National Transportation Safety Board - Aircraft Accident/Incident Database

Accident Rpt# CEN11CA608 08/05/2011 1420 CDT	Regis# N22592	Hot Springs, AR	Apt: Hot Springs Memorial Field HOT	
Acft Mk/Mdl AYRES CORPORATION S2R-G10	Acft SN G10-138	Acft Dmg: SUBSTANTIAL	Rpt Status: Unk	Prob Caus: Pending
Eng Mk/Mdl GARRETT TPE331-10-51	Acft TT 2930	Fatal 0 Ser Inj 0	Flt Conducted Under: FAR 137	
Opr Name: WESTERN PILOT SERVICE INC	Opr dba:		Aircraft Fire: NONE	
			AW Cert: SPR	

Narrative

The pilot reported that he prepared the airplane for a fire suppression mission with 450 gallons of water and 228 gallons of fuel. Using aircraft performance data, he calculated that his takeoff weight was 11,065 pounds (maximum take off gross weight was 11,500 pounds) and would be able to perform the mission given the reported temperature and calculated density altitude. The pilot lined up for take off with the wind sock showing about 3 knots down the runway, set the flaps for 15 degrees, and accelerated normally down the runway with engine instruments all showing full power indications.

After liftoff, the pilot raised the flaps and the airplane accelerated to 85 knots. He expected the acceleration to increase to 105 knots during the climb out, but the airplane was not accelerating normally and all engine instruments showed full power. The pilot felt the airplane drop about 50 feet straight down, so he squeezed the jettison trigger to release some water. There was no increase in airspeed or a positive rate of climb after this action, so the pilot released more water with no improvement in airspeed or climb. Nearing the ground, the pilot maneuvered the airplane to avoid hitting houses and trees and continued to release water, all the while attempting to point the airplane toward a clearing. Finally, the pilot pulled the nose back to intensionally stall to avoid hitting another house. The airplane clipped power lines, and the tail hit the edge of a house just before the airplane impacted the ground. The airplane sustained substantial damage to both wings, the fuselage, and the tail section. No mechanical anomalies with the airplane of engine were discovered that would have contributed to the accident.

After the accident, the pilot stated that the temperature was over 100 degrees with a density altitude near 5,000 feet (takeoff elevation 538 msl) and that there was very little relative wind during takeoff. Although the airplane was about 435 pounds under the maximum gross weight for take off, the pilot noted that he could have jettisoned the entire load of water when the aircraft was not climbing, thus reducing the total weight by 3,800 pounds and possibly allowing a positive climb rate. The high temperature, light relative wind for takeoff and the airplane's takeoff weight (near maximum gross) at the time of takeoff could have diminished the airplane's climb performance.

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Accident Rpt# ERA12WA086 11/25/2011 1640 UTC	Regis# HCCLH	Quito, EC		
Acft Mk/Mdl BELL 212-NO SERIES	Acft SN 30879	Acft Dmg: SUBSTANTIAL	Rpt Status: Unk	Prob Caus: Pending
Eng Mk/Mdl PRATT AND WHITNEY CANADA PT-6		Fatal 0 Ser Inj 4	Flt Conducted Under: FAR NUSN	
Opr Name: AEROMASTER AIRWAYS, SA	Opr dba:		Aircraft Fire: NONE	

Narrative

On November 25, 2011, about 1640 coordinated universal time (1140 local time), a Bell 212, Ecuadorian registration HC-CLH, operated by Aeromaster Airways, S.A., was substantially damaged when it impacted steep terrain next to the company's helipad in Quito, Ecuador. The two Ecuadorian pilots and two on board mechanics were seriously injured. Visual meteorological conditions prevailed for the local post-maintenance check flight conducted under Ecuadorian flight regulations.

Preliminary information from Ecuadorian authorities indicated that the helicopter had first conducted a series of hover checks before initiating a takeoff over a ravine. According to the pilot, during the takeoff, the helicopter lost rotor rpm, and he attempted a return to the helipad. However, the helicopter impacted the side of the helipad and descended 36 meters to the bottom of the ravine, where it came to rest on its left side in a creek.

The investigation is under the jurisdiction of the government of Ecuador. Further information can be obtained from:

Direcci¢n General de Aviaci¢n Civil
Junta Investigadora de Accidentes de la Rep£blica del Ecuador
Avenida Col¢n E5-56 y La R bida
Edificio Ave Mar¡a Torre Sur
Quito, Rep£blica del Ecuador
Tel: 593-02-223-8981
www.dgac.gov.ec

This report is for informational purposes, and only contains information released by the government of Ecuador.

