbine could be rotated freely, although roughly, through its full rotation by hand pressure. The hub and blade assembly appeared to be undamaged. The blades and hub were oil wetted. At least 11 blades exhibited metal spray on the convex surfaces. The power turbine shrouds were all intact and undamaged.

4. PT Accessory Drive Assembly:
   The accessory drive assembly housing appeared to be complete and undamaged. The two bevel gears remained intact and the bearings were found oil wetted. The accessory drive assembly boss was found undamaged.

5. Flex Drive Shaft:
   The flex drive shaft was found intact and its internal cable could be rotated, the ends of the cable were square and not rounded.

6. High Speed Shaft:
   During the on-scene activities at the wreckage site, the engine was removed with the high-speed shaft and input spur and input pinion still connected. During the engine inspection at Columbia, the high-speed shaft was inspected and no scoring was observed on the splined coupling.

D.5.1.3 Engine Number 2 Accessory Drive observations:

1. Starter:
   About 50% of the starter cover was missing due to thermal damage; the remaining portion of the cover could not be removed from the starter. The starter appeared to be undamaged and the exposed surfaces of the starter were covered with a light layer of soot. The starter dog drive teeth appeared to be undamaged and the Bendix operated. The starter could not be rotated with hand effort.

2. Front Frame Accessory Drive (FFAD):
   The FFAD bevel gears were found engaged and normal, no tooth wear was observed. Normal breakaway torque was noted on the locknut. The FFAD appeared to be undamaged, both bevel gears appeared to be undamaged and both shafts rotated with hand effort and rotated smoothly. All the drive shafts and gears were oil wetted. The compressor rotor input spline appeared to be undamaged. The dogs appeared to be undamaged.

3. Engine Accessory Gearbox (EAGB):
   Using hand pressure, the Airworthiness Group could not rotate the EAGB and/or fuel control unit through the radial drive shaft when they remained attached to the fuel pump. Upon disassembly from the fuel pump, the EAGB and fuel control unit rotated freely. The splines appeared to be undamaged. The input spline into EAGB appeared to be undamaged. When the input spline shaft was rotated by hand, all gears turned in unison indicating continuity. Output splines (fuel pump, oil pump, and CFP) appeared to be undamaged. No metal was noted on the EAGB magnetic plug. The fuel control spline appeared normal and its radial drive shaft remained intact. The oil pump was free to rotate and oil was noted in the EAGB.

   The EAGB magnetic plug was wet; oil was present and the gearbox turned with difficulty by hand. All seals were found thermally damaged. Material was found on the magnetic plug, the material was removed and identified as small pieces of carbon.

4. Dynamic (Centrifugal) Fuel Filter (DCFF):
   Examination of the DCFF found that it had been exposed to high thermal temperatures. The unit's input spline shaft (which is attached to the rotating element via a bronze bearing) appeared to be intact and rotated freely when hand pressure was applied. Disassembly of the unit revealed that no fuel was present and that its sealing O-ring was brittle and broken. The DCFFs

rotating element (centrifuge driving end assembly, fallout screen, inner shell and centrifuge driven end) could not be withdrawn from the housing by hand. Inspection found that the rotating element bearing was seized to the shaft. Using a punch and hammer, the rotating element was removed from the unit. No excessive wear was visually observed with the bearing. The filter element was removed from the assembly and no debris or contamination was found.

5. Engine Driven Pump:
   All elements of the engine driven pump were intact including the gears, internal splines, and the pumping element teeth. The O-rings were not present and assumed thermally consumed. Re-solidified globular-shaped metallic material consistent with babbitted bearing metal was found between the gears and bearings.

6. Fluids and Bearings:
   Evidence of oil was found throughout the engine, including the accessory drive gearbox. During the on-scene activities at the wreckage site, the oil tank remained attached to its respective engine. To facilitate removal of the engine, the oil tank was disconnected from the engine and placed within the main wreckage. Oil was observed within the tank; as the tank was removed, oil drained out of the tank onto the ground. The engine oil filter was found clean and dry. All five mainline bearings were in found in good condition and oil wetted.

7. Scavenge System:
   All three magnetic plugs (EAGB, #2 bearing, and the #3 bearing) remained in place and clean. There were small pieces/chunks of carbon on the PT acc drive chip detector. The PT accessory drive housing was found undamaged.

8. Flow Divider:
   The flow divider regulates the flow of metered fuel to the combustion chamber. An external visual inspection indicated no obvious damage (other than thermal) on the flow divider. The Airworthiness Group submitted the flow divider to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-019 in the Safety Board's public docket for accident, LAX08PA259).

D.6 JFC26 Fuel Control Units (FCU):

D.6.1 Description:

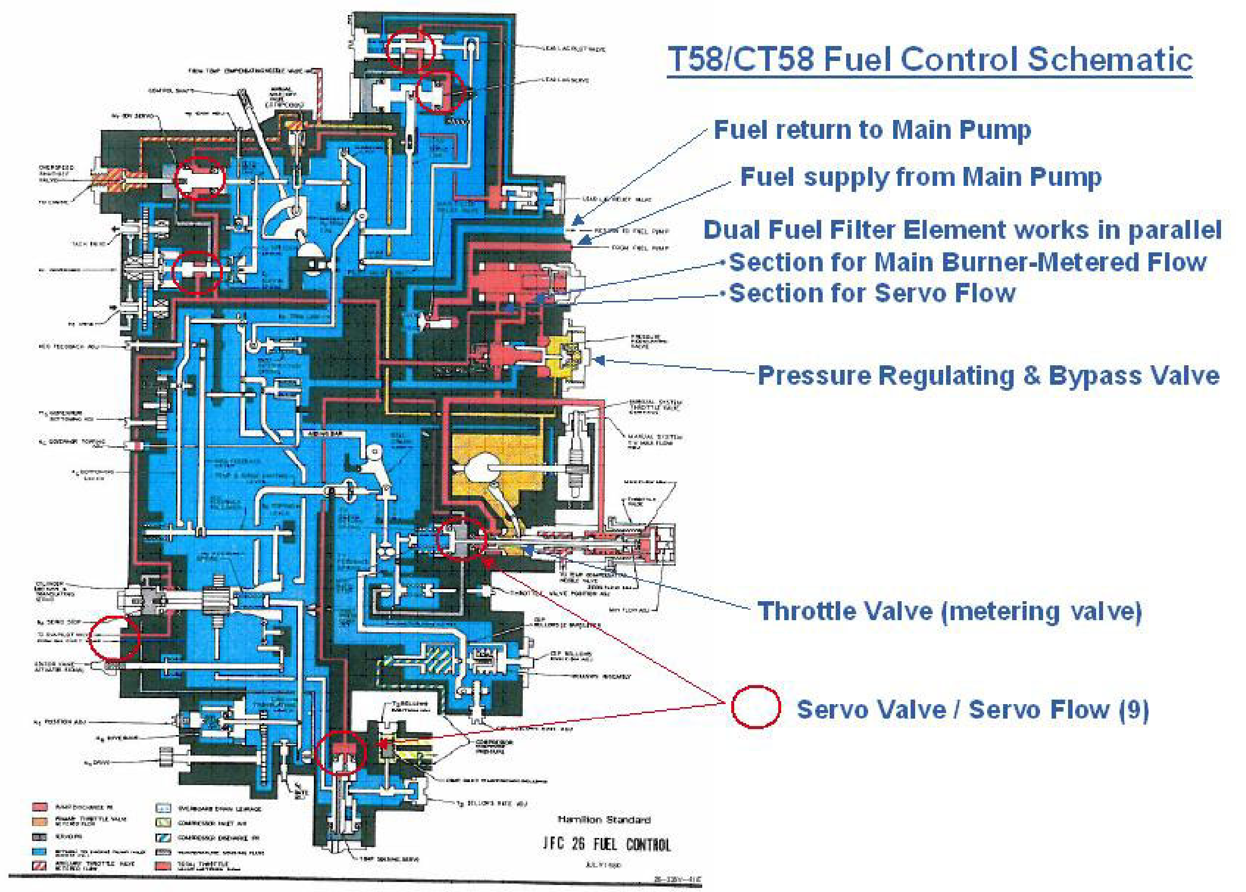
The JFC26 FCU (Figure 26) is a device designed to schedule the fuel flow required by the turboshaft engine to maintain a constant free turbine speed, and thus maintain a constant helicopter rotor speed[27]. Each engine contains one FCU that is connected to the engine accessory gearbox. The fuel control has a fuel metering section and a computing section. The metering section selects the rate of flow to the combustion chamber, based on information it receives from the computing section. High-pressure fuel is supplied to the FCU inlet by the engine-driven fuel pump. High-pressure fuel enters the dual element filter and then flows in two directions in parallel; one path of the filter feeds the metering section, and the other path feeds the computing section.

The metering section has a metering valve and a pressure-regulating valve (PRV), which maintains a constant pressure across the main metering valve by bypassing excess fuel back to the engine fuel pump inlet. The PRV senses the high-pressure fuel supply on one side of its diaphragm and senses the throttle valve downstream pressure on the other side. With the assistance of the loading spring on the low pressure side of the diaphragm, the PRV works to regulate a constant delta pressure across the throttle valve of about 60 psid regardless of supply pressure and bypasses excess flow back to main fuel pump inlet through the bypass valve. Metered fuel flow passes through the stopcock, which if opened, allows the fuel to pass to the control discharge port. The discharge port houses the over-speed shut-off valve. From the discharge port the metered fuel passes through the oil cooler, static filter, and flow divider on its route to the combustion chamber.

The computing section consists of nine servo valves throughout the control, three lead lag servos, over speed shut off servo, N_F governor servo, 3D cam servo, VG pilot valve, T_2 servo, and throttle valve servo. The servos are designed with half area servo heads to allow them to perform properly with dynamic changes in supply pressure. Supply pressure will change as pump speed and metered flow change. In the case of the VG pilot valve, the pilot valve will open to allow sufficient pressure to feed the VG actuator to the required direction to extend or retract the VG actuator. Pressure required to stroke the actuator is normally significantly lower than the pump high-pressure supply.

[27] The speed is determined by the position of the speed selector in the rotor speed-setting range.

