National Transportation Safety Board
Washington, D.C. 20594

Aircraft Accident Brief

Accident Number:	LAX01MA272
Aircraft and Registration:	Eurocopter AS350-B2, N169PA
Location:	Meadview, Arizona
Date:	August 10, 2001
Adopted On:	June 3, 2004

HISTORY OF FLIGHT

On August 10, 2001, about 1428 mountain standard time,[1] a Eurocopter AS350-B2 helicopter, N169PA, operating as Papillon 34, collided with terrain during an uncontrolled descent about 4 miles east of Meadview, Arizona. The helicopter was operated by Papillon Airways, Inc., as an air tour flight under *Code of Federal Regulations* 14 (CFR) Part 135. The helicopter was destroyed by impact forces and a postcrash fire. The pilot and five passengers were killed, and the remaining passenger sustained serious injuries. The flight originated from the company terminal at the McCarran International Airport (LAS), Las Vegas, Nevada, about 1245 as a tour of the west Grand Canyon area with a planned stop at a landing site in Quartermaster Canyon. The helicopter departed the landing site about 1400 and stopped at a company fueling facility at the Grand Canyon West Airport (GCW). The helicopter departed the fueling facility at 1420 and was en route to LAS when the accident occurred. Visual meteorological conditions prevailed, and a visual flight rules flight plan was filed.

No ground-based or airborne witnesses to the accident were identified; however, National Transportation Safety Board investigators interviewed the surviving passenger on March 7, 2002. She reported that she and the other passengers had worn headsets to hear the pilot's commentary and announcements.[2] She did not hear any bells or horns before the accident and did not hear a verbal warning from the pilot or overhear any radio transmission with the phrase, "mayday." The passenger stated that she remembered being "up in the air" and that she had "traveled quite some time" when everything "went quiet, the blades stopped turning and we fell." She did not remember seeing the pilot move any switches or buttons before the accident.

[1]Unless otherwise noted, all times in this report are mountain standard time. Because Nevada was on Pacific daylight time at the time of the accident, the times in Nevada and Arizona were the same.

[2]In her interview, the surviving passenger gave seating positions for the passengers that differed significantly from the manifest provided by Papillon.

During the investigation, the company procedures for passenger check-in and seat assignment were reviewed. When the passengers first arrive at the Papillon terminal for their flight, they are weighed and given a playing card that serves as their boarding pass; the playing card number represents the individual's seat assignment. The company uses a computer program, which takes the passenger weight value and assigns a seat for optimum weight and balance control. This program also generates a manifest. Passengers are then required to watch a short safety video before being bussed to their helicopter for the flight. Upon arrival at the helicopter, the passengers are met by the pilot and asked if there are any questions regarding the safety video. The pilot then points out the various safety features of the helicopter. Passengers are asked for their boarding pass, and the pilot ensures that the passengers sit in their assigned seats.[3]

The passengers in the accident helicopter were part of a group of 12 individuals traveling together who booked the Canyon Celebration tour flight offered by Papillon. During check-in, the 12-member group was assigned to two helicopters, six passengers in the accident helicopter and 6 in the other helicopter. According to Papillon's general manager, the route and tour are standard and performed five times a day. The route consists of departure from LAS eastbound along Tropicana Boulevard to Hoover Dam, over Indian Pass, and to the west end of the Grand Canyon to the entrance near "Green Route 4,"[4] then down to Quartermaster Canyon. The 1½-hour itinerary included a 30-minute stopover at either the Quartermaster Canyon or the Lower Ramada landing sites, which are leased from Native American tribes and used exclusively by Papillon. (The accident flight landed at Quartermaster Canyon.) The helicopters are shut down after landing, and the passengers are then given a picnic-type lunch, with nonalcoholic champagne. The passengers are then reboarded into the helicopters, and they depart. Depending on weight restrictions, some of the helicopters stop at an uncontrolled field, GCW, where Papillon maintains a fuel tanker truck. Hot fueling is conducted at this facility.[5]

Company flight following records indicated that Papillon 34 departed LAS about 1245 as the third helicopter in the six-ship flight conducting the Canyon Celebration tour, with about 5 minutes between departures. Papillon 34 was originally scheduled to depart at 1230.

The flight departed Quartermaster Canyon about 1400 and landed at GCW about 1410 with about 15 percent of the fuel remaining (about 30 minutes of flight time). (See figure 1.) The GCW fuel log indicated that the helicopter was serviced with 27.9 gallons of Jet A fuel, which increased the amount of fuel on board to about 34 percent of the capacity (about 70 minutes of flight time). Papillon 34 departed about 1420 for the return flight to LAS.

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[3]According to company procedures, the pilots can alter seat assignments, but only after recomputation of the weight and balance. Interviews with company pilots revealed that although this procedure gives them the authority to change the passenger seating, they rarely allow passengers to swap seats.

[4]The *Special Federal Aviation Regulation* (SFAR) 50-2 covering the airspace in and around the Grand Canyon specifies predetermined and charted routes that fixed and rotary wing aircraft must follow.

[5]Hot fueling is a procedure whereby the tanks are filled while the engine(s) are running. In the case of a helicopter, the rotors are turning at a ground-idle power setting while the procedure is conducted.

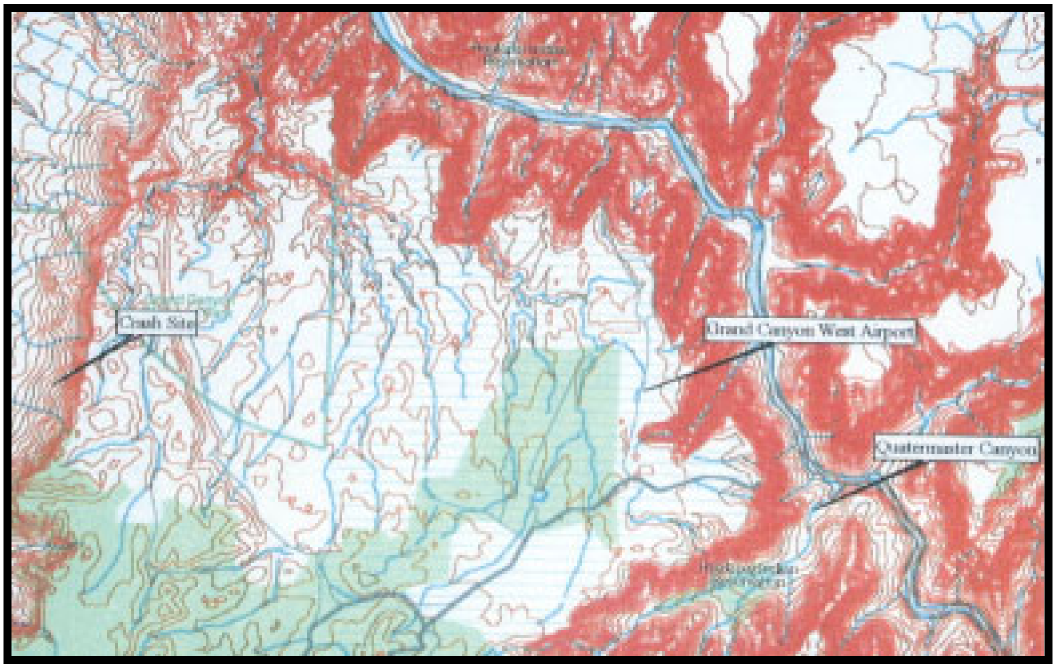

Figure 1. Topographic chart showing departure points and crash site.

None of the Papillon pilots who were interviewed recalled hearing any radio transmissions from the accident pilot after he had departed from GCW. A company pilot in N911KR recalled seeing the accident flight about 2 to 3 minutes in front of him at about 5,500 feet mean sea level (msl) in the vicinity of the Grand Wash Cliffs during the return flight to LAS. These cliffs were depicted on local aeronautical charts as part of the route structure for all returning flights and were used by the pilots as a navigational landmark. The pilot in N911KR reported that company procedures were to change radio frequencies prior to crossing the Grand Wash Cliffs. He did not recall hearing the accident pilot make that frequency change.

The area of the accident site is remote and has limited radar coverage. An examination of primary and secondary beacon targets derived from National Track Analysis Program radar data obtained from the Los Angeles Air Route Traffic Control Center was performed covering the period from 1424 to 1432. Based on the time stamp from the secondary beacon returns and the relationship of the targets to the accident location, only two secondary radar targets[6] could be associated with Papillon 34 just before the accident. No primary returns were detected. The target received at 1428:05 showed a mode C reported altitude of 5,400 feet msl, with a location over or just west of the cliff edge. The next target, received at 1428:17, showed a mode C reported altitude of 4,500 feet msl, with a location approximately 1,100 feet horizontally from the accident site. Statements from other pilots in the area indicated that, following the accident, other helicopters diverted to the crash site. Numerous other secondary targets appear over and around the accident location between 1429 and 1432.

[6]A secondary radar target is one in which the position, altitude, time, and a transponder beacon code is denoted on the air traffic controller's radar screen. A primary target consists of only a radar return from the aircraft structure and does not provide any altitude or beacon code information.

Upon crossing the Grand Wash Cliffs, all of Papillon's flights descend from 5,500 feet msl to 4,500 feet msl to avoid opposite direction traffic and to comply with the hemispheric flight rules in 14 CFR Part 91. As N911KR passed the Grand Wash Cliffs, the pilot looked to his left and noticed a column of black smoke. Upon closer investigation, he confirmed that the smoke was rising from the wreckage of a Papillon helicopter. He immediately notified the company of the accident and requested that emergency medical services be notified. He then landed at a suitable nearby spot and rendered whatever assistance he could to the survivor.

The accident site was located on a hillside about 5 miles east of Meadview, Arizona, at 35° 59.19' north Latitude and 113° 59.00' west Longitude. The main wreckage was located about ¼ mile west of, and 1,800 feet msl below, the rim of the Grand Wash Cliffs. Site elevation obtained from a digital handheld altimeter was 4,041 feet msl. The terrain slope was estimated to be 40° and sloped up toward the east culminating in the Grand Wash Cliffs.

PILOT INFORMATION

A review of Papillon company personnel and training records and Federal Aviation Administration (FAA) airman and medical record files revealed that the pilot held a commercial pilot certificate, with rotorcraft ratings for helicopters and instrument-helicopter, the most recent issuance of which was dated July 29, 1999. He also held a flight instructor certificate for helicopters. The most recent second-class medical certificate was issued to the pilot on July 3, 2001, without limitations. A review of FAA records found no accident or enforcement action history associated with the pilot.

The FAA airman records revealed that the accident pilot received a private pilot certificate with a rotorcraft-helicopter rating on June 25, 1993. His commercial certificate with a rotorcraft helicopter rating was first issued on March 25, 1996. He received a flight instructor certificate on June 11, 1997, and he was issued an instrument rating for helicopters on July 29, 1999.

The pilot was hired by Papillon for a line pilot position on September 14, 2000, and entered the company's Part 135 training program. According to the training records, he was given credit for completion of an AS350 helicopter line check and competency check in accordance with the requirements of 14 CFR 135.293 and 135.299 at his previous employer on August 3, 2000. He also received 20 hours of credit for previous ground training at his former employer. The pilot completed his initial ground training on September 16, 2000. He completed the ground and flight training for the SFAR Green Route 4 on September 22 and 26, 2000, respectively. He completed recurrent ground training on February 20, 2001, and his line and competency checks in the AS350, required by 14 CFR 135.293 and 135.299, on February 21, 2001.

According to company records, the pilot's total aeronautical experience consisted of 2,794 hours, all of which were accrued in helicopters, with 699 total hours flown in the AS350. In the 90 and 30 days preceding the accident, he had flown 224 and 86 hours, respectively.

A review of Papillon's ground and flight training program disclosed that the following topics were covered: settling with power; hovering in upwind, downwind, and crosswind conditions; and high-altitude takeoffs. The checkride training form used by the company contained a block for the instructor to sign off completion of the settling with power maneuver. There was also written material distributed to each pilot that described settling with power and the proper corrective actions. According to the chief pilot, settling with power was a required maneuver on all checkrides, and hydraulic transparency was a procedure discussed during the hydraulic emergencies segment of ground school.[7]

The investigation did not reveal anything unusual regarding the accident pilot's appearance, conduct, or performance during the 72 hours prior to the accident. The accident pilot reported for duty at Papillon operations in LAS at 0843 on the day of the accident. He was assigned to fly four tour flights that were scheduled to depart at 0945, 1230, 1515, and 1800. The dispatcher and several other pilots observed the accident pilot on the day of the accident, and they reported seeing nothing unusual regarding his behavior or demeanor.

The accident pilot was scheduled to fly the accident helicopter on all four of his scheduled flights. He completed the scheduled 0945 flight without incident. The 1230 flight was scheduled to carry six passengers. Five other Papillon helicopters were scheduled to depart on tour flights from LAS about the same time.

AIRCRAFT INFORMATION

Based on a review of maintenance and manufacturing records, the Eurocopter AS350-B2 helicopter, serial number 2477, had accrued a total time since new of 1,356 hours and 1,679 flight cycles. See figure 2 for a view of the AS350's main structural components. The helicopter was powered by a Turbomeca Arriel 1D1 engine, serial number 9155. According to factory records, this was the original engine installed in the helicopter at the time of manufacture. No previous overhauls were noted on the engine, which had accrued the same total time since new as the airframe. The engine cycles[8] were recorded as 1,696.55 for the gas producer, and 1,698 for the power section. In the previous 90 days, the helicopter had flown 39.0 hours, of which 37.6 hours were accrued in the past 60 days.

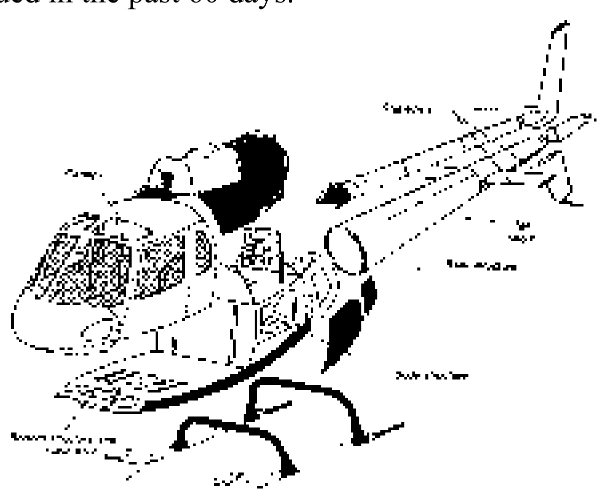

Figure 2. View of the AS350's main structural components.