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Incident Rpt# ERA11IA316 **05/27/2011 928 EDT**	**Regis# N749QS**	**Newburgh, NY**	**Apt: Stewart International Airport SWF**
Acft Mk/Mdl ISRAEL AIRCRAFT INDUSTRIES	**Acft SN 165**	**Acft Dmg: MINOR**	**Rpt Status: Unk** **Prob Caus: Pending**
Eng Mk/Mdl P&W CANADA PW306A	**Acft TT 3010**	**Fatal 0** **Ser Inj 0**	**Flt Conducted Under: FAR 091K**
Opr Name: NETJETS AVIATION INC	**Opr dba:**		**Aircraft Fire: NONE**

Narrative

HISTORY OF FLIGHT

On May 27, 2011, at 0928 eastern daylight time, an Israel Aircraft Industries Gulfstream 200, N749QS, managed by NetJets Inc., incurred minor damage when the right main landing gear collapsed during an emergency landing at Stewart International Airport (SWF), Newburgh, New York. The two certificated airline transport pilots and one passenger were not injured. The personal flight was conducted under the provisions of 14 Code of Federal Regulations Part 91K. Visual meteorological conditions prevailed and an instrument flight rules flight plan was filed for the flight destined to Westchester County Airport (HPN), White Plains, New York. The flight originated from Greenville Spartanburg International Airport (GSP), Greer, South Carolina, about 0730.

The pilot-in-command (PIC) stated that during approach to HPN, the landing gear lever was selected to the extend (down) position. Sounds associated with landing gear transit were heard; however, the landing gear cockpit indications displayed three red lights. He aborted the approach and entered a holding pattern to complete the appropriate checklist items. Approximately 20 to 40 seconds later, a "R HYD OVERHEAT" message illuminated on the engine indicating and crew alerting system (EICAS). At that time, hydraulic pressure was about 1500 psi (normal is 3000 psi), where it remained for the remainder of the flight. The flightcrew then completed the checklist items for a right hydraulic overheat.

The flightcrew subsequently performed the emergency gear extension checklist items and utilized the emergency gear blow-down bottle. The resultant cockpit indications were nosegear green, but the right and left main landing gear remained red. The flightcrew then declared an emergency and elected to divert to SWF due to a longer runway and less traffic. Before diverting to SWF, HPN tower personnel reported that they observed the airplane's three landing gear in the extended position. While enroute to SWF, the flight reviewed the right hydraulic system failure checklist. Upon landing on runway 27, the airplane remained level for 2 to 3 seconds and then began slowly tilting to the right. The airplane then settled on its right wing and slid to a stop on the runway.

PILOT INFORMATION

The PIC, age 45, held an airline transport pilot certificate, with a rating for airplane multiengine land; and a commercial pilot certificate, with a rating for airplane single-engine land. He also held a type rating for the Gulfstream 200. The PIC reported a total flight experience of 10,013 hours; of which, 3,244 hours were in the Gulfstream 200. He flew 105 hours and 25 hours during the 90-day and 30-day periods preceding the incident, respectively. The PIC's most recent Federal Aviation Administration (FAA) first-class medical certificate was issued on January 6, 2011.

The second-in-command (SIC), age 41, held an airline transport pilot certificate, with a rating for airplane multiengine land; and a commercial pilot certificate, with ratings for airplane single-engine land, rotorcraft helicopter, and instrument helicopter. He also held a type rating for the Gulfstream 200. The SIC reported a total flight experience of 5,800 hours; of which, 604 hours were in the Gulfstream 200. He flew 65 hours and 37 hours during the 90-day and 30-day periods preceding the incident, respectively. The SIC's most recent FAA first-class medical certificate was issued on December 30, 2010.

AIRCRAFT INFORMATION

The 11-seat airplane, serial number 165, was manufactured in 2007. It was powered by two Pratt & Whitney Canada PW306A engines, each capable of generating 6,040 pounds of thrust. The airplane was maintained under an approved inspection program. It's most recent inspection was completed on May 14, 2011. At that time, the airplane had accumulated 2,982 total hours of operation, and it had been operated an additional 23 hours since that inspection.

Additionally, maintenance work was performed on the airplane from May 22, 2011 to May 26, 2011. The maintenance work included inspection of the nosegear uplock actuator, cleaning the emergency gear blow-down valve mounting hardware, and replacing the landing gear selector valve rod end. Following the maintenance work, the landing gear was cycled twenty times with no anomalies noted. Subsequently, about 0500 on the day of the incident, the incident flightcrew departed Savannah, Georgia on a repositioning flight to GSP. They arrived at GSP uneventfully and did not report any problems with the landing gear system.

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METEOROLOGICAL INFORMATION

The weather reported at SWF, at 0930, was: wind calm; visibility 15 miles; few clouds at 15,000 feet; temperature 24 degrees C; dew point 17 degrees C; altimeter 30.01 inches of mercury.