Figure 26 Schematic depicting the JFC26 fuel control unit

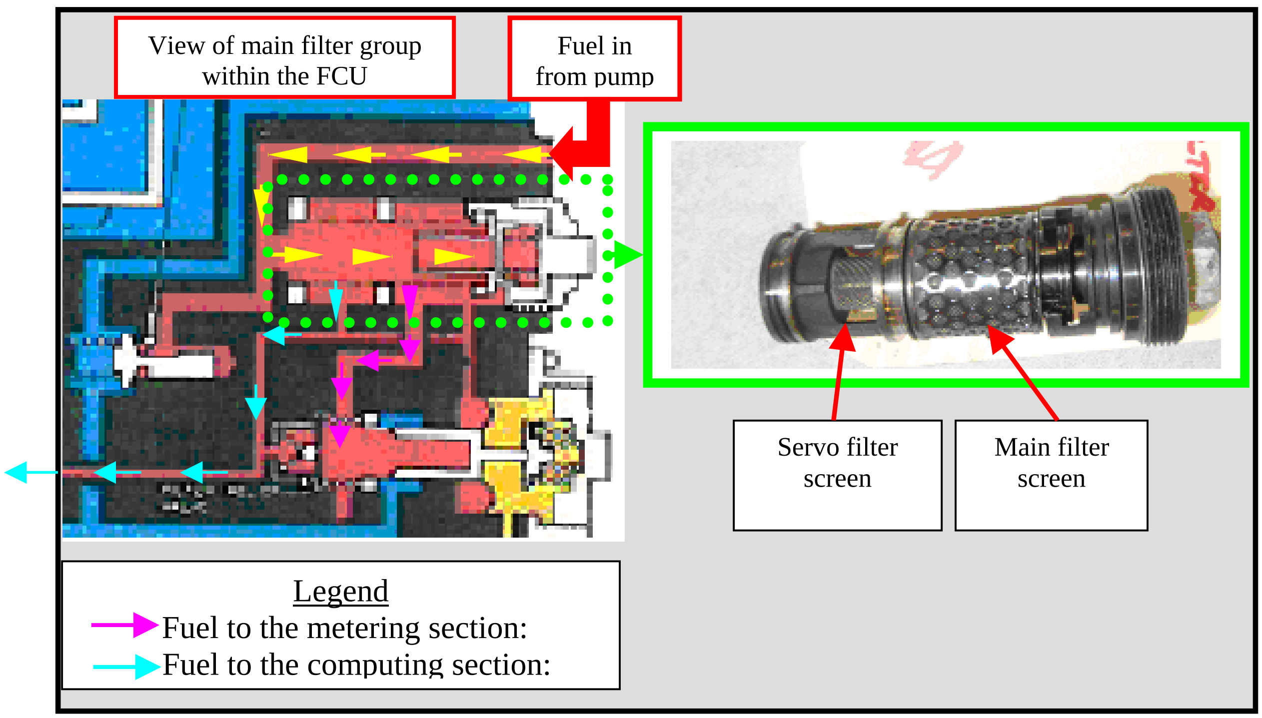


The major components of the fuel control system include: main and servo flow filter, pressure regulating valve, main and auxiliary throttle valve and servo, compressor discharge pressure sensing bellows, a power turbine over-speed-shutoff valve, a gas generator over-speed governor, an emergency throttle, and a stopcock.

D.6.1.1 Main Filter Group:

High-pressure fuel enters the dual element filter and then flows in two directions in parallel (Figure 27). One path of the filter feeds the metering section (Pressure Regulating & Bypass Valve), and the other path feeds the computing section (High Pressure Servo Supply port). The filter comprises a two-section screen-type 40-micron filter (a main screen and a servo screen), a cylinder, and a spring loaded bypass valve. Once inside the filter, the fuel is ported through the main screen to the PRV and ported through the servo screen (permanent filter) to feed all the control servos. A spring-loaded bypass valve is provided in the main filter group in the event that the metering section of the filter becomes clogged. If the filter goes into bypass mode (bypass valve cracks open at 25 psid) unfiltered fuel will be ported to the PRV and to the servo flow port.

Figure 27 Schematic of the FCUs main filter group



At the time of the accident, CHSI's engine fuel control filter maintenance program was based on the Sikorsky's Equalized Inspection Maintenance Program (EIMP) SA 4047-13 (manufacturers inspection program), which requires inspecting and cleaning the engine FCU filters every 150 hours. CHSI has since enhanced their maintenance program by inspecting the engine fuel control filter more frequently; as of February 15, 2009, they now perform the inspection at a 50 flight-hour interval[28]. As a comparison, the GE Aircraft Engines CT58 Turboshaft Maintenance Manual, section 73-20-1, page 216, dated July 31-2003, states that minor inspection/checks should be performed at intervals of 100 engine operation hours or 90 days calendar time, whichever occurs first.

The inspection and cleaning of the filter should be accomplished in accordance with the GE Aircraft Engines CT58 Turboshaft Maintenance Manual, section 73-20-1. The manual states; "position a small light inside the filter element and then visually inspect the element with a 10 power glass. It is necessary that an estimate of the degree of cleanliness be established. Count a representative sample of openings for a given area in the filter screen. Any element which has 70 percent or more of the available open area plugged is operating in partial or full bypass and therefore indicates the need to reduce the filter inspection/ cleaning time interval".

The maintenance manual also provides guidelines for establishing optimum filter inspection/cleaning intervals. According to the manual:

(a) Consistent plugging of 40-60 percent of available open areas. No change in the procedure is required.
(b) Consistent plugging of 61-70 percent of available open areas. Recommend 20% reduction in inspecting/cleaning interval. (e.g., current cleaning interval = 100 hours, reduce to 80 hours or less.)
(C) Consistent plugging of more than 70 percent of available open areas. Recommend 40% reduction in inspecting/cleaning interval.

CHSI's procedure is to remove and inspect the filter and replace it with a different (clean) one. The filter is subsequently cleaned using the Ultrasonic method recommended in the GE Maintenance Manual.

D.6.1.2 Pressure-Regulating Valve (PRV):

The two main purposes of the PRV are to maintain a constant delta pressure across the main metering valve and to bypass the unused or excess fuel flow from the fuel pump that the engine combustor and the control servo do not need.

For each position of the main metering valve piston, the PRV assumes a distinct position of equilibrium. Repositioning of the PRV, as a metering valve orifice area

[28] At the time of the accident the filter inspection was included in the CHI AAIP (Perkasie 133 inspection program) that the aircraft was on prior to going onto the CHSI CAIP (Grants Pass 135 inspection program). CHSI had revised it into their AIP and were working on the CAIP. The official revision date for when it was included in the CAIP is 2/15/09.

changes, results in the control of bypass fuel flow to maintain a constant pressure drop across the metering valve regardless of orifice area. For a balanced system in which engine fuel flow is constant at some rate selected by the computing section of the FCU, a specific main metering valve orifice area has been selected and the PRV is maintaining a constant ΔP across the orifice.

According to GE-Aviation, the main effect of a 'jammed' PRV is a bias to the fuel flow schedule. This bias will result in higher or lower gas generator speeds than demanded by the FCU depending on the direction of the flow shift. The main effect of a "sticking" PRV (one that is working but for one or more reasons has become difficult to move such as increased friction), the PRV will continue to function but at a slow or sluggish rate. Overshoot or undershoot on gas generator speed settings may occur and the levels of over/under shoot will depend on the level of "sticking" or increased friction of the valve and difference of the speed demand setting to the actual speed at initiation of the event.

D.6.1.3 Manual Throttle Valve:

The FCUs contain a manual throttle valve in the event that the main throttle control system becomes inoperative.

D.6.1.4 T_2 Servo and Bellows Group:

The T_2 bellows sensor is used to detect engine inlet temperature and provide a mechanical input to the fuel control to be used in scheduling engine fuel flow. The bellows will expand or contract with changes in inlet temperature, which through a sensing lever, servo poppet and servo valve, rotates the 3D cam to bias the fuel schedule. The T_2 servo and bellows group comprises the following: the aspirator and bellows group and the T_2 servo piston assembly (Figure 28). The T_2 bellows group is protected from the environment by a position adjusting cover[29]. The group consists of the bellows tube, an aspirator, and a housing (in which the bellows position adjusting screw, bellows assembly, compression helical spring, and spring retainer cap are contained). The bellows assembly, compression helical spring, and spring retainer cap are retained within the housing by a snap retaining ring that fits into the position adjusting screw. Air from the compressor inlet of the engine is drawn past the temperature sensing bellows by the action of the bellows aspirator. The T_2 servo valve assembly consists of a fluid flow restrictor, a spring-loaded poppet valve, and a servo piston, which engages the 3-D Cam by means of a

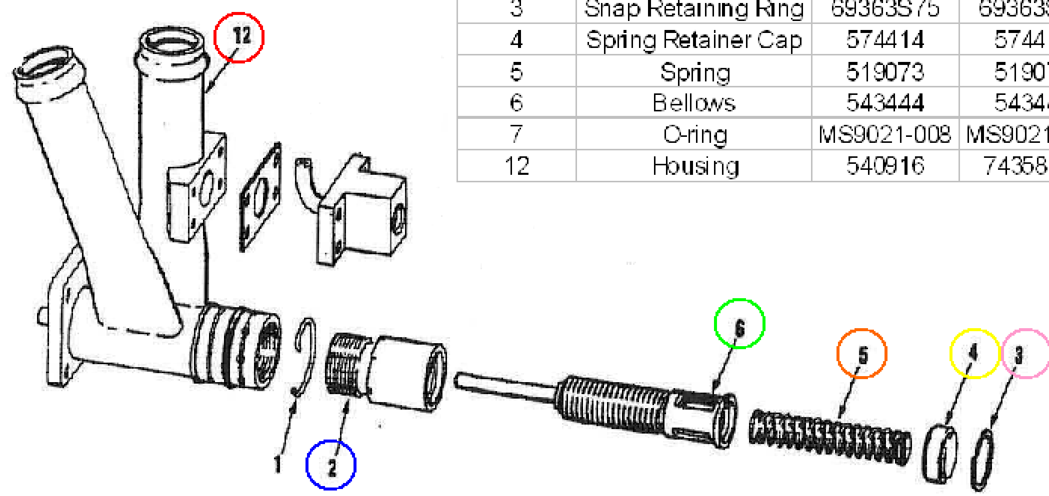
[29] According to note 1 in a Hamilton Sundstrand drawing, the Position Adjusting Cover can be manufactured from either Plastic, grade C (HS83, GR9 or GR10) or from Aluminum (AMS4117). The plastic version also has an option to bond on a separate piece to form the end face.