[7]Settling with power and hydraulic transparency are discussed later in this report.

[8]According to representatives at Turbomeca, the cycle count difference is due to the factory procedure for determining cycle counts, based on power levels and time.

Helicopter Maintenance History

General

The accident helicopter was manufactured by Eurocopter on May 5, 1991, and delivered to a private individual in Japan under the Japanese registration number JA6081. The airframe and engine accumulated 1,307 hours while in Japan. According to an engineering report supplied by Eurocopter France and historical records from Turbomeca, the helicopter sustained structural damage during a hard landing and ground resonance event on September 16, 2000. The helicopter was deemed to be not economically repairable and was sold to Heliquip International, Ltd., New Zealand, in a nonairworthy status. The Safety Board's Maintenance Records Group examined the original Japanese maintenance records, and no export airworthiness certificate from Japan to New Zealand was found.[9] The helicopter was registered in New Zealand under the registration number ZK-HXT on March 4, 2001, and issued a New Zealand Civil Aviation Authority (CAA) certificate of airworthiness and an export airworthiness certificate on April 17, 2001. Papillon Grand Canyon Helicopters purchased the helicopter from Heliquip International on April 4, 2001, and it arrived in a crate at the company's facility on June 20, 2001.

New Zealand Maintenance Records

The New Zealand maintenance records given to Papillon with the sale of the helicopter were examined. These records included logbooks for the airframe and engine and a separate record book detailing airworthiness directive (AD) compliance. A copy of Heliquip International Job Sheet [work order] number H1083, detailing the repair of the helicopter, and a New Zealand CAA export airworthiness certificate, number 1/21L/30, were also included. No Japanese export airworthiness certificate was found in the New Zealand records.

Heliquip International Job Sheet H1083 was dated January 3, 2001. Under the nature of work description an entry states, "Carry out C-Check and prepare for a NZ certificate of airworthiness." Thirty-nine specific items of work are listed on the job sheet, and separate sheets attached to this job sheet detail the replacement parts used during this work. The job sheet notes compliance with three service bulletins (SB). Items 23 and 24 of the job sheet note the replacement of the left and right center beams. Items related to the engine are listed in items 22 and 35 through 38 and include the following items: engine droop check, engine power check, rear bearing oil flow check, and engine vibration check. A rubber stamp entry appears at the bottom of page 4 of the job sheet certifying that the work was in accordance with Part 43 of the New Zealand CAA regulations.

The first entry in the aircraft logbook is dated April 4, 2001, and notes the completion of the C check. This entry states that the helicopter was imported from Japan and that the C check was "carried out following Japanese report of a ground resonance problem." The entry refers the reader to Job Sheet H1083 at Heliquip International for full details. The entry further states, "In brief: Helicopter dismantled including removal of tailboom, canopy, skid gear, flight controls, and all rotating components. The following components dismantled for inspection: main rotor shaft assembly, main rotor head, vibration absorber and B1 beam. New starflex, frequency

[9]Subsequent inquiries by the JAAIC determined that the Japan Civil Aviation Authority did not issue an export airworthiness certificate for the aircraft.

adaptors, main rotor blade sleeves, outer spherical bearings, and spherical thrust bearing bolts installed." The entry concludes with a certification that the work was in accordance with Part 43 of the New Zealand CAA regulations.

Three flights are noted in the journey log. The time carried forward at the start of the page shows 1,307.8 total hours and 1,595 cycles; these times correspond to the hours in the last entry in the journey log of the Japanese records. These flights were conducted on April 11, 12, and 14 of 2001, and were 1.1, 8.1, and 1.4 hours long, respectively. The ending total time was 1,318.4 hours, which is the time carried forward in the Papillon maintenance records as the beginning total time.

The last two entries in the airframe logbook are dated April 17, 2001. The first entry is the issuance by the New Zealand CAA of a standard airworthiness certificate. The second entry is the issuance by the New Zealand CAA of an export airworthiness certificate.

The engine logbook maintenance section has two entries. The first one is dated April 4, 2001, and notes that the engine was imported from Japan installed in airframe serial number 2477 (the accident helicopter). The entry stated the following:

> 100-hour inspection carried out [in accordance with the] manufacturers maintenance manuals. Oil changed to Mobil 254. All AD's & special inspections complied with. Engine test run and vibration check carried out with result of 10 mm/sec. Rear bearing oil flow check 100 mls. Engine test flown and power check carried out…droop check and found satisfactory after adjustment ½ turn anti-clockwise. Engine considered suitable for [New Zealand certificate of airworthiness] issue.

The entry concludes with a certification that the work was in accordance with Part 43 of the New Zealand CAA regulations.

Japanese Maintenance Records

The Japanese maintenance records given to Papillon with the sale of the helicopter were reviewed. Interpreters were retained to translate the records, and the history of the helicopter was determined. Written translations were made of the entries listed for the year 2000 in the airframe and engine logbooks and from June 2000 to the end of records in the journey log.

The last entry in the airframe logbook is dated June 29, 2000, and notes a balance adjustment to the tail rotor and tail rotor drive shaft. The last entry in the engine logbook is dated June 27, 2000, and notes the engine's reinstallation in the airframe following completion of SB 292720246. The last entry in the journey log is dated August 12, 2000, and concludes with a helicopter total time of 1,307.8 hours. The notation section of this entry notes that a hard landing occurred.

Records for Inclusion of Hard Landing Inspections

American Eurocopter specifies required hard landing inspection items in section 05.53.00.605-01 of its AS350-B2 Maintenance Manual. Two separate inspections are listed in that section, the first addresses hard landing events associated with autorotations and the second addresses hard landings under power. No maintenance or inspection activity was recorded in the Japanese records after the August 12, 2000, flight.

The specific entries in the New Zealand airframe and engine logbooks and Heliquip International Job Sheet H1083 were compared to the hard landing inspection items (hard landing under power) listed in the AS350-B2 Maintenance Manual. The only items not documented as having been accomplished are (1) tail rotor drive shaft alignment per work card 65.10.00.602 and (2) landing gear inspection per work card 32.13.00.601.

Turbomeca specifies required hard landing inspection items in chapter 71-00-04 of its Arriel 1D1 Maintenance Manual. Two separate inspections are listed in that chapter; the first addresses hard landing events associated with blade strikes and the second addresses hard landings under power without blade strikes.

The specific entries in the New Zealand airframe and engine logbooks and Heliquip International Job Sheet H1083 were compared to the hard landing inspection items (hard landing without blade strikes) listed in the Arriel 1D1 Maintenance Manual. The only item not documented as having been accomplished is a dye penetrant inspection of the starter generator and fuel control mounting flanges.

Recent Maintenance and Inspection History

Papillon's maintenance program utilizes the manufacturer's service recommended inspection program for both the airframe and engine. In addition, the company performs annual inspections in accordance with 14 CFR Part 43. The company also follows the manufacturer's recommended additional guidelines for sand and dust environments and elected to exclude the corrosive environmental guidelines and related inspections due to their exclusive operation in a dry, desert climate. The company also complies with all manufacturer SBs within the time specified.

After delivery to the Papillon maintenance facility, numerous maintenance inspections and modifications were accomplished. The modifications were performed either in accordance with a supplemental type certificate or field approvals under FAA Form 337. Papillon applied for a U.S. certificate of airworthiness during this maintenance and modification process. A conformity inspection was conducted by an FAA designated airworthiness representative (DAR),[10] and an airworthiness certificate in the normal category was issued on August 2, 2001.

Following completion of this work and a functional check flight on August 3, 2001, the helicopter was placed into revenue service at an aircraft and engine total time of 1,319.4 hours. Six routine scheduled maintenance inspections were performed between the date that the aircraft was placed into revenue service and the date of the accident. No unresolved discrepancies were noted in the maintenance records.

Airworthiness Directive and Service Bulletin Compliance

[10]A DAR is not a government employee. A DAR is a private individual who is appointed by the FAA to conduct certain activities related to the inspection and certification of aircraft on behalf of the FAA. These individuals typically charge a fee to the owners/operators of the aircraft for this service.

A comparison of the maintenance records with a listing of FAA ADs and manufacturer SBs applicable to the airframe and engine disclosed that the company had complied with all ADs and SBs.

Operator's Maintenance Personnel Training and Signoff Procedures

The maintenance records revealed that personnel working on and signing for discrepancies for the accident helicopter and other personnel at the company's Las Vegas base had no previous formal factory training on the AS350-B2 or Turbomeca Arriel 1D1. Furthermore, work performed on the accident aircraft was at times inspected by the same person that performed the work. A review of the selected records of several of the operator's AS350 fleet at the Las Vegas base disclosed that entries generally contained a description of the work performed, but did not include specific maintenance manual references for accomplishment of the work.

The maintenance and record-keeping requirements for operations under 14 CFR Part 135 are governed, in part, by 14 CFR 135.411, which prescribes rules in addition to those contained in other parts of the regulations for the maintenance, preventive maintenance, and alterations for each certificate holder. For aircraft with nine seats or less, the regulation allows the operator to select less stringent requirements concerning the number, job categories, and training of its maintenance personnel. Specifically, 14 CFR 135.411(a)(1) permits operators to use the more general maintenance personnel and record-keeping requirements found in 14 CFR Subparts 91 and 43.

The pertinent provisions of 14 CFR 43.9 allow FAA-certificated maintenance personnel to perform work on all aircraft without the requirement for make- and model-specific maintenance training. FAA-certificated maintenance technicians may also inspect and sign off their own work for return to service. The record-keeping requirements of this regulation also allow the maintenance technicians to record a description of the work performed without specific reference to maintenance manual sections, pages, or work cards.

For aircraft with 10 seats or more, 14 CFR 135.411 requires the operator to comply with the maintenance program provisions specified in 14 CFR 135.423 through 135.443. These parts require, in part, that the operator's maintenance personnel have make- and model-specific maintenance training, that trained and authorized inspectors other than the technician performing the work inspect any maintenance activity, and that more stringent record-keeping requirements be in place.

Weight and Balance

A weight and balance computation was performed for the helicopter with gross weight and center of gravity data points established for departure from LAS, GCW, and the accident site. In accordance with the helicopter's FAA-approved type certificate data sheet, the maximum allowable ramp weight and takeoff gross weight were 4,961 pounds. Based on the load manifest weights, the estimated fuel consumed to the point of refueling and the amount of fuel loaded at GCW, the last departure gross weight was estimated to be about 4,554 pounds (2,065.66 kilograms [kg]). Using the fuel flow figures in the AS350 rotorcraft flight manual (RFM), it was estimated that about 38 pounds of fuel were consumed after takeoff from GCW, for a computed gross weight of 4,515 pounds (2,047 kg) at the time of the accident. The lateral center of gravity was at zero for all conditions. The longitudinal center of gravity averaged 127 inches, with an allowable envelope between 125.5 and 136 inches.

Hydraulic System Shutoff Switch

A toggle switch is mounted on the end of the collective control lever, which allows the pilot to manually turn off the hydraulic system by depressurizing the pump output. Prior to 1990, the switch was guarded by perpendicular walls the height of, and on each side of, the toggle. In response to in-service reports (including one accident) in which inadvertent deactivation of the hydraulic system occurred by objects (principally sleeve cuffs and straps) catching on the switch, Eurocopter changed the design of the switch guard, which incorporated a horizontal, fixed-plate cover over the toggle switch to preclude inadvertent movement of the toggle. With the new guard, two separate motions are required to first reach the toggle, and then move it. SB 67-17R2 was issued by Eurocopter on October 25, 1990, and called for replacement of the old-style guards with the new ones. A review of Eurocopter manufacturing documents and airworthiness conformity documents in the maintenance records disclosed that the accident helicopter was equipped with the new-style guard, with SB 67-17R2 incorporated on the production line.

Fuel Tank Design and Certification Testing

The 540-liter capacity fuel tank is manufactured from a spin-molded polymide and does not have internal baffles. The tank is mounted in the fuselage behind the passenger compartment and below the main transmission deck, between the left and right baggage compartments. According to manufacturing drawing process specifications, the wall thickness varies according to measurement location on the tank. The manufacturing conformity acceptance test requires the tank to hold an internal pressure of 0.5 bars (7.14 psi [per square inch]).

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. The predecessor to the AS350-B2 is the AS350-B, which received an FAA type certificate on November 9, 1978. The AS350-B2 design variant was approved by the FAA on June 8, 1990. 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. There are no exemptions or equivalent level of safety findings listed for the fuel tank.

The regulatory change history of 14 CFR Part 27, "Airworthiness Standards: Normal Category Rotorcraft," that pertains to requirements for fuel tanks and tank crash resistance was examined. 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 tank's 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 tank's 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. The company provided report number 350A.06.2064, 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 tank's 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.

Engine Description

The Turbomeca Arriel 1D1 is a free-turbine, fixed-geometry, turboshaft engine, which is rated at 732 shaft horsepower (takeoff) and features a modular design. The gas generator section consists of a one-stage axial, a one-stage centrifugal compressor, an annular combustor with centrifugal fuel injection, and a two-stage gas generator turbine. The rotational speed of the gas generator is determined by the rate at which fuel is burned in the combustor. Gas generator exhaust gases drive a single-stage power (free) turbine, which drives a reduction gear assembly. Torque is transferred forward through a power turbine shaft to an accessory gearbox mounted at the front of the engine.

The Arriel engine provides the helicopter with a conventional fuel control system in which the collective pitch of the aircraft rotor establishes the power output of the engine. Generally, the helicopter rotor speed is held constant. As rotor collective pitch is changed, the load on the engine power turbine changes, tending to change its speed. The engine fuel control unit corrects for this by varying fuel flow, which changes the gas producer speed to supply the power required to maintain constant power turbine and helicopter rotor speeds.

METEOROLOGICAL INFORMATION

Company Weather Briefing Procedures

According to representatives of Papillon, the first pilot to arrive at LAS each morning was responsible for obtaining the current weather and forecast from the FAA flight service station. That information was then posted for the rest of the pilots to review upon their arrival. The chief pilot was responsible for updating the weather throughout the day, and he would notify the dispatcher if any weather phenomena threatened to become a hazard for company flights.

According to the Papillon chief pilot's records and statement, on the day of the accident, the weather was forecast to be sunny throughout the day with a slight chance of thunderstorms.

