National Transportation Safety Board
Office of Research and Engineering
Washington, D.C. 20594

Performance Study

Specialist Report
Marie Moler

A. ACCIDENT

Location:	Italy, Texas
Date:	July 6, 2016
Time:	1148 CDT
Airplane:	Bell 525, N525TA
NTSB Number:	DCA16FA199

B. GROUP

The Vehicle Performance group members were:

Chairman:	Marie Moler National Transportation Safety Board Washington, DC
Member:	Albert Brand Bell Helicopter Fort Worth, Texas
Member:	Eric Kinney Federal Aviation Administration Fort Worth, Texas

C. SUMMARY

On July 6, 2016, about 1148 central daylight time (CDT), an experimental Bell 525 helicopter, N525TA, broke up inflight and impacted terrain near Italy, Texas. The two pilots onboard were fatally injured and the helicopter was destroyed. The flight originated from Arlington, Texas, as a developmental flight test and was conducted under the provisions of 14 Code of Federal Regulations Part 91. Visual meteorological conditions prevailed at the time of the accident.

The objective of this study is to discuss the sequence of events leading up to the crash of N525TA. The *Aircraft Flight Path and Breakup* section will review the recorded data and describe the sequence of events leading to the inflight breakup. The subsequent section *Aircraft Response* will discuss the vibrational characteristics of the helicopter from an Aircraft Performance standpoint and will reference the Human Performance and Operations reports to explore the potential for the biomechanical response as a factor in the accident.

D. PERFORMANCE STUDY

The aircraft was equipped with a flight telemetry system which recorded flight data on the aircraft and streamed it to the test crew monitoring the flight from the ground. The aircraft was well instrumented as it was a flight test article. The lowest data collection rate was 31.25 Hz (pressure altitude, speed, e.g.) and the highest was 1000 Hz (rotor damper axial forces, hydraulic pressures, e.g.). The telemetry data that was used to analyze the accident is discussed in detail in the Flight Data Factual Report [1].

Data was recorded continuously throughout the day and then divided into identifying records for each test point performed. When the pilot initiated a new record, the test point timer started from zero. The aircraft was on test record 51 when the accident occurred, indicating it was the 51st flight test point on the day of the accident. Record 51 runs for 21.18 seconds. Two prior test points (records 50 and 48), completed shortly before the accident flight test, are also discussed in this report.

Weather Observations

METARS from Hillsboro Municipal Airport (KINJ) were recorded at 1136 CDT and 1156 CDT. The temperature was reported as 31°C (87°F) with a dewpoint of 23°C (73°F) and an altimeter setting of 29.98 inHg. Visibility was 10 statute miles with scattered clouds at 3,000 ft. Winds at 1136 CDT were reported at 16 kts from 190°, gusting to 22 kts. Winds at 1156 CDT were reported at 14 kts from 170°, gusting to 22 kts. KINJ is approximately 13 NM (15 statute miles) from the wreckage location.

Aircraft Flight Path and Breakup

Figure 1 shows the aircraft's flight path for record 51 with locations based on Global Positioning System (GPS) satellite data. Each point is annotated with the time shown as test time in seconds, and with corresponding height above ground in feet. The test record began at time zero, in the lower right of the figure. The aircraft's flight path was along a track of 320°. The telemetry system stopped recording data just after the 21 second mark. The location of the last verified data point is shown as 'last point' in Figure 1.

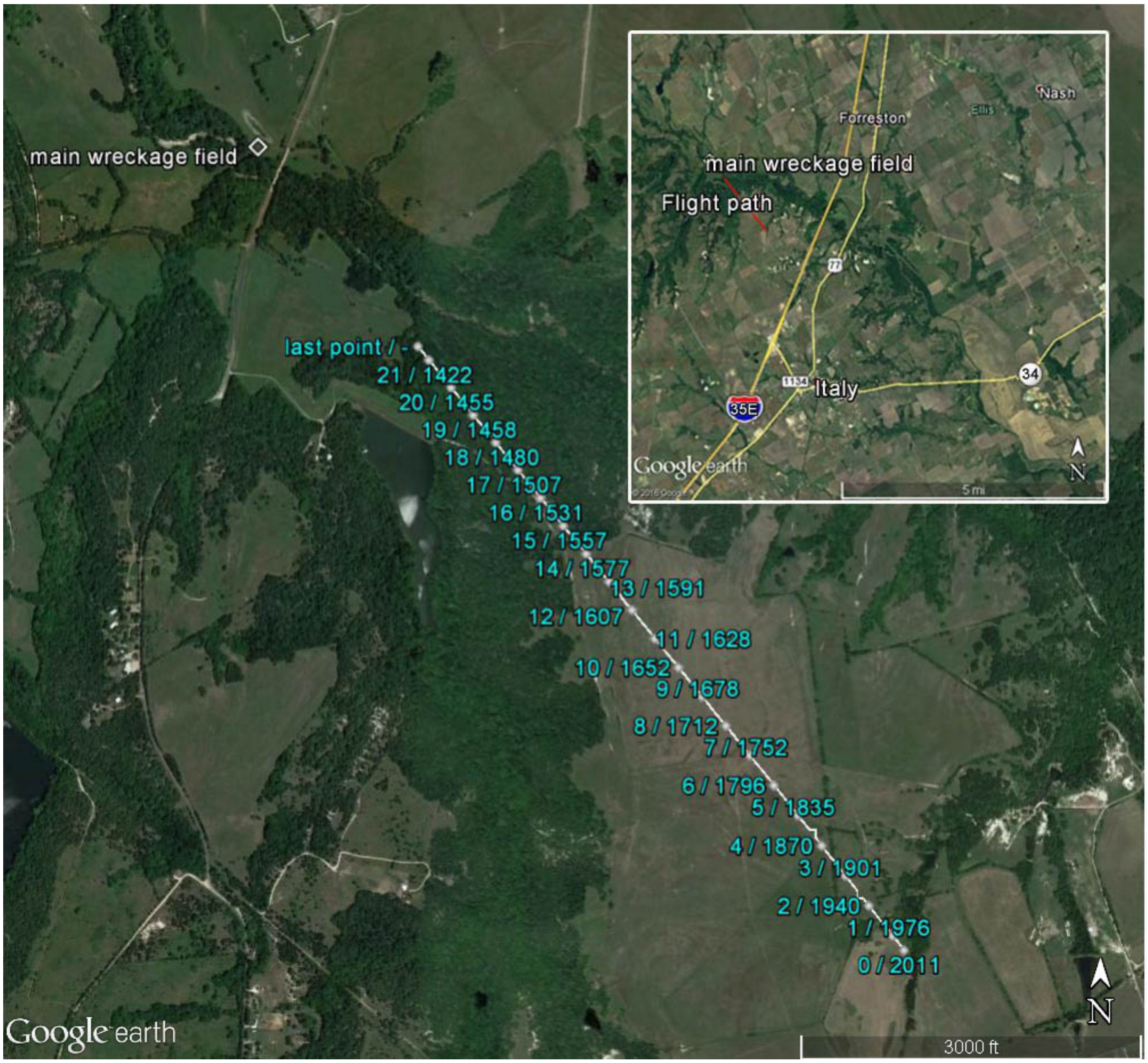

Figure 1. Aircraft flight path and main wreckage location. The last point occurs at 21.18s.

The main wreckage field was 2,200 ft from the last transmitted GPS point, along the flight path heading of 320° and included the main body of the aircraft, engines, transmission and the roots of the blue and green main rotor blades[1]. A debris field (Figure 2) extended between the last data point and the main wreckage covering approximately 4,000 ft (north to south) by 1,700 ft (east to west). The debris field indicates an inflight breakup of the aircraft that was confirmed by witness statements from the pilots of the chase helicopter. The tail boom and the orange, white, and red main rotor blades were in the larger debris field while the root ends of the blue and green blades were with the main wreckage of the aircraft.

