



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

February 14, 2019

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB No: ERA18LA053

A. ACCIDENT

Operator: Global Sky Air Charter Corporation
Aircraft: Sikorsky S-76A++, Registration N911FK
Location: Islamorada, Florida
Date: December 15, 2017
Time: 0652 eastern standard time

B. GROUP

Group Chairman:	Chihoon Shin National Transportation Safety Board Washington, District of Columbia
Member:	Neil Doh Federal Aviation Administration Burlington, Massachusetts
Member:	Javier Casanova Sikorsky Aircraft Corporation Jupiter, Florida
Member:	Bryan Larimore Safran Helicopter Engines Grand Prairie, Texas
Member:	Thomas O'Dea Monroe County Sheriff's Office Marathon, Florida
Member:	Frederic Aime Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile Le Bourget, France

LIST OF ACRONYMS

ALF	aft-looking-forward
ATT	aircraft total time
CFR	Code of Federal Regulations
CMM	component maintenance manual
ECL	engine control lever
EST	eastern standard time
FAA	Federal Aviation Administration
FCU	fuel control unit
FL	Florida
FOD	foreign object debris
lph	liters per hour
MTH	Florida Keys Marathon International Airport
NTSB	National Transportation Safety Board
Ng	gas generator speed
Np	power turbine speed
psi	pounds per square inch
RFM	rotorcraft flight manual
RFMS	rotorcraft flight manual supplement
rpm	revolutions per minute
S/N	serial number
STC	Supplemental Type Certificate
TCDS	Type Certificate Data Sheet
T5	turbine inlet temperature

C. SUMMARY

On December 15, 2017, about 0650 eastern standard time (EST), a Sikorsky S-76A++ helicopter, N911FK, owned by the Monroe County Sheriff's Office and operated by Global Sky Air Charter Corporation (doing business as Trauma Star), was substantially damaged by a fire in the main rotor gearbox compartment after landing near Islamorada, Florida (FL). The airline transport pilot was not injured. The flight was operated in accordance with Title 14 *Code of Federal Regulations* (CFR) Part 135 as an on-demand air medical flight. Visual meteorological conditions prevailed and a company visual flight rules flight plan was filed for the flight that departed the Florida Keys Marathon International Airport (MTH), Marathon, FL at 0637 EST.

The operator reported that the pilot was conducting an air medical trauma patient pickup and had landed at a pre-surveyed landing zone in Islamorada, Florida at 0650 EST. After landing, the pilot stated he positioned the engines controls to ground idle and applied the rotor brake before the onboard medical crew departed the helicopter. After the medical crew departed the helicopter and arrived at the parked ambulance, the flight nurse looked at the helicopter and noticed black smoke, followed by flames coming out of the main rotor gear box area.

The pilot reported that he noticed No. 1 engine temperature fluctuations and shut down the engine. The flight nurse waved her arms and shouted that the helicopter was on fire. The pilot did not observe any indications of a fire from the cockpit; however, he noticed flames when he partially exited the helicopter. He then shut down the No. 2 engine, discharged both fire bottles, and exited the helicopter. He stated the fire continued until the fire department extinguished it.

Examination of the helicopter by a Federal Aviation Administration (FAA) inspector revealed thermal damage to the main gearbox compartment and the engine inlets. The helicopter was relocated to the Monroe County Sheriff's Department hangar for further examination.

D. DETAILS OF THE INVESTIGATION

1.0 HELICOPTER INFORMATION

1.1 HELICOPTER DESCRIPTION

The Sikorsky S-76A++ helicopter has a four-bladed fully articulated main rotor system that provides helicopter lift and thrust.¹ A four-bladed flexible beam tail rotor system provides anti-torque and directional control. The helicopter flight controls are hydraulically assisted by a dual hydraulic system. The S-76A++ is equipped with two Safran Helicopter Engines (formerly Turbomeca) Arriel 1S1 turboshaft engines and a retractable tricycle landing gear. The S-76A++ helicopter is type certificated as a Transport Category A & B helicopter under FAA Type Certificate Data Sheet (TCDS) No. H1NE. The S-76A++ is a marketing designation; Note 11 within TCDS No. H1NE states the S-76A can be alternatively equipped with Arriel 1S1 engines under the provisions of Supplemental Type Certificate (STC) No. SH568NE.

¹ The main rotor blades rotate in a counter-clockwise direction when looking down at the main rotor disk from above.

The terms “left”, “right”, “up”, and “down” are used when in the frame of reference of looking forward from the aft end of the helicopter, i.e. aft-looking-forward (ALF). All locations and directions will be viewed from ALF unless otherwise specified. Additionally, clock positions are in the ALF frame of reference unless otherwise specified.

1.2 ENGINE DESCRIPTION

The Safran Arriel 1S1 turboshaft engine is composed of a two-stage compressor (1 axial and 1 centrifugal), an annular combustor, a two-stage gas generator turbine², and a single-stage power turbine³ (**Figure 1**). A hydromechanical fuel control unit (FCU) governs the gas generator speed (Ng) and power turbine speed (Np).⁴ The engine has a modular layout identified as follows: the accessory gearbox and power transmission shaft (Module 1); the axial compressor (Module 2); the gas generator high pressure section (Module 3); the power turbine (Module 4); and the reduction gearbox (Module 5) (**Figure 2**).