FLIGHT RECORDERS

The airplane was equipped with a cockpit voice recorder (CVR), which was forwarded to the NTSB Vehicle Recorders Laboratory, Washington, DC for readout. The CVR recorded 2 hours of audio data. A CVR Group convened and prepared a summary report of the most recent 30 minutes from the recording.

According to the summary report, at 0902, the flightcrew was performing a normal approach to HPN, with the PIC as the pilot flying in the left seat and the SIC as the pilot monitoring in the right seat. The SIC had initiated a landing gear extension and sounds were heard consistent with landing gear travel; however, the SIC noted that the three green lights did not illuminate in the cockpit. Additionally, at that time, a master warning (red EICAS message, "GEAR NOT DOWN") activated along with a landing gear warning horn. The flightcrew then received radar vectors for a hold to perform checklist items.

At 0903, as the PIC continued flying, the SIC initiated the Landing Gear Down Lock Indication Failure checklist contained in the quick reference handbook. He completed items No. 1 and No. 2, pertaining to the flaps and airspeed, respectively. Item No. 3 pertained to right hydraulic pressure, to which the SIC noted that there was a problem. Additionally, before addressing item Nos. 4 and 5, which would have instructed the SIC to cycle the landing gear selector handle, a master caution (amber EICAS message, "R HYD OVERHEAT") activated due to a right hydraulic overheat condition. Seconds later, the SIC also noted a burning smell. The PIC then expressed the need to get the airplane on the ground.

At 0905, the SIC suspended the Landing Gear Down Lock Indication Failure checklist, to initiate the Hydraulic System Overheat checklist. That checklist included the instruction to reduce the affected engine to idle power and the PIC reduced the right engine thrust to idle power.

At 0907, the SIC returned to the Landing Gear Down Lock Indication Failure checklist, item No. 3, and noted that if hydraulic pressure is "normal," proceed to item No. 4 (cycle gear); however, hydraulic pressure was not "normal." Additionally, the SIC also noted that item No. 7 instructed that the emergency landing gear extension should be performed if the hydraulic pressure was "low."

At 0908, the flightcrew switched duties due to the location of the emergency landing gear extension controls. The PIC read the Emergency Landing Gear Extension checklist, including item No. 3, which stated that the landing gear lever is in the down position. He then performed item No. 4, which was the release, turn, and lift of the emergency gear extension handle. Although item No. 5 stated that the landing gear was down and locked with a three-light indication, the flightcrew noted that only the nosegear was down and locked. The checklist did not include any additional instructions pertaining to a situation where all three landing gear were still not down and locked.

At 0911, the flightcrew switched duties again and discussed diverting to an airport with a longer runway. They declared an emergency and the PIC flew toward SWF at 0915.

At 0912, the airplane flew past the HPN control tower and tower personnel advised that all three landing gear appeared to be extended.

At 0919, the SIC began to read the Right Main Hydraulic System Failure checklist, but the flightcrew agreed that the hydraulic system had not failed.

At 0922, the flight was cleared for the visual approach to runway 27 at SWF and the SIC reported a left base leg at 0923 to the SWF tower.

At 0925, the PIC remarked that the hydraulic temperature was rising again and that they needed to get the airplane on the ground. At 0927, the flight was on short final approach at 1,000 feet. At 0928, about 3 seconds after touchdown, the right main landing gear collapsed.

WRECKAGE INFORMATION

Initial examination of the airplane by FAA inspectors revealed minor damage to the right wing, consistent with ground contact. The inspectors subsequently examined the airplane with representatives from the airplane manufacturer, management company, and pilots' union.

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The examination revealed that the landing gear selector handle was found approximately 1/8 to 1/4 inch from the full down position. The airplane was placed on jacks and supplied electrical power and hydraulic pressure from ground carts. The landing gear selector handle was then positioned full up, followed by full down, and the landing gear cycled successfully. During operation of the landing gear handle, there was no tactile feel of a detent as the make and model system utilized detents on the landing gear selector valve, rather than the landing gear handle. Subsequently, an emergency gear extension test (blow-down) was performed and all three landing gear moved to the down and locked position.

Examination of the hydraulic system did not reveal any failures. Additional tests were performed to attempt to simulate the in-flight event. When the landing gear selector handle was positioned to where it was found, a hydraulic bypass "hissing" sound was heard, accompanied by a rise in hydraulic temperature and reduction in hydraulic pressure. The main landing gear extended, but did not lock and the nose landing gear did not extend; however, when the emergency gear extension handle was activated, all three landing gear extended and locked.