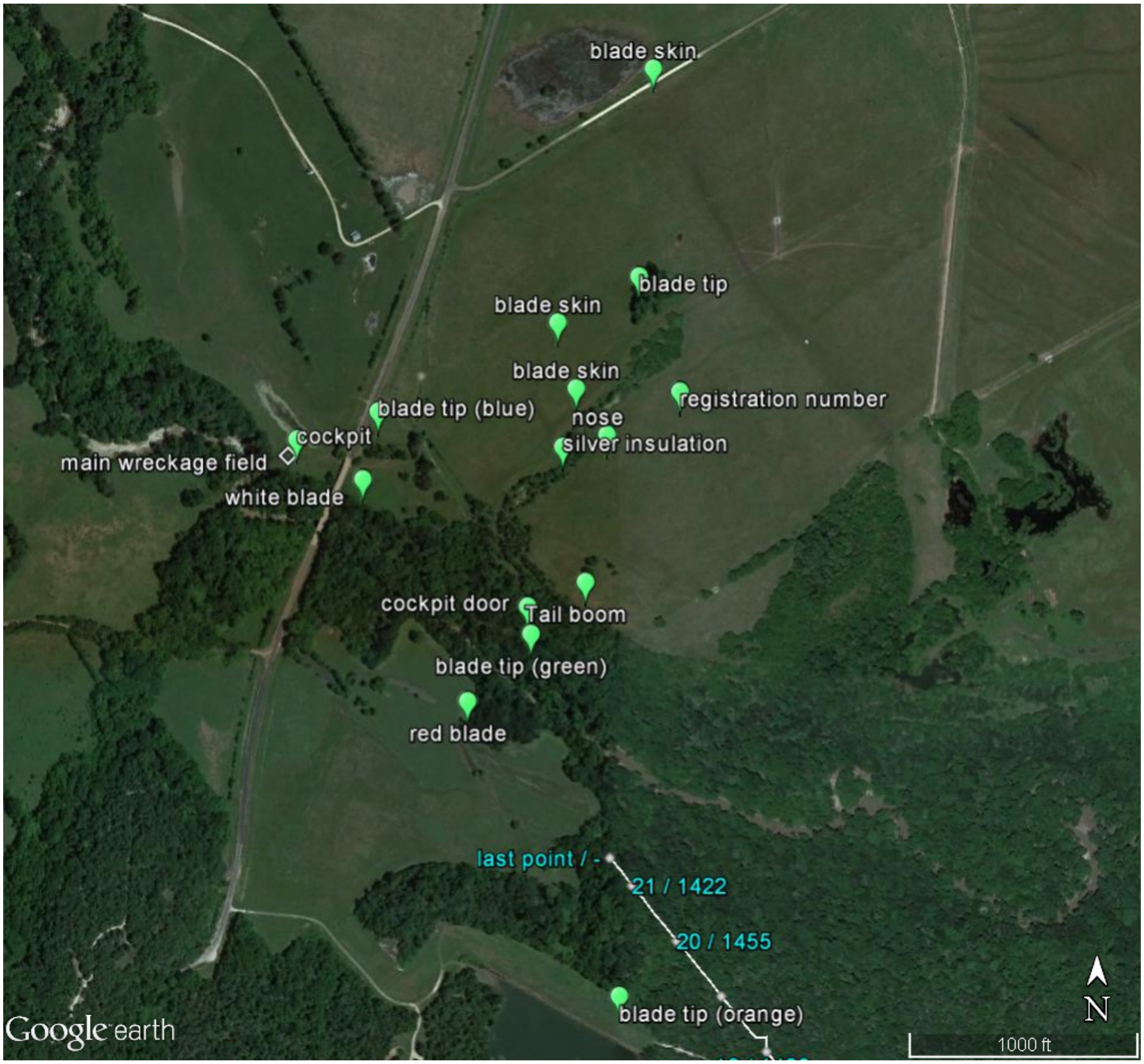

Figure 2. Aircraft flight path and main wreckage field.

The test record being performed was a one engine inoperative (OEI) test at 185 kts and a heavy gross weight and forward center of gravity (cg) configuration. 185 kts had been identified as a potential aircraft 'do not exceed' speed or V_{NE}[2]. The nine test records prior to record 51 were also

[1] The five blades were labeled by color in sequence of rotation – blue, orange, red, green, white.
[2] The do not exceed speed is "potential" because the aircraft was not yet certified.

OEI records where airspeed was increased in 10 kts increments. Figure 3 shows the airspeed, altitude, and heading of accident flight and two prior successful OEI tests[3]. The entry conditions for this test point had been previously evaluated with regard to the aero-servo-elasticity stability (ASE). ASE refers to the dynamic stability of the airframe, rotors, and control system in flight.

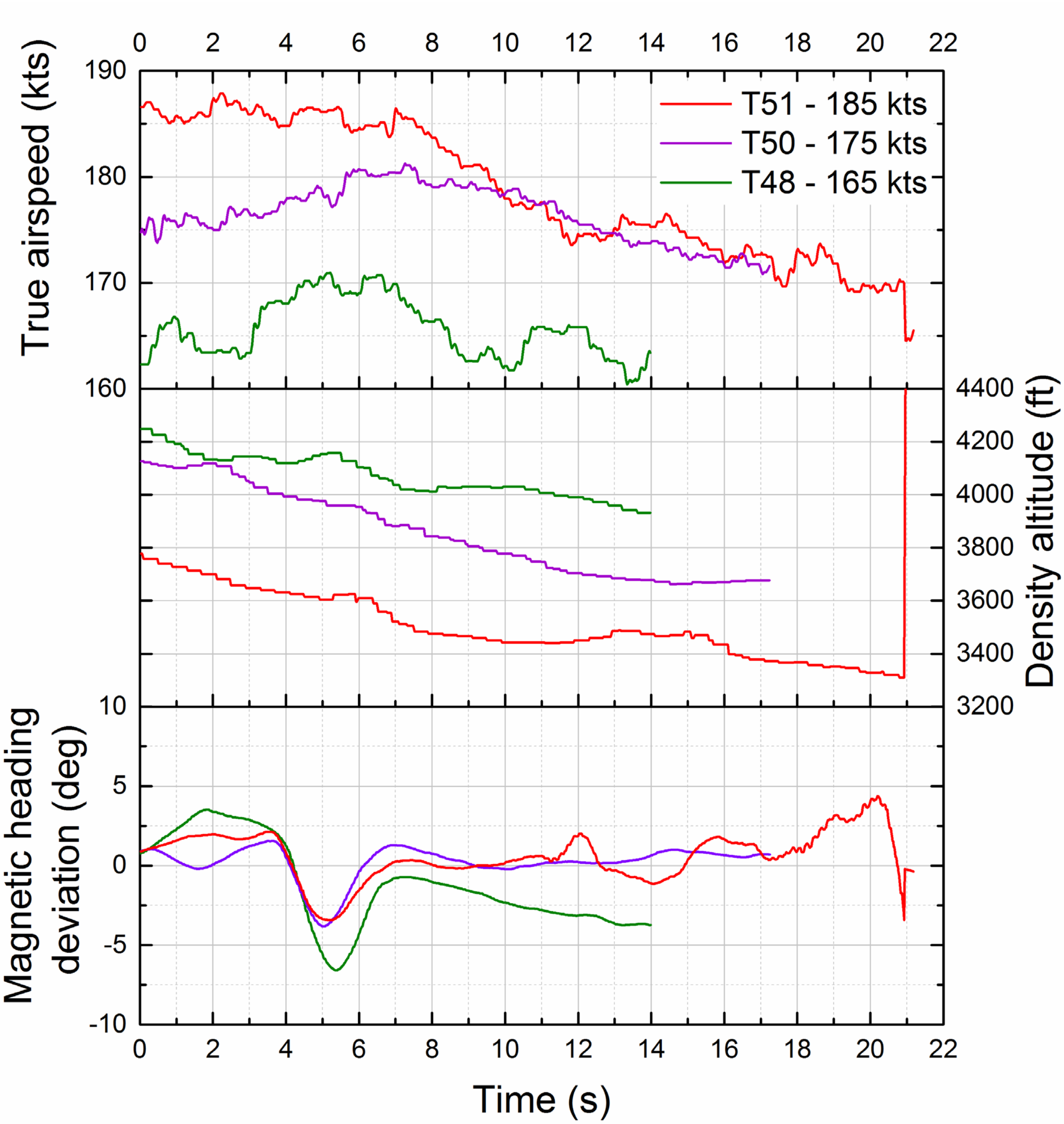

Figure 3. Accident flight and prior two successful OEI test records. In each test, the OEI condition was initiated at about the 4 second mark.