The intended purpose of the position adjusting cover is to prevent dust or debris from contaminating the T_2 bellows assembly; it is not intended as a means of retention for the T_2 bellows assembly. An internal retaining ring provides retention of the bellows assembly. The function of the fuel control unit is unaffected without the position adjusting cover in a clean environment, providing the retaining ring remained in place."

gear rack on the side of the piston. Motion of the bellows is transmitted to the T_2 servo poppet valve by the temperature sensing bellows lever. The poppet valve bleeds servo pressure from the large end of the servo through a nozzle, translating the servo in accordance with bellows motion. The rack on the servo piston mates directly with the spline on the 3-D Cam and rotates the cam as a function of engine compressor air inlet temperature.

Figure 28 Schematic of the T2 Aspirator and Bellows Group

Figure #	Part	IPB	X Section
2	Adjusting Screw	574415	742574-1
3	Snap Retaining Ring	69363S75	69363S75
4	Spring Retainer Cap	574414	574414
5	Spring	519073	519073
6	Bellows	543444	543444
7	O-ring	MS9021-008	MS9021-008
12	Housing	540916	743589-1



Aspirator and Bellows Group

D.6.1.5 3-D Cam:

The FCU contains a 3-D Cam, which is positioned axially by the N_G governor servo and rotationally by the compressor air inlet temperature servo. The axial and rotational positions, as reflected by the cam contour, are transmitted by suitable linkage to the control rollers. The axial and rotational positions, as reflected by the cam contour, are transmitted by linkage to the control roller and stator vane actuator positions.

D.6.1.6 High Pressure Relief Valve:

The FCU contains a high-pressure relief valve that is located downstream from the control inlet to protect the components of the control against excessive fuel pressures. When the fuel pressure becomes high enough to overcome the spring force (about 1000 psi), the valve opens and relieves excess pressure.

D.6.1.7 Fuel Flow through FCU:

As part of this investigation, the NTSB asked GE Aviation to provide the NTSB with information summarizing the fuel flow and corresponding rates through the FCU for engine speed at idle, mid-range, and topping (Figure 29).

Figure 29 - Information on fuel flow through the FCU

CT58 Fuel Control Flow Split, FCU 6003T91
Design Specification M50T1197-S5

	Idle ~60% Ng	Mid Range ~80% Ng	Topping ~102% Ng	
Pump Flow; Wp pph	1600	2200	2800	
Control Internal Leakage; Ws pph	260	500	925	
Servo Flow / Flow thru Servo Filter	16%	23%	33%	% of Pump Flow
Flow thru Main Filter; Wp - Ws pph	1340	1700	1875	
Metering Section Flow	84%	77%	67%	% of Pump Flow
Flow thru Main Filter; Wmf pph	1340	1700	1875	
Burn Flow; Wf pph	85	450	800	
Flow to fuel nozzles	6%	26%	43%	% of Main Filter Flow
Flow thru Bypass Valve; Wbp pph	1255	1250	1075	
Wmf - Wf (flow back to pump)	94%	74%	57%	% of Main Filter Flow

Above flows are approximate and will vary with actual component performance; fuel type & temp; time since new, etc.
Flows are for normal steady state conditions only. Flow split will change during transients such as accels and decels.

D.6.2 FCU Investigative Findings:

From August 7 - 11 2008, the Airworthiness Group visually accessed the FCUs during the on-scene portion of the accident investigation. The assessment revealed that a FCU having part number 725725-6 and serial number 72835BR remained attached to the Number 1 engine (left) and that an FCU having part number 725725-5 and serial number 49882 remained attached to the Number 2 engine (right). The exterior surface of both FCUs had evidence of discoloration, sooting and ash deposits consistent with being exposed to high external temperatures.

On August 11, 2008, both CT58-140 engines (including their respective FCUs) were removed from the accident site by helicopter[30] and transported by road to Columbia Helicopters, Inc., located in Aurora, Oregon for further examination.

The Airworthiness Group convened at the Columbia Helicopters, Inc facility to conduct the examination and teardown of the engines. On August 13, 2008, the Group examined the Number 1 engine, which included the removal of its FCU. On August 14, 2008, the Group examined the Number 2 engine, which included the removal of its FCU. The examinations were conducted under the supervision of the NTSB and witnessed by representatives from the Federal Aviation Administration, Columbia Helicopters, Inc, General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service and CHSI.

On August 15, 2008 the NTSB instructed a representative from Columbia to package and ship the FCUs to the NTSB offices in Washington D.C. As directed by the NTSB, the FCUs (and associated parts) were shipped by Columbia to the NTSB offices in Washington D.C and received on August 26, 2008[31]. The Airworthiness Group removed and identified the "as-received" components at the NTSB offices on August 28, 2008[32]. The FCUs (and associated parts) were then shipped to the FAA located in Windsor Locks, Connecticut.

The Airworthiness Group convened in Windsor Locks, Connecticut at the Hamilton Sundstrand facility on September 18 and 19, 2008 to evaluate the condition of the FCUs. Representatives from Hamilton disassembled both FCUs under the supervision of the NTSB and witnessed by representatives from General Electric Aviation Engines, Sikorsky Aircraft, United States Forest Service, and CHSI.

D.6.3 FCU History (Left Engine):

A review of the maintenance records provided by CHSI, revealed that on June 6, 2008, aircraft N612AZ experienced slight fluctuations in the Number 1 engine's NG while at high torque settings. On June 7, 2008, CHSI addressed this issue by replacing the FCU on the Number 1 engine. FCU having part number (P/N) 725725-6 and serial number (S/N) 24928 was removed and was replaced with an FCU having P/N 725725-6 and s S/N 72835BR. At the time of the accident, this FCU had been installed on the engine for about 123 flight hours.

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[30] The engines were removed from the accident site via long line operation, using a Lama aircraft on August 11, 2008.

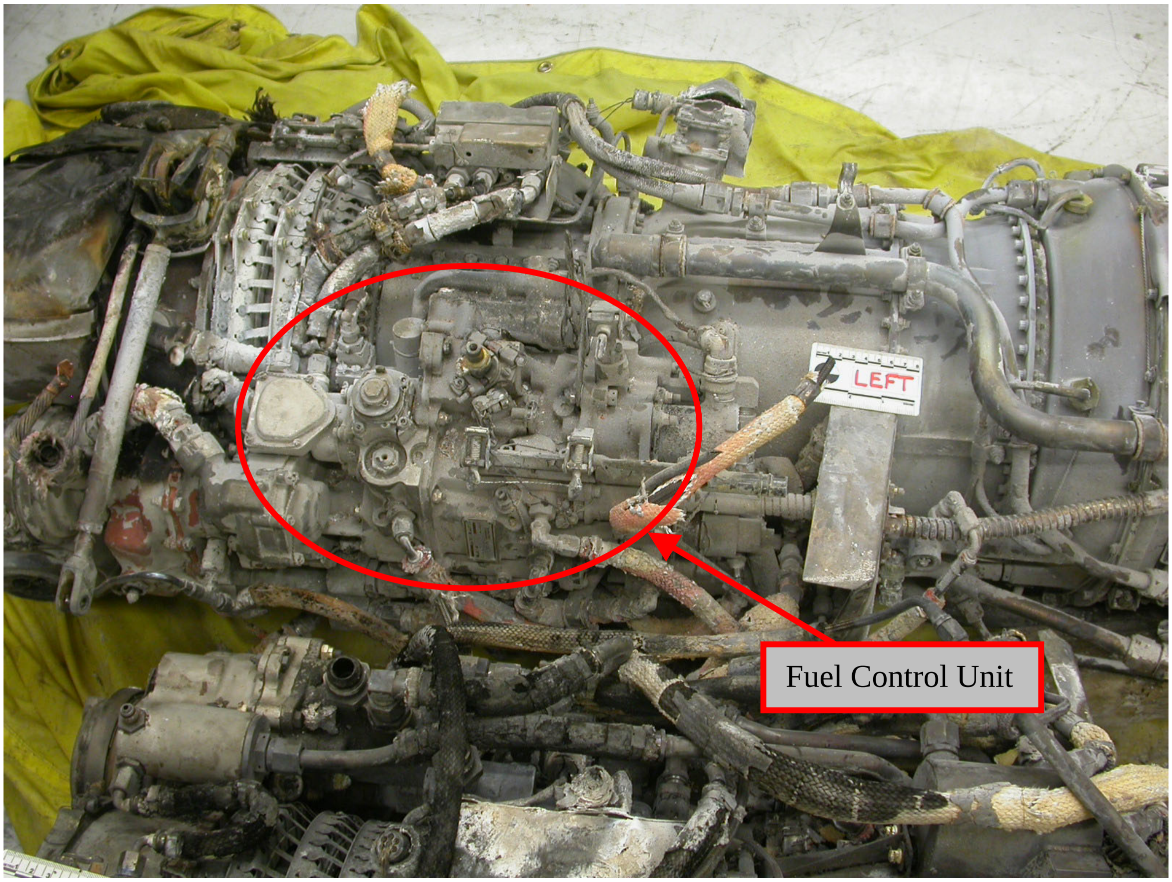
[31] Columbia Helicopters, Inc video recorded the packaging of the all engine and FCU parts.

[32] Two shipping containers (light weight cardboard box) containing the left and right engine FCUs arrived, via FedEx Express, in Washington D.C. on August 26, 2008. The NTSB in the presence of representatives from GE – Aviation and Carson Helicopters Services Inc. opened the two shipping containers. Upon opening the shipping containers, an inventory of the hardware revealed that the following components from FCU number 1 were not present: metal position adjusting cover, snap retainer ring, spring retainer cap, spring and bellows. A review of the video recording taken by Columbia Helicopters, Inc revealed that the missing parts were not present at the time of packaging and therefore were not packaged and shipped to the NTSB.

D.6.4 FCU Serial Number 72835BR (Left Engine) Examination:

As received at Columbia Helicopters Inc., FCU having S/N 72835BR remained intact and attached to the left engine (Figure 30). The FCU was removed from the engine and placed on a bench for further examination. The Airworthiness Group determined that this FCU could not be functionally tested (mechanically or hydraulically) due to the thermal damage it sustained during the post-crash fire. The Airworthiness Group visually examined and documented the following features on the units: physical and structural condition, position of control inputs, cleanliness of filters, and condition of mechanical linkages.

Figure 30 Photograph of the left engine's FCU, serial number 72835BR, as installed on the engine



Overall, the FCU remained intact, all attachment hardware and safety wire were present and secured, and all of its external component visually appeared intact and in place. White oxide residue was present on the exterior of the main housing including within the T_2 aspirator supply tubes. Upon disassembly of the unit, the inside of the FCU was found dry (no fuel present) and coated with a thin layer of black material consistent with residue from coked fuel. Charred remnants of elastomer and polymer seals remained in the areas where seals would normally be present. No metal contamination was visually observed within the upper or lower control housings and all of the internal components were observed to have no mechanical damage. The machined surfaces of the main housing and internal components were observed with no scoring, galling, scratch marks or mechanical damage.

