NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety
Gardena, California 90248

May 5, 2011

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

WPR10FA371

This report contains ten embedded photographs

A. ACCIDENT

Operator: AirMethods Corporation
Location: Tucson, Arizona
Date: July 28, 2010
Time: 1342 Mountain Standard Time (MST)
Aircraft: N509AM, American Eurocopter AS350B3

B. AIRWORTHINESS GROUP

Eliott Simpson - Chairman National Transportation Safety Board Gardena, California	Scott Tyrell - Member Federal Aviation Administration Dallas, Texas
Archie Whitten - Member Turbomeca, USA Grand Prairie, Texas	Joe Syslo - Member American Eurocopter Grand Prairie, Texas
Don Lambert - Member AirMethods Corporation Denver, Colorado	David Lok - Member Helicopter Services of Nevada Boulder, Nevada

C. SUMMARY

On July 28, 2010, at 1342 MST, an American Eurocopter AS 350 B3, N509AM, descended rapidly and collided with terrain in an urban area of Tucson, Arizona. The helicopter was operated by Air Methods Corporation, as LifeNet 12, on a repositioning flight, under the provisions of Title 14 Code of Federal Regulations Part 91. The commercial pilot and two medical flight crewmembers were killed. The helicopter was substantially damaged, and consumed by a post impact fire. Visual meteorological conditions prevailed, and a company flight plan had been filed. The repositioning flight originated at Marana Regional Airport, Tucson, at 1332, and the intended destination was Douglas, Arizona.

D. DETAILS OF THE INVESTIGATION

1.0 Overview

The American Eurocopter AS350B3 is a single engine helicopter, equipped with a three blade main rotor system, conventional two blade tail rotor, and powered by a Turbomeca Arriel 2B1 turboshaft engine. The accident helicopter (S/N 4698) was manufactured by American Eurocopter in the United States in 2009. It was configured with a right pilot seat and three forward-facing medic seats positioned against the aft cabin wall. A swivel-stretcher extended from the left forward cabin aft, with a plastic shield separating it from the pilot. The overall length of the helicopter was 42.45 feet, with a main rotor disk diameter of 35 feet. (Figure 1) The total time on the helicopter at the time of the accident was 352 hours. The engine had a total time of 352 hours, N1 602.28 cycles, N2 234.90 cycles, and 689 starts.

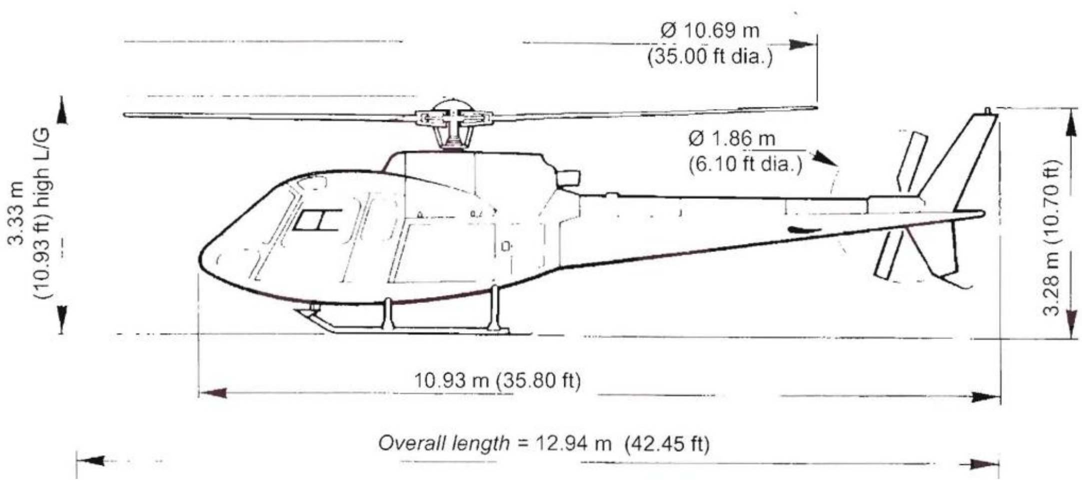

Figure 1: AS350-B3 Physical dimensions.

1.1 Site Examination

The helicopter came to rest upright, on a heading of 055 degrees magnetic, along the east verge of a north-south residential street. (Figure 2) The center fuselage structure was straddled across a five-foot-tall cement-block wall, with the nose resting on a garden shed. Both the shed and the wall were fragmented by the impact, with fire consuming the shed, its contents, and vegetation within a 15-foot radius of the fuselage. (Figure 3) The top of a tree located 5-feet east of the helicopter nose had been severed at the 12-foot-level. A 40-foot-tall set of power lines paralleled the street on the opposite verge, about 50-feet west of the main wreckage. No impact marks or damage were observed to the power lines, or any other adjacent tree or structure. Fragments of the fuel tank were found about 30 feet to the southwest, and the odor of jet fuel was present throughout the site.