The landing gear selector valve rigging was visually inspected, photographed, discussed, and no anomalies were noted at that time. The rigging was not further examined; however, the landing gear selector valve, landing gear emergency blow-down valve, and right main landing gear side brace actuator were forwarded to their respective manufacturer's facilities for testing and examination under government supervision. Utilizing the individual component acceptance test procedures, testing of the units did not reveal any mechanical malfunctions, nor did the subsequent teardown examinations. According to a representative of the airplane manufacturer, most of the Gulfstream models had a landing gear selector handle that must be moved left out of a detent, then down, then right into a detent to extend the landing gear. However, the Gulfstream 200 was formerly an Israel Aircraft Industries Galaxy, and the detent mechanism was not located on the handle itself, but on the landing gear selector valve instead. The representative added that the hydraulic bypass from an intermediate position of the landing gear selector valve would result in the failure of the landing gear down locks to engage due to back pressures; however, the examinations could not positively determine why the emergency landing gear extension (blow-down) worked during the ground test, but not during the incident flight. One possibility was that a hydraulic mule was used for the ground test, whereas the actual airplane hydraulic system was used in flight.

ADDITIONAL INFORMATION

Checklists

After the incident, the airplane manufacturer revised several of its normal and abnormal checklists. Specifically, terms such as "normal" and "low" were replaced with actual numerical values. Additionally, the Landing Gear Down Lock Indication Failure and Emergency Landing Gear Extension abnormal procedures were revised to include more guidance on ensuring the landing gear handle was at the full extent of its downward travel. Lastly, the Emergency Landing Gear Extension abnormal procedure was expanded to include a situation where an emergency blow-down procedure failed to extend and lock all three landing gear.

Landing Gear Selector Valve Rigging

Subsequent to the on-scene examination, manufacturing drawings were obtained of the landing gear selector valve. Review of the drawings and specifications revealed that when positioned to the full gear down position, the landing gear selector valve arm must be rigged within plus 10 degrees to minus 5 degrees from the zero reference point to prevent hydraulic bypass. Teardown testing of the incident valve revealed a larger design margin of plus 20 degrees to minus 5 degrees from the zero reference. Comparison of the drawings to photographs of the landing gear selector valve as found revealed that when the landing gear selector handle was placed in the full down position, the landing gear selector valve arm was approximately plus 12 degrees from the zero reference point.

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Accident Rpt# WPR10LA295 06/16/2010 1440 MDT Regis# N134WC Donnelly, ID

Acft Mk/Mdl KAMAN AEROSPACE CORP K-1200 Acft SN A94-0006 Acft Dmg: SUBSTANTIAL Rpt Status: Unk Prob Caus: Pending

Eng Mk/Mdl LYCOMING T-53 Fatal 1 Ser Inj 0 Flt Conducted Under: FAR 133

Opr Name: WOODY CONTRACTING INC Opr dba: Aircraft Fire: NONE

AW Cert: SPR

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Narrative

HISTORY OF FLIGHT

On June 16, 2010, about 1440 mountain daylight time, a Kaman K-1200 helicopter, N134WC, impacted the terrain about five miles west of Donnelly, Idaho. The commercial pilot, who was the sole occupant, was killed in the accident sequence, and the helicopter, which was owned and operated by Woody Contracting Inc., sustained substantial damage. The 14 Code of Federal Regulations Part 133 long-line logging flight had been airborne for about an hour and 40 minutes. The flight was taking place in visual meteorological conditions. No flight plan had been filed.

According to the four witnesses near the site, the helicopter, which was using a 200 foot long-line, had just lifted a large log off the ground when the accident sequence began. Two of the witnesses were looking directly at the helicopter as it lifted the log off the ground, and the other two looked immediately toward the helicopter after hearing a very loud noise that coincided with the beginning of the accident sequence. The pilot of the helicopter, who intended to take the log to a collection landing area about one-quarter mile away, had already attempted to lift the log off the ground once, but other logs interfered with the lift, so he set it back down. The pilot then asked the hooker to reset the choker nearer to the end of the log, which according to one of the witnesses would cause the log to hang more straight down during the lift. The hooker therefore reset the choker nearer to the butt end of the log, whereupon the pilot transmitted that, "That is more like it." Soon thereafter, the pilot again began to lift the log. According to the two witnesses that had a good view of the log itself, the accident sequence began just after the log was completely off the ground.

The two witnesses who were looking directly at the helicopter perceived the accident sequence to have begun when they saw what appeared to be an outboard section of one of the rotor blades separate and fly through the air. This was followed a "split second" later by a very loud bang or pop. To the two witnesses who were not looking directly at the helicopter, the sequence began when they heard the very loud noise. Both of these individuals, one of whom was about one-quarter mile away, and the other (the hooker) who was within 50 feet of the helicopter, immediately turned to look directly at the helicopter. The first thing that both of these individuals saw was a piece (pieces according to the hooker) of the rotor blade flying through the air. Almost immediately after the separation of the rotor blade piece (pieces), other portions of the helicopter's blades and structure began to separate. The helicopter then went out of control, descended toward the terrain, ultimately coming to rest inverted. There was no fire.