D.6.4.1 Manual Throttle Valve:

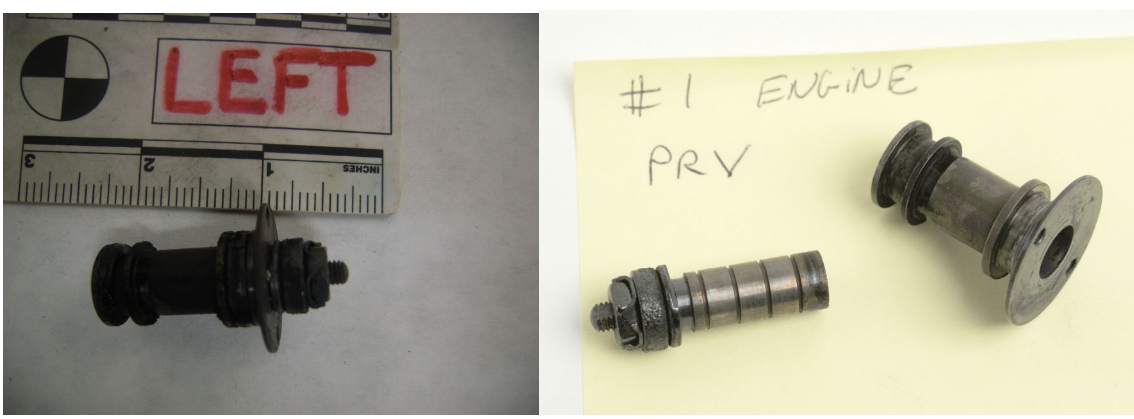
From August 7 through 11 2008, the Airworthiness Group conducted the on-scene phase of the investigation at the accident site. Prior to the engine removal, the Airworthiness Group visually inspected the position of the manual throttle valve. Inspection revealed that the main throttle spindle remained positioned against its full open throttle position stop. The emergency throttle linkage remained connected; it was observed in the closed position. The emergency throttle measured 1.76 inches retracted indicating it was in the shut off position.

D.6.4.2 Pressure-Regulating Valve (PRV) Observations:

The PRV assembly was removed from the FCU at Columbia Helicopter Inc. Visual inspection revealed that the PRV assembly remained intact. Its exterior surface was found discolored and had dry remnants of thermally damaged seals in its grooves (Figure 31). No further examination of the PRV was conducted at Columbia Helicopter Inc.

Figure 31

Photograph of the PRV removed from the left engine's FCU, serial number 72835BR

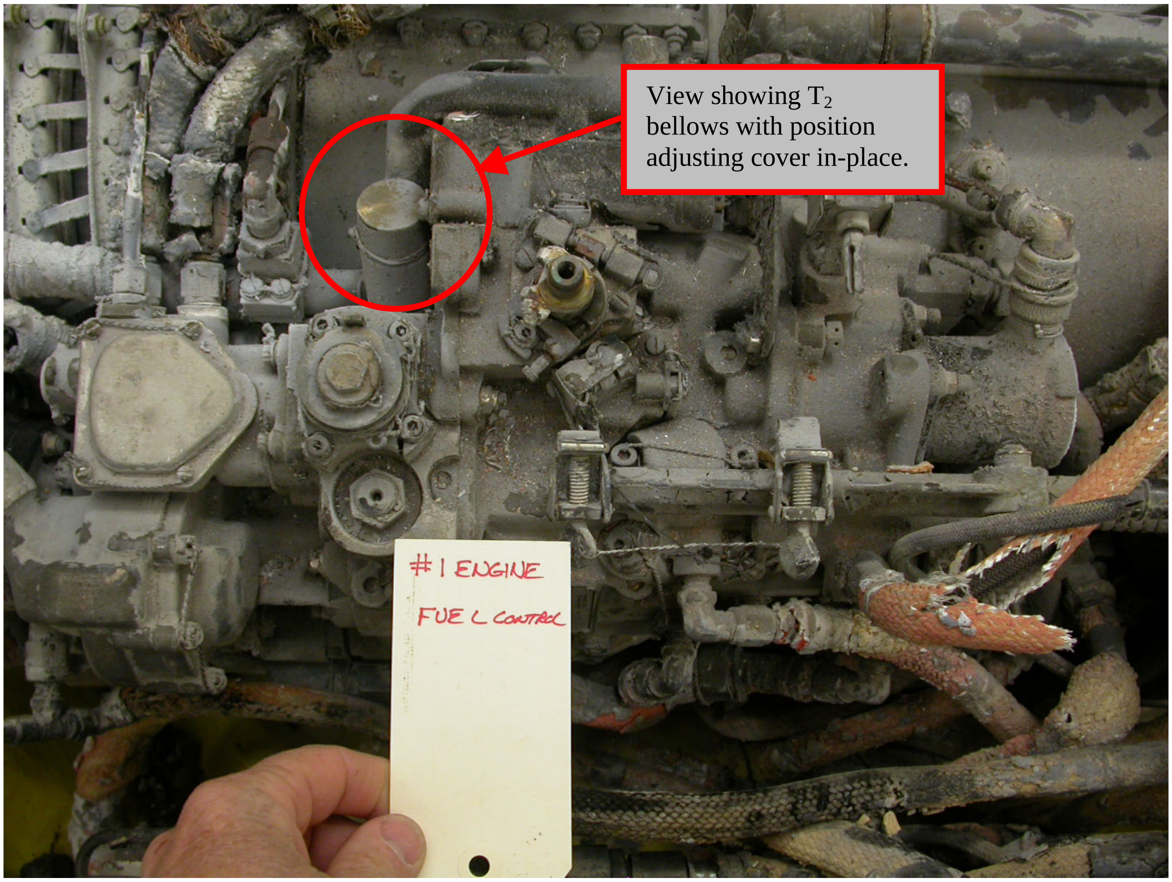


The PRV was further examined at Hamilton Sundstrand at which time its spool and sleeve assembly were separated from each other using an arbor press (very little force was required). After the spool was removed from the sleeve, erosion was observed on the edge of the spool near the metering edge. According to a representative from Hamilton Sundstrand, the erosion appears normal and is indicative of a high time valve spool. No evidence of contamination was observed in the spool's balance grooves and, other than soot, the sleeve appeared clean. Linear scratches were observed on the spool and according to Hamilton Sundstrand these scratches looked like a normal wear pattern. Witness marks of coked fuel were seen on the outer diameter (O.D.) of the spool that matched the opening of the metering windows in the sleeve. The Airworthiness Group submitted the PRV assembly (spool and sleeve) to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 09-08-121 in the Safety Board's public docket for accident, LAX08PA259).

D.6.4.3 T_2 Servo and Bellows Group Observations:

Visual examination of the T_2 servo assembly was conducted at Columbia Helicopters Inc. The examination reveled that its position adjusting cover (metal) remained intact and lock wired in place (Figure 32).

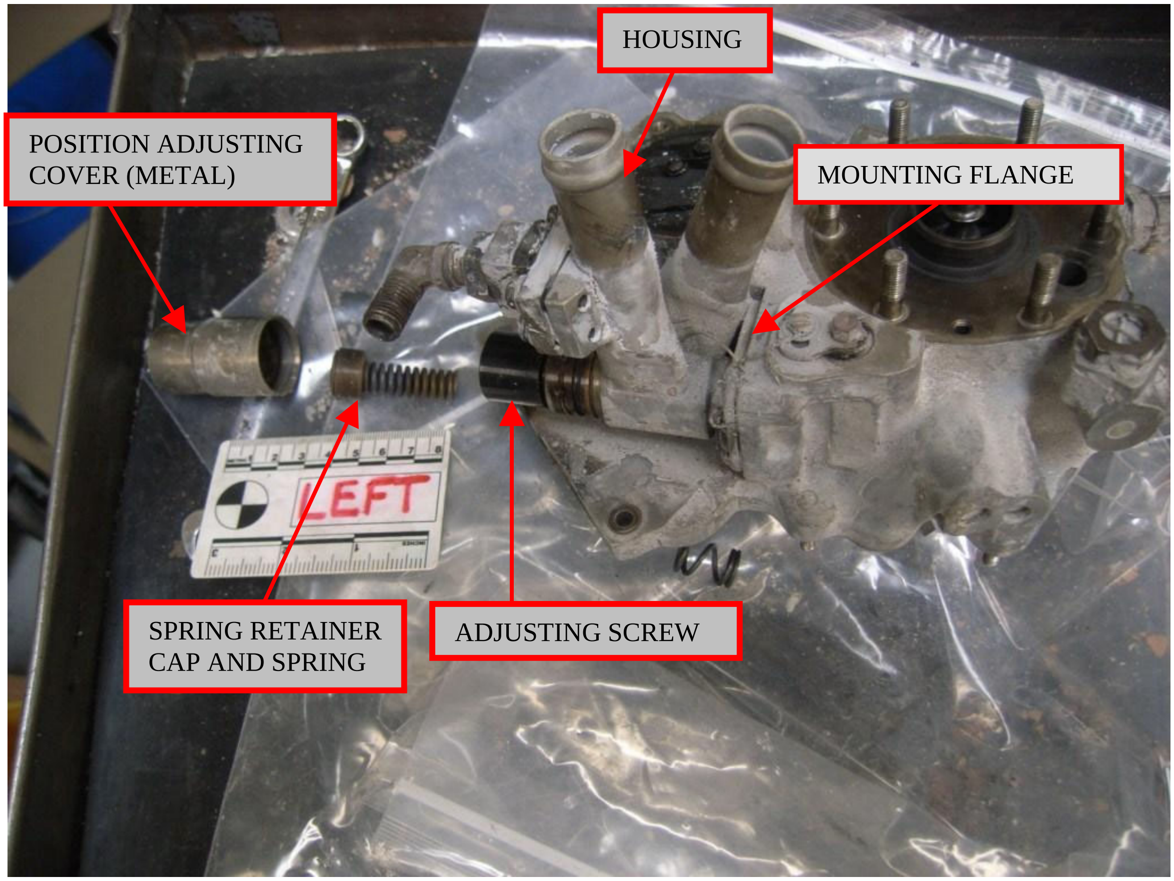
Figure 32 Photograph of the left engine's FCU, serial number 72835BR



As directed by the NTSB, a representative from Columbia cut the lock wire and removed the position adjusting cover from the bellows housing exposing the aspirator and bellows group. The aspirator bellows assembly was observed to be loose within its housing. When the housing assembly was rotated down, the following components fell out of the housing: snap retainer ring, spring retainer cap, spring and bellows (Figure 33). These are the parts that later went missing as described in footnote 32.