Figure 2: Photo of accident location viewed from above and to the north. Power lines are to the left, with fuel tank fragments indicated by green arrows. The tail boom is visible lying on the right side of the road.

Figure 3: Photo of accident location viewed from above. The center fuselage structure is situated on the brick wall, which runs diagonally across this image. All three main rotor blades and the engine are visible in the center of image.

The helicopter's canopy, cabin, and rear structure were consumed by fire from the floor cross-members through to the engine exhaust heat shield. The cross-members and associated flight controls sustained crush damage concentrated around the left side of the cabin floor. The landing gear skids were fragmented in multiple areas. The front cross tube sustained crush damage primarily to its left side, and was bent outwards. The aft cross member remained intact, and the right aft skid was fire consumed.

The left foot pedal of the tail rotor control was found in the forward position, and it was deformed forward and down. (Figure 4) The collective control was in the full down position, with the rotary throttle set to 'VOL' (flight position). The cyclic control exhibited buckling deformation to its shaft at the cabin floor, with the control pointing forward and to the right. The flight control linkages located beneath the cabin floor were deformed and impinged against the cross members. The remaining linkages aft of the cabin were consumed by fire through to the flight control servo actuators. The rotor brake and fuel shutoff valve control assemblies sustained thermal damaged, with both controls pointing to the forward position.

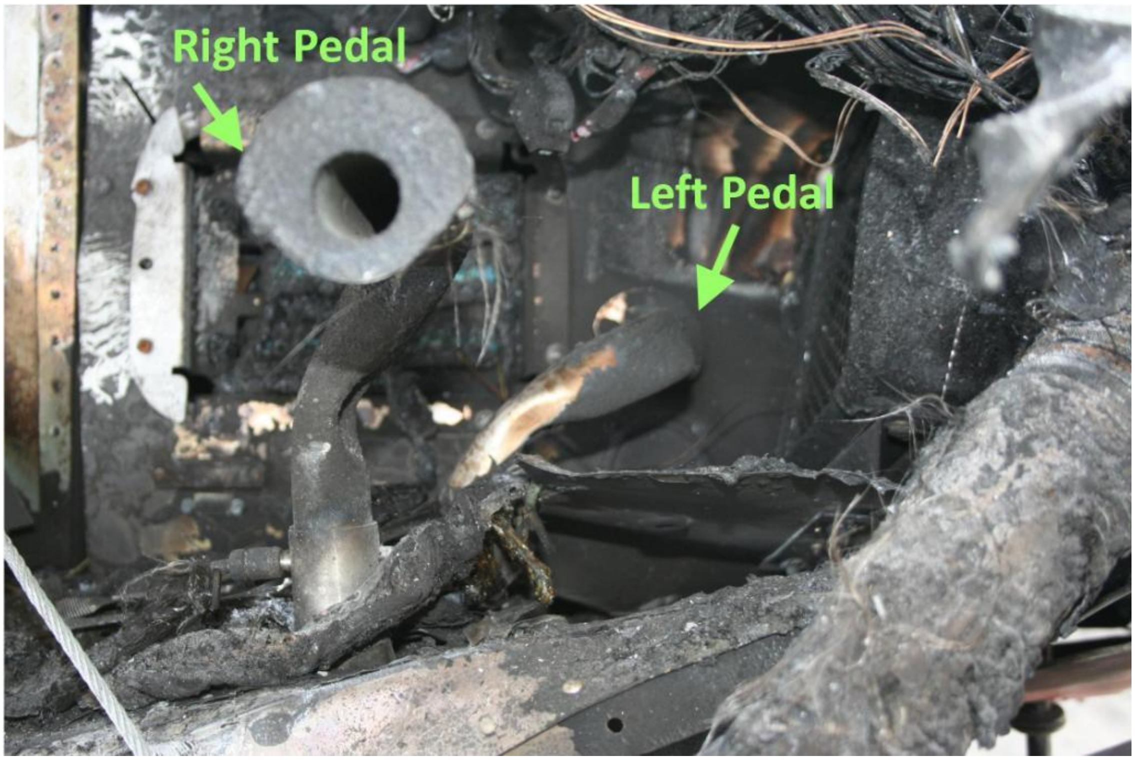

Figure 4: Photo of tail rotor foot pedal controls viewed from the right side of the cabin, the front of the helicopter is to the right.