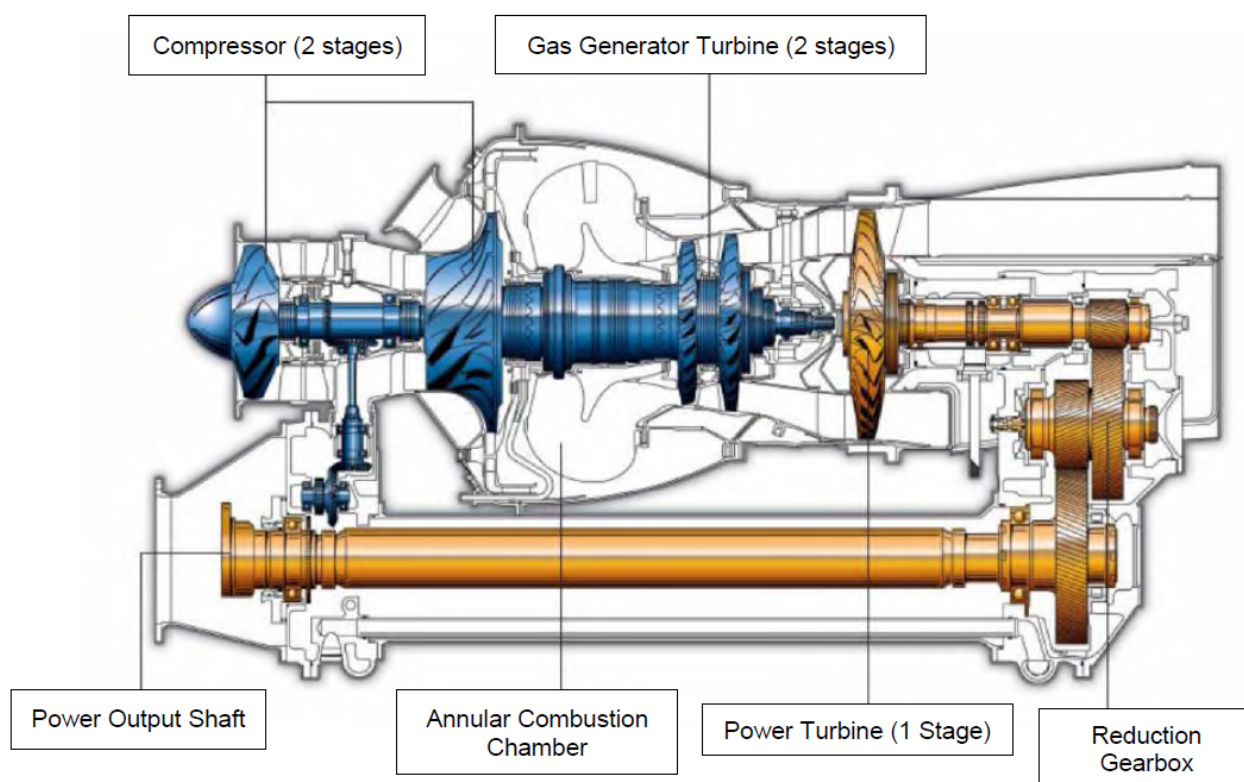


Figure 1. Arriel 1 engine cross-section.

² The gas generator turbine is also known as a compressor turbine or a high pressure turbine.

³ The power turbine is also known as a free turbine.

⁴ Ng is also known as N1. Np is also known as N2 and Nf.