Figure 33

Photograph of the left engine's FCU, serial number 72835BR, showing T2 Servo and Bellows Group



Visual examination at Hamilton Sundstrand revealed that one end of the adjusting screw remained attached to its mating housing assembly; its other end was found fractured 360-degrees around its circumference. The outer lips of the adjusting screw's snap ring grooves appeared to have been extruded as if the snap ring and plug were forced outward. Examination revealed that the T_2 servo assembly-mounting flange appeared warped. When measured with a flat straight edge (machinist's parallel bar) light could clearly be seen between the mounting flange and the FCU housing. The Airworthiness Group submitted the housing assembly (containing the adjusting screw) to the National Transportation Safety Board Materials Laboratory for metallurgical examination of the fracture surface on the adjusting screw and an evaluation of the "warped" mounting flange (Reference NTSB Materials Laboratory Factual Report number 08-121 in the Safety Board's public docket for accident, LAX08PA259).

D.6.4.4 Main Filter Group Observations:

According to CHSI, the filter was last inspected during a Period 1 Phase 2 inspection on June 30, 2008 at an aircraft total time of 35,328.7 flight hours. At the time of the accident, the main filter group had accumulated about 71.2 total flight hours since being inspected. As per the CHSI inspection program at the time they had 78.8 hours remaining until the filter needed to be re-inspected.

During the examination at Columbia Helicopters Inc., a visual examination of the main filter screen revealed that the main filter group was dry and exhibited signs of heat distress. Overall, the filters visually appeared clean (no contamination) and were not blocked.

The main filter group was re-examined at Hamilton Sundstrand. The filter was disassembled in their laboratory, and when held up to a light, light was visible through the filter elements of both the main screen and the servo screen. Visual examination, using a microscope, revealed that both the main screen and the servo screen had some kind of fiber strands present on their inside surfaces. The Airworthiness Group submitted the main filter assembly to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 08-121 in the Safety Board's public docket for accident, LAX08PA259).

D.6.4.5 3-D Cam Observations:

Investigation of the 3-D Cam was conducted at Columbia Helicopter Inc., and then again at Hamilton Sundstrand. Position measurements of the 3-D Cam were documented at both facilities. The objective of the measurements was to establish an approximation of the last speed being sensed by the cam before it came to rest (rest meaning that the 3-D Cam was no longer traveling along its servo axis). The measurements were not to specifically determine the position of the temperature & surge limiting lever and stator vane actuator follower. According to a Hamilton Sundstrand representative, the "as received" position of the 3-D Cam is not an accurate indication of the final engine speed or the engine speed prior to the event. Cam rest position can vary greatly.

In normal operation, during spool down of speed towards the "off" direction, the "Rest Position" of the 3-D Cam in its axial direction is not to any fixed point or hard stop. The servo contains built-in friction from its dynamic seals, feedback spring, fluid and component temperature and in conjunction of its actual servo pressure to offset the friction for force balance, the servo will most likely not end up in the same position each time it cycles down to the "off" direction. In the event where a 3-D Cam servo may come to final rest at high operating conditions, it will require a reduction in servo fuel pressure to decay in the servo cavity and then rely on the Cam Translating Lever Spring to attempt to drive it alone (without additional servo pressure) to a point where the spring force comes into force balance with existing opposing forces from the seals, levers, spring, or other damage that may occur.

The 3-D Cam speed servos and temperature servos were very hard to rotate by hand pressure. According to a Hamilton Sundstrand representative, the "as received" 3-D Cam was positioned to its full hot position (maximum T_2); it was not centered within the unit. According to a Hamilton Sundstrand representative, the position of the "as received" 3-D Cam is an indication that the T_2 bellows was heated to a point where it drove the 3-D Cam to its full hot position [33].(Nominal setting is about 59 degrees F T_2.).

At Columbia, measurements were taken from the face of the FCU housing to the face of the 3-D Cam using a 6-inch scale. At Hamilton Sundstrand, measurements, using a depth micrometer, were taken from the cam face to the housing surface face of FCU in the as received position and then with the cam pushed fully down. Test results showing an approximation of the sensed speed of the "as received" 3-D Cam are indicated in (Table 2).

D.6.5 FCU History (Right Engine):

A review of the maintenance records provided by CHSI, revealed that FCU having part number 725725-5 and serial number 89674BR was removed on 5/12/08, due to "the engine hanging at 50% torque", and sent to Columbia Helicopters Inc. for repair. The FCU was replaced with an FCU having part number 725725-5 and serial number 49882. At the time of the accident, this FCU had been installed on the engine for about 194 flight hours.

According to Columbia Helicopters Inc., FCU having serial number 89674BR was found contaminated with metal during disassembly.

D.6.6 FCU Serial Number 49882 (Right Engine) Examination:

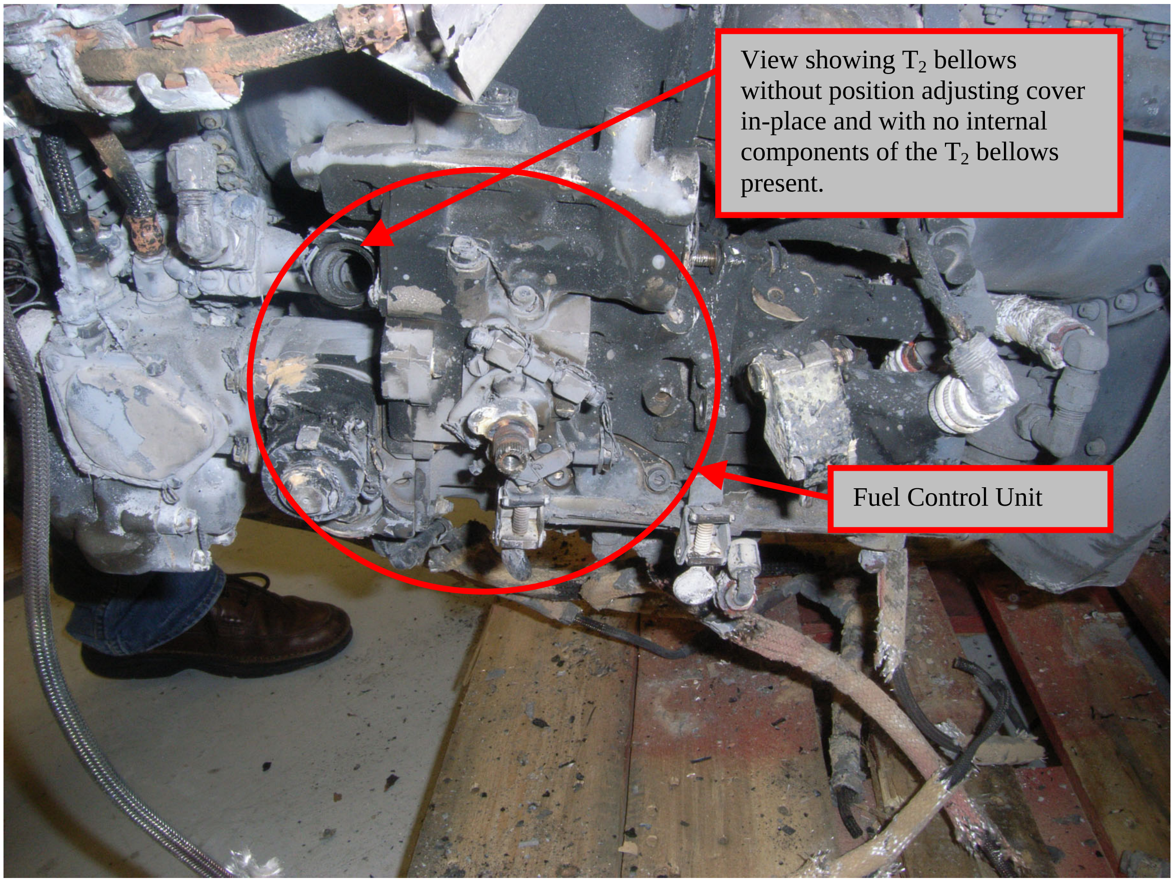
The FCU was removed from the engine and place on a bench for further examination. The Airworthiness Group determined that this FCU could not be functionally tested (mechanically or hydraulically) due to the thermal damage it sustained during the post-crash fire.

As received at Columbia Helicopters Inc., FCU having part number 725725-5 and serial number 49882 remained intact and attached to the right engine (Figure 34). The Airworthiness Group determined that this FCU could not be functionally tested (mechanically or hydraulically) due to the thermal damage it sustained during the post-

[33] The liquid in the bellows is sensitive to temperature from –65 to + 160 degrees F, expanding or contracting to a balanced steady state condition, as required. The bellows follower pin transmits the bellows action to a lever arm that actuates a spring-loaded poppet valve. The valve movement determines the amount of servo pressure that is bled from the T_2 servo chamber into the body, thereby allowing high pressure to translate the T_2 servo piston, which is geared to rotate the 3-D Cam. Thus, the compressor air inlet temperature signal establishes the rotational position of the 3-D Cam.

crash fire. To verify the integrity of the FCU, the Airworthiness Group visually examined and documented the following features on the units: physical and structural condition, position of control inputs, cleanliness of filters, and condition of mechanical linkages.

Figure 34 Photograph of the right engines FCU, serial number 49882, as installed on the engine



Overall, the FCU remained intact, all attachment hardware and safety wire were present and secured, and all of its external components (other than the T_2 bellows group) visually appeared to be intact and in place. White oxide residue was present on the exterior of the main housing including within the T_2 aspirator supply tubes. Upon disassembly of the unit, the inside of the FCU was found dry (no fuel present) and coated with a thin layer of black material consistent with residue from coked fuel. Charred remnants of elastomer and polymer seals were found in the areas where the seals would normally be present. No metal contamination was visually observed within the upper or lower control housings and all of the internal components were observed to have no mechanical damage. The machined surfaces of the main housing and internal components were observed with no scoring, galling, scratch marks or mechanical damage.

D.6.6.1 Manual Throttle Valve Observations:

From August 7 – 11, 2008, the Airworthiness Group conducted the on-scene phase of the investigation at the accident site. While on-scene, the Airworthiness Group visually inspected the position of the manual throttle valve. Inspection revealed that the main throttle spindle remained positioned against its full open throttle position stop. The emergency throttle linkage remained connected; it was observed in the closed position.