The aft 15-foot-long section of the tailboom remained in line with the fuselage, but was separated in the area of the exhaust heat shield. The tailboom junction frame and attachment members were consumed by fire. The horizontal stabilizer remained attached to the tailboom, with the left stabilizer bent upwards about 15 degrees mid-span. The lower surfaces of the stabilizer and the tailboom section in the area of the doubler fittings sustained crush damage and dimpled indentations consistent with road surface contact. The upper vertical fin remained attached to the tailcone, which had separated from the tailboom just forward of the fin attachment fittings. The lower fin had separated at its root, and sustained vertical accordion crush damage through its entire length. The tail guard had separated from the fin and was found about 40 feet southwest of the tail. The underside of the guard exhibited dimpled indentations and gouges on its left edge, consistent in shape to contact with the road surface. A corresponding skid-length indentation was noted in the road surface adjacent to the tail cone, on a heading of 045 degrees.

The tail rotor gearbox remained intact and affixed to the tailboom structure. Oil was observed through the gearbox sight window, and rotation of the tail rotor yoke resulted in a corresponding rotation of the tail rotor drive shaft at the tailboom heat shield. Both blades remained firmly affixed at the rotor hub, were largely intact, and their associated pitch change links were noted connected to both the rotating plate, and blade control horns. The yellow blade sustained a leading edge tar-covered nick at the tip. A corresponding cut in the tarmac road surface was noted adjacent to the blade. The balance weight mount was bent inboard; road tar transfer and scrapes in the paint were noted on the surface of the mount and the yellow blade control horn. Corresponding indentations were noted in the adjacent road surface, about 2-feet to the south.

(Figure 5) The tail rotor push-pull tube was continuous from the rotor hub bellcrank through to the tailboom heat shield. Forward of the heat shield, the remaining control tube and drive shaft were fire consumed through to the area of the steel forward drive shaft. The cylinder casing and control lines of the yaw pedal servo actuator were fire consumed; the associated servo rod end and input control bolts remained in place.

Figure 5: Photo of tail rotor blades and gearbox affixed to aft tailboom. The lower right hand corner of image displays ground scars corresponding to balance weight contact.

The main rotor blades had sustained varying degrees of fire damage and could not be identified by color. All three blade leading edges remained relatively intact, with the blades still connected through their full length to the hub blade attach pin. The 'Starflex' hub, elastomer blocks, swashplate, and sleeves were partially consumed by fire, and located within the immediate vicinity of the rotor mast assembly. Gouging was noted on the outer shell of rotor hub spring housing, adjacent to the blade/sleeve retaining bolts, and all three vibration absorber springs were located in the adjacent debris area.

One of the main rotor blades (S/N 31257) exhibited nicks and scrapes to the lower leading edge surface consistent with road surface contact, and sustained fire damage from the root to 5-feet outboard from the root. The entire trailing edge of the second blade (S/N 31175) was consumed by fire; the leading edge displayed no indications of abrasions. The outer surfaces of the third blade were consumed by fire, and as such the serial number could not be identified. (Figure 6)

Figure 6: Photo of main rotor blades after removal from immediate accident site.

The main rotor drive gearbox suspension system had been consumed by fire. The main rotor gearbox input shaft remained attached to the engine shaft flange through the drive shaft. The main gearbox flex coupling was disassembled at the accident site for recovery. Rotation of the gearbox input shaft by hand resulted in a corresponding rotation of the rotor hub. The three main rotor servo actuators sustained varying degrees of fire damage and remained affixed to their mounting points on the main gearbox. The actuator control pitch links, scissor link, and swashplate were consumed by fire; the upper pitch link eyebolts remained attached at the blades. The dual hydraulic lines for each servo were fire damaged; all lines remained affixed to their respective ports on the servos and gearbox casing. Both the belt and direct drive hydraulic pumps were located and had sustained thermal damage revealing their inner mechanisms.

The engine and its associated gearbox remained attached and were located within the primary fuselage structure. The engine remained in-line with the fuselage, and had come to rest on its left side.

External examination of the engine at the accident site revealed that the fuel inlet union, located on the lower right-hand-side of the engine, had become detached from the boss on the compressor case. (Figure 7) The fuel supply line remained attached to the union and to the hydro-mechanical unit (HMU). The intermediate gasket was located in the fuselage debris, directly below the union. The remaining wreckage and ground area were extensively examined utilizing a series of magnets and sifting grates, but did not reveal the presence of the two five-point bolts and self locking nuts for mounting the union to the compressor case flange.