Weather Reports, Forecasts, and Pertinent Data

A Safety Board meteorologist conducted a study of the atmospheric conditions at the accident site during the timeframe of the occurrence.

The closest official weather observation station to the accident site is the Kingman, Arizona, airport, which is 43.7 nautical miles south. The station's elevation is about 700 feet lower than the impact location. At 1356, the station reported winds 190° at 10 knots gusting to 15 knots; visibility 10 miles; scattered clouds at 7,500 feet; temperature 36° Celsius (C), dew point 17° C; altimeter setting 30.07 inches of mercury (Hg).

At 1436, the station reported winds 320° at 12 knots; visibility 10 miles; clear skies; temperature 34° C; dew point 16° C; altimeter setting 30.05 inches of Hg. The peak wind and a wind shift were recorded at 1416. The peak wind was 320° at 28 knots. The station pressure at 1356 and 1416 was recorded as 26.57 inches of Hg and 26.54 inches of Hg, respectively.

Weather Surveillance Radar-88D (WSR-88D) Doppler weather radar images from the Las Vegas, Nevada (KESX), station were reviewed. The KESX radar is located 46.9 nautical miles from the accident site on an approximate bearing of 248°. At the accident site, the KESX radar beam center is about 8,900 feet with a beam width of about 4,700 feet. The images show weak weather radar echoes (weather radar echo intensities greater than 25 decibels) in the accident area. The images show intense weather radar echoes (weather radar echo intensities greater than or equal to 50 decibels) about 20 to 30 nautical miles east of the accident site.

Although weak weather radar echoes were noted in the area of the accident, Geostationary Operational Environmental Satellite (GOES) 10 visible satellite imagery for 1400, 1430, and 1500 showed clear skies in the accident area. Convective activity was noted in the 1430 image about 20 nautical miles east and south of the accident site.

The radar Base Radial Velocity images centered on the accident site for the times 1420, 1426, 1432, and 1438 generally showed radial velocities of 5 to 10 knots away from the radar (westerly component) in the accident area. However, radial velocities of about 45 knots toward the radar (easterly component) were seen in the 1420 and 1432 images.

The 1420 image showed a radial velocity of about 5 knots away from the radar at the accident site. A radial velocity of about 45 knots toward the radar was detected about 2 nautical miles north of the accident site. The 1432 image showed a radial velocity of about 0 knots at the accident site. A radial velocity of about 45 knots toward the radar was seen about 5 nautical miles northeast of the accident site.

A National Weather Service Senior Meteorologist at the Las Vegas, Nevada, Weather Service Forecast Office, was interviewed by Safety Board investigators via telephone on November 9, 2001, concerning the weather radar images. He reported that the weak weather radar echoes seen in the accident area were probably small cumulus clouds and ground clutter. The 45-knot, in-bound radial velocities seen in the Base Radial Velocity images were likely spurious data and nonmeteorological.

Using data from all the sources, the pressures and temperatures for the listed altitudes were estimated for the time and area of the accident. These pressures and temperatures are shown in table 1 below.

Table 1. Pressure and temperature valves for various altitudes on day of the accident.

Altitude (feet msl)	**Pressure** (milibars)	**Temperature** (degrees C)
4,000	878	36
5,003	850	34
7,146	789	24

Papillon company pilots who landed at the accident site within minutes of the accident were asked about the atmospheric conditions. One pilot noted that the winds were calm and the temperature was approximately 106 degrees Fahrenheit (41.1^{o} C) and that there was no turbulence. A second pilot said that the weather at the time was hot, light winds, and very light to moderate turbulence. When asked for a clarification of the turbulence, the pilot noted that he got "jostled a little" but nothing to cause a problem. A third pilot said that the weather was clear, sunny, hot, and that there was no turbulence at all, especially when crossing the ridge.

FLIGHT RECORDERS

The helicopter was not equipped with a cockpit voice recorder (CVR) or a flight data recorder (FDR), nor were they required by FAA regulations. The Papillon maintenance department had intended to install a maintenance-trend monitoring system on the helicopter, but had not yet scheduled this work. The system considered for installation would have sampled various engine parameters and would have stored these in nonvolatile memory for maintenance technician retrieval.

WRECKAGE AND IMPACT INFORMATION

The accident site was located on the western side of an escarpment known as Grand Wash Cliffs in a steeply sloped, mountainous area covered with sparse brush, hard dirt, loose rocks, and Joshua trees indigenous to the high Arizona desert. Geologically, the Grand Wash Cliffs consist of small plateaus stair-stepped up from a flat desert floor that culminate into a mesa. See figure 3 for a profile view of the accident site.

Figure 3. Profile view of the accident site. Note downsloping terrain.

The accident site was located at an elevation of 4,041 feet msl. The average elevation of the top of the mesa formed by Grand Wash Cliffs is about 5,500 feet msl. The accident site was just outside the SFAR 50-2 (Green Route 4) area of the Grand Canyon National Park.

An easterly energy path consisting of ground scars and wreckage debris was identified oriented in the direction of the cliff face and rising terrain. The energy path was approximately 180^{o} from Papillon 34's last known direction of flight. All ground scars were confined to the immediate vicinity of the wreckage mass within the approximate diameter of the main rotor.

Papillon tour helicopters and other operators based at Las Vegas generally cross the Grand Wash Cliffs westbound at 5,800 msl. The accident helicopter's last known direction of flight was from GCW toward LAS, on a median magnetic heading of 265^{o}.

The Safety Board's Structures Group examined the wreckage on site from August 12 through August 17, 2001. FAA inspectors from the Las Vegas Flight Standards District Office (FSDO) arrived on-scene along with rescue personnel and made initial on-scene observations during the initial law enforcement rescue response phase. The inspectors photographed ground scars and impact marks before rescue personnel disturbed the area during the course of victim recovery operations. The on-scene investigation revealed no evidence of an in-flight fire.

Three separate aerial and ground surveys were conducted from points on the plateau at the cliff edge along possible approach paths to the accident site. No ground disturbances or other items associated with the helicopter were observed during these aerial and ground surveys.

Due to thermal effects from the postcrash fire, bolt torque on all of the examined components could not be determined.

The main wreckage area distribution, including helicopter debris, tree strikes, and ground impressions, were contained in an area approximately 40 feet in diameter. See figure 4 for an aerial view of the crash site.

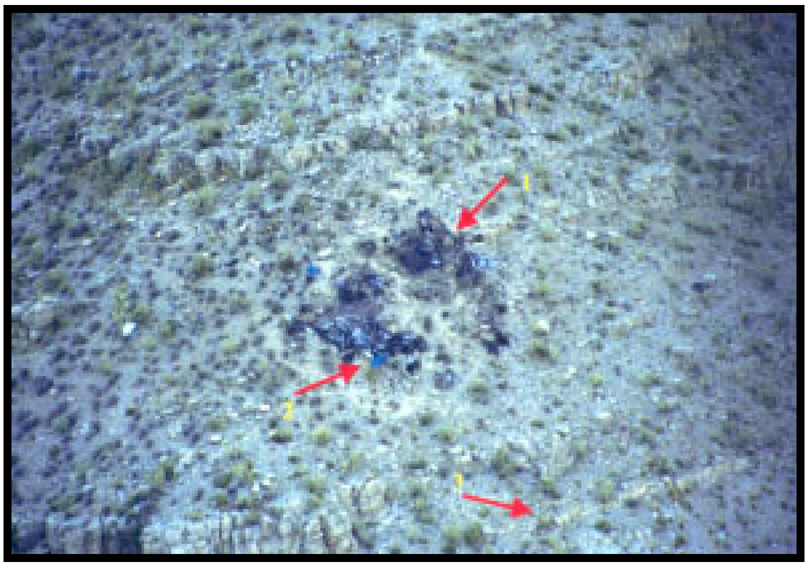

Figure 4. Aerial view of the crash site. Top arrow denotes first identified point of contact (IPI). Middle arrow denotes resting point of main wreckage. Lower arrow denotes right door. Note that the crash site is compact.

The first identified point of impact (IPI) contained an outcropping of boulders. Various portions of the center fuselage belly panel, canopy glass, and all of the landing gear with the exception of the aft cross tube were found on and immediately around the boulder outcropping.

The main wreckage came to rest about 20 feet downslope of the IPI. Using the IPI as the center point, the median bearing of the upslope debris path was approximately 085°. Charring and soot patterns propagated from the IPI down hill in a V-shaped pattern.

The farthest piece of the helicopter wreckage from the IPI was the right door. It was located about 400 feet directly downslope from the IPI and was found in multiple pieces. The fracture surfaces on the door pieces were angular, irregular, and granular in character. The upper and lower hinges were distorted, with the door-side hinge gudgeons[11] split and elongated. The hinge pins

[11]A gudgeon is a socket for the pin of a hinge.

were attached to the door jettison arm and rod mechanism and were found adjacent to their respective hinge gudgeons in positions dimensionally similar to the pins being fully seated in the gudgeons. The bottom forward corner of the right front door contained longitudinal scrape marks consistent with the rock-covered slope between the door and the IPI. The corresponding fuselage doorpost was found in the vicinity of the IPI and had separated from the structure, with distortion evident in the vertical direction (lengthwise on the member).

The left front door was found up slope from the IPI. (See figure 5.) It had separated from the fuselage with a portion of the forward fuselage doorpost attached to the upper and lower hinges. The doorpost section was torn from the fuselage structure. The lower hinge was bent and distorted, whereas the upper hinge was intact and undamaged. The hinge pins were fully seated in their corresponding gudgeons and also remained attached to the door jettison arm and rod mechanism, with the internal cockpit-side jettison handle in a near normal position. The door-operating handle was found in the closed and latched position.

Figure 5. View of crash site showing left door (in foreground) found upslope from IPI.

The pitot tube, which had been mounted on the nose of the helicopter, was found about 2 feet uphill of the IPI.

Ground impressions, scars, and tree strikes were upslope of the IPI over an arc between magnetic bearings from 080^{o} to 110^{o}. Two Joshua trees, centered on, and upslope of, the IPI exhibited angled cuts in the severed trunks. From the IPI, a ground scar and a Joshua tree were observed to the left at the 080^{o} bearing, and another Joshua tree was observed to the right at the 110^{o} bearing. Both trees were on an arc with about a 20-foot radius.

According to Eurocopter technical documentation, the rotor disk diameter is approximately 35 feet, and the main rotor blades rotate in a clockwise direction when viewed from above. The Joshua tree to the left (080^{o}) of the IPI was lying on its side showing upward cuts near the base adjacent to an arcing, thin ground scar. The Joshua tree to the right (110^o) of the IPI had an upward cut, which was flat and smooth on the end facing north and splintered on the opposite side (toward the south). Both of the cut Joshua trees were within a rotor disk radius of the center of the IPI. Damage patterns on the main rotor blades were principally on the lower surfaces and leading edges of the blades.

The right forward cross tube and segmented pieces of the skid tube to the aft cross tube leg were located at the IPI. The skid tubes contained large dents and scrape marks that were not parallel to the longitudinal axis of the skids. The skid tubes also exhibited bends about midspan. The right spring steel extension was separated from the skid tube. The left spring steel extension was bent up 70^o from its original position on the airframe. The right forward skid toe exhibited fire damage. The left forward skid toe was not fire damaged. The aft cross tube and skid heels were located with the main wreckage.

Examination of the fuselage wreckage disclosed that the floor of the cockpit was inverted in the main wreckage area. A portion of the aluminum tail boom aft of the horizontal stabilizer was separated, with the fracture features uneven and jagged in nature. The separated portion of the tail boom and the nose of the helicopter came to rest adjacent to each other. The yellow main rotor blade[12] was lying on top of the tail boom. The blue and red main rotor blades were oriented 180^o from the tail boom, laying on top of each other (the blue blade was on top of the red blade) and adjacent to the engine. (See figure 6.) The engine came to rest behind the main gearbox debris in its normal relationship to the fuselage.

Figure 6. View looking upslope from IPI. The tail rotor blade is shown in right half of photograph.

[12]Helicopter rotor blades are color coded for maintenance identification purposes.

The tail rotor blades were bowed and had minor leading edge damage. One strike tab was bent aft and exhibited chordwise scratching. The other strike tab was bent inboard and also exhibited chordwise scratching.

The primary fuselage structure is of aluminum construction. The postimpact fire consumed the majority of the cabin area aft of the pilot's seat. The secondary structure is mostly of composite construction and consisted of the canopy, doors, fairings, and windows; these structures were also almost totally consumed by fire.

The helicopter was equipped with the following doors: two hinged cabin doors on either side for pilot and forward passenger entry, one subdoor on the right side, and a large sliding door on the left side. The two front doors are hinged to the doorpost adjacent to the front windows and open forward. All of the doors were accounted for at the accident site.

The postimpact fire consumed all of the aluminum control tubes aft of the pilot's seat to the transmission deck. The collective pitch lever and cyclic stick suffered thermal damage, and all of the associated electrical switches on the grips were thermally consumed. No determination could be made of the position of the hydraulic cutoff switch. The cyclic stick was frozen and could not be moved. The collective pitch lever was manually raised and lowered, with corresponding movement noted to the associated collective pitch lever control system rods in the forward cockpit area that were not consumed by fire.

The center console was thermally consumed, and the positions of the switches in this area including the hydraulic system test switch and the electrical master switch could not be determined. The rotor brake lever was found in a normal operating position. The throttle lever was found in the Normal Flight (gate detent) position.[13]

Flight control continuity could not be established from the pilot's station to the hydraulic servos due to thermal destruction of the aluminum push-pull and torque tubes; however, all of the 17 steel flight control rod end fittings associated with these tubes were accounted for, with no unusual damage or operating signature pattern evident on the bolts, nuts, safety devices, or clevis bolt hole bores. Some of the flight control rods with their associated end fittings were intact in that portion of the fuselage not destroyed in the postimpact fire.

The left lateral and fore/aft main rotor hydraulic servos remained attached to the flared housing (top portion of the main rotor gear box [MGB]). The right lateral main rotor hydraulic servo and the flared housing attachment lug were consumed in the postimpact fire. The upper attachments for the hydraulic servos were separated from the main rotor nonrotating swash plate at the rod end bearings. The housing of the left-lateral hydraulic servo was partially consumed in the postimpact fire, and the rod end fracture face was angular and granular in appearance. The fore/aft hydraulic servo was also partially consumed; however, no fracture surface remained. The housing of the right-lateral hydraulic servo was completely consumed in the postimpact fire, and no fracture surfaces remained.