D.6.6.2 Pressure-Regulating Valve (PRV) Observations:

The PRV assembly was removed from the FCU at Columbia Helicopter Inc. Visual inspection revealed that the PRV assembly remained intact its exterior surface was found discolored and had dry remnants of thermally damaged seals in its grooves. No further examination of the PRV was conducted at Columbia Helicopter Inc.

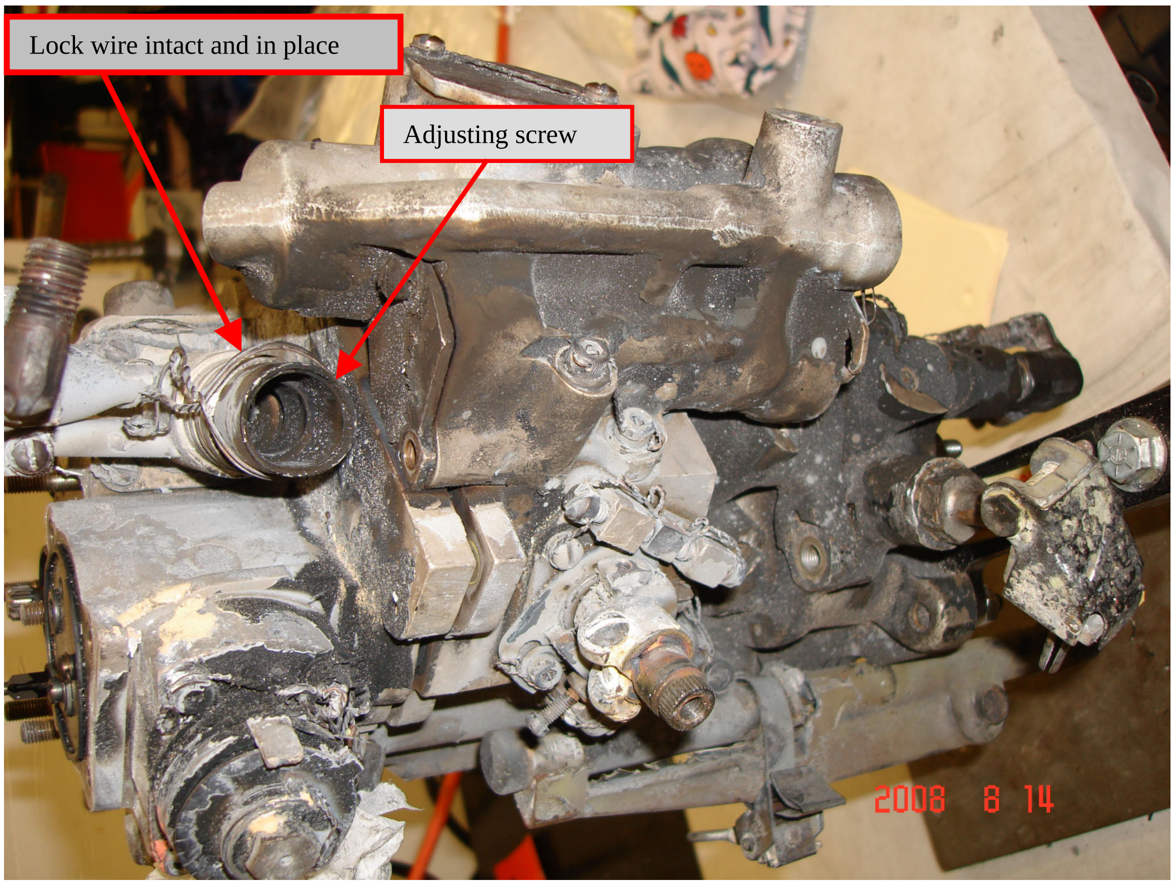
The PRV assembly was further examined at Hamilton Sundstrand. Prior to separating the spool from the sleeve, the remaining dry, hardened seals were cleaned out of their grooves and disposed. An arbor press was used to separate the spool from the sleeve with very little force required. Small pieces of black material, which fell off of the PRV assembly during separation, were collected. The material was submitted to the National Transportation Safety Board Materials Laboratory for metallurgical examination.

Examination of the PRV assembly, using a microscope, revealed unidentified fiber strands resting in the second balance groove from the metering end of the spool. No erosion was observed on the metering end of the spools O.D. Witness marks of coked fuel were observed on the O.D. of the spool that matched the opening of the metering windows in the sleeve. The Airworthiness Group submitted the PRV assembly (spool and sleeve) to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 08-121 in the Safety Board's public docket for accident, LAX08PA259).

D.6.6.3 T_2 Servo and Bellows Group Observations:

Visual examination of the T_2 servo assembly was conducted at Columbia Helicopters Inc. The examination reveled that its position adjusting cover was not present (Figure 35), but its lock wire remained in place and intact. As received at Columbia Helicopters Inc., the aspirator bellows assembly group (snap retainer ring, spring retainer cap, spring and bellows) was not present.

Figure 35 Photograph of the right engine's FCU, serial number 49882, after removal from the engine



Visual examination, at Hamilton Sundstrand, revealed that one end of the adjusting screw remained attached to its mating housing assembly; its other end was found fractured 360-degrees around its circumference. The opening of the adjusting screw appeared oval shaped with sooting consistent with exposure to fire and heat. The outer lips of the adjusting screw's snap ring grooves appeared to have been extruded as if the snap ring and plug were forced outward. Examination revealed that the T_2 servo assembly-mounting flange appeared warped. When measured with a flat straight edge (machinists parallel bar) light could clearly be seen between the mounting flange and the FCU housing.

The Airworthiness Group submitted the housing assembly (containing the adjusting screw) to the National Transportation Safety Board Materials Laboratory for metallurgical examination of the fracture surface on the adjusting screw and an evaluation of the “warped” mounting flange (Reference NTSB Materials Laboratory Factual Report number 09-08-121 in the Safety Board’s public docket for accident, LAX08PA259).

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During the examination at Columbia Helicopters Inc., a visual examination of the main filter revealed that the main filter group was dry and exhibited signs of heat distress. Overall, the filters visually appeared clean (no contamination) and were not blocked.

The main filter group was re-examined at Hamilton Sundstrand. The filter was disassembled in their laboratory, and when held up to a light, light was visible through the filter elements of both the main screen and the servo screen. Visual examination, using a microscope, revealed that both the main screen and the servo screen had some kind of fiber strands present on their inside surfaces. The Airworthiness Group submitted the main filter assembly to the National Transportation Safety Board Materials Laboratory for metallurgical examination (Reference NTSB Materials Laboratory Factual Report number 08-121 in the Safety Board’s public docket for accident, LAX08PA259).

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In normal operation, during spool down of speed towards the “off” direction, the “Rest Position” of the 3-D Cam in its axial direction is not to any fixed point or hard stop. The servo contains built in friction from its dynamic seals, feedback spring, fluid and component temperature and in conjunction of its actual servo pressure to offset the friction for force balance, the servo will most likely not end up in the same position each time it cycles down to the “off'” direction. In the event where a 3-D Cam servo may come to final rest at high operating conditions, it will require a reduction in servo fuel pressure to decay in the servo cavity and then rely on the Cam Translating Lever Spring to attempt to drive it alone (without additional servo pressure) to a point where the spring force comes into force balance with existing opposing forces from the seals, levers, spring, or other damage that may occur.

The 3-D Cam speed servos and temperature servos were very hard to rotate by hand pressure. A Hamilton Sundstrand representative stated that the 3-D Cam remained at its low speed setting and that the cam remained nominally centered within the unit (Nominal setting is about 59 degrees F T_2.).

At Columbia, measurements were taken from the face of the FCU housing to the face of the 3-D Cam using a 6-inch scale. At Hamilton Sundstrand, measurements, using a depth micrometer, were taken from the cam face to the housing surface face of FCU in the as received position and then with the cam pushed fully down. Test results showing an approximation of the sensed speed of the "as received" 3-D Cam are indicated in (Table 2).

D.6.7 JFC26 Fuel Control Units (FCU) Contamination Events:

During the course of this investigation, the NTSB has become aware of a condition that can cause power fluctuations (N_G), erratic operation or slow acceleration of CT58-140 engines. The condition has been linked to foreign solid contaminants, smaller than 40 microns) entering and lodging within the engine fuel control pressure regulating valve (PRV), in turn causing the PRV to bind or seize.

Specifically, the NTSB has learned of two cases in which pressure-regulating valves (PRV) were found to contain contamination. One PRV had been removed from a CHSI. JFC26 FCU in 2004 and the other PRV had been removed from a JFC26 FCU from a Hayes Helicopter that crashed in Canada in 2002 (Reference Transportation Safety Board of Canada report number A02P0320).

The NTSB reviewed a report generated by GE Aviation to Columbia Helicopters, dated December 5, 2005 regarding Hamilton Sundstrand's engineering evaluation of a PRV that was sent to GE Aviation by Columbia Helicopters Inc. (the overhaul and repair facility). According to the GE Aviation report, the Carson Helicopter Inc. FCU was removed from the helicopter due to engine N_G fluctuations. Testing, by Columbia Helicopters Inc., revealed that the FCU exhibited a stuck PRV. The GE report states: "*Lab analysis cites silica fibers (fiberglass), and hard angular oxides trapped in the clearance area between the inner diameter and outer diameter of the valve assembly for the seizure of the PRV*". "*The particle size is 2.5 to 25 micron, which is smaller than the 40 micron size of the fuel filter, but definitely large enough to stick the valve and cause scoring*".