[3] Record 49 was not a successful OEI record and was aborted because of two engine torque spikes typical of wind gust encounters.

The OEI procedure calls for the pilots to capture the desired airspeed, start the data record, and to select OEI training mode on the Garmin Touch Control panel that interfaces with the General Electric engine control software. The General Electric engine control software would then reduce the power to both engines simulating the transient and steady state response of a one engine inoperative situation. Power required is a function of the flight condition as reflected in the rotor torque and rotor rotational speed.

If the engine power available is insufficient to maintain the flight condition a reduction in rotor rotational speed (reported in revolutions per minute or RPM, or as a percentage of Nr)[4] will occur. As rotor speed decreases, main rotor blades can increasingly flap out of plane. For the V_{NE} airspeed of 185 kts, a single engine is insufficient to maintain the flight condition. Therefore, management of the rotor Nr is critical to recovering from the loss of an engine. Pilot response at high speed is to lower the collective to reduce torque on the rotor and/or to pull the longitudinal cyclic back to reduce the airspeed. Both actions result in reducing the power required by the main rotor and allowing Nr to recover. Once rotor speed has recovered to the target value of 103% Nr, the test point would be considered complete.

Figure 4 shows the two prior successful OEI test records at 175 kts airspeed (labeled T50 for record 50) and 165 kts airspeed (labeled T48 for record 48). For both tests, the main rotor Nr degraded from 100% at about 3.5 to 4 s, consistent with initiation of the OEI test. Collective was reduced to 51% for T50 and 43% for T48. The Nr stopped decreasing at around 90% before recovering by ten seconds to 97% for T50 and nearly 100% for T48 before continuing to rise to the 103% completion point. The time from Nr decay to initial Nr recovery for both tests was between two and three seconds.

[4] Helicopter parameters and control settings are often specified or reported in percentages. The pilot collective stick which varies the mean pitch angle for all blades, is recorded as a value from 0% (resulting in minimum blade pitch) to 100+% (resulting in maximum blade pitch). The pilot cyclic stick varies the cyclic pitch angle of the blades and thus the tilt of the rotor disk and aircraft angular (pitch and roll) response. The longitudinal cyclic stick varies from 0% (resulting in maximum aftward tilt of the rotor disk) to 100% (resulting in maximum forward tilt of the rotor disk). Lateral cyclic stick varies from 0% (resulting in maximum left tilt of the rotor disk) to 100% (resulting in maximum right tilt of the rotor disk). Rotor speed is described both in terms of actual RPMs and a percentage of normal operating rpm. Normal operating rpm is 100% Nr.

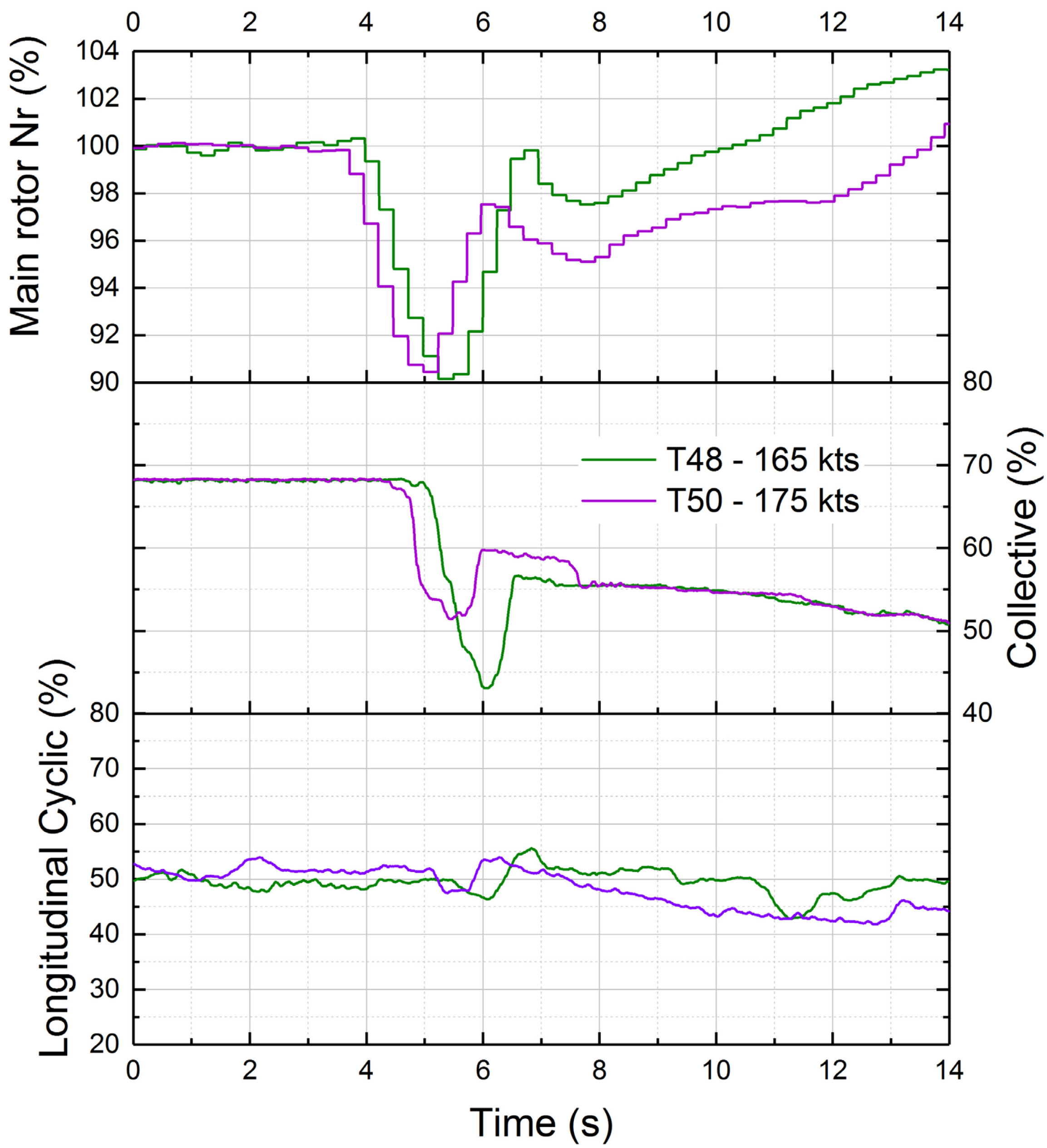

Figure 4. Prior two successful OEI test records.