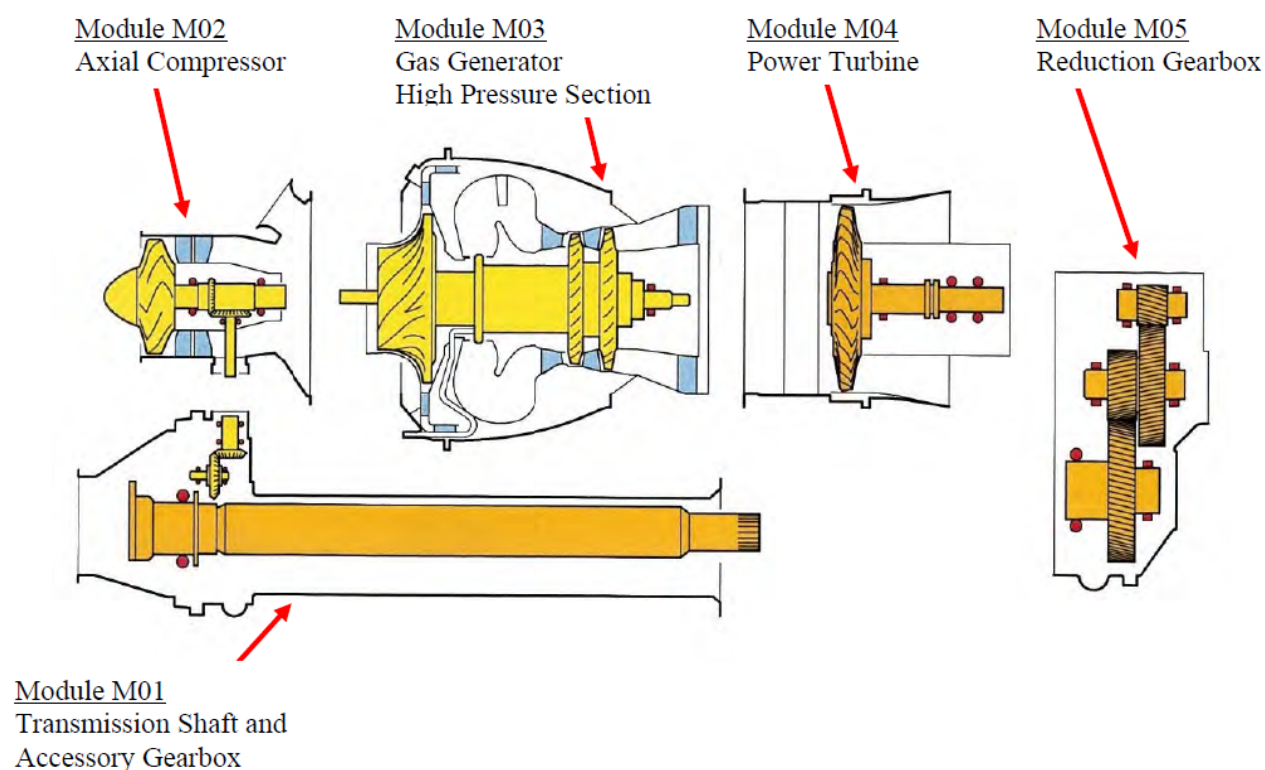


Figure 2. Arriel 1 engine module layout.

1.3 HELICOPTER HISTORY

The accident helicopter, serial number (S/N) 760197, was manufactured in 1981. According to helicopter records, the airframe had accumulated an aircraft total time (ATT) of about 11,432.1 flight hours on the day of the accident. The engines installed on the accident helicopter were S/Ns 15014 and 3029, positioned as the No. 1 (left) and No. 2 (right) engines, respectively. The No. 1 engine, installed on July 23, 2017, had accumulated about 10,412.5 hours. The No. 2 engine, installed on April 13, 2016, had accumulated about 5,101.3 hours.

The helicopter was originally equipped with Arriel 1S engines and had been designated as an S-76A+. The change to Arriel 1S1 engines resulted in a change to the helicopter designation, to an S-76A++. The helicopter was originally equipped with an electrically activated rotor brake system. A manually activated rotor brake system replaced the electrically activated system under the provisions of Sikorsky Special Service Instruction No. 76-28 Revision A.

2.0 HELICOPTER EXAMINATION

On January 18, 2018, representatives from the NTSB, FAA, Sikorsky, Safran, and Global Sky Air Charter convened at a Monroe County Sheriff's Office hangar at MTH to document the accident helicopter. The main rotor blades, transmission fairing, both engines' inlet barrier filters, rotor brake calipers, and the rotor brake disk shield had been previously removed. Both engines remained installed on the airframe. According to the operator, the main rotor blades were removed prior to relocation of the helicopter to the hangar. Additionally, the No. 2 transmission input shaft was disconnected from the main

gearbox. Photographs that documented the helicopter before and after removal of the transmission fairing and engine inlet barrier filters were provided to NTSB investigators.

2.1 NO. 1 ENGINE OBSERVATIONS (S/N 15014)

The engine exterior did not exhibit evidence of thermal damage. The compressor turned freely when manually rotated. The No. 1 transmission input shaft turned freely in the freewheeling direction when manually rotated. Turning the No. 1 transmission input shaft in the driving direction resulted in rotation of the main rotor head. The painted exterior surfaces of the No. 1 transmission input shaft exhibited a shiny appearance (**Figure 3**). Both electric chip detectors exhibited no evidence of debris. The reduction gearbox magnetic plug did not exhibit ferrous debris. The accessory gearbox magnetic plug could not be removed. A check of the rigging between the No. 1 engine control lever (ECL) in the cockpit and the fuel control unit (FCU) for the No. 1 engine revealed no anomalies. When the ECL was set to idle, the FCU throttle indicated a position of about 27 degrees.

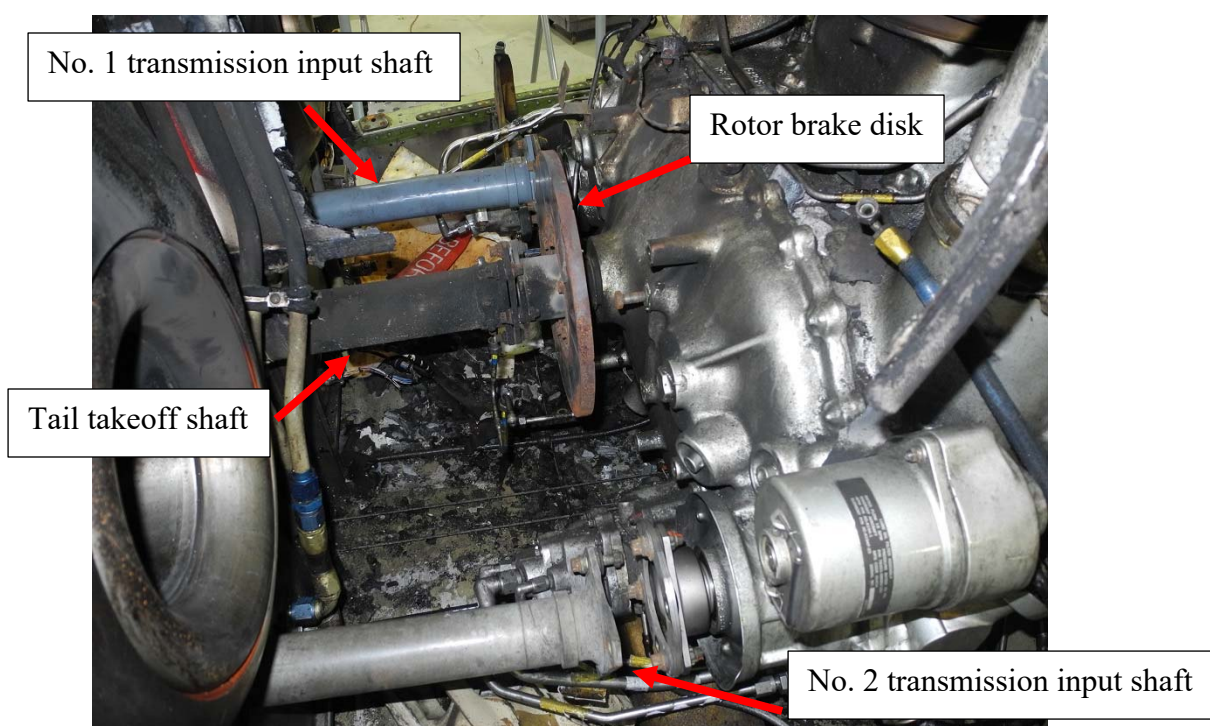


Figure 3. The transmission input shafts, rotor brake disk, and tail takeoff shaft.