The NTSB reviewed report A02P0320, which was generated by the TSB of Canada. According to the report, during an engine check flight, "*while the helicopter was climbing through about 1000 feet above sea level at about 65 knots, the crew became* *aware of an intensifying whining sound which was followed by a single, loud bang. Immediately the number 1 engine lost power and the number 2 engine did not automatically compensate for the power loss*." Investigation revealed that a failure of the helicopter main gearbox #1 input pinion forward bearing, caused the Number 1 engine to lose power due to loss of load, overspeed and shutdown. According to the report, the Number 2 engine did not respond quickly enough to the increased load demand on it from the main rotor, resulting in a hard autorotation landing on a road. According to the report, "*The combination of the misadjusted stator vane actuator, the fuel control unit topping settings, and a sticking pressure regulating valve prevented the number 2 engine, when number 1 engine lost power, from assuming the total load.*" "*To examine the operation and performance of the various fuel delivery components of each engine in their undisturbed state, the components were attached to a slave engine and run in a test cell at an independent, approved engine overhaul facility in Richmond. The components included the FCU, flow divider, fuel purifier, and stator vane actuator (SVA). The Ng and Nf tach-generators were also tested; all four operated normally. The test cell runs were unremarkable with two minor exceptions. When the number 1 engine components were run, the engine "rumbled" during acceleration tests, likely as a result of poor airflow. When the number 2 engine components were run, the engine ran too cool, requiring adjustment to the SVA linkage, and the normal topping limit was not reached." "The same components were then bench-tested and disassembled by an FAA-approved facility in the United States under the direct supervision of TSB investigators. With the exception of the FCU from the number 2 engine, all the components tested within specification limits and were unremarkable.* According to the report, "*The number 2 FCU fuel filter contained a significant amount of mixed contaminants; however, it could not be determined if the filter had gone into bypass. Further disassembly revealed that the pressure-regulating valve (PRV) in the FCU was jammed with contaminant significantly different to that found in the filter. Earlier bench tests had showed that the PRV was sticking; a sticking or immoveable PRV would cause unstable SVA operation, engine starting difficulties, and inconsistent topping settings. Collective experience from US operators of this FCU show that sticking or jammed PRVs also lead to unpredictable and degraded engine performance.*" "*Microscopic and infra-red analysis of the debris found in the PRV determined that it comprised particulates of chip board, bleached cellulose, paint, and metal; the FCU filter contaminant comprised cellulose, paint, human hair, and unidentifiable fibers. Laboratory examination of the debris found in the airframe fuel filter and the aft fuel tank boost pump revealed particulates of mainly chip-board, cellulose, paint, silk, human hair, and polyethylene. The source(s) of these various contaminants, or the time of their introduction, could not be determined. The aft fuel tank had been removed, repaired, and replaced on 27 November 2002*."

The Airworthiness Group queried the Aviation Safety Communiqué (SAFECOM[34]) database for GE CT58 engine events involving fuel controls. The search

[34] This database fulfills the Aviation Mishap Information System (AMIS) requirements for aviation mishap reporting for the Department of Interior agencies and the US Forest Service. Categories of reports include incidents, hazards, maintenance, and airspace. The system uses the SAFECOM Form AMD-34/FS-5700-14 to report any condition, observation, act, maintenance problem, or circumstance with personnel or the aircraft that has the potential to cause an aviation-related mishap.

of the database revealed six SAFECOM reports from the beginning of the year (2003) to the day of the accident. The reports were reviewed for repetitive items, maintenance trends, and discrepancies that referenced, a loss of engine power, N_G fluctuations or fuel pressure fluctuations. The review identified several events in which a fuel control unit was replaced as the corrective action. Overall, there were multiple engine or engine indications that led operators to replace the FCU. The following provides a summarized list of reasons for the replacement of the FCUs:

1. Pilot notices a reduction in power in an engine
2. Pilot notices a loss of power in an engine and fluctuation on the fuel gauge.
3. Pilot notices an uncommanded fuel pressure fluctuation with corresponding N_G and engine temperature T5 increases.
4. Upon restart, the engine would not start
5. At engine start, engine went to full power then back to idle.
6. Pilot noticed the Number 1 engine torque was 90 and the Number 2 engine torque was at 30

The SAFECOM reports provide the corrective action, (replacement of FCUs), however, they do not provide the findings of the components that were replaced.

The NTSB also received information from CHSI detailing a history of fuel control unit replacements on their S-61 aircraft. Table 3 provides the list of replacements.

D.7 Tables:

Table 1 -SkyConnect data (Page 1)

registration	longitude	latitude	speed	heading	altitude	fix	PDOP	HDOP	UTC	local	usageType	source
N612AZ	-123.2748	40.78867	87	53	3294	3D	1	1	2008-08-05 21:39:08.000	2008-08-05 14:39:08.000	0	SkyConnect
N612AZ	-123.2262	40.81933	80	42	5463	3D	1	1	2008-08-05 21:41:19.000	2008-08-05 14:41:19.000	0	SkyConnect
N612AZ	-123.217	40.8475	29	25	4039	3D	2	1	2008-08-05 21:43:25.000	2008-08-05 14:43:25.000	0	SkyConnect
N612AZ	-123.233	40.81617	103	209	5207	3D	1	1	2008-08-05 21:45:38.000	2008-08-05 14:45:38.000	0	SkyConnect
N612AZ	-123.3002	40.774	103	230	1706	3D	2	1	2008-08-05 21:47:44.000	2008-08-05 14:47:44.000	0	SkyConnect
N612AZ	-123.3227	40.776	4	51	1132	3D	3	2	2008-08-05 21:49:52.000	2008-08-05 14:49:52.000	0	SkyConnect
N612AZ	-123.3003	40.7745	68	63	2195	3D	1	1	2008-08-05 21:51:58.000	2008-08-05 14:51:58.000	0	SkyConnect
N612AZ	-123.253	40.79867	80	74	4190	3D	1	1	2008-08-05 21:54:04.000	2008-08-05 14:54:04.000	0	SkyConnect
N612AZ	-123.2145	40.83133	35	16	5564	3D	1	1	2008-08-05 21:56:17.000	2008-08-05 14:56:17.000	0	SkyConnect
N612AZ	-123.2223	40.85083	64	194	4423	3D	2	1	2008-08-05 21:58:23.000	2008-08-05 14:58:23.000	0	SkyConnect
N612AZ	-123.175	40.81883	101	109	5449	3D	1	1	2008-08-05 22:00:29.000	2008-08-05 15:00:29.000	0	SkyConnect
N612AZ	-123.0955	40.802	115	108	4869	3D	1	1	2008-08-05 22:02:37.000	2008-08-05 15:02:37.000	0	SkyConnect
N612AZ	-123.0245	40.77983	107	103	4856	3D	1	1	2008-08-05 22:04:48.000	2008-08-05 15:04:48.000	0	SkyConnect
N612AZ	-122.9465	40.78133	109	69	4511	3D	1	1	2008-08-05 22:06:54.000	2008-08-05 15:06:54.000	0	SkyConnect
N612AZ	-122.8693	40.78817	101	87	3668	3D	1	1	2008-08-05 22:09:00.000	2008-08-05 15:09:00.000	0	SkyConnect
N612AZ	-122.8088	40.7885	39	127	3501	3D	1	1	2008-08-05 22:11:09.000	2008-08-05 15:11:09.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	58	3278	3D	1	1	2008-08-05 22:13:02.000	2008-08-05 15:13:02.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3281	3D	2	1	2008-08-05 22:38:39.000	2008-08-05 15:38:39.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3284	3D	2	1	2008-08-05 22:40:48.000	2008-08-05 15:40:48.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3278	3D	2	1	2008-08-05 22:46:04.000	2008-08-05 15:46:04.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3264	3D	2	1	2008-08-05 23:58:01.000	2008-08-05 16:58:01.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3278	3D	2	1	2008-08-06 00:00:08.000	2008-08-05 17:00:08.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3271	3D	2	1	2008-08-06 00:02:14.000	2008-08-05 17:02:14.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3284	3D	2	1	2008-08-06 00:04:20.000	2008-08-05 17:04:20.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3274	3D	1	1	2008-08-06 00:06:27.000	2008-08-05 17:06:27.000	0	SkyConnect
N612AZ	-122.7973	40.78467	80	131	3323	3D	2	1	2008-08-06 00:08:08.000	2008-08-05 17:08:08.000	0	SkyConnect
N612AZ	-122.865	40.77234	119	279	3940	3D	2	1	2008-08-06 00:10:17.000	2008-08-05 17:10:17.000	0	SkyConnect
N612AZ	-122.9548	40.78267	115	273	5259	3D	2	1	2008-08-06 00:12:25.000	2008-08-05 17:12:25.000	0	SkyConnect
N612AZ	-123.047	40.8	122	299	5673	3D	2	1	2008-08-06 00:14:35.000	2008-08-05 17:14:35.000	0	SkyConnect
N612AZ	-123.1248	40.84133	117	309	5020	3D	2	1	2008-08-06 00:16:43.000	2008-08-05 17:16:43.000	0	SkyConnect
N612AZ	-123.1945	40.8815	111	305	6637	3D	2	1	2008-08-06 00:18:51.000	2008-08-05 17:18:51.000	0	SkyConnect
N612AZ	-123.254	40.909	87	77	6227	3D	2	1	2008-08-06 00:21:03.000	2008-08-05 17:21:03.000	0	SkyConnect
N612AZ	-123.2493	40.91667	78	337	6266	3D	2	1	2008-08-06 00:23:12.000	2008-08-05 17:23:12.000	0	SkyConnect
N612AZ	-123.2548	40.915	2	155	6168	3D	2	1	2008-08-06 00:25:24.000	2008-08-05 17:25:24.000	0	SkyConnect
N612AZ	-123.2213	40.88833	97	168	6138	3D	1	1	2008-08-06 00:27:39.000	2008-08-05 17:27:39.000	0	SkyConnect
N612AZ	-123.2117	40.82767	111	179	5551	3D	1	1	2008-08-06 00:29:49.000	2008-08-05 17:29:49.000	0	SkyConnect
N612AZ	-123.2245	40.76667	101	201	2949	3D	2	1	2008-08-06 00:31:55.000	2008-08-05 17:31:55.000	0	SkyConnect
registration	longitude	latitude	speed	heading	altitude	fix	PDOP	HDOP	UTC	local	usageType	source
N612AZ	-123.2417	40.70667	93	196	3757	3D	2	1	2008-08-06 00:34:19.000	2008-08-05 17:34:19.000	0	SkyConnect
N612AZ	-123.2692	40.66283	87	355	4767	3D	2	1	2008-08-06 00:36:25.000	2008-08-05 17:36:25.000	0	SkyConnect
N612AZ	-123.2492	40.71367	109	1	3829	3D	2	1	2008-08-06 00:38:31.000	2008-08-05 17:38:31.000	0	SkyConnect
N612AZ	-123.2615	40.7375	72	302	2037	3D	2	1	2008-08-06 00:40:37.000	2008-08-05 17:40:37.000	0	SkyConnect
N612AZ	-123.2663	40.73367	8	129	2057	3D	2	1	2008-08-06 00:42:47.000	2008-08-05 17:42:47.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	6	1585	3D	2	1	2008-08-06 00:44:53.000	2008-08-05 17:44:53.000	0	SkyConnect
N612AZ	-123.2607	40.733	0	6	1581	3D	2	1	2008-08-06 00:45:31.000	2008-08-05 17:45:31.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	6	1578	3D	2	1	2008-08-06 00:47:41.000	2008-08-05 17:47:41.000	0	SkyConnect
N612AZ	-123.2607	40.733	0	6	1585	3D	2	1	2008-08-06 00:49:51.000	2008-08-05 17:49:51.000	0	SkyConnect
N612AZ	-123.2637	40.736	70	319	1683	3D	2	1	2008-08-06 00:51:57.000	2008-08-05 17:51:57.000	0	SkyConnect
N612AZ	-123.2658	40.73817	87	319	1713	3D	1	1	2008-08-06 00:52:04.000	2008-08-05 17:52:04.000	0	SkyConnect
N612AZ	-123.2965	40.79417	111	334	3576	3D	2	1	2008-08-06 00:54:10.000	2008-08-05 17:54:10.000	0	SkyConnect
N612AZ	-123.2937	40.85783	117	17	5410	3D	2	1	2008-08-06 00:56:16.000	2008-08-05 17:56:16.000	0	SkyConnect
N612AZ	-123.265	40.91117	80	55	6256	3D	2	1	2008-08-06 00:58:28.000	2008-08-05 17:58:28.000	0	SkyConnect
N612AZ	-123.2498	40.912	58	117	6069	3D	2	1	2008-08-06 01:00:34.000	2008-08-05 18:00:34.000	0	SkyConnect
N612AZ	-123.2548	40.9165	37	180	6129	3D	2	1	2008-08-06 01:02:40.000	2008-08-05 18:02:40.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	65	5942	3D	2	1	2008-08-06 01:04:17.000	2008-08-05 18:04:17.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	65	5961	3D	2	1	2008-08-06 01:06:25.000	2008-08-05 18:06:25.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	65	5974	3D	2	1	2008-08-06 01:08:31.000	2008-08-05 18:08:31.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	65	5935	3D	2	1	2008-08-06 01:10:41.000	2008-08-05 18:10:41.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	65	5948	3D	2	1	2008-08-06 01:12:49.000	2008-08-05 18:12:49.000	0	SkyConnect
N612AZ	-123.2522	40.91383	10	185	5991	3D	2	1	2008-08-06 01:14:55.000	2008-08-05 18:14:55.000	0	SkyConnect
N612AZ	-123.2445	40.91217	86	85	6129	3D	2	1	2008-08-06 01:15:25.000	2008-08-05 18:15:25.000	0	SkyConnect
N612AZ	-123.2742	40.8945	107	260	6256	3D	2	1	2008-08-06 01:17:38.000	2008-08-05 18:17:38.000	0	SkyConnect
N612AZ	-123.3262	40.84983	117	192	5246	3D	2	1	2008-08-06 01:19:44.000	2008-08-05 18:19:44.000	0	SkyConnect
N612AZ	-123.294	40.78	122	151	2818	3D	2	1	2008-08-06 01:21:50.000	2008-08-05 18:21:50.000	0	SkyConnect
N612AZ	-123.2577	40.73267	37	262	1742	3D	3	1	2008-08-06 01:23:56.000	2008-08-05 18:23:56.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	308	1565	3D	3	1	2008-08-06 01:25:18.000	2008-08-05 18:25:18.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	308	1562	3D	3	1	2008-08-06 01:27:26.000	2008-08-05 18:27:26.000	0	SkyConnect
N612AZ	-123.2662	40.73817	91	318	1722	3D	3	1	2008-08-06 01:29:32.000	2008-08-05 18:29:32.000	0	SkyConnect
N612AZ	-123.2957	40.79716	121	337	3556	3D	3	1	2008-08-06 01:31:43.000	2008-08-05 18:31:43.000	0	SkyConnect
N612AZ	-123.2847	40.86717	121	22	5308	3D	3	1	2008-08-06 01:33:54.000	2008-08-05 18:33:54.000	0	SkyConnect
N612AZ	-123.2533	40.915	29	135	6053	3D	3	1	2008-08-06 01:36:03.000	2008-08-05 18:36:03.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	58	5948	3D	3	1	2008-08-06 01:37:13.000	2008-08-05 18:37:13.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	58	5951	3D	3	1	2008-08-06 01:39:22.000	2008-08-05 18:39:22.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	58	5978	3D	3	2	2008-08-06 01:41:36.000	2008-08-05 18:41:36.000	0	SkyConnect
N612AZ	-123.2518	40.90633	119	182	5627	3D	3	1	2008-08-06 01:43:11.000	2008-08-05 18:43:11.000	0	SkyConnect