For comparison, Figure 5 shows the same Nr, collective, and longitudinal cyclic parameters for the accident record compared to the prior tests. For record 51, Nr began to decay at about 3.5 s, similar to the prior tests. Collective was reduced to 58% and the Nr stabilized near 92% but did not return to 97% or higher as in the previous tests. After 6 s, higher frequency components can be seen in the collective and longitudinal cyclic inputs that were not present in the earlier tests. After 7 s, the Nr stopped recovering and by 18 s, it had decreased to below 80%.

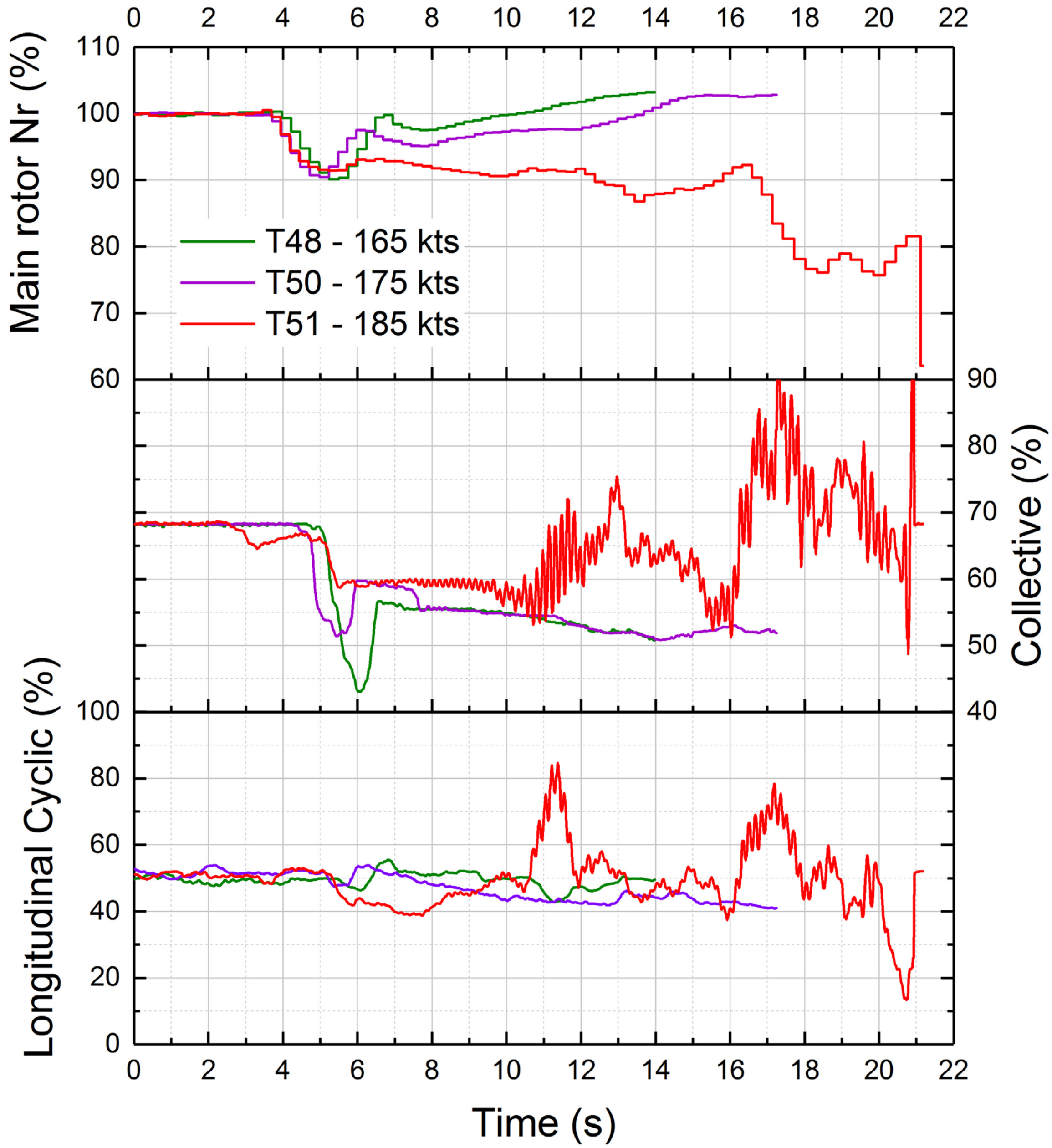

Figure 5. Accident Nr and control input with earlier flight test records.

Figure 6 shows the control inputs from the collective and cyclic and the aircraft's response and orientation during the accident test point. When collective was increased between 10 and 13 s and again between 16 and 17.5 s, the main rotor rotational speed slowed. The middle plot shows lateral cyclic and roll. After 10 s, cyclic input activity increased as did the roll response. The bottom plot shows longitudinal stick and pitch. The aircraft's roll and pitch responded to cyclic input throughout the accident flight.

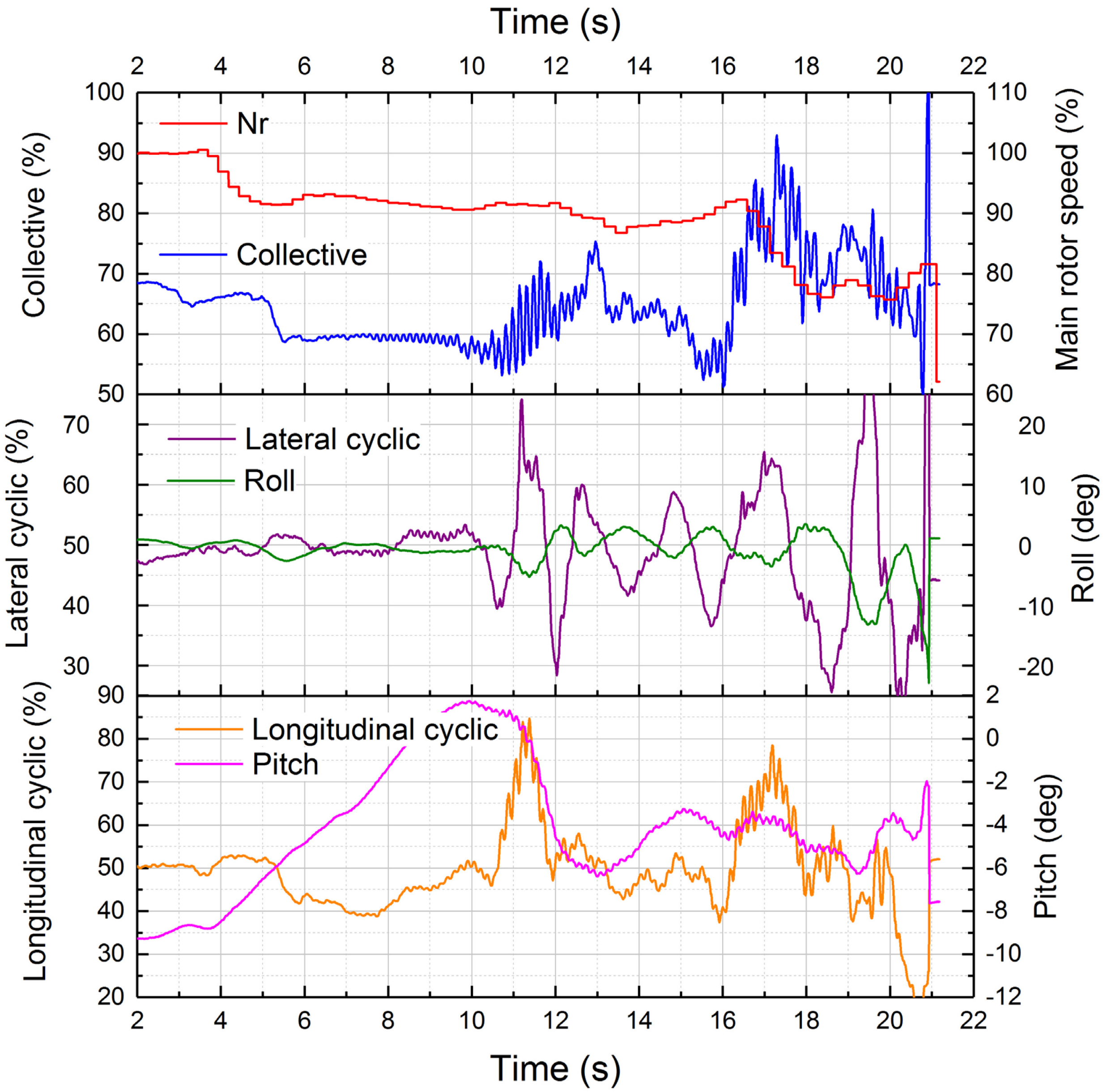

Figure 6. Control input and aircraft response during the accident test record.