[13]The lever-type throttle control is mounted on the control quadrant below and behind the lower center console and has the following operating positions: Stop (fuel flow off), Normal Flight (gate-detent), and Emergency Override.

The antitorque pedals on the pilot's side were separated from the structure, and each pedal was bent forward.[14] At the tail boom separation point, movement of the pitch change push-pull tubes resulted in pitch change of the tail rotor blades.

Thermal processes consumed the fuel tank. Both boost pumps were thermally damaged. All aluminum fittings and lines at the fuel tank, boost pump, and fuel filter areas were thermally destroyed. The steel fuel fittings and connections located in the engine compartment aft of the engine firewall were found with the main wreckage and were exposed to thermal processes. The fuel level transmitter in the fuel tank was separated in three pieces. The fuel shutoff valve was found in the open position.

All three arms of the Starflex were fractured and separated from the main body of the Starflex. All three main rotor head sleeve sets were thermally consumed/destroyed by fire, leaving only the fiberglass rovings. The main rotor blade retaining pins were in place. The spherical thrust bearings[15] were thermally damaged; however, all of the steel laminates making up the spherical thrust bearings were present. Two of the three frequency adapters (elastomer blocks) were thermally consumed/destroyed. The third frequency adapter was thermally damaged and remained on the sleeve in its normal position.

All three of the main rotor blades were principally intact; however, the blades exhibited trailing-edge, spanwise buckling and separation of the lower and upper surface skins as well as broomstrawing[16] of the tips. All three main rotor blades exhibited chordwise scratching on the lower surfaces as well as denting of the leading edges.

The yellow blade came to rest on top of the tail boom. The yellow blade's trailing edge was separated near the tip. The remaining trailing edge of the blade was delaminated over most of its span. The blue and red blades came to rest adjacent to the engine, with the blue blade lying on top of the red blade. The blue blade exhibited fire damage with the trailing-edge portion split open and burned. The red blade had leading-edge, tip-end impact damage, with the composite fibers fanned out in a broomstraw pattern at the blade tip.

The main rotor pitch change rod[17] for the blue main rotor blade was intact and bent. The main rotor pitch change rods for the yellow and red main rotor blades were thermally destroyed leaving only the rod ends, which were connected at the pitch change horn and rotating swashplate.

The engine-to-MGB drive shaft was separated from the MGB input pinon coupling tube. The flexible coupling[18] between the shaft and engine splined adapter flange was intact and connected

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[14] This helicopter was configured for tour operation with the left cyclic, collective, and antitorque pedals removed. The copilot's seat was also removed and replaced with a two-place bench seat.

[15] The spherical thrust bearings consist of laminated sandwich structure of steel cups and thin elastomer sheets.

[16] The damage pattern called broomstrawing is characterized by longitudinal spanwise fracturing/splintering of the composite unidirectional fibers with the fanning out of these fibers in a flared pattern reminiscent of a broom straw.

[17] The pitch change rods connect the rotating swashplate to the pitch change horns of the main rotor hub.

[18] The flexible coupling is designed to absorb minor misalignments in the drive shaft segment between the engine power takeoff and the MGB. The unit is composed of a series of thin stainless steel disks, which are stacked between the triangular three-armed flanges of the adjacent shaft connections. Each flange arm of one shaft is separated by 120^{o} and the unit is assembled with the adjacent shaft's flange arms at 60^{o} from the first flange arms. Six bolts are used to secure the connection between the shaft flange arms and the stack of steel disks.

with all six bolts to the flange arms of both the shaft and the splined engine side flange. The flexible coupling was distorted in an S-bending pattern. The engine-side shaft flange splines were smeared around the circumference to a depth of about ¼ inch.

Portions of the flexible coupling on the MGB input coupling tube flange were retained by two of its three bolts. The remaining attachment hole was elongated and bent forward. No bolts or flexible coupling segments remained on the shaft side of the MGB end of the drive shaft. The attachment holes on the shaft flange arms were elongated and gouged opposite the direction of rotation, and the arms were bent aft toward the engine. Dents and nicks were observed on the side of the arms opposite the direction of rotation. During detailed examination of this area of the wreckage, several pieces subsequently identified as components of the missing shaft side flexible coupling were found in the ash debris. A bolt was found in a fragment of flexible coupling band. The bolt had failed in shear, and the nut was not recovered. Another flexible coupling fragment was found in association with a nut that was split open. The three remaining nuts, bolts, and associated flexible coupling fragments that make up the whole coupling were not recovered.

The postimpact fire consumed the engine-to-MGB drive coupling housing and gimbal (universal) joint; however, all four gimbal joint pins were recovered with their safety pins in place. The postimpact fire also consumed the MGB magnesium housing. The internal gears were exposed with magnesium ash deposits present, and no prefire damage or other unusual operating signature was observed on the gear teeth.

The forward portion of the tail boom structure was mostly consumed in the postimpact fire; however, portions were recovered containing the tail rotor compensator mechanism, and the horizontal stabilizer.

The tailskid was bent up into the lower portion of the vertical stabilizer. The lower vertical stabilizer was bent out to the right about 30^{o}. The upper vertical stabilizer had one perforation at the trailing edge.

The tail rotor drive system was comprised of an intermediate drive shaft, a long drive shaft, and the tail rotor gearbox (TGB). The intermediate drive shaft was separated and found about 5 feet from the main wreckage. The forward end of the intermediate drive shaft, where it attaches to the engine, remained attached through a flexible coupling. The coupling was fractured and exhibited "fanning" of the laminates. No rotational scoring was observed on the splined adapter or the flexible coupling at the aft end of the intermediate drive shaft. No rotational scoring was observed on the steel splined fitting on the forward end of the long drive shaft. The long drive shaft was fractured in three places and exhibited thermal distortion and flattening. All five tail rotor drive shaft support bearings were recovered at the site. Bearing 1 (nearest to the TGB) exhibited fire damage; however, the bearing rotated with no noticeable binding. The long drive shaft aft of bearing 1 was attached to the TGB with the flexible coupling and was intact. Bearings 2 through 5 exhibited moderate fire damage and were seized.

No visible damage was apparent to the exterior of the TGB. The tail rotor was manually rotated with no binding of the TGB evident. Both tail rotor pitch change rod ends were bent on the shaft just below the spherical bearings. One of the tail rotor pitch change rod ends had fractured and exhibited an angular and granular fracture face. The pitch change horns were damaged and bent.

The forward landing gear cross tube was fractured at the right side fuselage attachment point. The separated portion was bent forward. Both of the forward landing gear shock absorbers had separated from the forward cross-tube attachment points. The aft landing gear cross tube remained intact and attached to the cabin floor. The two aft attachment clamps were found rotated approximately 90^{o} forward of their normal position. The right leg of the aft cross tube was bent slightly aft in relation to the left leg.

The skid tubes were fractured and separated at the attachment points of the cross tubes. The right forward skid toe exhibited fire damage. The left forward skid toe was not fire damaged. The steps appeared undamaged and were located at the IPI.

Members of the Safety Board's Powerplants Group documented the engine at the accident site on August 12, 2001. The engine was found resting on its right side,[19] in its normal location and attitude with respect to the aircraft fuselage. A ground burn pattern was evident at the IPI and extended about 20 feet downslope, where the main body of the wreckage lay. The aircraft fuel cell, forward tail boom, fuselage body structures, and the cabin aft of the cockpit had been consumed by fire.

The engine bay deck had deformed around the engine's lower right and bottom sides; only the left and rear sides were accessible to inspection. The engine was burned in areas unprotected by the deck material but was otherwise intact, with no discernable evidence of catastrophic mechanical failure or in-flight fire. Visible portions were coated with fire extinguisher chemicals and pieces of resolidified metal material. Re-formed metal droplets noted at component edges were consistent with the resting attitude of the engine. The area of greatest fire damage was observed at the front of the engine between 12 and 5 o'clock, where the accessory gearbox and engine inlet were exposed to the ground fire encompassing the airframe and transmission. Portions of the engine fuel and air system installed in this area had been consumed by fire. The front engine mount had been destroyed. One clamp of the two-clamp rear engine mount-attaching device remained secured to the engine deck; the second clamp was fire damaged and no longer intact. The throttle quadrant pointer on the fuel control unit was positioned at 55° (full open, the normal flight position). It was possible to access the axial compressor rotor and free turbine by hand; neither could be rotated manually. All visible fuel and air connections were intact. The engine was no longer connected at its output drives.

TESTS AND RESEARCH

Engine and Fuel Control Unit Examinations

Engine and fuel control unit teardown examinations were conducted at the Turbomeca Engine Company facility in Grand Prairie, Texas, from August 28 to 30, 2001, under Safety Board supervision.

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[19]All references to position are looking forward from aft.

Engine Teardown

Examination of the engine found indications of normal operation at impact, and no evidence of preimpact failure. No condition that would have prevented normal operation of the fuel control was found upon examination and testing of all recovered external components.

Free turbine and gas generator rotation were checked at the free turbine wheel and the axial compressor, respectively; both drive trains were seized.

The accessory gearbox front support casing had separated circumferentially 1 to 2 inches aft of the forward flange, with portions lost to thermal damage. The gearbox remained attached between 3 and 5 o'clock. A raised and distorted area was noted on the outer diameter between 1 and 4 o'clock just aft of the separation, with corresponding rotational scoring on the inside diameter between 10 and 2 o'clock. The forward flange hardware was intact, with portions of the (airframe provided) coupling tube aft mating flange still captured under the bolt heads.

The gas generator drive train remained seized after the axial compressor assembly was removed.

Both compressor casings suffered some light thermal damage, but there was no obvious case distortion. The 1st-stage compressor stators were intact. There was damage to the leading edges of 10 of the 13 axial compressor rotor blades, and the tips were slightly bent opposite the direction of rotation. Three blades were missing small amounts of blade material from the leading edge tips. There was blue discoloration 360° around the inner diameter of the compressor cover coincident with the plane of centrifugal rotor vane rotation. The leading edges of the centrifugal rotor vane displayed blue discoloration corresponding with the cover damage. There was one nick in the inducer region of one vane leading edge.

The 1st-stage turbine wheel blade path showed light rub indications, and there were evenly distributed, shiny metallic deposits on the suction (low pressure) sides of the 1st-stage nozzle vanes. There were indications of light tip rub on the 1st-stage turbine wheel blades, and light, evenly distributed metallic deposits on both high pressure and suction (low pressure) sides of all blades. The 2nd-stage nozzle vane leading edges showed evenly distributed metallic deposits and no mechanical damage. There was a rub mark along the 2nd-stage blade path (360° contact).

The 2nd-stage wheel airfoils were found intact. Light, evenly distributed, metallic deposits were found on both high pressure and suction (low pressure) sides of the 2nd-stage wheel airfoils. The turbine rear bearing was intact with free and smooth rotation. The bearing cage oil delivery port was unobstructed. The free turbine assembly rotated freely after separation from the turbine reduction gearbox. The reduction gearbox gear train was seized but was freed after application of penetrating oil. The oil tubes were unobstructed. The continuity of the gear train was confirmed. The bearings were heat discolored but were intact and rotated freely.

The freewheel shaft was heavily coated with soot. Rotational scoring was noted on the aft freewheel housing flange; this damage was consistent with the scoring noted on the inner wall of the front support casing. Freewheel unit sprag clutch operation was confirmed. The unit was disassembled with no anomalies noted.

The lock securing the nut to the forward end of the externally splined freewheel shaft displayed metal displacement (smearing), and the initial 1/4 inch of the splines in this area showed severe 360° scoring damage, with corresponding scoring noted on the female splines of airframe drive shaft portion.

Both oil strainers were found intact and free of debris. The oil pump was disassembled with no anomalies found. No evidence of contamination was found in the oil filter. The reduction gearbox magnetic plug was inspected and found clean. A small amount of oil was present at this fitting, and it had a clean appearance and no burnt odor. This oil was collected for analysis. The electric magnetic plug located at the oil return to the aircraft tank was found separated from the engine. The electric magnetic plug located at the rear bearing return was intact. Both plugs were inspected and found clean. The pressure oil line to the front section, rear bearing, reduction gear, and power turbine (four-way union) was intact. The breather-to-tank-to-general air vent was fire damaged but intact.

Fuel Control Unit

The fuel control unit (FCU) incorporates a gas generator control, power turbine governor, fuel pump and filter unit in a single, double-drive accessory. Portions of the gas generator control components were consumed or damaged by the ground fire. Teardown examination of the FCU disclosed no anomalies other than thermal exposure damage. No fuel was found in the unit.

At the request of Safety Board investigators, Turbomeca (France) supplied the investigation with an engineering analysis of the failure modes and subsequent effects on engine power generation for failures of the FCU components that were thermally consumed. The analysis found no component failure effects that could have negatively influenced engine power generation.

Metallurgical Analysis of Blade Material Deposits

A blade removed from the 2nd stage turbine wheel assembly was submitted to the Safety Board's Materials Laboratory, Washington, D.C., for chemical analysis of the shiny deposits noted throughout the turbine gas path. These deposits were shiny, smooth, and could not be removed with a knife or pencil eraser. X-ray energy spectroscopy analysis identified the material as aluminum.

Engine Oil Analysis

Analysis of the oil sample removed from the reduction gearbox found it contained iron, copper, chromium, and nickel levels below the maintenance action thresholds for these elements.

Airframe and Drive Train Component Examinations

The Structures Group reconvened at the Eurocopter facilities in Grand Prairie, Texas, from August 29 to 31, 2001, for detailed component examinations.

Main Rotor Drive Shaft and Mast Components

The main rotor shaft was intact, and exhibited thermal damage and discoloration. The upper bearing of the rotor shaft exhibited thermal damage, with no evidence of scoring or threading observed on the shaft. The lower bearing was a four-contact type bearing and was thermally damaged. Manual rotation of the bearing was not possible due to molten debris deposits in the bearing and thermal damage. The two bearing seals were thermally damaged.

All three clevises of the nonrotating swashplate were intact and contained the steel hydraulic servo rod end fittings. All three of the rod end-fitting bolts were in place with their safety pins intact.

The pitch change rods connect the rotating swashplate to the pitch change horns of the main rotor hub. The pitch change rods for the yellow and red blades were thermally destroyed. The third pitch change rod remained attached to the blue main rotor blade and was thermally distorted.