The first stage compressor did not exhibit evidence of soft or hard body foreign object debris (FOD) damage.⁵ The free (power) turbine was observed through the engine exhaust and exhibited no evidence of blade shedding. A borescope inspection of the first and second stage gas generator turbines revealed no evidence of anomalous damage. The No. 1 engine was retained for further examination.

⁵ Hard body impact damage is characterized by a serrated appearance and deep cuts or tears to the blade leading and trailing edges. Hard body impact damage can result from impact with metal parts, concrete, asphalt, and rocks. Soft body impact damage is characterized by a large radius of curvature or curling deformation to the blade and can cause curling deformation of the blades in the direction opposite of normal rotation. Soft body impact damage can result from impacts with pliable objects such as birds, ice, tire rubber, and plastic objects.

2.2 NO. 2 ENGINE OBSERVATIONS (S/N 3029)

The engine exterior did not exhibit evidence of thermal damage. The compressor and the No. 2 transmission input shaft could not be manually rotated in either the freewheeling or the driving direction. The exterior surfaces of the No. 2 transmission input shaft exhibited a matted appearance consistent with exposure to fire. Both electric chip detectors exhibited no evidence of debris. The reduction gearbox and accessory gearbox magnetic plugs did not exhibit ferrous debris. A check of the rigging between the No. 1 ECL and the FCU for the No. 2 engine revealed no anomalies. When the ECL was set to idle, the FCU throttle indicated a position of about 27 degrees.

The first stage compressor did not exhibit evidence of soft or hard body FOD damage. The power turbine was observed through the engine exhaust and exhibited no evidence of blade shedding. A borescope inspection of the first and second stage gas generator turbines revealed damage to the turbine blades. The No. 2 engine was retained for further examination.

2.3 ROTOR BRAKE OBSERVATIONS

The rotor brake disk remained installed on the tail takeoff pinion. Raised material was observed on the rotor brake disk surface. The raised material had the appearance of brake pad material fused to the rotor brake disk surface. Soot deposits were observed on the exterior of the No. 1 tail rotor drive shaft. A hydraulic fluid level was not present when observing the sight gauge for the rotor brake accumulator.

Hydraulic pressure was applied to both rotor brake calipers using a manual handheld hydraulic pump. When about 200 pounds per square inch (psi) was applied, the pistons extended and no leaks were observed. The rotor brake disk, calipers, accumulator, rotor brake lever, and the hydraulic lines between the accumulator and calipers were retained for further examination.

2.4 MISCELLANEOUS OBSERVATIONS

Soot deposits were observed on the pitch change links, dampers, and pitch horns for the 'black' and 'red' main rotor blades.⁶ The upper surfaces of the engine cowling exhibited two areas of heavy soot deposits; the heavy soot deposits were aligned with the space between the Nos. 1 and 2 oil cooler ducts and the main gearbox oil cooler duct. The left and right intake fairing, engine cowling, the internal ducting for both engine oil coolers, and the rotor brake cowl exhibited thermal damage.

The airframe oil supply lines from the oil tank were intact for both the No. 1 and No. 2 engines. The airframe oil return lines to the oil tanks were thermally damaged and fractured for both the No. 1 and No. 2 engines. When the No. 1 engine oil filler cap was removed, there was no visible evidence of oil near the filler. When the No. 2 engine oil filler cap was removed, there was visible evidence of an oil level near the filler.

⁶ Each main rotor blade and its associated components are assigned a color for identification purposes. The assigned colors are presented in the order of advancing rotation (when seated in the pilot seat and observing the blades pass from right to left): 'blue', 'red', 'black', and 'yellow'.

3.0 ENGINE EXAMINATION

On February 6-7, 2018, representatives from the NTSB, FAA, Safran, and Sikorsky convened at Safran Helicopter Engines' facilities in Grand Prairie, Texas to examine the engines. Both engines were disassembled and both FCUs were functionally tested.

3.1 NO. 1 ENGINE DISASSEMBLY (S/N 15014)

Removal of the turbine inlet (T5) thermocouples revealed no anomalous damage. The electrical resistance of each of the three T5 thermocouples were measured and were within the acceptable range per Safran specifications. No evidence of cracks or fractures were observed on the external lines removed from the engine. Manual rotation of the power transmission shaft resulted in a corresponding rotation of the power turbine with no evidence of binding. The reduction gearbox was removed; the splined connection between the power turbine and the reduction gearbox exhibited no anomalous damage. The reduction gearbox splined nut index mark was present and did not exhibit misalignment.⁷ All power turbine blades were present and none exhibited evidence of anomalous damage. The power turbine rotated freely with no evidence of binding. The power turbine inlet stator vanes did not exhibit anomalous damage (**Figure 4**). Soot deposits were observed on the inner diameter wall, adjacent to the tip end of the stator vanes.