Increasing the collective increases the torque load on the main rotor. However, in the high speed flight with OEI mode active, the engine power is insufficient for the load and therefore the main rotor rotational speed decreases. The red blade was instrumented to record flapping and the blade's out of plane flapping motion is shown in Figure 7. As the rotational speed decreased after 16 s, the out of plane flapping motion increased. At approximately 20.7 s, a large aft cyclic input reached peak value. At 20.4 s, the string potentiometer that measured blade flapping motion stopped functioning due to excessive flapping. Within two rotations of losing the flapping signal, one or more of the main rotor blades severed the tail boom from the aircraft and all data recording and telemetry ended. This sequence of events is discussed in greater detail in the Airworthiness Factual report [2].

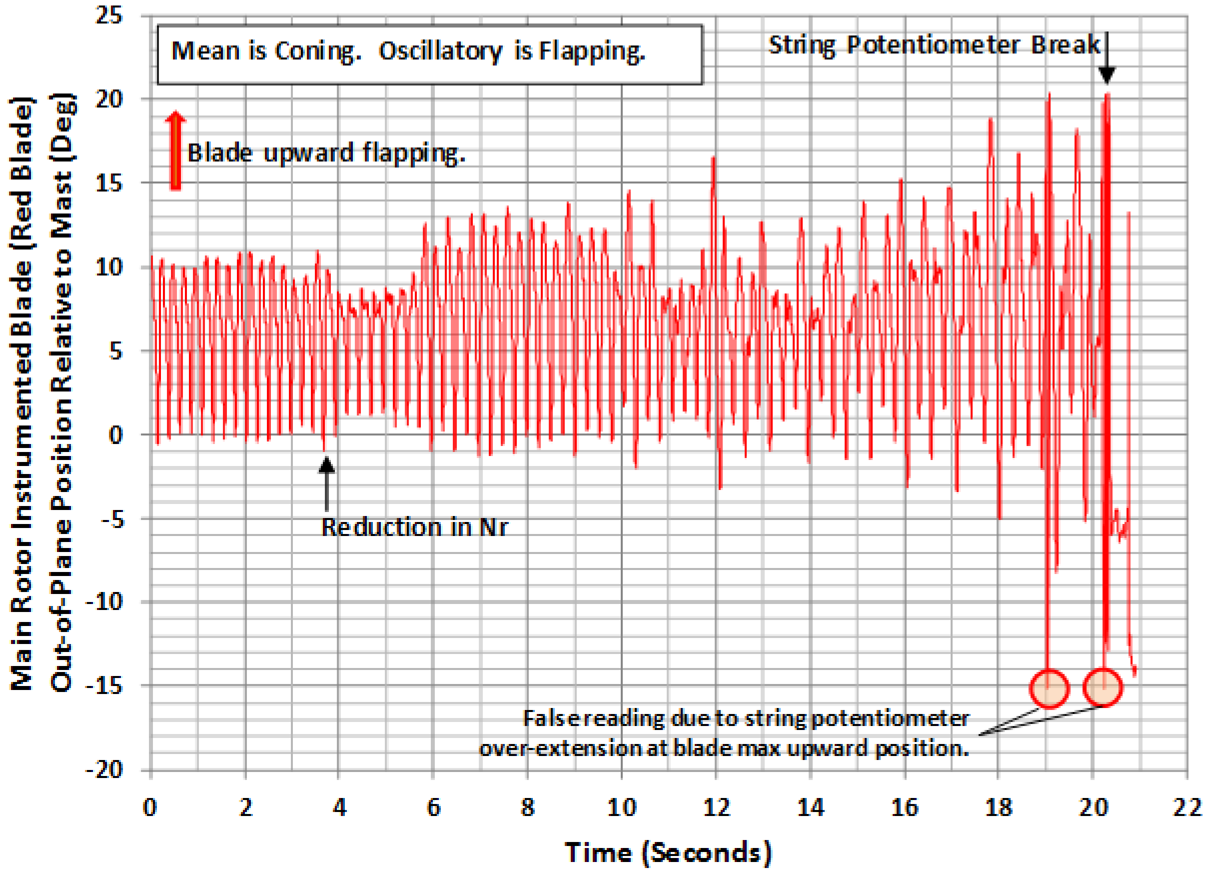

Figure 7. Red blade out-of-plane position during the accident test record. Plot courtesy of Bell Helicopter.

In conclusion, the inflight breakup was initiated when, during an OEI test flight at high forward speed, rotor rotational speed decayed below 80%, which allowed significant main rotor blade flapping. Twenty-one seconds after record 51 began, flapping main rotor blades severed the tail boom from the aircraft.

Aircraft Response

Introduction of 6 Hz Vibration

Figure 5 shows an oscillation in the collective and cyclic control inputs during the accident record, sometime after 6 s. The oscillation was not present during the previous test records. The oscillation indicates a vibration in the structure and controls near 6 Hz. This vibration, which will be referred to as 'the 6 Hz' vibration throughout this report, was not a single mode of vibration at exactly 6 Hz throughout the flight. The frequency of the vibration changed slightly through the record as rotor speed changed and various airframe and rotor modes were excited. The 6 Hz vibration was introduced at a very low amplitude and it became apparent in many data channels sometime between 6 and 8 s. This 6 Hz vibration was distinctive, grew in amplitude, and affected the entire helicopter and the crew.

Figure 8 shows the recorded tail rotor torque and tail rotor Nr (the tail rotor is mechanically linked to the main rotor so tail rotor Nr tracks main rotor Nr). After 7 s, the vibration was well defined and the amplitude began to grow. At 10 s, the amplitude grew before decaying again after 12 s. This growth was described as a 'blossom' in the vibration and the term will be used throughout this study. The appearance of the 6 Hz torque oscillation corresponded with a rotor Nr of about 92%. Since the collective was oscillating at 6-Hz, the control laws would send a corresponding (6-Hz) command to the tail rotor as anti-torque compensation.