The drive scissors were thermally destroyed. All bolts were accounted for and were safe-tied. The stationary scissors were thermally destroyed. All bolts were accounted for and were safe-tied. The Uniball[20] was thermally discolored. Three dents dimensionally and geometrically similar to the contact sleeve of the stationary star were observed on the ball. The axes of these dents were on three separate diagonal planes. The flared (conical) housing also functions as the top cover for the MGB and was partially destroyed by thermal processes.

Examination of the swashplate bearing disclosed that it was thermally affected and could not be rotated. For this reason the bearing was cut open to examine it for preimpact defects. The bearing cage was thermally damaged, and the seals were thermally consumed. The bearing race and balls were thermally damaged, with ash deposits found. No brinelling or galling was noted on the race or the balls.

Main Rotor Hub and Blades

Components of the main rotor hub assembly examined included the main rotor sleeves, antivibration assembly, and the Starflex.

The main rotor sleeves were thermally consumed; however, elements of the sleeves' fiberglass rovings were evident and remained attached to the main rotor blades. The six rotor blade attach pins and the six bolts at the sleeve and spherical stop were in place. The three

[20]The Uniball is a ball joint mounted on the main rotor mast, which allows the stationary star to pivot in response to cyclic control inputs. The rotating star is on a bearing mounted on the stationary star, which allows it to copy all of the movements of the stationary star and transmit control inputs via the pitch change links to the main rotor blade pitch horns.

antivibration springs and weights were present. Two of the three springs were retained in their respective housings. The third spring was in place; however, there was thermal distortion to the housing and the cover was open.

All three of the Starflex arm tips were found in the immediate wreckage area. The tips fractured in the plane of rotation from their respective arms. The tips were subsequently matched to their respective arms by comparing the fracture patterns. The fracture of the yellow arm showed a 45^{o} fracture from the outboard leading edge to the inboard trailing edge. The blue arm of the Starflex showed a complex fracture pattern. The first fracture was 1 inch in length and ran from the outboard trailing edge toward the inboard leading edge. The remainder of the fracture was perpendicular to the Starflex arm. A peeled away surface was observed on the upper trailing edge portion of the Starflex arm. The bottom portion of the blue arm was also peeled away. The fracture of the red arm showed a 45^{o} fracture. However, the fracture was opposite of the yellow arm. The fracture ran from the inboard leading edge to the outboard trailing edge of the Starflex arm.

Main Rotor Gearbox

The MGB was made up of three modules: epicyclic, bevel reduction gear, and the oil pump. The freewheeling unit is integrated into the engine power takeoff transmission/drive shaft. All three modules were thermally damaged with foreign debris present. The debris consisted of molten aluminum deposits, magnesium ash, dirt, and rocks. The transmission filter, oil lines, flexible hydraulic lines, and electrical wiring attached to the case of the MGB were thermally destroyed.

The epicyclic was composed of the planetary gears, stationary ring gear, and the sun gear. According to the Eurocopter Illustrated Parts Manual, the required hardware consisted of seven epicyclic to mast junction bolts, one main center bolt, and six secondary bolts; all of the bolts were present and locked in place. The conical housing was thermally damaged and partially destroyed. The planetary gears and the stationary ring gear were thermally damaged. Neither the planetary gears nor the stationary ring gear showed evidence of galling, bending, or missing teeth. An attempt was made to rotate the planetary gears. Due to thermal damage, only one of the planetary gears could be manually rotated. The sun gear was thermally fused to the vertical shaft/bevel reduction gear and could not be removed. The sun gear's spring clip and two spacers were accounted for and in place.

The bevel reduction gear was thermally damaged and did not show evidence of galling, bending, or missing teeth.

The oil pump housing was thermally destroyed. The internal impeller gears displayed thermal damage without evidence of galling or bent or missing teeth.

Hydraulic System Components

The hydraulic pump is belt driven by the engine-to-MGB drive shaft. The belt, which was mounted in an area of almost complete thermal destruction, was not located. The hydraulic pump body was thermally damaged and partially destroyed, preventing a functional check. The regulator unit and hydraulic fluid reservoir were mounted in areas that were exposed to extreme thermal damage and were not found.

The main rotor servos are single-cylinder servo controls with accumulators that provide hydraulic power assistance for transferring pilot control inputs to the nonrotating swashplate. The servos consist of a fore/aft servo, left-hand (L/H) lateral servo, and right-hand (R/H) lateral servo. In addition, another servo is used for tail rotor pitch control.

The fore/aft servo body was thermally damaged and partially destroyed. The hydraulic line fittings were secured to the servo body. The accumulator was separated from the servo body, and the remaining threads were thermally damaged. The servo input rod end was connected to the input valve and was safe-tied.

The L/H lateral servo body was thermally damaged, and the upper portion was thermally destroyed. A portion of the upper servo body was removed during the examination to prevent binding and to allow movement of the piston rod.

The R/H lateral servo body was thermally destroyed.

The tail rotor servo was thermally damaged and partially destroyed. The fittings for the lines remained connected to the servo. The control rod to the tail rotor was separated from the servo. The input rod was connected to the servo input; however, the aft end of the control cable separated from the sheath.

The Structures Group reconvened in Southern California from December 4 to 5, 2001, to conduct additional component examinations on the hydraulic servos. An x-ray examination of the three main hydraulic servos was conducted at General Inspections Laboratories, Inc., Cudahy, California, on December 4th. A teardown examination was conducted on December 5th at Hawker Pacific Aerospace, Sun Valley, California.

The R/H lateral servo body was thermally destroyed and all that remained was the piston body. Neither an x-ray nor a teardown examination could be conducted on the R/H lateral servo.

The fore/aft and L/H lateral servos were x-rayed prior to a teardown to verify that there were no internal mechanical discrepancies and to record the internal position of the piston. All internal components were observed in place, with no internal mechanical anomalies noted. The external examination of the fore/aft and L/H lateral servos revealed that all the safety wire and nuts and bolts were in place. Due to the thermal damage, the valve bodies could not be opened; therefore, a component teardown was not possible.

Tail Rotor System Components

The TGB was disassembled. When turned manually, the input and output shafts rotated freely. There were no galled, bent, or missing teeth evident to either the input pinion gear or the Gleason spiral bevel gear. The oil appeared normal in color and aroma.

Witness marks were noted on the forward edge of the pitch change bell crank. A separate mark was observed on the adjacent forward TGB mount structure. No witness marks were observed on the pedal stops. An exemplar helicopter was examined to determine positional orientation of the observed marks in comparison to cockpit antitorque pedal input. The comparison revealed that alignment of the marks was only possible with control positions beyond full right anti-torque pedal input.

No visual defects or binding of the assembly was noted with the pitch change spider.

The fracture face of the tail rotor yellow pitch change link spherical bearing end exhibited brittle fracture characteristics. The spherical bearing end of the red pitch change link was bent.

Witness marks were observed on the inboard side of the blade control horn and corresponding marks were observed on the tail rotor hub. An exemplar helicopter was examined to determine what blade pitch would result in positional alignment of the observed marks. The comparison found that the marks would require full positive pitch (full right pedal) and a flapping motion of the tail rotor blade.

Additional witness marks were observed on the inboard laminated tail rotor bearings and corresponded with marks on the tail rotor hub. The inboard balancing weights on the lower outboard edge contained witness marks, which corresponded with witness marks on the rotating portion of the pitch change spider. Alignment of these marks required full positive pitch of the blades.

Cockpit Caution/Warning Panel Light Bulb Filament Analysis

The light bulb filaments in the cockpit caution/warning panel were intact. The panel was disassembled for a visual examination under a microscope and a 10x magnifying glass; no filament stretch was observed. The panel was sent to the Safety Board's Materials Laboratory in Washington, D.C., for further examination and documentation. With the exception of two light bulbs[21] for the pitot indicator, no filament stretch was observed. The filament for one of the pitot bulbs exhibited minor stretch when compared to the other bulbs in the panel. The second pitot bulb exhibited minor stretch in the area where the filament was supported by the post; however, no stretch was observed in the remainder of the filament.

MEDICAL AND PATHOLOGICAL INFORMATION

The pilot sustained fatal injuries in the accident and an autopsy was conducted by the Mojave County Medical Examiner's Office. Tissue and fluid samples from the pilot were submitted to the FAA's Civil Aerospace Medical Institute in Oklahoma City for toxicological analysis. The results were negative for carbon monoxide, cyanide, and all screened drug substances. No ethanol was found in the vitreous fluid.

[21]Each caution/warning light position within the caution/annunciator panel has two light bulbs for redundancy purposes.

ORGANIZATIONAL AND MANAGEMENT INFORMATION

Tour Operator Procedures

All flight tours of the Grand Canyon transited the Grand Canyon National Park Special Flight Rules Area (GCNP SFRA) Airspace. The Las Vegas FSDO issued Order 1380.2A, "GCNP SFRA Procedures Manual," to all operators approved to operate within this airspace. Order 1380.2A contained, in part, the following information:

- reporting points, radio frequencies, and procedures;
- weather minimums;
- aircraft exterior lighting requirements;
- qualifications and training requirements;
- routes, altitudes, and procedures;
- air tour route descriptions for the 20 authorized routes;
- flight plans; and
- commercial air tour limitations.

Operational Ground Procedures

The on-duty dispatcher was responsible for ordering fuel, filing flight plans, coordinating with the ticket counter, tracking each flight, and maintaining radio communications with company aircraft.

Operational In-flight Procedures

Following departure from LAS, each Papillon flight toured Lake Mead and the Hoover Dam prior to entering the GCNP SFRA. When Papillon operated within the GCNP SFRA, all of its operations that departed from LAS were conducted on Green Route 4. The majority of that route traversed areas of mountainous and rugged terrain.

About 45 minutes after takeoff, each flight landed for 30 minutes at either Quartermaster Canyon or Lower Ramada. These were Papillon's designated landing sites located on the floor of the canyon. A picnic lunch was provided, and the passengers were given the opportunity to take photographs.

If additional fuel was required prior to returning to LAS, the flights would stop at GCW where Papillon operated its own fueling pad. It was at the pilot's discretion whether he would obtain fuel prior to or after landing at one of the designated landing sites. Passengers disembarked the helicopter during fueling; however, the engine continued to operate at idle power.

Company History

The company was founded in the spring of 1965 by the present owner and originally certificated as Grand Canyon Helicopters. The firm operated from the South Rim of the Grand Canyon. When the owner purchased Hawaiian-based Papillon Helicopters, he incorporated both operations under the name Papillon Grand Canyon Helicopters. In 1993, the company owner founded the Tour Operators Program for Safety to promote safety and bolster the public image of Grand Canyon helicopter tour operators. The company's name was changed to Papillon Airways, Inc., in October 1995.

The company was primarily a tour operator but also held contracts with the Arizona Fish and Game Agency, the Forestry Service, and several utility companies. The company had operations at LAS and at Grand Canyon Airport with the latter being the largest. The company also operated a fueling facility at GCW. At the time of the accident, Papillon operated a total of 36 helicopters with a staff of 55 line pilots. Company helicopters flew over 250,000 passengers each year with a seasonal peak of about 250 departures per day.

FAA Oversight

An interview was conducted by Safety Board investigators with the FAA principal operations inspector (POI) assigned to the Papillon certificate. He had been Papillon's POI since September 1, 1993, and he worked at the FAA's FSDO in Las Vegas, Nevada. He had about 6,500 rotorcraft hours and had recently retired from the Army National Guard.

The POI stated that he had oversight responsibilities for all helicopter operations in LAS and the Grand Canyon. Those certificates operated 85 helicopters and 34 fixed-wing airplanes. He was assigned an assistant in July 1996, and he stated that he felt that the two of them were able to provide adequate oversight to their assigned operators.

Records provided by the FAA indicated that the POI and his assistant performed the following surveillance inspections at Papillon from March 1998 to August 2001:

- three operations manuals inspections,
- twenty ramp inspections,
- six training program inspections,
- two crewmember/dispatcher training records inspections,
- three facility inspections,
- two flight following inspections,
- one de-icing procedures inspection,
- ten check airman oral examination inspections,
- nine check airman aircraft inspections, and
- nine check airman line inspections.

Records provided by the FAA indicated that the POI and his assistant performed the following technical/administrative activities at Papillon from November 2000 to August 2001:

- two operations specifications revisions,
- twelve airman oral examinations,
- eleven airman aircraft evaluations,
- fifteen airman line evaluations,
- one minimum equipment list revision,
- one incident investigation,

- one administrative enforcement action, and
- one enforcement investigation closed out with no action.

FAA Surveillance Policy of Flights Conducted Outside of the SFAR

The FAA provides national surveillance guidelines for aviation safety inspectors in the September 14, 2001, order number 1800.56B,[22] entitled “National Flight Standards Work Program Guidelines” (NPG). This order delineates the policy, including inspection intervals and activities, for the execution of annual surveillance work programs to be followed by flight standards inspectors. The order was distributed to the Las Vegas, Nevada, Flight Standards District Office.

The FAA POI reported that his surveillance of Papillon was performed in compliance with the NPG. The POI indicated that, according to the NPG, there is no requirement for him to perform direct (in-flight) surveillance of Papillon’s tour routes outside the airspace designed by the SFAR.[23] Because the NPG did not mandate this type of direct surveillance, none was performed under either the required or planned work activity program during the preceding year.

ADDITIONAL INFORMATION

Helicopter Performance

Company Procedures for Determining Performance Limits

According to the chief pilot, the company used limiting takeoff temperatures and temperature ranges in the form of tables for each of the landing sites used by the operator. The tabular data were based on performance charts found in the FAA-approved Rotorcraft Flight Manual (RFM). These tables depicted information that pertained to situations in which the helicopter hovered in-ground effect and also hovered out-of-ground effect (HOGE). There was also a cockpit reference chart that provided pilots with the maximum gross weight for takeoff based on the actual reported temperatures at LAS and GCW.

Performance Capability - Fight from Grand Canyon West to Accident Site

With the assistance of Eurocopter engineering and flight test personnel, a performance capability profile was determined for the helicopter from the time it departed the GCW until the accident. Portions of this performance profile were generated by computer modeling using Eurocopter Performance Prediction Software (Sycomore Version 1.45). The balance of the performance capability profile was generated by using the performance charts contained in the RFM applicable to the accident helicopter’s serial number.