Figure 4. Power turbine inlet stator vanes for the No. 1 engine.

The first stage (axial) compressor exhibited a black coloration but did not exhibit anomalous damage. The first stage compressor was manually rotated and the gas generator turbine rotated freely with no evidence of binding. An even coating of soot was observed on both the second stage (centrifugal) compressor and the interior surfaces of its cover. The combustion liner exhibited no evidence of anomalous damage or debris.

⁷ Within the reduction gearbox is a splined nut which secures the input gear and transfers drive from the muff coupling to the input gear. The nut is indexed by a final torque (index) mark, also referenced as a “slippage” mark. The amount of “slippage” (or amount of overtorque) can be referenced by the quantity and direction the index mark on the splined nut is offset from the index mark on the input gear.

All first stage and second stage gas generator turbine blades were present and none exhibited anomalous damage (**Figures 5 and 6**). The curvic coupling between the first and second stage gas generator turbines exhibited no anomalous damage. The first and second stage nozzle guide vanes exhibited no evidence of anomalous damage. Soot deposits exhibiting a “wavy” pattern was observed on the first stage nozzle guide vane surfaces. The second stage nozzle guide vane surfaces exhibited abnormal bluing in a “U” shaped pattern.

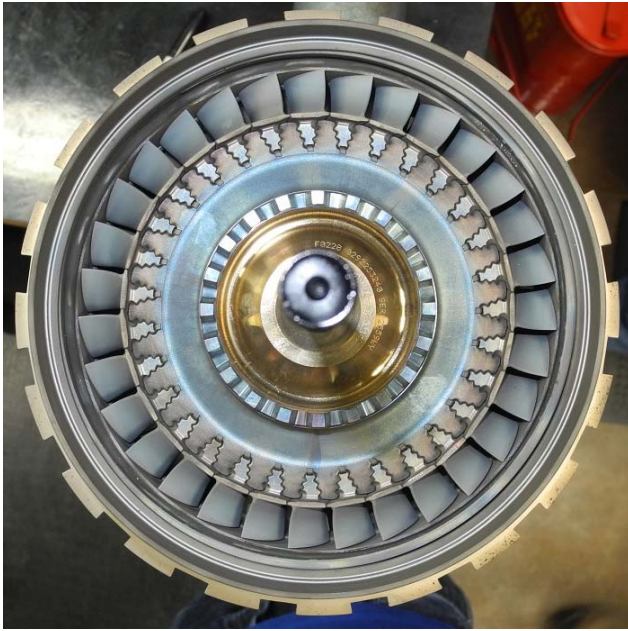


Figure 5. The first stage gas generator turbine blades on the No. 1 engine.

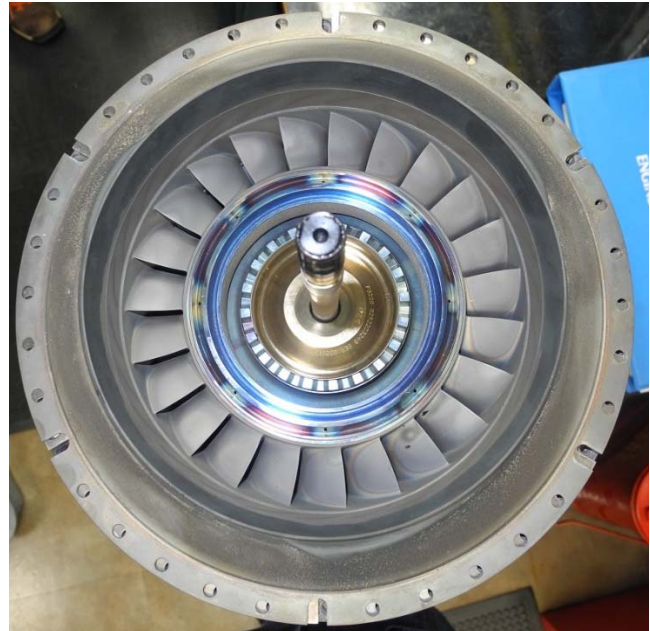


Figure 6. The second stage gas generator turbine blades on the No. 1 engine.

Continuity of drive was confirmed between the accessory drive and the starter-generator pad. All drained oil exhibited a tan coloration. The accessory and reduction gearboxes were not disassembled. The FCU was removed for functional testing.

3.2 NO. 2 ENGINE DISASSEMBLY (S/N 3029)

The power transmission shaft and the gas generator were not able to be manually rotated. Removal of the T5 thermocouples revealed the two of the three thermocouples, left and right, were partially melted. The majority of the third, center, thermocouple was missing and exhibited thermal damage. The reduction gearbox was removed and the gearbox and power transmission shaft rotated freely with no evidence of binding; the power turbine still could not be manually rotated. The reduction gearbox splined nut index mark was present and did not exhibit misalignment. The overspeed sensor and muff coupling were removed and did not exhibit anomalous damage. After the power turbine section was removed, the power turbine was able to be manually rotated. The power turbine bearings exhibited a chattering noise when manually rotated. The turbine inlet stator vanes had a dull gray coating that was flaking off its surfaces (**Figure 7**).



Figure 7. Damage observed on the turbine inlet stator vanes for the No. 2 engine.