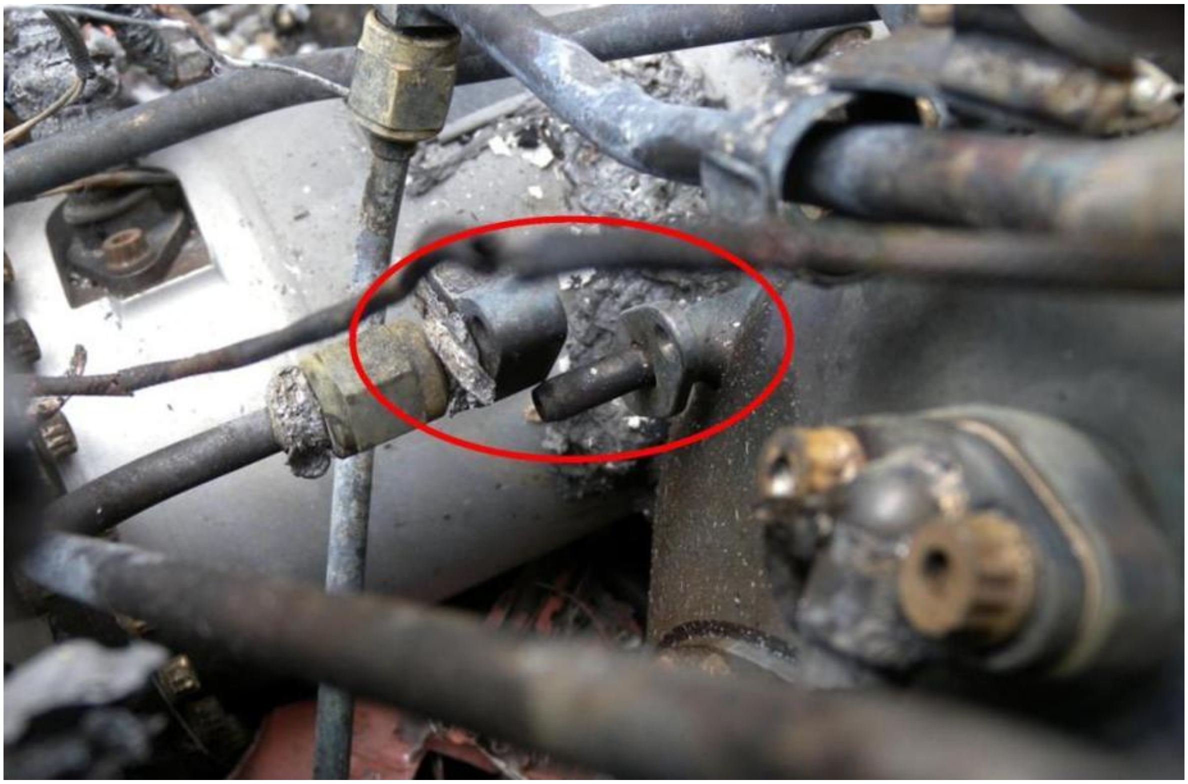

Figure 7: Photo of fuel union and compressor case as found at accident site. The nipple is in the center of the image, protruding from the compressor case on the right. The fuel union is separate from the flange and to the left of center.

Both the main rotor and tail rotor output shafts remained connected to their respective drive shafts. Examination of the tail rotor output shaft flex-coupling revealed that its lock plate remained in place, but had not been bent into a locking position; the coupling remained firmly affixed to the shaft and exhibited no indication of slippage. The engine was partially disassembled, and shipped to the Turbomeca facility for further examination.

There was no evidence found of any pre-existing failures of the helicopter airframe.

Post-accident visual examination of fuel samples drained from the fueling truck which serviced the accident helicopter revealed that it was of a color consistent with Jet-A fuel.

1.2 Fuel Injection System

Fuel is delivered directly into the combustion chamber through use of a radial fuel supply, centrifugal fuel injection system. (Figure 8) The injector assembly consists of a stationary distributor (manifold) and a rotating injection wheel located within the combustion chamber. The manifold contains a series of holes, which deliver fuel to the wheel. Holes within the injection wheel, which is mounted between the compressor and turbine shaft, act as fuel spraying jets. Pressure integrity between the wheel and the manifold is achieved by labyrinth seals, with rotation of the injection wheel resulting in the extraction of fuel through centrifugal force.