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[22] The National Flight Standards Work Program Guidelines, order number 1800.56B, was superceded by order number 1800.56C, on September 13, 2002. None of the changes made in order 1800.56C addressed 14 CFR Part 135 en route surveillance procedures or special emphasis areas for tour flights.

[23] The accident occurred during the return portion of the tour flight, outside the Grand Canyon National Park SFAR designated airspace.

Certain factual data concerning the atmospheric conditions, terrain clearance altitudes, and the helicopter's gross weight during the accident sequence were used in the development of the performance capability profile.[24] The factual data used in the profile development included the following:

- The aircraft gross weight at the time of the accident, which was estimated to be about 4,515 pounds (2,047 kg), with the center of gravity within the longitudinal and lateral limits for this weight.
- The minimum practical msl altitude that the helicopter could have crossed the western edge of the Grand Wash Cliff feature as it flew westbound, which was 5,300 feet.
- The elevation that the aircraft was at when it collided with the ground with an eastbound energy path, which was 4,080 feet.
- The ambient outside air temperature at 5,300 feet, which was estimated to be 34^{o} C and 36^{o} C at the impact altitude.[25]
- Based on the above values, the engine power available was limited by the engine (Ng)[26] limitations, not by transmission or torque limits.
- Use of the antitorque pedals requires some of the engine power output, with full right pedal application pulling 90 kilowatts (kw) of power out of the drive train. At sea level, takeoff power is equal to 530.9 kw, and maximum continuous is 466 kw.
- The minimum power required airspeed (V_Y), which was 55 knots.

The HOGE performance capability as the helicopter crossed the ridgeline westbound was determined from the AS350-B2 flight manual. Using takeoff power, the hover ceiling was determined to be a helicopter weight of 2,072 kg. At the impact elevation, the hover ceiling was determined to be a weight of 2,148 kg. According to Eurocopter's analysis, the helicopter would have had only a marginal capability to slow to zero airspeed at the ridgeline elevation.

Safety Board investigators asked Eurocopter engineers to determine the vertical climb capability of the helicopter under the conditions of a near-zero, forward-speed vertical descent. Since the performance charts in the RFM do not address vertical climb capability, the flight test department computer software referenced previously was employed to predict this capability. A calibration of the software data output was performed by inputting data to predict the hover ceiling performance discussed previously; the values returned by the program were within a few kilograms of the values obtained from the flight manual charts. At the accident site elevation conditions, the software predicted a theoretical positive climb capability of 421 feet per minute (fpm).[27] At the ridgeline elevation conditions, the software predicted climb capability of 41 fpm. The engineers noted that this vertical climb rate capability assumes starting at a stabilized hover, and if the helicopter was in a vertical descent of greater than 421 fpm, insufficient power was available to arrest the rate of descent.

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[24]The detailed factual basis for these data can be found in the pertinent sections of this narrative dealing with those specific areas.

[25]Company pilots who landed at the accident site within minutes of the crash reported outside ambient air temperatures on the ground of 41^{o} C.

[26]Ng is the gas generator section of the engine. Power is "Ng limited" when the atmospheric temperature and pressure conditions preclude the engine developing full-rated power.

[27]This assumes an outside air temperature of 36^{o} C. If the calculations are performed using the company pilot reports of 41^{o} C at the accident site, the vertical climb capability reduces to 43 fpm.

According to the Eurocopter engineering analysis, for the conditions approximating those between 5,300 feet and the impact elevation, any airspeed decrease below V_Y would require an increase in power; as the speed decreases toward zero, the power required to maintain altitude is just under or equal to the power required to HOGE or available from the engine. Table 2 illustrates the various weight limits and power margins for the stated atmospheric conditions.

Table 2. Various weight limits and power margins for stated atmospheric conditions.

Condition	HOGE Weight Limits	Power Margin	Vertical Climb Ability
5,300 feet at 34^o C	2,072 kg	2.1 kw	+ 41 fpm
4,041 feet at 36^o C	2,148 kg	21.5 kw	+421 fpm
4,041 feet at 41^o C	2,073 kg	2.2 kw	+43 fpm

The engineering analysis also noted that any tailwind component would significantly decrease the hover and vertical climb performance capabilities of the helicopter.

Settling with Power

Various helicopter aerodynamic texts and research articles[28] address the phenomenon of settling with power. Also known as vortex ring state, the settling with power phenomenon is characterized by an unstable, chaotic, and disorganized rotational airflow around and through the main rotor, which results in a net decrease in the thrust produced by the main rotor. This rotational airflow can be visualized as a donut-shaped area of air swirling up and around the blade tips that induces a downward air through the rotor disc and a resultant decrease in lift. In this condition, the use of collective pitch by the pilot will not slow the rate of descent, but it will aggravate the descent rate by increasing the rotational airflow around and through the rotor disc. Descent rates in fully developed vortex-ring state conditions can reach values from 4,000 to 6,000 fpm. Depending on disc loading, various helicopters can enter the state under some or all of the following conditions:

- powered vertical rates of descent from 300 to 700 fpm;
- flightpath angles of around 70^o (although in wind tunnel studies significant main rotor thrust instability has been found under certain conditions at flightpath angles between 30^o and 70^o);
- airspeeds at, near, or below the translational lift boundary speed (although wind tunnel work suggests the phenomenon can occur at speeds as high as 60 knots in some designs);
- high-density altitudes near or beyond the HOGE ceiling; and
- autorotational or quick stop flare maneuvers while flying downwind.

The typical actions required to escape this condition include reducing the collective pitch and entering an autorotation until the descent rate is at or above 1.25 times the hovering induced

[28]See: R.W. Prouty, Helicopter Aerodynamics, revised edition dated August 21, 2000; U.S. Navy, Aerodynamics for Naval Aviators, NAVWEPS 00-80T-80; U.S. Department of Transportation, Basic Helicopter Handbook, revised 1978; U.S. Air Force, Guide to Mishap Investigation, AFP 127-1; U.S. Army, Fundamentals of Flight, FM1-203; Laboratoire ONERA-Ecole de L'air, Induced Velocity Model in Steep Descent and Vortex Ring State Prediction.

downward airflow through the rotor disc (typically 1,500 to 3,000 fpm). Increasing the forward airspeed will also moderate or eliminate the condition.

A related condition is known as power settling, which occurs when the power required for any hover or arresting a rate-of-descent condition exceeds the power available in the drive train to the main rotor.

Retreating Blade Stall

Eurocopter France was asked by Safety Board investigators to research certification flight test reports to determine if a retreating blade stall was ever encountered during flight test work. According to the company, the helicopter was flown at maximum weight to V_D (design maneuvering airspeed), which is 1.11 times V_{NE} (never exceed airspeed), without encountering a retreating blade stall. For the AS350-B2 at sea level and 2,250 kgs gross weight, V_{NE} is 155 knots, with a 3-knot reduction for every 1,000 feet above sea level.

Hydraulic Control System Servo Transparency Phenomenon

In its submission to the Safety Board, Papillon discussed possible scenarios and/or causes to this accident. One scenario that Papillon presented involved a phenomenon known as cyclic transparency or jack stall.

Discussions and correspondence were conducted between Safety Board investigators and Eurocopter engineers and flight test pilots concerning the hydraulic control system transparency phenomenon. While typical hydraulic systems in other helicopters and fixed-wing aircraft operate at pressure ranges from 1,000 to 3,000 psi, the hydraulic system in the AS350 series helicopters operates at 600 psi. According to Eurocopter representatives, this system pressure was chosen during design as a balance between the loads generated by the rotor system that must be overcome and acceptable load limits on the rotor system dynamic components (that is, control rods, links, and fittings).

At the request of Safety Board investigators, Eurocopter France provided a copy of AS350 certification flight test report No. H/EV 17.530. The report documents a series of test flights conducted in 1985 to explore the points where the hydraulic transparency effect (also called control reversibility) occurs. From the data collected, a series of graphic plots were developed and included in the flight test report, which predict the G loads for phenomenon onset for a given weight, density altitude, and helicopter speed.

Eurocopter representatives stated that the transparency phenomenon is nonviolent and transitory, lasting only 2 to 3 seconds at most due to the “self-correcting actions of the pilots” to reduce the G loads and/or the natural static and dynamic stability “response of the helicopter.” They also stated that the controls are fully operable throughout the entire transparency event; however, the force required to effect movement of the flight controls against the rotor system dynamic feed back loads would increase significantly. Eurocopter stated that the force feedback for each control channel would be dependent in part on the amount of G loads experienced; however, the company estimated that about 22 pounds of force would be required to move the collective in the UP or increased pitch direction, with the same amount to move the cyclic to the left. Additionally, flight tests data indicate that servo transparency could not be encountered if the collective is less than 50% raised.

Safety Board investigators reviewed Section 3.2 of the RFM for the AS350, which covers hydraulic system failures. This section lists the approximate force increase needed to move the flight controls following a hydraulic system failure. The system has an accumulator, which would provide for about 45 seconds of normal system operation following a pump failure before the system would revert to manual reversion.[29] As examples, the force required for movement of the collective would go from minimal (boosted condition) to 40 pounds, while the force for cyclic movement would go from minimal to about 20 to 25 pounds, depending on the direction of control input.

An additional review of the RFM that was in effect at the time of the accident was performed to determine what information was presented concerning the hydraulic transparency phenomenon. The review found that the phenomenon was only described in paragraph 7.2 of section 4.1. Paragraph 7.2, "Maneuvers," states the following: "Maximum load factor in turns is felt in the form of servo-control 'transparency'; this phenomenon is smooth and presents no danger." There were no informational charts in this section to illustrate the phenomenon onset as a predictable relationship between gross weight, speed, density altitude, and maneuvering-induced load factors. One additional related note was found in section 2 of the RFM, Limitations, Paragraph 7.3, "Maneuvering Limitations," which states, "Do not exceed the load factor corresponding to the servo control reversibility limit."

At the request of Safety Board investigators, Eurocopter provided the predicted onset G loads for hydraulic transparency for the accident helicopter at the weight and atmospheric conditions existing at the time of the accident. At 100 knots, the transparency phenomenon would have occurred at +1.7 Gs, and at 120 knots it would have occurred at +1.5 Gs.[30]

On December 19, 2003, the FAA issued Special Airworthiness Information Bulletin (SAIB) No. SW-04-35 which "alerts owners and operators of Eurocopter France AS350B, BA, B1, B2, B3, D, AS355E, and EC120B model helicopters that the pilot can encounter a phenomenon known as *Servo Transparency, Servo Reversibility, or Jack Stall.*" The bulletin stated the following:

> Pilot and operators may misunderstand this phenomenon. This aircraft phenomenon occurs smoothly, and can be managed properly if the pilot anticipates it during an abrupt or high load maneuver such as a high positive g-turn or pull-up. The factors that affect Servo Transparency are high airspeed, high collective pitch, high gross weight, high G-loads, and high density altitude.
>
> The maximum force that the servo actuators can produce is constant and is a function of hydraulic pressure and of the servo characteristics. The system is designed to exceed the

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[29]The time the accumulator can provide hydraulic system pressure is dependent on the amount of nitrogen pressure precharge in the accumulator and on how much control movement is made by the pilot after the pump failure or system deactivation.

[30]The basis for this number can be found in Eurocopter's Engineering Certification Flight Test Report No. H/EV 17.530.

requirements of the flight limitations in the approved flight manual. With excessive maneuvering and under a combination of the above listed factors, the aerodynamic forces can increase beyond the opposing hydraulic servo forces and Servo Transparency can occur. An improperly serviced/maintained hydraulic system can also effect the onset of Servo Transparency.

Servo Transparency begins when the aerodynamic forces exceed the hydraulic forces and is then transmitted back to the pilot's cyclic and collective controls. On clockwise turning main rotor systems, the right servo receives the highest load when maneuvering, so Servo Transparency results in uncommanded right and aft cyclic motion accompanied by down collective movement. The pilot control force to counter this aerodynamically-induced phenomena are relatively high and *could give* an unaware pilot the impression that the controls are jammed. If the pilot does not reduce the maneuver, the aircraft will roll right and pitch-up.

The amplitude of the induced control feedback loads is proportional to the severity of the maneuver, but the phenomenon normally lasts less than 2 seconds.

The SAIB presented the following nonmandatory recommendations to operators:

- Review Chapter 14, Aeronautical Decision Making, Rotorcraft Flying Handbook FAA-H-8083-21.
- Properly service the hydraulic system before each flight.
- The pilot should follow (*not fight*) the control movement. Allow the collective pitch to decrease (monitoring Rotor RPM, especially at very low collective pitch settings) to reduce the overall load. You should be aware that as the load is reduced, hydraulic assistance will be restored and force being applied to the controls could result in undesired opposite control movement. Follow the aircraft limitations in accordance with the Aircraft Flight Manual.
- Understand that Servo Transparency is a natural phenomenon for any flyable helicopter. BASIC AIRMANSHIP should prevent encountering this phenomenon by avoiding combinations of high speed, high gross weight, high-density altitude, and aggressive maneuvers, which exceed the aircraft's approved flight limitations.

Pilot Tour Narration and Gratuities

Papillon does not employ tour guides who accompany the flights. Part of the pilots' additional responsibilities includes providing tour narration and commentary regarding notable landmarks and other features of the Grand Canyon environment. Each passenger is provided a headset prior to departure, and the pilot narrates the tour as he points to all the features along the route of flight. For non-English speaking passengers, the company provides a prerecorded audiotape narration of the tour utilizing a cassette system installed in the helicopter.

The investigation noted that several tour operators in Hawaii and Alaska were found to have adopted a system whereby the principal tour narration is given by a prerecorded tape.

Interviews with company pilots disclosed that although the operator prohibited the pilots from soliciting gratuities from passengers, the gratuity amount voluntarily given the pilots by passengers was usually proportional to the quality of the entertainment. The investigation determined that gratuities provided significant additional income for the pilots.

Emergency Contingencies for the Routes Flown

During the investigation, Safety Board investigators had several opportunities to fly in Papillon company helicopters over the routes flown for the passenger tour flights. Over much of the SFAR-mandated route structure, the nature of the terrain is composed of precipitous and rugged canyon topography with few locations that would be suitable for emergency autorotations. In the latter stages of the route approaching the Quartermaster Canyon and Lower Ramada landing sites, there are prolonged periods where the helicopters operate over the Colorado River in areas of the canyon where the near sheer vertical walls of the canyon rise from the river edge, and the only possible emergency landing site is in the water.[31]

Accident Pilot's Prior Tour Flights

Several passengers who had flown on a tour flight with the pilot about 1 month prior to the accident provided information on their experiences. One passenger provided a copy of a videotape taken during the tour.