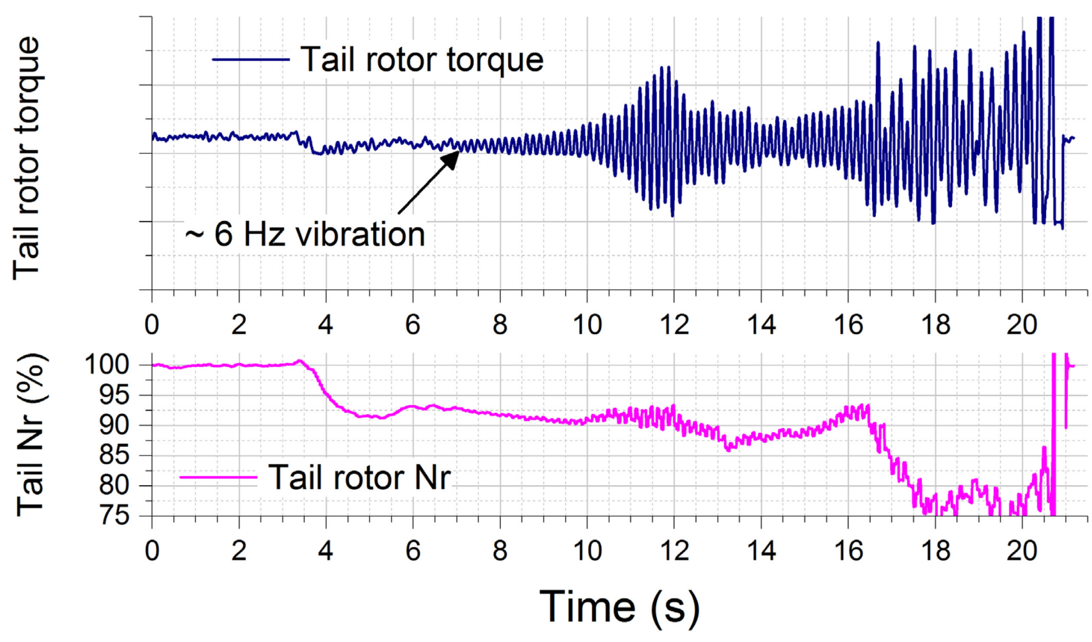

Figure 8. Tail rotor torque and tail rotor Nr during the accident test record.

The torque on the main rotor is shown in Figure 9. The filtered sub-plot is used to better differentiate the 6 Hz vibration, and highlights the vibration at 10 s.

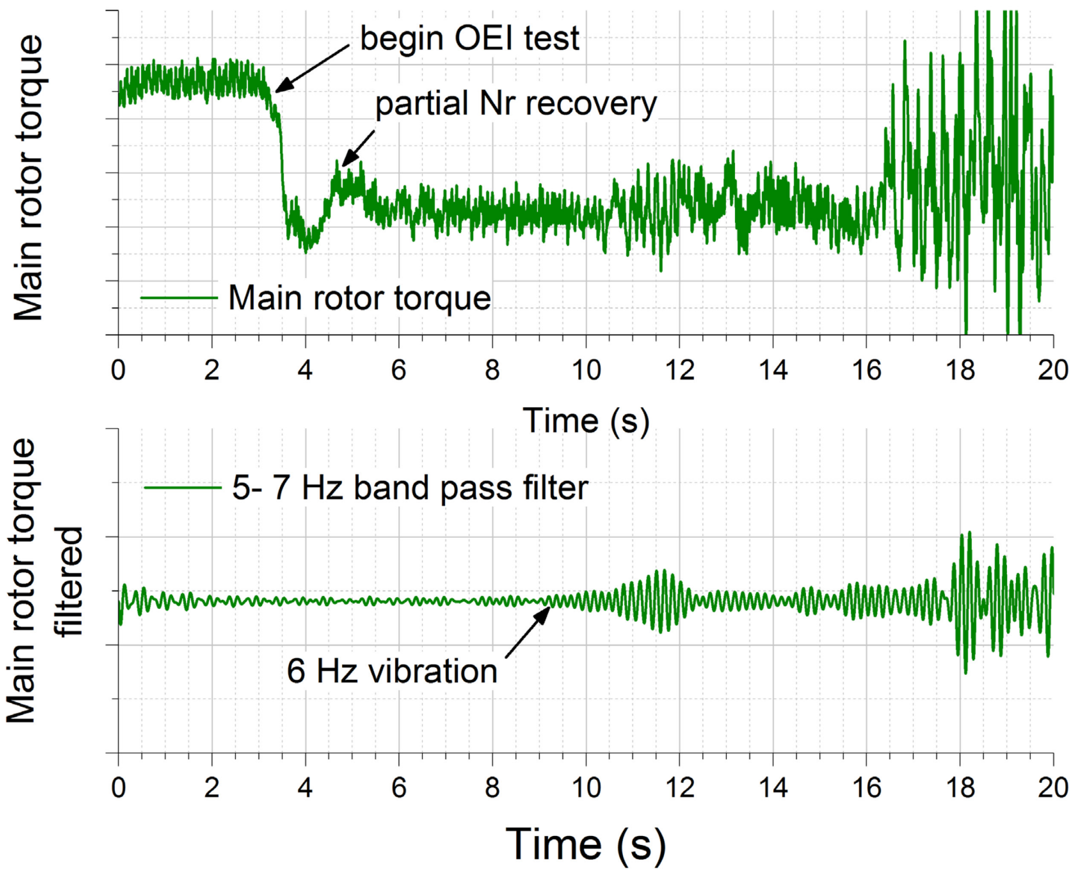

Figure 9. Main rotor torque during the accident test record.

Figure 10 shows main rotor Nr and pilot seat vertical acceleration[5]. As the rotor speed stayed near 92%, the 6 Hz vibration grew in amplitude from around 7 s until 11 s. At 11 s the Nr slowed below 90% and the 6 Hz amplitude decreased. At 13.5 s, the Nr began to increase from 86%. As the Nr again approached 92% the 6 Hz amplitude increased. The amplitude of the vertical oscillation was near ± 2.5 g at 6 Hz around 11 s and again after 16 s. For comparison, earlier test records showed variations in vertical acceleration no greater than ± 0.3 g. The 6 Hz vibration appeared in the control inputs, especially the collective, starting before 7 s (Figure 11). The effects of this vibration on the pilots is discussed in the Human Performance Report [3].

[5] Vertical and lateral accelerations were recorded for the pilot's seat. Longitudinal acceleration was not.

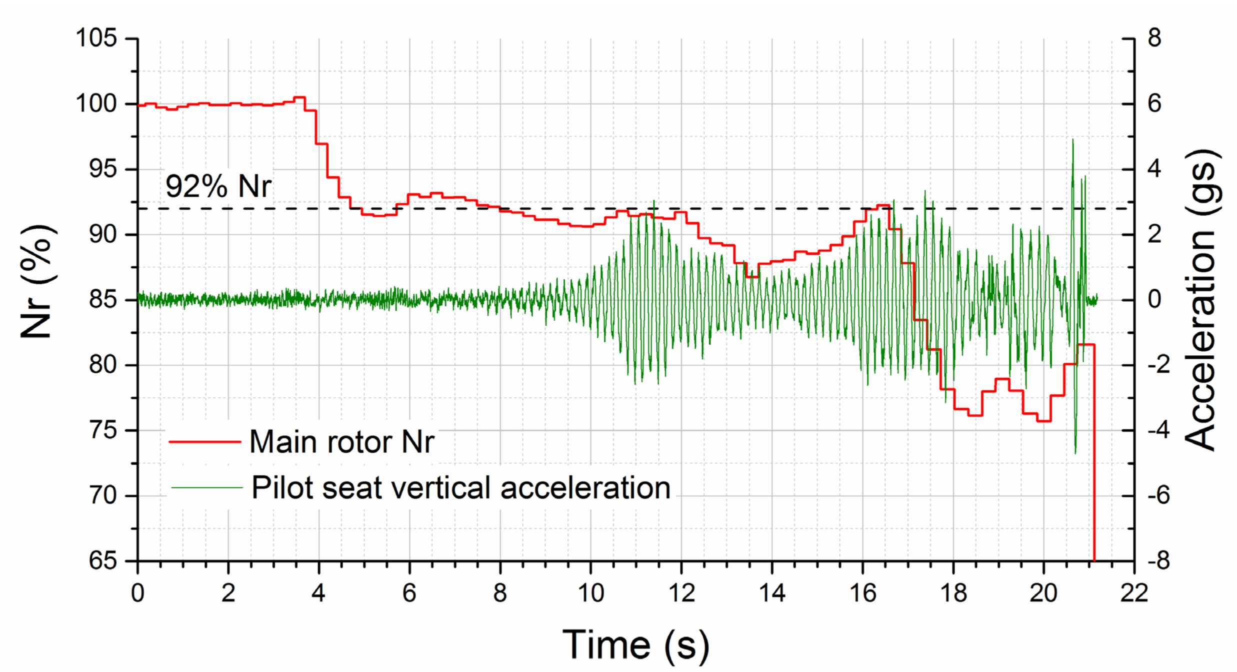

Figure 10. Main rotor Nr and pilot seat vertical acceleration during the accident test record.