According to all four witnesses, there were no unusual engine or rotor sounds prior to the initiation of the accident sequence. They all stated that everything looked and appeared normal prior to the separation of the first rotor blade piece. Two of the witnesses made the point of saying that they had worked around helicopters for a long time, and felt that they would have recognized any abnormal change in engine or rotor sound. One witness made the point that he had heard unusual engine and rotor sounds from other helicopters over the years, but that this time everything sounded perfectly normal.

Of the two witnesses who were looking directly at the helicopter, the one closest to the event was asked if it appeared that the pilot ascended faster than normal or tried to jerk the log off the ground. He responded that the pilot had not done either of those things, and that the lift looked like any other lift involving a single large log. The witnesses that were in position to judge how close the helicopter was to any trees were adamant that its rotor blades had not come in contact with a tree or any other obstacle.

PERSONNEL INFORMATION

The pilot, a 64 year old male, held a Commercial pilot certificate, with a helicopter rating and an airplane single engine land rating. His last FAA airman's medical, a class 2, was performed on 3/18/2001, and was issued without limitations or waivers. His total pilot time as of the date of the accident was estimated to be 27,600 hours, with 12,600 hours in the Kaman 1200. His last flight review was on December 8, 2008, in a Robinson R-22.

AIRCRAFT INFORMATION

The aircraft was a 1996 Kaman Aerospace K-1200 KMAX helicopter, serial number A94-0006, registered as N134WC, and powered by a Honeywell/Lycoming

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T53-17A1 engine, rated at 1,500 horsepower. The helicopter's last progressive inspection under its Approved Airworthiness Inspection Program (AAIP) was on June 15, 2010. At the time of the accident, the airframe had accumulated 24,053 hours.

METEOROLOGICAL INFORMATION

At the time of the accident, the helicopter pilot was working under a 12,000 foot overcast ceiling, in winds that varied from calm to variable at six knots. Visibility was more than ten miles. The temperature was about 45 degrees F.

WRECKAGE AND IMPACT INFORMATION

The helicopter lifted the log from the ground about the 6,300 foot level of the southeast facing down-slope of North Business Mountain, at about 44 degrees, 45.771 minutes north by 116 degrees, 11.138 minutes west. At the termination of the accident sequence, the airframe came to rest inverted over the top of the log it had lifted, about 410 feet southeast of the lift location, at an altitude of about 6,180 feet. The airframe wreckage included the main fuselage, the nose and cockpit area, the main and nose landing gear, the engine, transmission, and right rotor mast pylon, both left and right stabilizers, and the tail boom, including the vertical fin and rudder.

Both horizontal stabilizers were fully attached, and both vertical stabilizers were fully attached to their respective horizontal stabilizers. Both horizontal stabilizers had rotated about 180 degrees so that the leading edge of both were facing aft, and the bottom tip of both vertical stabilizers were facing toward the top of the fuselage. The outboard two-thirds of the trailing edge of the right horizontal stabilizer was bent down to about 90 degrees, and trailing edge of the top half of the associated vertical stabilizer was bent outward to about 90 degrees. The top half of the right vertical stabilizer was bent inward toward the fuselage at about 45 degrees, but the bottom half was in its normal alignment. The left horizontal stabilizer was undamaged, and its associated vertical stabilizer was undamaged except for an area near the forward portion of the bottom tip where it had been crushed inward about 10 inches by impact with a limb of the log that was being lifted. The tail boom had twisted counter-clockwise (looking forward) just aft of the tail boom mounting points, and its structural skin had been almost entirely ripped apart at that location. All five tail boom mounting bolts were connected through their fittings, and the fittings themselves were undamaged. The remainder of the tail boom, aft of where it had twisted/torn, was undamaged, and the rudder was fully connected and free to pivot. There was no evidence of any flight control surface anomaly or malfunction.

The belly hook was in the open position, and the bow shackle on the upper end of the long-line was released from the hook. The long-line itself was wrapped around the fuselage and left stabilizer, then strung up into some nearby birch trees, and then back down to the remote hook. The remote hook was connected to the choker, which was still attached to the log about three feet from its butt end. There were no marks or damage on the long-line that would have been consistent with it coming in contact with the rotor blades.