The gas generator was removed from the engine assembly and the axial compressor was removed from the gas generator. The axial compressor was able to be manually rotated but its rotation was not smooth. Soot deposits were observed on the axial compressor surfaces, which also exhibited a black coloration. Soot deposits were also observed on the centrifugal compressor surfaces. Removal of the centrifugal compressor revealed a shiny, silver-colored rub mark near the 6 o'clock position on the interior surface of the centrifugal compressor cover (**Figure 8**). A dull, silver-colored marking was observed on the exterior surface of the centrifugal compressor cover near the 6 o'clock position, corresponding in location to the marking on the interior of the cover. Shiny, silver-colored rub marks were observed on the exducer of the centrifugal compressor blades matching the location of the rub marks observed on the interior surface of the cover (**Figure 9**).

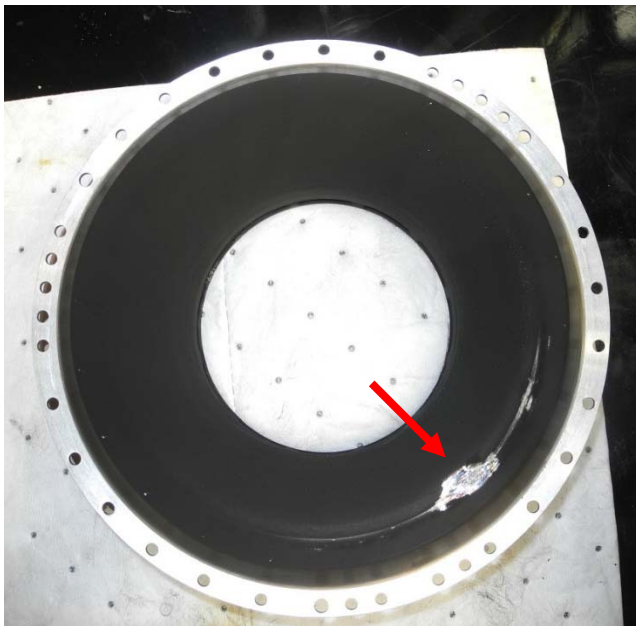


Figure 8. Rub mark on the centrifugal compressor cover of the No. 2 engine.

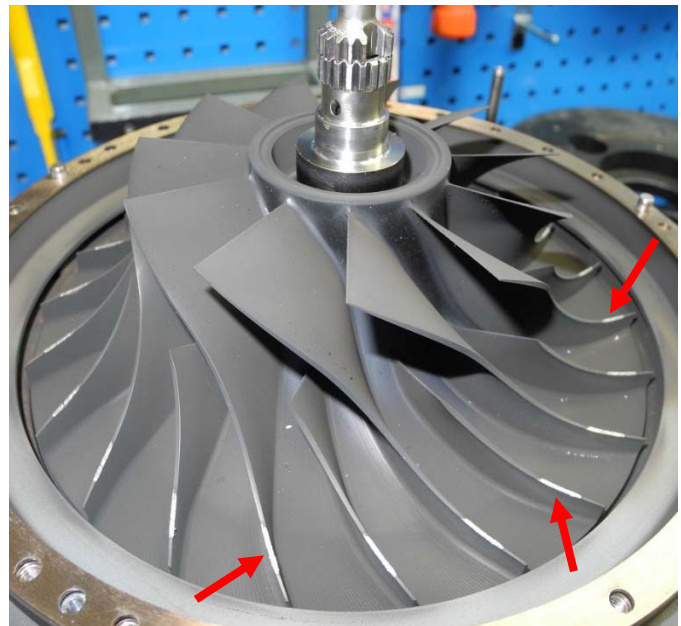


Figure 9. Rub marks on the centrifugal compressor of the No. 2 engine.

All second stage gas generator turbine blades were present but their tips were missing (**Figure 10**). The surface coating of the second stage nozzle guide vane surfaces exhibited flaking with a couple of the vanes' mid-section exhibiting partial melting (**Figure 11**). All first stage gas generator turbine blades were present and their surface coating was missing at their tip ends, with the missing coating being biased toward the trailing edge of the blades (**Figure 12**). The first stage gas generator turbine blades were contacting the shroud near the 1 o'clock position. The first stage nozzle guide vanes exhibited a brown coloration but did not exhibit evidence of melting. The curvic coupling between the first and second stage gas generator turbines exhibited no anomalous damage. The combustor liner contained debris similar to the flaked coating observed on the turbine inlet guide vanes.



Figure 10. The No. 2 engine second stage gas generator turbine blades exhibiting missing tips.



Figure 11. A close up showing partial melting of the second stage nozzle guide vanes for the No. 2 engine.



Figure 12. Damage to the first stage gas generator turbine blades.

Continuity of drive was confirmed between the accessory drive and the starter-generator pad. All drained oil exhibited a tan coloration. The accessory and reduction gearboxes were not disassembled. The FCU was removed for functional testing.

3.3 NO. 1 ENGINE FCU (S/N B828B)

The No. 1 engine FCU was installed on a test bench to simulate operation of the FCU when installed on an engine. A normal start procedure was initiated and the FCU was brought to flight (≥ 52 degrees throttle position) with no anomalous response from the FCU. When the FCU throttle was set to 27 degrees, Np shaft speed remained at 0 revolutions per minute (rpm) and the fuel flow was at about 40.2 liters per hour (lph).⁸ A load, simulated from the test bench, was placed on the Np drivetrain and manipulated to assess the FCU anticipator⁹ response; the anticipator responded normally.