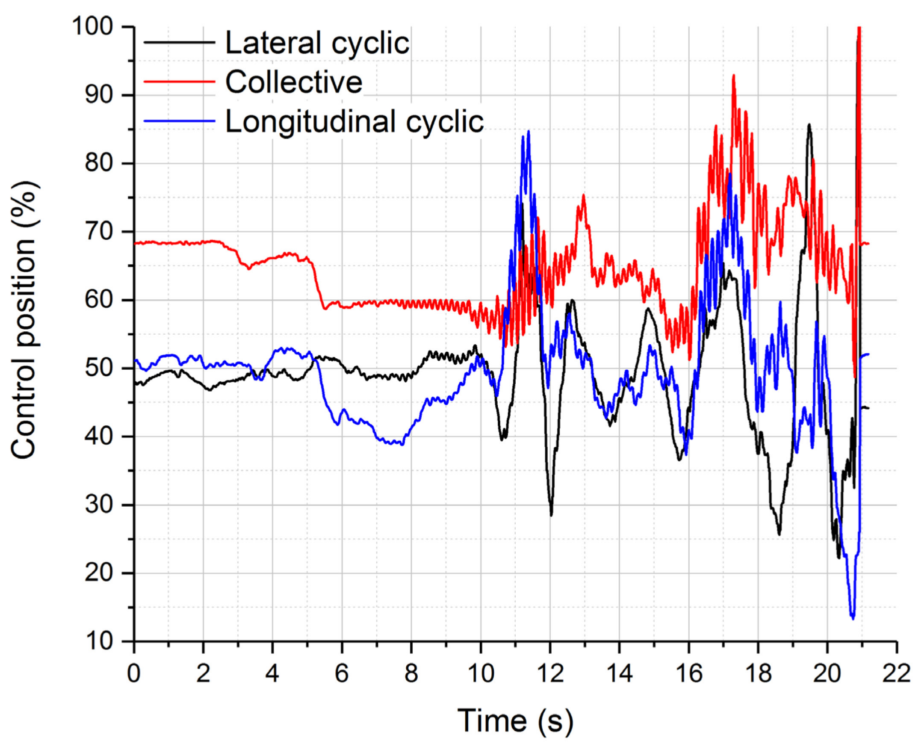

Figure 11. Control inputs during the accident test record

In summary, a 6 Hz vibration that was not present in the earlier OEI tests manifested sometime after 6 or 7 s into the accident test record. The appearance of the vibration corresponded with a main rotor Nr of 92%. The 6 Hz signal was seen throughout the aircraft – in both rotor systems, the airframe, the pilot seats, and the control inputs.

Source and Development of the 6 Hz Vibration

The focus of much of the investigation was on the source of the 6 Hz vibration. The predicted frequency of the main rotor 1st in-plane scissors mode is 6.8 Hz at 100% Nr and drops to 6.2 Hz at 92% Nr. These values are reported as fixed-system (non-rotating) frequencies. For a rotor mode, this is the frequency at which the rotor hub conveys motion into the fixed system. An illustration of the in-plane scissors mode in the main rotor system is shown in Figure 12. The lead-lag motions of the blades act in such a way that adjacent blades move together and apart in a scissoring motion. The excitation of the scissors mode was confirmed by plotting the relative lag angle between the red and green blades as a function of azimuth. The scissors mode produces a fore-aft motion of the main rotor mast due to aerodynamic forces.

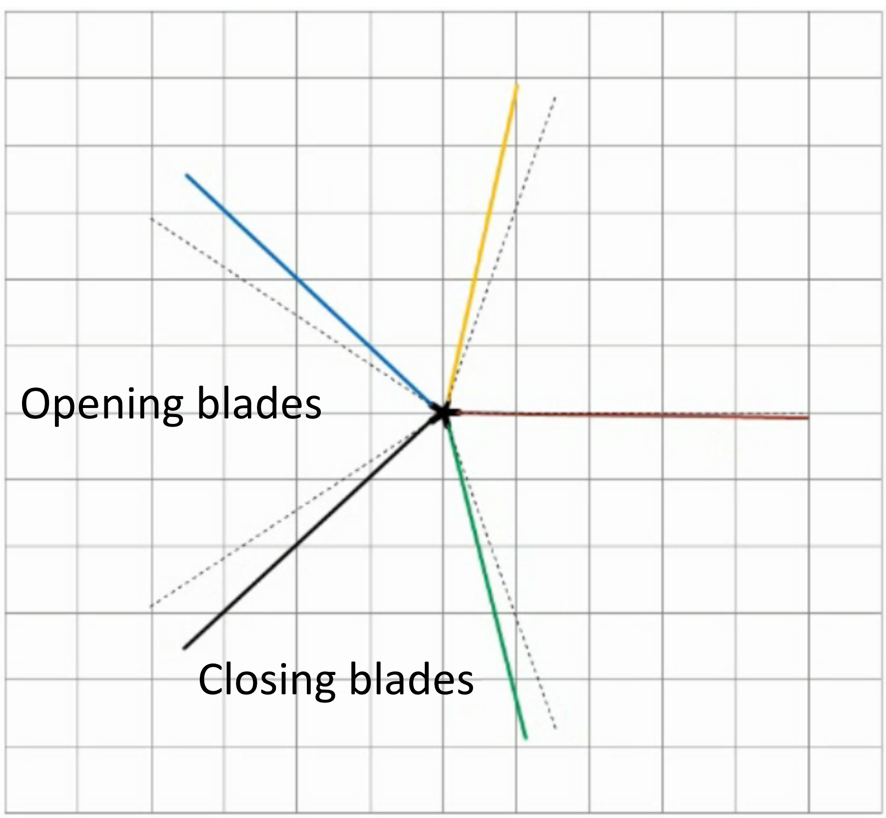

Figure 12. Scissors mode in main rotor system. Illustration courtesy of Bell Helicopter.

The main rotor scissors mode had been encountered in two previous tests. In the prior tests, when the main rotor Nr was 100%, the scissors mode was seen at 6.7 Hz[6]. The tests were at a lower airspeed (145 – 152 kts), but included high blade loading due to increased load factor in a banked turn. In both tests, the vibration quickly damped out as the load factor was reduced. Further testing at 100% Nr demonstrated that the scissors mode was well damped. During the accident flight, the main rotor scissors mode was manifested at a lower frequency in the fixed system due to the lower rotor speed.

Figure 13 shows the frequency of the main rotor scissors mode (upper red line), the first vertical bending mode[7] (black line), the main rotor cyclic regressing mode[8] (lower red line), and the tail rotor cyclic regressing mode (green line). The frequencies of the rotor modes vary with rotor speed. The existence of a mode at a given frequency on this plot is not indicative of vibratory amplitude at that frequency. The cyclic regressing modes of the rotors and vertical bending mode of the fuselage have small amplitude responses prior to the main rotor scissors mode becoming the dominant mode.

The frequency and amplitude of the vertical oscillations of the pilot seat during the accident test flight are shown as a series of colored circles. Each circle is labeled with the test time in seconds (from 2 to 19 s) and the color of the circle reflects the amplitude of the frequency response. The amplitude color scale is shown to the right of the figure.