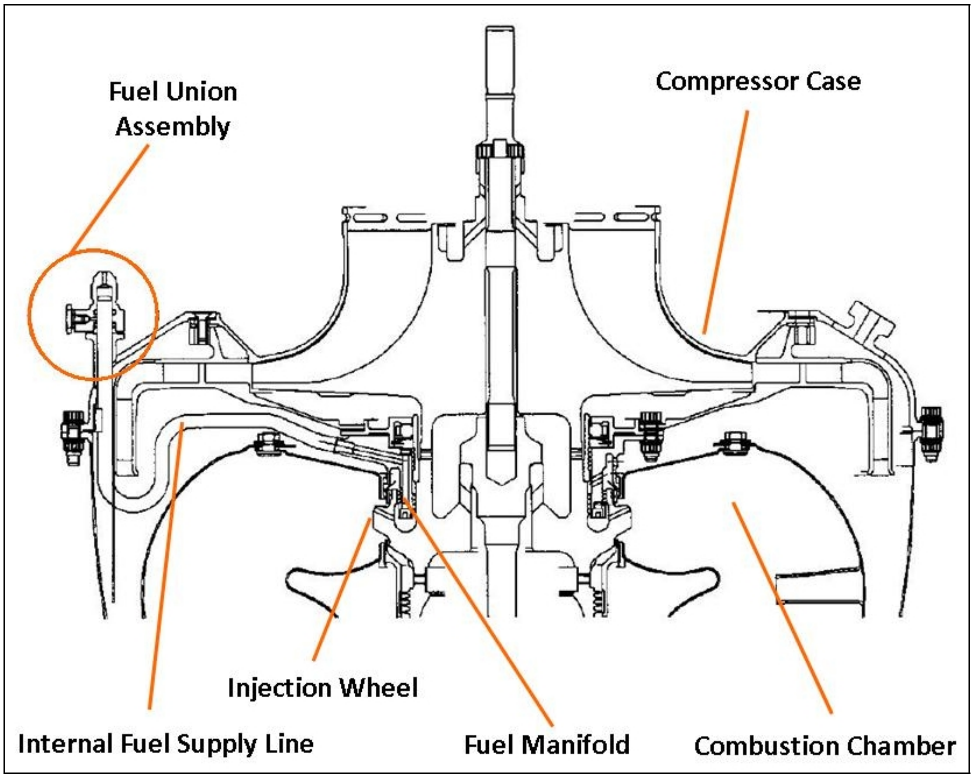

Figure 8: Cross-section view of fuel supply routing and manifold on engine compressor case.

An internal supply line provides fuel to the manifold. The supply line routes from the manifold and along the outer combustor casing, where it protrudes through a boss on the compressor case. The exposed line protrudes in the form of a nipple through a flange on the case boss. An engine fuel inlet union, consisting of a mounting flange and seal, provide the interface between the internal fuel line nipple, and external fuel supply lines. The union is affixed to the compressor case flange with two five-point bolts (P/N: 72-43-00-01-304) and two self locking nuts (P/N: 72-43-00-01-302). Vacuum seal is provided by the use of a metallic intermediate gasket at the flange, and fuel seal is provided by two o-ring seals on the internal fuel supply line nipple. A leak test plug is provided in the body of the union to test the seal integrity of the interface. The union is connected to the external fuel supply line through a B-Nut fitting. (Figure 9)

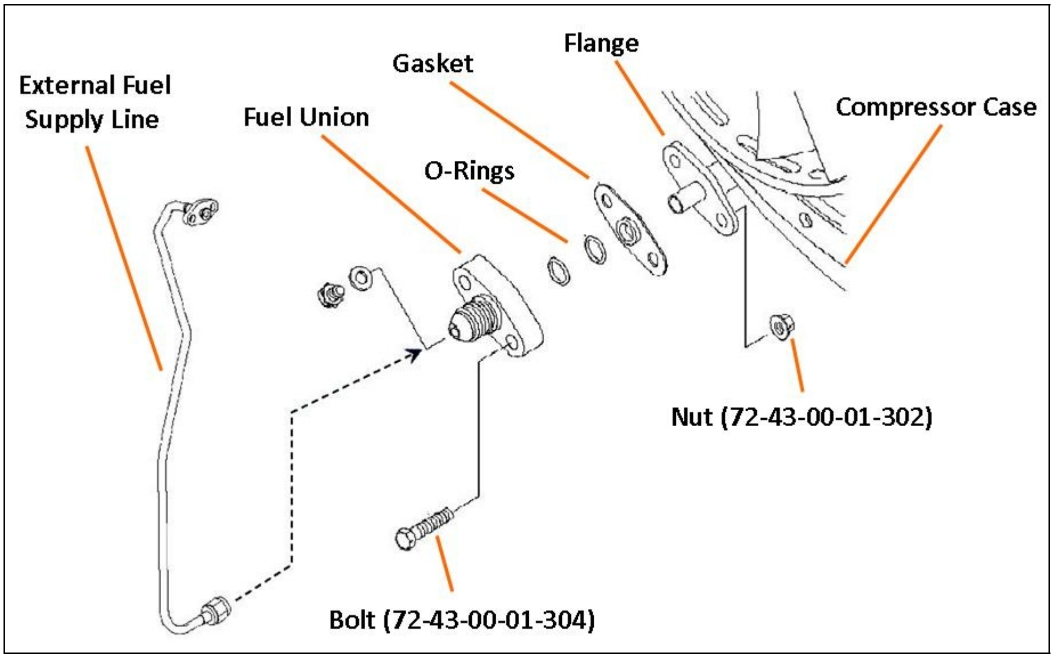

Figure 9: Exploded view of fuel supply routing on engine compressor case.