3.4 NO. 2 ENGINE FCU (S/N B926B)

The No. 2 engine FCU was installed on a test bench to simulate operation of the FCU when installed on an engine. A normal start procedure was initiated and the FCU was brought to flight (≥ 52 degrees throttle position) with no anomalous response from the FCU. When the FCU throttle was set to 27 degrees, within the idle range, Np shaft speed remained at 0 rpm and the fuel flow was at about 39.9 lph.⁸

A load, simulated from the test bench, was placed on the Np drivetrain. Fuel flow rate was about 43 lph, Ng shaft speed was about 3,000 rpm, and Np shaft speed was less than <15 rpm. The FCU throttle position was increased from 27 degrees to 52 degrees and the fuel flow rate increased to a peak of about 280 lph. A corresponding increase of Ng and Np shaft speeds were observed. Ng shaft speed increased to and stayed at about 5,100 rpm; Np shaft speed peaked to about 1,150 rpm before reducing down to 0 rpm.

The threshold for Np and Ng governing on the FCU was found to be about 48 degrees throttle position. Manipulation of the throttle lever revealed the FCU responded normally when in the governed range and Np shaft speed remained at 0 rpm (with load) when the throttle was set below 48 degrees.

4.0 ROTOR BRAKE COMPONENT EXAMINATION

On June 19, 2018, representatives from the NTSB, FAA, and Sikorsky convened at Sikorsky's facilities in Trumbull, Connecticut to examine the rotor brake calipers, accumulator, and master cylinder. Aircraft Braking Systems Corp. component maintenance manual (CMM) No. 32-43-42 was referenced for this examination.

⁸ The Np, Ng, and fuel flow results were unique to the test bench and were not values expected to be read on the cockpit instrument gauges in the accident helicopter.

⁹ The FCU contains an anticipator, linked to the main rotor collective flight control, to increase and decrease fuel flow with increases and decreases in collective input, respectively.

4.1 RIGHT BRAKE CALIPER ASSEMBLY (S/N 00023R)

Both brake pucks remained retracted at the beginning of the examination. Air was bled from the caliper using about 10 psi of hydraulic pressure via a manual hydraulic pump. For the first test, about 230 psi¹⁰ of hydraulic pressure was applied to the caliper, resulting in extension of both brake pucks. A slight hydraulic leak was observed at the forward seal subassembly. Hydraulic pressure was removed and the aft puck retracted; it was not known if the forward puck retracted as it was not observed by the group. For the second test, hydraulic pressure of 230 psi was applied to the caliper. The aft puck extended. When hydraulic pressure was removed, the aft puck retracted, but the forward puck did not retract and remained extended. There was no evidence of hydraulic leaks at the forward seal subassembly. A third test was performed with hydraulic pressure of 230 psi applied to the caliper. The forward puck did not move, but the aft puck extended. Removal of hydraulic pressure resulted in retraction of the aft puck, but not of the forward puck. The caliper was drained of hydraulic fluid and the forward puck was manually retracted without difficulty. Air was bled from the caliper using about 10 psi of hydraulic pressure, but the forward puck extended during the bleeding process. Testing of the right brake caliper was stopped at this point.

The caliper was disassembled into its two halves, identified as the forward and aft housings. The transfer tube remained with the aft housing and its o-ring appeared to be in good condition. The forward and aft puck assemblies were disassembled from their respective housings (**Figures 13 and 14**). There was no evidence of grooves or gouges between the piston and bore. Residual hydraulic fluid was observed within the bore and on the piston surfaces. The piston o-ring had no evidence of anomalous damage and remained elastic to the touch. The spring and pin were disassembled from the forward and aft puck assemblies and revealed no evidence of anomalous damage. The aft spring force was measured to be about 120 pounds at 0.476 inches of spring compression.¹¹ The forward spring force was measured to be about 125 pounds at 0.476 inches of compression.



Figure 13. The forward caliper housing.
(photo courtesy of Sikorsky)

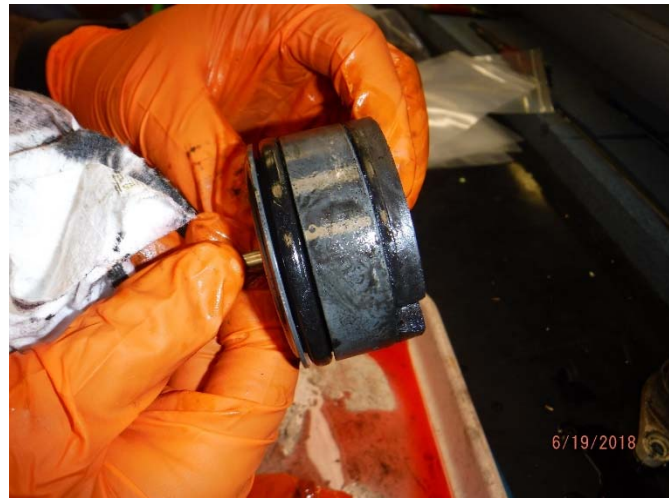


Figure 14. The puck from the forward caliper.
(photo courtesy of Sikorsky)

¹⁰ The master cylinder hand pump produces hydraulic pressure of about 230 psi on the helicopter rotor brake system.

¹¹ CMM No. 32-43-42 requires the forward and aft springs to exhibit a force of 108-132 pounds when the spring is compressed to 0.476 inches.