Table 1, SkyConnect data (Page 2)

registration	longitude	latitude	speed	heading	altitude	fix	PDOP	HDOP	UTC	local	usageType	source
N612AZ	-123.281	40.83283	132	203	3734	3D	3	1	2008-08-06 01:45:24.000	2008-08-05 18:45:24.000	0	SkyConnect
N612AZ	-123.2813	40.7605	122	152	2411	3D	3	1	2008-08-06 01:47:33.000	2008-08-05 18:47:33.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	325	1585	3D	3	1	2008-08-06 01:49:42.000	2008-08-05 18:49:42.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	325	1588	3D	3	1	2008-08-06 01:50:08.000	2008-08-05 18:50:08.000	0	SkyConnect
N612AZ	-123.2607	40.73317	0	325	1565	3D	3	1	2008-08-06 01:52:18.000	2008-08-05 18:52:18.000	0	SkyConnect
N612AZ	-123.254	40.736	103	86	1739	3D	3	1	2008-08-06 01:54:23.000	2008-08-05 18:54:23.000	0	SkyConnect
N612AZ	-123.1622	40.756	134	69	3337	3D	2	1	2008-08-06 01:56:31.000	2008-08-05 18:56:31.000	0	SkyConnect
N612AZ	-123.069	40.77133	117	84	3802	3D	2	1	2008-08-06 01:58:38.000	2008-08-05 18:58:38.000	0	SkyConnect
N612AZ	-122.9773	40.7785	121	81	4764	3D	2	1	2008-08-06 02:00:47.000	2008-08-05 19:00:47.000	0	SkyConnect
N612AZ	-122.8828	40.78567	121	84	3773	3D	2	1	2008-08-06 02:02:53.000	2008-08-05 19:02:53.000	0	SkyConnect
N612AZ	-122.8053	40.78717	29	121	3356	3D	2	1	2008-08-06 02:05:00.000	2008-08-05 19:05:00.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	98	3274	3D	2	1	2008-08-06 02:06:11.000	2008-08-05 19:06:11.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	98	3274	3D	2	1	2008-08-06 02:08:17.000	2008-08-05 19:08:17.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	98	3278	3D	2	1	2008-08-06 02:10:23.000	2008-08-05 19:10:23.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	98	3284	3D	3	1	2008-08-06 02:12:36.000	2008-08-05 19:12:36.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	98	3287	3D	2	1	2008-08-06 02:14:42.000	2008-08-05 19:14:42.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3294	3D	3	1	2008-08-06 02:19:04.000	2008-08-05 19:19:04.000	0	SkyConnect
N612AZ	-122.8043	40.787	0	0	3284	3D	3	2	2008-08-06 02:21:14.000	2008-08-05 19:21:14.000	0	SkyConnect
N612AZ	-122.804	40.78183	76	213	3524	3D	2	1	2008-08-06 02:23:20.000	2008-08-05 19:23:20.000	0	SkyConnect
N612AZ	-122.9578	40.809	115	297	6148	3D	2	1	2008-08-06 02:26:59.000	2008-08-05 19:26:59.000	0	SkyConnect
N612AZ	-123.0508	40.84167	134	292	5469	3D	1	1	2008-08-06 02:29:11.000	2008-08-05 19:29:11.000	0	SkyConnect
N612AZ	-123.142	40.87717	119	289	4423	3D	2	1	2008-08-06 02:31:20.000	2008-08-05 19:31:20.000	0	SkyConnect
N612AZ	-123.2233	40.901	122	322	6302	3D	2	1	2008-08-06 02:33:26.000	2008-08-05 19:33:26.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	138	5945	3D	2	1	2008-08-06 02:35:32.000	2008-08-05 19:35:32.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	138	5961	3D	2	1	2008-08-06 02:36:04.000	2008-08-05 19:36:04.000	0	SkyConnect
N612AZ	-123.2522	40.91417	0	138	5955	3D	2	1	2008-08-06 02:39:17.000	2008-08-05 19:39:17.000	0	SkyConnect
N612AZ	-123.2517	40.913	0	236	5928	3D	3	2	2008-08-06 02:41:25.000	2008-08-05 19:41:25.000	0	SkyConnect

Table 2 Speed Position Analysis and Comparison on 3-D Cam Servo:

	Columbia (Depth, Engine RPM, N_G)	**Hamilton Sundstrand** (Depth, Engine RPM, N_G)
FCU 1	Measurement depth[35]: = 0.725” N_G servo displacement: = 0.322” 2600 rpm--62% N_G	Measurement depth: 0.718 Depth at bottom: 0.960’ Difference: 0.242” 2700 rpm = about 64% N_G
FCU 2	Measurement depth: = 0.900” N_G servo displacement: = 0.147 1900 rpm – 45% N_G	Measurement depth: 0.896 Depth at bottom: 0.958’ Difference: 0.62” 1950 rpm = about 46% N_G (sub idle)
Delta	0.175”	0.178”

Note: According to a Hamilton Sundstrand representative, the position of the cam is not an indication of the final engine speed. Cam rest position can vary greatly.

[35] The measurement depth from the FCU housing to the 3-D Cam face (0.725 and 0.900 inches) is compared to a Columbia datum reference of 1.047 inches. The delta (0.322 and 0.147 inches) is the N_G servo displacement. Hamilton Sundstrand Overhaul Manual SEI-185, Figure 3-5, Gas Generator (N_G) Servo Calibration Curve, was then used to convert this displacement into a corresponding N_G -Control rpm.

Table 3 Carson S-61 FCU events

Year	Aircraft	FCU Serial Number	Reason for replacement
2006	N81661	90030	Engine torque split of 10% and fuel pressure fluctuations.
	N81661	Unknown	Engine slow to respond
	N81697	Unknown	Engine slow to respond
2007	N4503E	Unknown	Engine slow to respond
	N81661	95604	Engine spooled down to idle and did not respond to speed selector.
	N4503E	Unknown	Engine slow to respond
	N4503E	49882	Engine fuel pressure fluctuation
2008	N612AZ	29428	Aircraft yawed during power check
	N4503E		Engine fuel pressure fluctuation
	N612AZ	89674	Engine torque limited to 50%

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