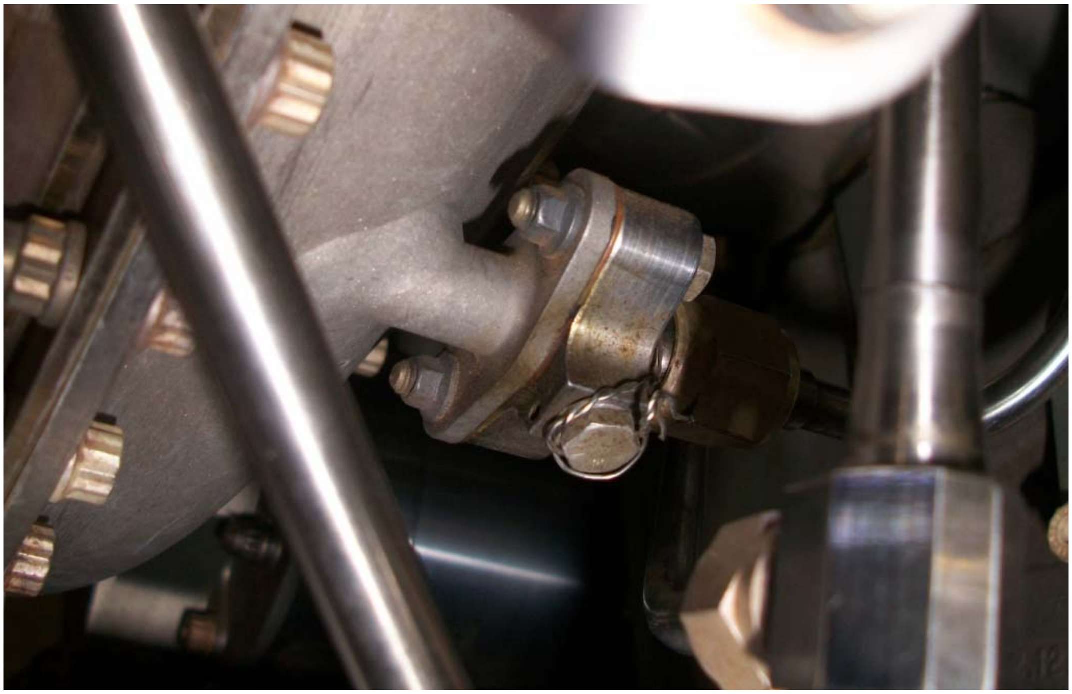

Figure 10: Photo of exemplar installation of fuel union at the compressor case. Note about 1 1/2 threads visible on exposed bolt heads.

1.3 Engine and HMU Examination

The engine and HMU were examined by the group at the Turbomeca facility in Grand Prairie, Texas. The HMU remained intact, and had sustained thermal damage to the engine back-up control ancillary unit (EBCAU). The HMU had become separated from the accessory case at its mount point. The mounting clamp rings exhibited buckling signatures consistent with impact overload.

The reduction gearbox (module 5) was removed from the engine to facilitate examination. Gearbox drive continuity was confirmed from the input to the output shaft, through hand rotation. The axial compressor casing sustained crush damage to the lower lip of the inlet flange. (Figure 11) Corresponding damage was observed on the tips of the axial compressor wheel around its full circumference. The power turbine wheels were intact; and could be rotated smoothly by hand.

The power turbine blades were intact, exhibited no obvious indications of damage when observed through the exhaust stack, and could be rotated. (Figure 12) The gearbox input power shaft was rotated, with a corresponding smooth rotation of output shaft. Rotation of the entire gas generator assembly revealed continuity through to the pad outlets on the accessory gearbox.

The freewheel shaft was rotated by hand, and the freewheel unit was observed to lock in the drive direction and slip in the freewheel direction. The tail rotor drive shaft, located within the module 5 unit, exhibited bending and twist just forward of the output drive spline. Rotation of the gearbox input shaft by hand resulted in a corresponding rotation of the output shaft with no binding noted.

The engine and HMU were further disassembled, and no internal mechanical anomalies were identified.