According to the passengers, once the tour started, the pilot was talking all the time. He was very informative, and they felt he knew his history and geography very well. They went over the Hoover Dam and Lake Mead. About 20 minutes into the flight, the pilot turned his head toward the back and was talking to the passengers as the helicopter flew toward a cliff. The people in the back were trying to get the pilot's attention and point out that he was flying toward a cliff, but he pretended he did not understand what they were saying, as if this was all being done on purpose. All this time, the pilot was turned around and talking to the passengers in the back seat, while the passengers were all pointing up trying to get him to climb. One witness said she finally picked up the microphone and said, "they are really scared….turn around and pull up the helicopter," and he did. She could not estimate how far they were from the cliff when the pilot terminated the maneuver.

One of the passengers stated that there were particularly exciting episodes during the tour that were frightening to some of the others. As part of the tour, they flew over a site that was used in the commercial motion picture film *Thelma and Louise,*[32] and the pilot pointed out the cliff. The pilot stopped for fuel before he landed in the canyon for the picnic lunch. After lunch, no more stops were made. During the return to LAS, the pilot asked if they wanted to know what it was like to drive a car off of a cliff. She stated that they all said "no" to this question; however, he proceeded to fly very fast toward the edge of the cliff and then dove the helicopter as it passed the edge. The passenger reported that it was "frightening and thrilling at the same time but it scared the others to death."

[31] According to Papillon, no flotation devices are carried on the helicopters.
[32] The scene referenced shows a car driving off of a high cliff and diving toward the desert floor.

Safety Board investigators reviewed the videotape supplied by the passenger. The two episodes referenced by the passenger were on the tape. No voices could be discerned due to the engine and rotor noise. In the first event, the pilot was observed turning his head toward the rear seat passengers as the helicopter approached a clifflike terrain feature about 50 to 100 feet below the top. The passengers were observed to be gesturing up and then the pilot turned around and initiated a climb. The helicopter appeared to clear the top by 50 to 100 feet. Regarding the *Thelma and Louise* event, after the lunch stop, the video showed a view of the helicopter flying an estimated 100 feet over Grand Wash Cliff plateau. Just past the cliff edge, the pilot initiated a diving descent. The amount of nose pitch down was estimated to be in the range of 10^{o} or less. Observations of the sound change in the engine and rotor noise were consistent with a lowering of the collective and unloading of the rotor system during this maneuver.

ANALYSIS

General

The pilot was properly trained and certificated in accordance with Federal regulations and company requirements. The pilot's rest and duty times were within the requirements of Federal regulations.

Interviews with company pilots and supervisors revealed that they considered the accident pilot one of their "very best." He was consistently praised for his knowledge of the helicopter and its systems. They stated that even the mechanics and other maintenance personnel within the company praised his knowledge, skills, and abilities. There was no evidence that suggested his emotional state materially influenced his flying abilities or job performance.

Although the investigation was concerned about the large number of operators and aircraft assigned to the FAA POI, the records review and the results of this investigation did not reflect an inadequate level of surveillance by the POI and his assistant of Papillon Airways. However, the investigation revealed that FAA's local and national policy did not contain adequate guidance or requirements for direct surveillance of thousands of air tour flights that occur outside of the SFAR.

Examination of the helicopter maintenance records and work orders from New Zealand and interviews with the Papillon maintenance technicians who worked extensively on the helicopter after its arrival in the United States revealed that repair work on the helicopter was of very high quality and not related to the accident.

Analysis of the atmospheric conditions disclosed no evidence of any meteorological phenomenon that was contributory to the accident other than the high-ambient temperatures and resultant high-density altitude. The winds at both the top of the cliff and at the accident site were about 5 to 10 knots from the west.

Accident Flightpath and Accident Site Location

Due to noise and traffic restrictions, Grand Canyon tour aircraft are required to follow preapproved routes. The accident helicopter was following a route that crosses east to west over a plateau formed by the Grand Wash Cliffs, and once past the western face of the cliff, descends 1,400 feet[33] and continues west across level desert terrain. Passing over the cliff edge along this route, the terrain drops off dramatically to barren desert floor 3,000 feet below. According to Papillon representatives, helicopter pilots are trained to identify potential landing sites along a route, and the investigation determined that Papillon pilots flying this route would most likely attempt to perform an emergency autorotation[34] landing on the flat terrain to the west of the cliff as opposed to attempting any landing in the hazardous terrain in proximity to the cliff.

[33] This is a required elevation change, to avoid opposite-direction traffic and to comply with the hemispheric flight rule provisions of 14 CFR Part 91.

[34] Autorotation is a stable flight condition in which the rotor system is turned by airflow, not engine power, and altitude and forward airspeed are exchanged for increases in rotor rpm. It is essentially a tradeoff of rotor kinetic energy for lifting force (reduced rate of descent) and forward airspeed.

The total time between the last radar return associated with the accident helicopter being on course at cruise altitude and the first observation of heavy smoke above the crash site was about 3 minutes.[35] The crash site evidence suggests that during the final minutes of flight, the accident helicopter passed over the cliff edge, and rather than continuing straight ahead along the route, the helicopter reduced forward airspeed and reversed its heading. This maneuver would have placed the helicopter below the cliff edge, facing into the escarpment.

Although the plateau's western edge has elevations as high as 6,000 feet msl, the area where the helicopter is believed to have crossed it is about 5,300 feet msl. Based on the limited available radar data, the helicopter flew across the plateau at about 100 feet agl (5,400 feet msl). After crossing the plateau and still on a westerly heading, a normal descent to 4,500 feet msl was required. This descent provided the tour pilots an opportunity to demonstrate the helicopter's rapid descent capabilities. Several of the operator's tour pilots, including the accident pilot, nicknamed this maneuver the *Thelma and Louise* descent. Interviews with passengers who had flown with the accident pilot before the accident and other people familiar with the way he flew his tour indicated that he preferred to cross the cliff edge at low altitude and high speed. They stated that he would then enter a rapid descent by lowering the collective and nosing the helicopter over into a dive. The evidence of a compact crash site located at the base of the cliffs is consistent with a maneuver of this type in which the descent was not arrested.

The accident helicopter came to rest on an easterly heading (180^{o} from the original direction of flight) on a rocky 40^{o} slope, at an elevation of 4,041 feet msl, about 1,400 feet below the point where it crossed the cliff. This area was unsuitable for a helicopter landing, either as an intentional elective landing or due to any conceivable emergency situation. The terrain directly ahead of the pilot, as he crossed the Grand Wash Cliffs, featured many suitable areas to initiate a forced landing. The altitude difference between crossing the plateau and these suitable landing areas was about 3,000 feet msl and could have significantly extended the helicopter's power off glide distance and time.

Analysis of the accident site, terrain features, and options available to the pilot indicated that the helicopter came to rest in the least advantageous location available. Postaccident interviews with other company pilots determined that they would choose to continue straight ahead toward the level terrain[36] as opposed to attempting any sort of landing at the accident site. None of the pilots interviewed could conceive of any emergency where they would attempt an emergency landing at the accident site, given the flat terrain directly ahead on the original heading. They also pointed out that an alert and proficient pilot would never intentionally turn the helicopter toward any type of sloping terrain, especially one that would expose the helicopter to a 5- to 10-knot tailwind.[37]

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[35] A company helicopter was flying 3 minutes behind Papillon 34 and had it in visual contact about a minute before it crossed the Grand Wash Cliffs.

[36] In addition to company pilots, many others were asked about the site as a potential emergency landing place and, given the nature of the terrain ahead on the flight path, none could conceive of any emergency that would induce a pilot to attempt a normal or emergency landing at the crash site.

[37] According to the Safety Board's meteorological study of the atmospheric conditions, the winds from 3,000 to 6,000 feet were from the west at 5 to 10 knots.

Analysis of the crash site evidence indicates that there was insignificant forward velocity and a very high, vertical descent rate at impact. The helicopter's skid-type landing gear was found sitting upright and facing into the cliff face at the IPI on a large outcropping of boulders.[38] A burn pattern and limited debris was found at this location. There was a second burn pattern about 20 feet downslope where the helicopter had come to rest after rolling over and where much of the aircraft had been consumed by fire. The two burn patterns and the wreckage distribution indicates that the fuel cell was compromised at initial impact. The rearward rotation of the skid cross tubes and the outward collapse of the skids indicate that the aircraft was in an upright and slightly flared[39] attitude at impact and that high-compressive forces (high-vertical rate of descent) were experienced. The undersides of the skids showed only minor longitudinal scrape marks, and the wreckage and terrain scars were located within the immediate vicinity of the IPI. While the degree of airframe thermal destruction precluded a more definitive estimate of vertical loading during the impact, the G loads were estimated to range between 10 and 16. No lateral scrape marks were found on the skids or the ground, indicating that the helicopter was not turning or spinning at ground contact.

Engine Power at Impact

The investigation revealed that the engine and rotor system were developing a significant amount of power at the time of impact. The collective pitch control was in the full UP (maximum pitch) position[40] and the antitorque pedals were positioned full right.[41] These control positions are consistent with a high-power demand from the pilot and a high-power output from the engine to the rotor system. Both pedals were bent forward at the tips, which is consistent with pilot contact on the controls.

Terrain scars identified near the IPI were consistent with the damage observed to the leading edges and tips of the main rotor blades. The helicopter Starflex rotor head and drive train flexible couplings exhibited the torsional damage signatures typically associated with sudden rotor stoppage under considerable driving power.[42] The severity of the damage indicates

[38] This orientation represents a 180° deviation from the route and from the helicopter's last known heading.

[39] A flare, or nose up helicopter attitude is achieved by moving the cyclic control aft. This input decelerates the aircraft and increases lift/rotor energy (the rate of descent and the forward airspeed are reduced; rotor rpm is increased). A flare maneuver is typically performed to cushion a touchdown. In this case, the evidence suggests the helicopter's pitch attitude at impact was about 5° less than the slope of the hillside or slightly nose down.

[40] This position equates to the pilot commanding the maximum lift or climb capability from the main rotor system. Determination of this control position was made with a high degree of confidence based on the collective demand pointer on the fuel control unit; this is cable driven and not likely to have been influenced by vertical impact forces on the collective torque and push/pull tubes.

[41] In European-manufactured helicopters, a full-right antitorque pedal position results in the maximum tail rotor blade angle, which is the highest setting available to counteract engine torque effect. Determination of this control position was made based on the witness marks found on the blade control horns and tail rotor hub. These marks could only have been made with the tail rotor rotating and the blades just beyond maximum positive pitch (right pedal). Although the impact-related down bending separation of the tailboom would tend to move the blades to a right pedal position, the investigation believes that the pilot had right pedal input, and the tailboom downward bending moved the tail rotor blade pitch past the limit stop.

[42] Sudden rotor stoppage is the sudden arrest of a helicopter drive system due to rotor blade contact with a solid object while being driven by the engine. Typical damage signatures include rotor blade leading edge impact damage, rotor head torsional damage consistent with the direction of rotation and with the sequence of blade strikes, torsional shearing of engine-to-transmission couplings, starflex arm fracture patterns, and rotor blade stack up. Such signatures are not seen if the engine has failed: a clutch mechanism disengages a failed engine.

that power was being transferred from the engine and that the main rotor was rotating at a high rotor rpm when contact occurred. The internal condition of the engine provides additional evidence that there was significant engine power development at impact.

Teardown examination found indications of normal engine operation at impact. The axial compressor wheel blade tips were bent in the direction opposite to rotation, a classic rotational signature. Frictional heat damage had resulted in blue discoloration 360° around the inside diameter of the compressor cover. There was corresponding blue discoloration on all centrifugal rotor vane tips. These signs of rotational contact resulted from abnormal contact of stationary structures with the compressor rotor during engine operation, when vertical deceleration forces were experienced at impact. High rotational speed is required for heat buildup to occur from only momentary contact.

Ingestion of foreign material often occurs when a turbine engine continues to operate during an accident sequence. In this case, there is evidence that impact forces compromised the helicopter's inlet protection and that foreign aluminum-based material was ingested shortly before engine shutdown. Small nicks and tears characteristic of foreign-object damage[43] were found on the leading edges and tips of the axial compressor blades. Metallic deposits found spattered on both gas generator and power turbine airfoils were confirmed as aluminum. Because the Turbomeca Arriel 1D1 engine does not have any internal aluminum-based parts or coatings, the presence of aluminum in the turbine confirms the ingestion of foreign material. The metal deposits also indicate that there was ingestion shortly before engine shutdown because such deposits will burn away with continued operation.[44] The smooth appearance and fused state of the deposits confirm that engine combustion was taking place either during or immediately prior to their contact, which strongly suggests that the engine continued to operate while airframe structural breakup was taking place. Aluminum airframe structures located just forward of the engine inlet are likely sources of foreign-ingested material.

Rotational contact had also occurred between the power turbine output shaft and the engine front support casing. Circumferential scoring was present 360° around the casing inner diameter at the location axially coincident with the aft engine freewheel housing flange. This flange had corresponding rub damage along all of its flange tips. The contact had caused a roughly 45° arc of raised and heat-distorted metal to appear on the casing outer diameter. The asymmetry of this contact indicates a misalignment and suggests that the rub occurred during the drive train separation in response to high vertical loadings on the structure.

Other than the compressor impact-related damage, all engine airfoil parts were in good condition, with no obvious sand erosion or hot section over-temperature damage found. The possibility of a wear-induced stall event having occurred is ruled out by the condition of these

[43] Hard-body impact damage is characterized by a serrated appearance and cuts or tears to the airfoil's leading and trailing edges. Severity will vary depending on impact velocity, secondary damage, and relative hardness of the objects and the components they strike. Common foreign objects causing hard-body impact damage include metal parts, concrete, asphalt, and rocks.

[44] If particles of aluminum or other metal of similar melting point pass through the engine when it is operating at a stable speed of idle or above, the particles will be wholly or partially molten when they exit the combustor and will tend to adhere to downstream parts. If the surface of a part is above the melting point of aluminum (1200° F) the aluminum will fuse, and have an even, smooth appearance. Molten aluminum contacting cooler surfaces will have a rough appearance, and will sometimes be present in globular form. This type of deposit will be easily scraped away with a fingernail or knife (temporary adhesion). Both deposit forms will burn away quickly with continued operation.

components. No engine anomalies were found other than the ground fire-produced thermal damage. The gas generator drive train seizure was caused by physical interference to gas generator accessory gear train rotation by the thermally distorted gearbox casing.[45] The power turbine drive train seizure resulted from a temporary lockup of the reduction gear train due to thermal degradation of the gearbox oil.[46] The absence of power turbine overspeed-related damage suggests an engine fuel supply interruption occurred almost coincident with the drive train arrest.