The transmission was in its correct location, but the KAFlex coupling between the engine and transmission had sheared, with the associated adapter rings still being attached to the transmission and engine respectively. The forward engine mounts had separated from the helicopter's airframe structure, and the aft engine mounts had been torn from the engine diffuser housing. The right pylon was attached to the transmission, but was partially separated along the outboard edge of the transmission top cover and pylon base. The mast still protruded from the top of the pylon, and was imbedded in the dirt under the inverted fuselage. The rotor hub assembly had fractured releasing blades 94A and 94B. The majority of the structure of blades 94A and 94B was found southwest of the fuselage about 470 feet and 440 feet respectively. The most outboard six and one-half feet of blade 94B was located at a later date about two-tenths of a mile south of the fuselage.

The left pylon, including the upper cover and the planet gears, had separated from the transmission where the pylon bolts to the transmission top cover, and was located about 190 feet east of the fuselage section. It had come to rest inverted (mast tip imbedded in the soil), with about 12.5 feet of blade 169B and 3.5 feet of blade 169A still attached. The damper tube black end was attached to blade 169B, and the fractured damper tube white rod end was attached to blade 169A. The U-crank was still attached to blade 169B, but the U-crank on blade 169A had broken off and was lying alongside the blade section, connected only by the idler assembly. The grips and teeter pins associated with both blades were intact. The majority of the remainder of blade 169A was located in an area about 250 feet west of the fuselage. The remainder of blade 169B was more widely scattered, with much of the blade not recovered, including any structure that could be positively identified as being outboard of station 260.

The servo flap for blade 94A was attached to both of its mounting brackets, and both brackets remained securely attached to the blade afterbody. The leading edge/spar of the servo flap for blade 169A was attached to both of its mounting brackets, and both brackets were securely attached to the blade afterbody. The servo flap afterbody for blade 169A had separated along a straight span-wise line about one inch aft of its pivot point along its top skin, and about two inches aft of the pivot point along its lower skin. The servo flap afterbody itself was found lying near the tail of the helicopter. Most of the servo flap for blade 169B was

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found about 230 feet north of the fuselage. Its leading edge/spar, including the stainless steel protection strip, was all present, and it retained about two-thirds of the outboard portion of its afterbody. The inboard end of the leading edge/spar itself was attached to the inboard mounting bracket and pitch change linkage, but it had fractured about four inches outboard of the pivot bearing, and was retained only by the pivot rod. Its outboard pivot bearing had been pulled from the associated mounting bracket, and the mounting bracket itself was found securely attached to about an 18 inch section of blade afterbody. The servo flap for blade 94B was fragmented and recovered in pieces. Its inboard fitting was securely attached to the blade body, but the inboard end of the flap leading edge/spar had separated at the inboard pivot bearing, and was retained only by the pivot rod and the pitch change linkage. The leading edge/spar protection strip was all present, although deformed and bent forward along its outboard most eight inches. The leading edge/spar structure was present, except for about its most outboard four inches. The outboard hinge fitting was recovered, but it was completely detached from the servo flap pivot bearing. Its forward one-half was attached only to a piece of blade honeycomb structure barely wider than the fitting itself, and its forward most three inches was bent up and aft. The leading edge of the fitting and its forward attachment bolt and nut were damaged and deformed toward the trailing edge, in a manner consistent with impact from a blade rotating counter to it.

After the on-scene investigation was completed, the wreckage was recovered to the facilities of SP Aircraft, in Boise, Idaho, where the NTSB Investigator-In-Charge (IIC), oversaw a layout inspection of the entire wreckage, including the engine. At the completion of that inspection, blades 94B and 169B, which had witness marks indicating that their leading edges had come in contact with each other, were shipped to the Kaman Aerospace Corporation, in Bloomfield, Connecticut. There they underwent a series of X-ray inspections, overseen by a representative from the Federal Aviation Administration (FAA) Manufacturing Inspection District Office at Windsor Locks, Connecticut (MIDO -41). The blades were subjected to the X-ray process in order to determine whether there were any preexisting anomalies within the interior of the blades, and to specifically detect any existing span-wise/chord-wise cracks or span-wise delamination in the laminated spruce spar. At the completion of the X-ray inspection, blades 94B and 169B were shipped to the NTSB's Materials Laboratory in Washington, D.C., for further examination and analysis. Upon completion of the Materials Laboratory examination process, blades 94B and 169B were returned to the facilities of SP Aircraft, where the investigative team performed another inspection of the major components of the helicopter's drive train, flight control system, and rotor blades. That examination was overseen and directed by an NTSB Materials Research Engineer. In addition, four panel-mounted instruments were removed from the wreckage and sent to the facilities of Howell Instruments, Inc., in Fort Worth, Texas, for inspection and attempted data retrieval. That activity was overseen by an NTSB Air Safety Investigator from the NTSB's South Central Region.