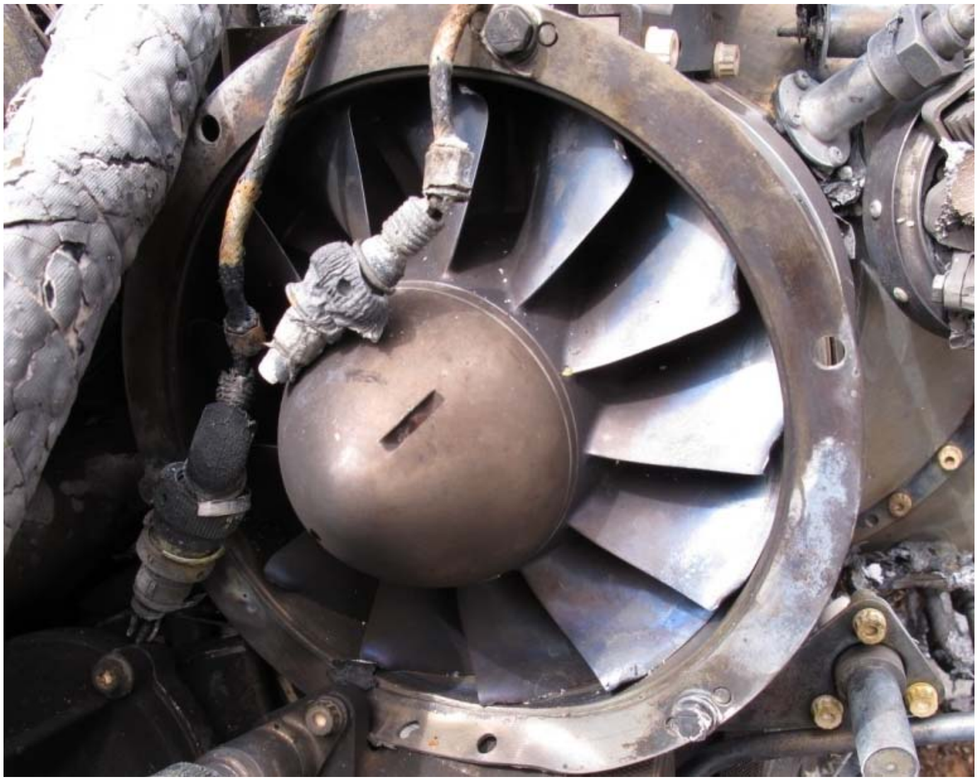

Figure 11: Photo of axial compressor and casing

Figure 12: Photo the of power turbine blades viewed through the exhaust pipe

1.4 Fuel Inlet Union Tests

A group of engine test runs were performed on an exemplar Arriel 2B1 engine at the Turbomeca facility in Bordes, France. The tests were conducted under the supervision of investigators from the Bureau d'Enquêtes et d'Analyses (BEA). The purpose of the tests was to assess the engines operating abilities with the fuel inlet union incorrectly affixed to the engine case flange.

The series of engine runs was performed in a test cell with the fuel inlet supply line union partially attached to the compressor case flange in the following configurations:

- Union inserted over inlet fuel nipple with no attachment nuts and bolts installed.

- Union partially inserted over inlet fuel nipple to first o-ring seal, with no attachment nuts and bolts installed.

- Union inserted, with attachment nuts and bolts installed, and hand-tightened only.

- Union inserted over inlet fuel nipple, utilizing only one attachment nut and bolt, tightened by hand.

The tests were performed at varying engine power levels, simulating startup and flight modes. The data revealed that with the union installed without its associated mounting nuts and bolts, it was possible to start and run the engine with no observable fuel leak. During the test with the union nuts and bolt tightened by hand, the engine ran for 3 minutes and 32 seconds before the nuts began to unscrew from the bolts.

The tests further revealed that with both nuts and bolts removed, the union would ultimately eject from the boss and nipple, resulting in an expulsion of about 0.5 liters of fuel, followed by a subsequent engine shutdown. See the attached Turbomeca report for further details.

The group examined comparative dimensions for new and used union nuts and bolts[1]. The results revealed that when tightened by hand, a new bolt would engage about two threads into the nut before friction was felt, resulting in an exposed bolt shank length of about 17 millimeters. By comparison, a used nut and bolt were similarly tightened, resulting in an exposed bolt shank length of about 13.5 millimeters. (Figure 13) The threaded end of the used bolt was also observed to extend beyond the end of the nut by about half a thread. Comparative dimensions taken from an exemplar engine installation revealed that with the nuts and bolts installed and tightened to the specified torque on the union flange, approximately 1.5 threads were exposed beyond the nut, resulting in a bolt shank length of about 13 millimeters. (Figure 13)

[1] The total service time and tightening cycles of the used hardware was unknown.

Figure 13: Photo of new and used union attachment nuts and bolts, with the nuts tightened by hand.

1.5 Maintenance

The most recent maintenance was completed the day prior to the accident and included the replacement of the engine fuel injection manifold. The mechanic who performed the maintenance reported referencing the Turbomeca maintenance technical instruction number X 292 M1 303 2, Update No. 4: January 30th, 2008 entitled, 'Replacement of the Injection Manifold.'

The manual details the steps required to remove the manifold; these include removal of the engine from the helicopter, and the subsequent disassembly of the engine. The manual further specifies instructions which include the removal of the fuel line union from the compressor case, and subsequent reassembly, utilizing a new union intermediate gasket and o-ring seals. Turbomeca authorizes the reuse of hardware, provided it falls within a set of criteria specified in the engine service manual. See the Maintenance Group Chairman report for further details.