4.2 LEFT BRAKE CALIPER ASSEMBLY (S/N 00029)

Both brake pucks remained retracted at the beginning of the examination. Air was bled from the caliper using about 10 psi of hydraulic pressure via a manual hydraulic pump (**Figure 15**). During the bleeding process, the forward puck extended but the aft puck did not extend. The forward puck was able to be manually moved to the retracted position without difficulty. For the first test, about 230 psi of hydraulic pressure was applied to the caliper, resulting in extension of both pucks. No evidence of leaks was observed. When hydraulic pressure was removed, the aft puck retracted but the forward puck remained extended. The forward puck was manually retracted without difficulty. For the second test, as the manual hydraulic pump was increasing in hydraulic pressure, the forward puck extended at about 46 psi and the aft puck extended at about 130 psi. There was no evidence of leaks when hydraulic pressure of 230 psi was attained. When hydraulic pressure was removed, the aft puck retracted but the forward puck remained extended. Testing of the left brake caliper assembly was stopped at this point.

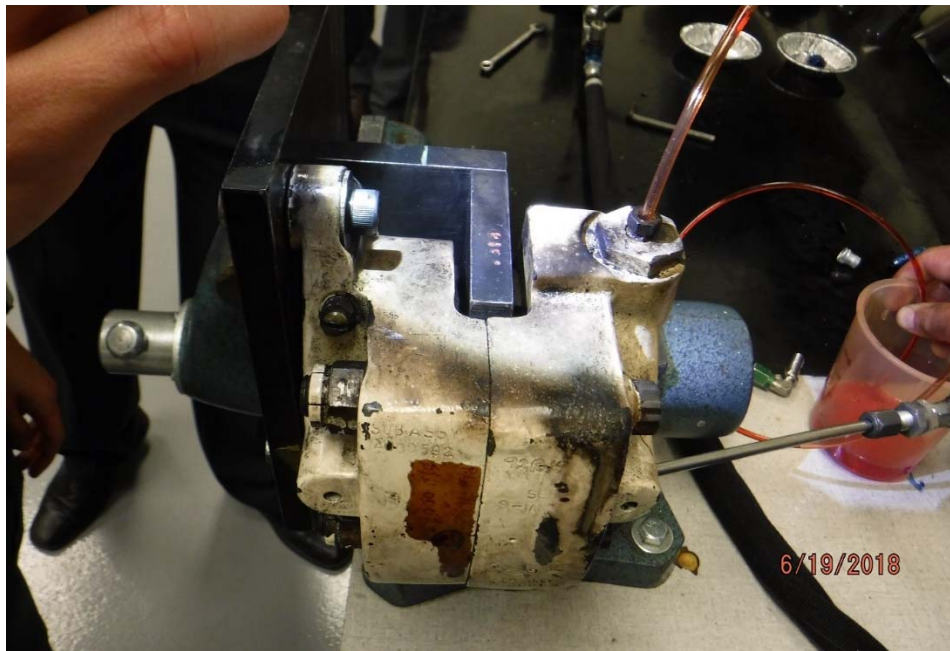


Figure 15. The left brake caliper assembly mounted on the test setup.

The caliper was disassembled into its two halves. The transfer tube remained with the aft housing and its o-ring appeared to be in good condition. The forward and aft puck assemblies were disassembled from their respective housings. There was no evidence of grooves or gouges between the piston and bore. Residual hydraulic fluid was observed within the bore and on the piston surfaces. The piston o-ring exhibited no evidence of anomalous damage and remained elastic to the touch. The spring and pin were disassembled from the forward and aft puck assemblies and revealed no evidence of anomalous damage. The aft spring force was measured to be about 125 pounds at 0.476 inches of spring compression. The forward spring force was measured to be about 110 pounds at 0.476 inches of spring compression.

4.3 ACCUMULATOR

The accumulator was set up on a test bench and hydraulic pressure applied to the accumulator. Hydraulic pressure, read from a gauge, increased to about 297 psi but would not go higher as pressure was increased.¹² A switch on the accumulator was functionally checked using a multimeter.¹³ With no hydraulic pressure on the accumulator, the multimeter showed the switch was in the opened position. After application of about 12 psi of hydraulic pressure to the accumulator, the multimeter showed the switch had closed.

4.4 MASTER CYLINDER

A hydraulic supply line was installed from the retained accumulator to the master cylinder. The accumulator was filled with hydraulic fluid to a level between the middle and top of the accumulator sight glass. A pressure gauge was installed on the output side of the master cylinder. When the master cylinder lever was pushed forward (actuated), there was an increase in the indicated gauge pressure. The indicated gauge pressure decreased as the lever was pulled aft (to the stowed position), but pressure could be built up with multiple strokes of the lever. When the lever was pushed forward and pressure increased on the gauge, the pressure relief button was pressed with no effect on the indicated gauge pressure.

5.0 OTHER FINDINGS

S-76A rotorcraft flight manual (RFM) No. SA 4047-76-1, found in the cockpit of the accident helicopter, contained rotorcraft flight manual supplement (RFMS) No. 29B at revision No. 5 that was applicable to helicopters equipped with Arriel 1S1 engines installed in accordance with STC No. SH568NE. RFMS No. 29B contained rotor brake limitations for when the rotor was either stopped or turning. The rotor turning limitation stated in part:

Rotor brake application limited to one (or two) engine(s) operating at idle or both engines shut down.

Maximum rotor speed for normal rotor brake application is 65% Nr.

A decal on the rotor brake lever of the accident helicopter contained a caution that stated the following:

Normal rotor brake stops are authorized only at 65% NR or less with both engines shut off or one engine at idle.

Chihoon Shin
Aerospace Engineer – Helicopters

¹² A Sikorsky engineering specification for the accumulator requires a cracking pressure of 300±30 psi.

¹³ When there is >10 psi of hydraulic pressure in the accumulator, the switch closes, resulting in illumination of the rotor brake caution light in the cockpit. When hydraulic pressure is <7 psi, the switch opens, resulting in extinguishing of the rotor brake caution light.