Hydraulic and Flight Control System Malfunctions

Examination of the wreckage and analysis of the accident site location did not reveal any evidence of a preimpact hydraulic failure or anomaly. The caution light warning panel was removed from the wreckage for further examination. An analysis of the filaments for the light bulb corresponding to the hydraulic system failure light revealed that the filaments were not stretched, indicating that they were not illuminated at impact. Careful examination of all the light bulbs for systems other than the pitot heat revealed that no filaments exhibited stretch, further indicating that no system failures had occurred. Both of the pitot heat bulbs exhibited minor stretch, indicating that the advisory light for the pitot heat system was illuminated at the time of ground impact.

The investigation revealed that none of the in-flight emergencies relating to the failure of the hydraulic system would have required the pilot to immediately execute an emergency landing in sloping hazardous terrain with a prevailing tailwind. Most anomalies and emergencies of the hydraulic control system result in stiff controls requiring a high degree of effort by the pilot to effect movement. Most hydraulics-off emergencies are terminated by the pilot executing a running landing[47] into a suitable flat area with the actual touchdown being performed at or above translational lift speed[48] (normally 15 to 20 knots). Several suitable areas, including an airstrip, were available to the pilot for such a landing within 8 nautical miles (about 5 minutes flying time) of the accident site.

As previously mentioned, the investigation revealed that the engine was operating at considerable power at impact Other witness marks and evidentiary findings indicate that the pilot's antitorque pedals were in the full-right position (consistent with counteracting the full-torque effects of a running engine, at low-forward airspeed, and with the rotor blades at maximum climb or lift pitch), and the collective lever was in the full-up position (pilot commanding the rotor blades into the maximum lift configuration). Ground scar evidence at the initial impact point clearly establishes that the helicopter was not rotating or spinning and, thus, was under positive pilot directional control. These indications of positive directional control also establish that the tail rotor and yaw control system was fully functional.

[45] The location of the heat-damaged parts was consistent with the areas of greatest ground fire damage.

[46] The gearbox rotated freely following application of penetrating oil.

[47] A running landing, also known as a run-on landing. During a running landing, instead of coming to a stationary hover before touchdown, the pilot conducts an approach and landing like a conventional fixed-wing airplane. In addition to a hydraulic failure, the loss of antitorque control (tail rotor failure) is the other principal reason a pilot would be required to conduct a running landing.

[48] Translational lift speed is the forward velocity of the helicopter where the inflow of fresh undisturbed air into the top of the rotor disk greatly improves the rotor efficiency and results in a significant reduction in the power required for a climb or level flight.

Helicopter Performance Capability

The investigation considered the criticality of the high-density altitude and the helicopter's weight. These considerations revealed that any interruption in power, however momentary or slight, would have demanded that the pilot initiate an immediate autorotation in order to maintain the critically important rotor speed in the green range. The investigation also considered the possibility of a deceleration event, which could have induced an attempted autorotation, and then the engine recovering to a high operating speed at the time of impact. Although this scenario is possible, it is considered highly unlikely. The 3-minute or less timeline of the accident sequence does not favor the possibility that the engine could have decelerated to idle and accelerated back to full power. An examination of historical accident/incident data indicates that most turbine engine malfunctions resulting in a loss of power require the pilot to lower the collective to maintain rotor rpm and enter autorotation.

All forced landings are ideally executed into the wind and rarely with a tailwind. Optimal forced landing areas were available downslope, away from the accident site, on a westerly heading toward lowering terrain, which would also have provided the pilot more time to prepare for a landing or troubleshoot the problem. A westerly heading would have also maintained the helicopter into the prevailing headwind. The investigation notes that the extremely hazardous nature of the accident site topography makes it improbable that the pilot intentionally selected the accident site as a landing spot in response to any emergency, particularly when several other adequate locations would have been available to the pilot. The investigation also could not determine why the helicopter arrived at the accident site location at a high rate of descent with an apparently fully functioning helicopter with a running engine and driving rotor system.

For a complete understanding of possible accident scenarios, it is important to consider the following performance capabilities of the helicopter for the weight and atmospheric conditions existing at the accident site:[49]

- The helicopter was marginally capable of HOGE at the top of the plateau.
- The helicopter was marginally capable of HOGE at the altitude of the accident site.
- The available power margin at the accident site altitude translates into a helicopter vertical climb capability from a hover of only 421 fpm at best. If the helicopter were descending vertically (or nearly so) at a greater rate, the pilot could not arrest the descent.
- Use of the right antitorque pedal utilizes drive train power that would otherwise be available for hover or vertical climb capability. Full-right pedal input consumes almost 20 percent of the available drive train power and would significantly affect the power available margin to the main rotor system.

One possible scenario that is consistent with the evidence is the condition known as settling with power; however, this scenario assumes that the pilot elected to execute a descending 180°

[49] The temperatures used for these calculations were the most conservative chosen. Pilots landing at the accident site immediately following the crash estimated significantly higher ambient air temperatures, which would translate into performance capabilities more degraded than the numbers used in the report.

turn with an abrupt slowing of forward speed back toward the cliff after crossing the face of the Grand Wash Cliffs. In the absence of a survivor providing an explanation or any flight recorder data, the investigation could not determine why the pilot would have initiated such a maneuver; therefore, a determination of a settling with power event cannot be made.

A hydraulic systems failure, in the worst of cases, could possibly produce a right descending turn; however, the magnitude and severity of the turn and descent with this scenario is unlikely to result in the impact geometry seen in the wreckage. In the event of either a failure of the hydraulic system or the system being inadvertently deactivated by the cockpit switches, once the system accumulators bleed down to zero pressure, the dynamic forces in the main rotor will drive the collective down and the cyclic to the full-aft and full-right position. As the servos and cockpit controls move to these positions, the helicopter will enter a right descending turn if the pilot does nothing to counter this motion.[50] The scenario also requires the highly improbable condition that the pilot had his hands off the controls and was distracted to the point that he failed to notice an extremely abrupt attitude change and a violent movement of the cyclic flight control that would most likely have hit his right leg very hard. Since there was no filament stretch in the hydraulic system failure warning light in the enunciator panel, the investigation concludes that no system failure occurred.

During the analysis of the helicopter's flight profile from the last fueling stop through the route to the accident site, comparisons were made to determine the possible effects of hydraulic servo transparency to the flight profile. Servo transparency can cause the collective to move downward, and the cyclic to move rearward and to the right. During this phenomenon, both collective forces and cyclic forces create flight control forces that cause the helicopter to pitch up and to the right, and to slow. A review of flight test reports indicates that servo transparency could be encountered at high gross weights, high forward airspeeds, and during maneuvers that produce high flight loads on the rotor system. The tests also showed that servo transparency could not be encountered if the collective is less than 50% raised, and that servo transparency usually lasts only a few seconds. Analysis indicates that the flight profile of the helicopter included a region where servo transparency might have been encountered if the helicopter was maneuvered with a high-speed pull up following the dive over the cliff. However, since servo transparency effects would have caused a pitch up and a slowing of the helicopter, rather than a dive, and given the impact geometry of accident site, servo transparency is not considered to be a likely factor in the accident.

In the absence of any evidence to indicate a preimpact mechanical malfunction, and given the density altitude, helicopter performance considerations, and virtually all of the signatures evident at the IPI and in the wreckage, the investigation revealed that a probable scenario involves the pilot's decision to maneuver the helicopter in a flight regime, and in a high density altitude environment, which significantly decreased the helicopter's performance capability, resulting in a high rate of descent from which the pilot was unable to recover prior to ground impact. Additionally, although no evidence was found to indicate that the pilot had intended on performing a hazardous maneuver, the high rate of descent occurred in proximity to precipitous terrain, which effectively limited remedial options available.

[50]Although the hydraulics-off control forces are relatively high, in the 20- to 25-pound range, they are completely manageable.

Flight Recorders

The almost complete destruction of the helicopter by impact forces and postcrash fire hindered the investigation and required the expenditure of significant resources. The accident helicopter was not required to be equipped with an FDR or a CVR; however, the Safety Board notes that this investigation would have benefited from having recorders installed in the helicopter.

Since January 2000, the Safety Board has investigated numerous accidents involving turbine-powered aircraft that were not required to be equipped with either a CVR or an FDR. These accidents involved aircraft operating under 14 CFR Parts 91 and 135. Included among these accidents was the October 25, 2002, accident involving a Raytheon (Beech) King Air, which crashed on approach to Eveleth-Virginia Municipal Airport, Eveleth, Minnesota, killing all eight persons on board, including Senator Paul Wellstone. The airplane was not equipped with either a CVR or an FDR at the time of the accident, nor did Federal regulations require it to be so equipped.

The Safety Board has investigated several cases in which the aircraft was not required to be equipped with a flight recorder,[51] but in which a CVR was installed voluntarily on the aircraft. The Board has found that data from these CVRs provided invaluable information during its investigations. Specifically, in the beginning phases of an investigation, CVR data may reveal operational issues that are not readily apparent from the physical evidence found at an accident site. Such critical data regarding the aircraft's operation and the flight crew's actions not only help to immediately narrow the focus of an investigation but may also be the sole source of evidence for a probable cause and recommendations that could prevent future similar occurrences.

In addition, Safety Board investigators have repeatedly found that CVRs installed in conjunction with FDRs provide data instrumental in reconstructing events leading to the accidents. Specifically, CVRs have provided insight into the operational environment within the cockpit, and FDRs have provided information regarding the aircraft's performance. Using data from both recorders, investigators have been able to determine the aircraft's motion and flight crewmember response to it or, conversely, how crewmember actions affected the airplane's performance. The CVR and the FDR each provide a different but complementary perspective on the events leading to an accident.

Although CVRs and FDRs are required on most larger passenger-carrying aircraft,[52] smaller aircraft are excluded by the current regulations and are not required to be equipped with any crash-protected recorder: specifically, single-pilot certificated turbine-powered aircraft and dual-certificated cargo/passenger aircraft. When neither CVR nor FDR data are available, Safety

[51] For private operations, CVR and FDR requirements are in 14 CFR 91.609. For charter operations, CVR requirements are in 14 CFR 135.151, and FDR requirements are in 14 CFR 135.152. For scheduled transport operations, CVR requirements are in 14 CFR 121.359, and FDR requirements are in 14 CFR 121.343, 14 CFR 121.344, and 14 CFR 121.344a.

[52] The Safety Board has previously addressed the issue of insufficient recorded data from larger aircraft (that are currently required to be equipped with a both a CVR and an FDR) in Safety Recommendations A-00-30 and -31.

Board investigators can sometimes compensate in part with radar data or air traffic control recordings. However, these data do not provide the same level of detail about the aircraft's flightpath, flight conditions, or operations that can be provided by CVR and FDR data. Furthermore, when accidents occur in areas outside radar coverage, such as the case in this accident, these data are not available.

Many turbine-powered aircraft, including the AS350, Bell 407, and Bell 206L helicopters are single-pilot-certificated aircraft that are heavily used in Part 135 passenger charter and other commercial operations. Under 14 CFR Parts 91 and 135 operations, a CVR is required only if the aircraft are certificated to operate with two pilots. Moreover, because these aircraft are configured with fewer than 10 passenger seats, they are not required to have an FDR installed on board. Although CVR data from single-pilot operations may not provide the same level of operational information as would be expected when two or more crewmembers are present, the CVR can capture events that are important to accident investigators. Specifically, investigators routinely extract aircraft event data from a CVR recording—some of which might have been available from an FDR, if one had been installed—such as warnings and alerts, engine power settings, and main rotor rotation speed. Significantly, despite their single-pilot certification, aircraft in this category are often operated with two pilots due to insurance requirements or for other reasons, such as safety-of-flight, making the presence of a CVR even more valuable to Safety Board investigations.

In a safety recommendation letter sent to the FAA on December 22, 2003, the Safety Board stated that it identified the need to install crash-protected recording devices on all turbine-powered aircraft. The Board stated that it recognizes the economic impact of requiring both a CVR and an FDR on smaller aircraft and consequently proposes that all smaller turbine-powered aircraft be equipped with a single crash-protected recorder: the video image recorder. Such recorders obtain not only audio information like that from CVRs and event data like that from FDRs, but also information about the environment outside the cockpit window. The Safety Board made the following recommendations to the FAA:

- Require the installation of a crash-protected image recording system on all turbine-powered, nonexperimental, nonrestricted-category aircraft manufactured after January 1, 2007, operating under 14 *Code of Federal Regulations* Parts 91, 135, and 121. (A-03-62)

- Amend the current regulations for 14 *Code of Federal Regulations* Parts 91, 135, and 121 operations to require all turbine-powered, nonexperimental, nonrestricted-category aircraft that have the capability of seating six or more passengers to be equipped with an approved 2-hour cockpit voice recorder that is operated continuously from the start of the use of the checklist (before starting engines for the purpose of flight), to completion of the final checklist at the termination of the flight. (A-03-63)

- Require all turbine-powered, nonexperimental, nonrestricted-category aircraft that are manufactured prior to January 1, 2007, that are not equipped with a cockpit voice recorder, and that are operating under 14 *Code of Federal Regulations* Parts 91, 135, and 121 to be retrofitted with a crash-protected image recording system by January 1, 2007. (A-03-64)

- Require all turbine-powered, nonexperimental, nonrestricted-category aircraft, that are equipped with a cockpit voice recorder, that are manufactured prior to January 1, 2007, and that are operating under 14 *Code of Federal Regulations* Parts 91, 135, and 121 to be retrofitted with a crash-protected image recording system by January 1, 2010. (A-03-65)

PROBABLE CAUSE

The National Transportation Safety Board determines that the probable cause of this accident was the pilot's decision to maneuver the helicopter in a flight regime and in a high density altitude environment which significantly decreased the helicopter's performance capability, resulting in a high rate of descent from which recovery was not possible. Factors contributing to the accident were high density altitude and the pilot's decision to maneuver the helicopter in proximity to precipitous terrain, which effectively limited remedial options available.

Adopted: June 3, 2004