Once this maintenance action had been completed a post maintenance check flight was performed by the duty pilot the evening prior to the accident flight. The check flight lasted 7.5 minutes and upon its conclusion, the helicopter was signed off as airworthy.
Shipping records indicated that an 'install kit' consisting of replacement hardware and consumable materials was shipped to the maintenance facility prior to replacement of the injection manifold. The kit contained a set of new union nuts and bolts. The remnants of the kit were examined after the accident; the new set of union nuts and bolts were found unused, and still packaged within the kit.

1.6 Digital Engine Control Unit

The engine was equipped with a digital engine control unit (DECU), which records only when specific engine control system discrepancies are encountered. The recordings are intended for maintenance use only.

The DECU was extensively damaged by post impact fire. The unit was sent to the NTSB Office of Research and Engineering for data extraction. The recovered data was analyzed by Turbomeca in France under the supervision of the BEA. The data contents of the DECU are included in the Turbomeca Investigation Report T10-CR0744A-1, which is included as an attachment to this report.

The data revealed that the last exception code was recorded eleven engine cycles prior to the accident flight, with no exceptions recorded during the accident flight. According to Turbomeca, under the circumstances of a fuel inlet union separating in flight, the DECU logic will trigger an 'Engine Flameout' event, but will not record that event in the DECU memory. In an effort to test this principle, Turbomeca configured an Arriel 2B1 engine in a test cell, and disconnected the fuel inlet union line with the engine operating at NG values ranging between 68% and 98%. The DECU memory was subsequently downloaded after the tests, and no exceptions were noted. A detailed explanation of the DECU operation and testing parameters entitled, 'Turbomeca Technical Memo 2010/225' dated January 10, 2011, is included as an attachment to this report.

1.7 Fuel Tank Design and Certification Testing

The 143-gallon capacity fuel tank is mounted in the fuselage below the main transmission deck, behind the passenger compartment, between the left and right baggage compartments. The tank is manufactured from a spin-molded polyamide and does not have internal baffles.

All versions of the AS350 that hold FAA-type design approvals are under Type Certificate Data Sheet (TCDS) H9EU. The first of the AS350 series, the AS350-C, initially received an FAA type certificate design approval on December 21, 1977. Subsequent predecessors to the AS350-B3, the AS350-B, and AS350-B2 design variants were approved by the FAA on November 9, 1978 and June 8, 1990 respectively. The AS350-B3 was approved by the FAA on May 7, 1998. The certification basis for all design variants under TCDS H9EU consist of 14 CFR 21.29 and Part 27, effective February 1, 1965, plus the requirements contained in amendments 27-1 through 27-10.

For amendments 27-1 through 27-10, the only certification requirement for fuel tank crash resistance was found in 14 CFR 27.965, which was effective February 1, 1965, and was unchanged through amendment 27-10. The regulation states, in part, each fuel tank must be able to withstand, without leakage, an internal pressure equal to the pressure developed during the maximum limit acceleration with that tank full, but not less than 3.5 psi for conventional tanks; or 2.0 psi for bladder tanks. With amendment 27-12 (effective May 2, 1977), 14 CFR 27.965 was amended. The basic internal pressure requirements listed previously were unchanged; however, other vibration proof tests were imposed against potential in-service leakage sources not related to deceleration events.

Specific fuel system crash resistance standards were first introduced into 14 CFR Part 27, with amendment 27-30, effective November 2, 1994. Section 27.952 imposes drop-test requirements to prove that the design can withstand certain static and dynamic deceleration loads, depending on the fuel tanks location within the helicopter structure. The minimum static and dynamic loadings (in units of gravity [or G] and direction of force application) specified for the fuel tanks location in the AS350-B2 is: (1) 1.5 Gs in upward direction, (2) 8.0 G.s in forward direction, (3) 2.0 Gs in sideward direction, and (4) 4.0 Gs in downward direction.

Documents were obtained from Eurocopter France concerning the fuel tank design and any testing performed to establish static and/or deceleration load limits. Eurocopter provided report number 350A.06.2064,[2] dated June 1, 1977, which documents a series of drop tests performed from September 1976 to February 1977. In the test series, an accelerometer instrumented, conforming production fuel tank filled to 80 percent capacity with water and the tanks associated mounting structure was subjected to a series of drop tests from varying heights to induce deceleration loads in the vertical (Z) and horizontal/longitudinal (X) axes. According to the report, the tank sustained a loading in the vertical downward Z axis of 6.1 Gs without deformation to the tank. In the X axis, neither the tank nor the mounting structure ruptured or exhibited deformation up to 3.85 Gs. Between 3.85 and 10.0 Gs, the tank exhibited deformation without rupture; however, the mounting structure was deformed. The tank and the mounting structure withstood an ultimate loading of 10.35 Gs.

[2] See the report for NTSB case number LAX01MA272, which is available at http://www.ntsb.gov/aviationquery/index.aspx.