The aforementioned series of inspections and examinations determined the following:
1. All fractures associated with the separation of the left pylon from the transmission were consistent with overstress, with no indication of preexisting fatigue cracking found.
2. Both the forward and aft left pylon supports failed in overstress, with no indication of preexisting fatigue cracking found.
3. The pylon-to-pylon shear tie failed in overstress on its left side and on the upper channel of its right side. Its right side lower channel cracked, but did not fully fracture. There was no evidence of preexisting fatigue cracks.
4. Turning input to the transmission freely turned the right main rotor shaft and the right azimuth assembly.
5. Both the azimuth assembly and the five planet gears at the bottom of the left pylon turned freely.
6. None of the components of the left side planetary gear system exhibited any visible damage to suggest misalignment, binding or excessive wear.
7. The left and right transmission chip detectors were free of any contamination or metallic particles.
8. All the pivots, bellcranks, rod ends, and bearings associated with the cyclic and collective controls and the horizontal stabilizer control systems were still connected, except where separated by overstress fractures. The aforementioned components all moved freely except where binding occurred due to impact damage.
9. All four main rotor blades fractured in multiple locations, with the servo flap push-pull tubes generally fracturing adjacent to the rod end at the inboard ends of the blades, where they were connected to the U-crank and idler assemblies. All the fractures in the servo flap push-pull tubes were consistent with overstress separation. Blades 94A and 94B separated from the mast assembly at, or just inboard of, the lead-lag pivot. Blades 169A and 169B separated from the blade root about two feet outboard of the end of the blade grip.
10. Both blade 169A and 94A had fractured in two similar locations (about station 50 to 60, and again about station 230 to 240), and neither blade showed evidence of coming in contact with another blade or airframe structure outboard of about station 65. The fractures at both locations on both of the blades were consistent with separation in overstress, and no evidence of a pre-crash anomaly was detected at any of the fractures. Inboard of Station 60, both blades displayed evidence of contact with one another. The bottom skin of blade 94A retained a 1 to 2 inch-wide curved scrape near blade station 52, which was consistent with contact with hardware on the left main rotor. Also, a fragment of glass-fiber composite skin and honeycomb that was wedged between the grip and the top of blade 169A was matched to a location on the lower skin of blade 94A between blade stations 60 and 65. In addition, on blade 169A there was damage to the U-crank and U-crank pivot bolt hole, along with other contact witness marks consistent with blade 94A impacting the U-crank of blade 169A, and pushing the U-crank toward the center of the mast.
11. Blade 94B separated from the right mast by overstress fractures in the hub assembly, and the blade itself fractured in two locations. The most inboard blade

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fracture, which revealed surfaces consistent with an overstress failure, was approximately at the same location as the inboard fractures on blades 169A and 94A (station 55). The outboard end of the blade section inboard of the fracture showed severe leading edge contact damage, and the bottom surface of the blade area just outboard of the fracture contained curving scrape marks consistent with contact with hardware on the left main rotor. The second fracture of the wood spar, at station 193, was associated with the blade body being fragmented into a number of pieces. The fracture in the spar itself at station 193 was consistent with an overstress tension failure near the leading edge, and consistent with an overstress compression failure near the trailing edge. The stainless steel leading edge protection strip, which was still attached to a piece of the blade consisting of station 239 to station 289 (the blade tip), was fractured at station 210, with deformation at the trailing edge of the strip indicating fracture under forward bending. The leading edge protection strip was distorted consistent with impact between stations 185 and 220 (on both sides of the fracture), with the most severe deformation adjacent to the fracture at station 210. The lower surface of the protection strip exhibited a smoothly curving witness mark initiating at station 243 and extending inboard to station 185, consistent with contact with blade 169B.
12. Blade 169B, inboard of a transverse fracture running from station 117 to about station 152, was still attached to the left mast/hub assembly. Outboard of the transverse fracture, blade 169B was the most severely fragmented, and much of the blade was either not recovered or not able to be identified. The bottom surface of the blade inboard of the transverse fracture did not show any evidence of contact marks that would have indicated that it had come in contact with hardware of the right main rotor system. At station 117, the wood spar fracture surfaces were jagged, consistent with tension overstress, toward the leading edge and lower surface, and were flatter, consistent with compression overstress, toward the upper trailing edge. A portion of the stainless steel blade leading edge protection strip, running from station 180.5 to station 225.5, was recovered (in two pieces). The strip, which was indented and deformed in a manner consistent with impact with the leading edge of blade 94B, also retained a vertical indentation 0.75 inch wide and 2 inches high at station 198, consistent with impact from the outboard 0.675-inch-wide servo flap hinge fitting on blade 94B. No identifiable pieces of blade 169B outboard of station 260 were recovered.
13. The X-ray inspection process did not reveal any evidence of pre-existing blade spar cracks or delaminations, nor did it detect any internal blade anomalies not directly associated with impact damage.
14. Inspection of the engine revealed rotational and thermal damage consistent with the production of power both at the initiation of and during the accident sequence. Examination of the engine inlet revealed extensive foreign object damage to the axial compressor and inlet guide vanes, and a number of airframe fittings and hardware pieces were found lying in the inlet against the remains of the inlet guide vanes. The first stage compressor blades were all missing, along with the first stage stator vanes. The second and third stage compressor blades showed heavy bending opposite the direction of rotation and the remaining stator vanes were bent in the direction of rotation. The power turbine, reduction gearbox, and output shaft were free to rotate, and all rotated smoothly by turning the remains of the KAflex coupling and output shaft of the engine. The power turbine was intact, and had fine light colored debris adhering to the surfaces in the area of the 2nd stage power turbine wheel. There was also a spray pattern consistent with melted metal that had adhered to the 2nd stage nozzle, and a silver-gray colored material had collected on the inside of the blade tip shrouds. The power turbine governor drive shaft was removed, and found to be intact and in good condition. Continuity was confirmed between the power turbine group and the gearbox, and to the power turbine governor, by rotating the engine output shaft and witnessing the rotation of the splined gearshaft.
15. Of the four panel-mounted instruments that were removed for inspection and attempted data retrieval (H1900K-22 EGT Indicator, H1901-1 NG RPM Indicator, H1973-1 Load Indicator, H1943-1 Torque Indicator), all had sustained varying degrees of impact damage, up to and including multiple fractures of printed circuit boards. According to a representative of Howell Instruments, no recorded data points were able to be recovered from the EGT Indicator, the Load Indicator, or the Torque Indicator. A Peak Engine Speed (NG) of 101.4 percent was recovered from the NG RPM Indicator, but, according to Howell Instruments, that data point could not positively be determined to be associated with the accident sequence. A Maximum Load Warning Limit of 2,724 kilograms (5,992.8 pounds) was recovered from the Load Indicator, but that was a data point preset by software. Exceedance flags were triggered on both the Load Indicator and the Torque Indicator, but there were no ball codes recorded to further define the exceedance event, and according to Howell Instruments, it cannot be determined if the flags were associated with the accident sequence, some other sequence, or the result of corrupted data.

MEDICAL AND PATHOLOGICAL INFORMATION

An autopsy was performed under the authority of the Valley County Coroner's Office, with the cause of death being blunt force trauma, and the manner of death being accidental.

The FAA's Civil Aerospace Medical Institute (CAMI) performed a forensic toxicology examination on specimens retrieved from the pilot. The results of that examination were negative for carbon monoxide and cyanide in the blood, negative for ethanol in the vitreous, and negative for listed drugs in the urine.

ADDITIONAL DATA AND INFORMATION

TURN WEIGHT

At the request of the NTSB IIC, the operator weighed the 200-foot long-line, the 30 foot choker, and the log that was being lifted. According to the operator, the

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long-line and associated hardware weighed 146 pounds, the choker weighed 15 pounds, and the log itself weighed 6,330 pounds. Based on these weights, the external load being lifted was 6,491 pounds. According to the helicopter's manufacturer and the limitation printed on the side of the helicopter, the maximum permissible external load was 6,000 pounds.

According to the calculations of the operator, the helicopter itself weighed 5,797 pounds at the time of the accident. With the addition of the 6,491 pound total weight of the external load, the helicopter's estimated total gross weight at the time of the accident was 12,288 pounds, which is 288 pounds over maximum allowable gross weight.

BLADE HISTORY

K-Max rotor blades are put into service as matched pairs. Some blades have been re-matched when their original opposite blade was removed from service. When that was done, the new pairing was re-designated as a new serial number pair, and the time-since-new assigned to that pairing going forward was the time-since-new of the higher time of the two blades.

The blade pairings on N134WC were serial numbers 94A/94B, and serial numbers 169A/169B. Blade pair 94A/94B was an original pairing with a zero time installation date of 15 February 1998. Blade pair 169A/169B was a re-matched pair. Blade 169A was originally serial number 65A, with an initial zero time installation date of 1 December 1996. Blade 169B was previously serial number 145B, and originally had been blade serial number 47B, with an initial zero time installation date of 26 July 1995. Records indicated that at the time of the accident, blade pair 94A/94B had accumulated 4,803.2 hours since new, and 1,226.5 hours since overhaul. The records indicated that blade pair 169A/169B had accumulated 9,633.2 hours since new, and 256.1 hours since overhaul. According to records provided by Kaman, at the time of the accident, blade pair 169A/169B had accumulated the eighth highest time since new of the known operational inventory. The highest time blade pair (72A/72B) had accumulated 11,225.0 hours since new (about 1,592 more hours than blade pair 169A/169B).

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