[6] The frequency of rotor modes can vary slightly as the lead-lag dampers are affected by environmental conditions.
[7] The first vertical bending mode is the lowest frequency mode at which the aircraft fuselage oscillates about its lateral axis (both nose and tail flex up, then both nose and tail flex down relative to the center portion of the fuselage). The 1st vertical bending mode of the aircraft was determined to be 5.4 Hz for the static fuselage by a shake test earlier in the test program.
[8] A cyclic mode occurs when rotor blades lead and lag in such a way that the hub of the rotor begins to orbit about its axis of rotation. The mode is regressing if the time it takes the hub to make one full cycle is slower than one full rotation of the blades around the hub.

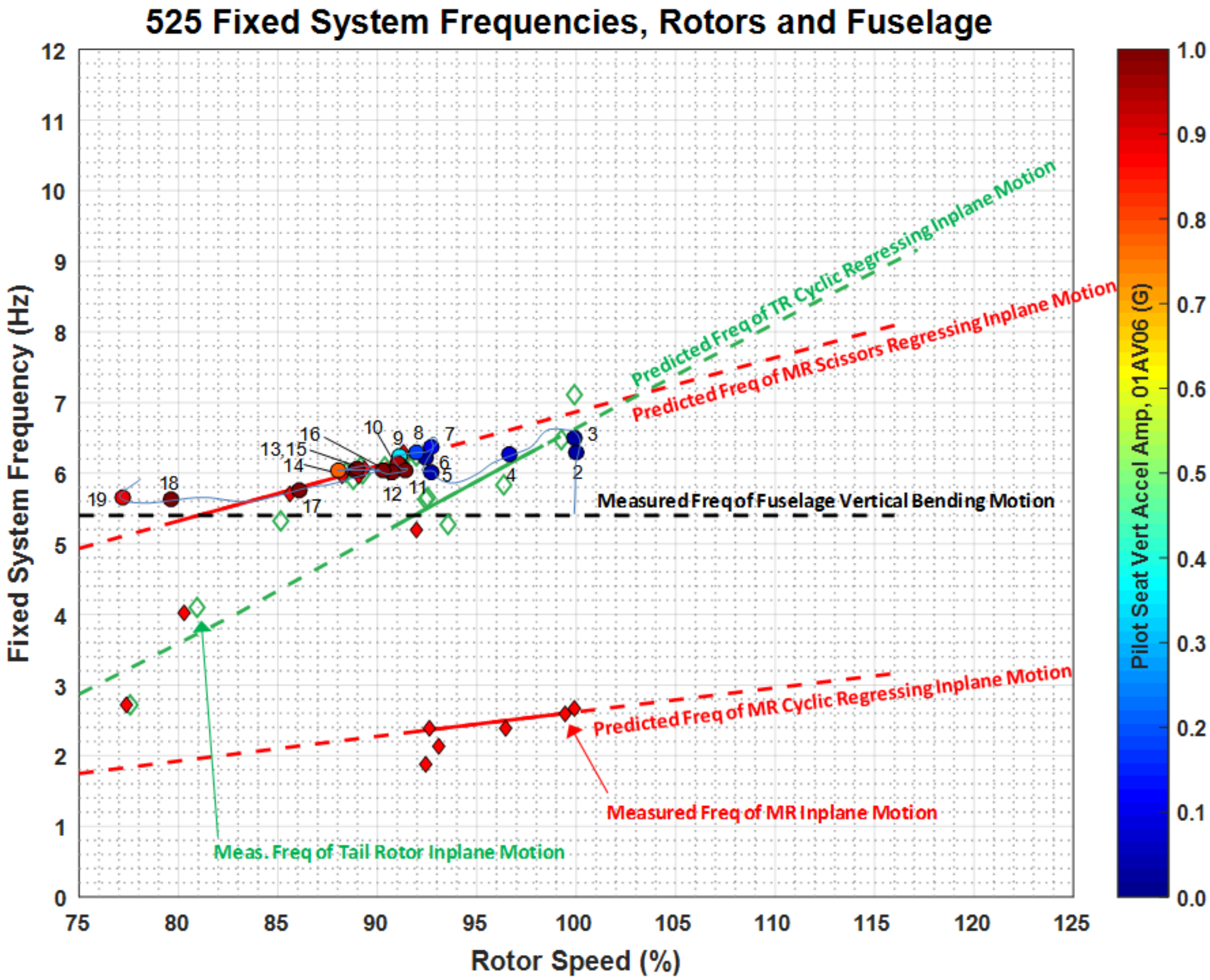

Figure 13. Fixed system frequencies versus rotor Nr. Plot courtesy of Bell Helicopter.

Early in the test record, the frequency of the main and tail rotor systems followed their cyclic regressing modes as shown by the green diamonds (tail rotor in-plane motion) and red diamonds (main rotor in-plane motion). Initially, as the rotor speed was between 100% and 93%, the pilot seat vertical vibration frequency followed the frequency of the tail rotor cyclic regressing mode. The amplitude of this response was quite low: less than 0.2 g's and consistent with prior tests.

As the rotor speed decreased towards 92% Nr, the measured in-plane motion of the main rotor blade shifted from a dominant cyclic regressing mode to a dominant scissors regressing mode (upper red dotted line) and by 6.5 s the pilot seat vertical acceleration responded at the main rotor scissors mode frequency, indicating that the fuselage of the aircraft was responding to the scissors mode. The main rotor's shift to the scissors mode produced a frequency around 6 Hz that began dominating the vibratory signature of the tail rotor, the fuselage, and by 7 seconds (referring to Figure 5) affected the controls.

From 5 s to 11 s, the main rotor speed stayed between 90% and 93% Nr and the amplitude of the pilot seat vertical acceleration increased markedly from less than ±0.1 g to greater than ±1g. After

12 s, a collective control increase (Figure 5) resulted in a further reduction in rotor speed which coincided with a reduction in the pilot seat vibration seen in Figure 10 near the 14 second mark. As the test record continued, the amplitude of the vibration grew again in all channels where it was present.

The tail rotor cyclic regressing mode and the first vertical bending mode of the aircraft (black line) coalesce near 5.4 Hz at 92% Nr. This coalescence coincides with the shift in pilot seat vertical vibration frequency to the main rotor scissors mode frequency. It is possible that the coalescence at 5.4 Hz initiated the main rotor scissors mode response. However, the growth in aircraft vibration at 92% Nr is not a response to the tail rotor or airframe modes (at 5.4 Hz), but is a response to the scissors mode at near 6Hz. Two sources were determined to have increased the amplitude of the aircraft's 6 Hz frequency response:

1. Biomechanical feedback into the collective control
2. Cyclic stir in the swashplate driven by the Attitude Heading Reference System (AHRS)

Determining the separate contributions of the biomechanical feedback and the AHRS to the increase in amplitude was not possible with the flight data, but evidence of their effects will be discussed.

As shown in Figure 11, the 6 Hz vibration also manifested in the control inputs, starting around 6.5 s. The scissors mode through aerodynamics produces a fore-aft motion of the main rotor mast. The airframe responded to this forcing frequency with an oscillatory ~6 Hz vertical motion at the pilot seat.

The evidence for biomechanical feedback is seen in the trace of pilot collective control after 6.5 seconds, which shows the pilot's control stick moving at the 6-Hz scissors mode frequency. The pilot's collective control oscillations result in further amplification of the main rotor scissors mode, further amplifying the vertical seat vibration, and further increasing the collective stick oscillation. The biomechanical feedback loop appears to attenuate after 10 s (and again at 16 s as seen by reductions in the pilot seat vertical vibration and collective control oscillation at those times. The command between the collective input and the actuators was governed by a control law that limited inputs greater than 22°/s. The control law rate limit in the collective control is seen as the difference between the pre-limit and post-limit collective command signal shown in Figure 14. Further discussion of the biomechanical feedback is in the Human Performance Report [3].

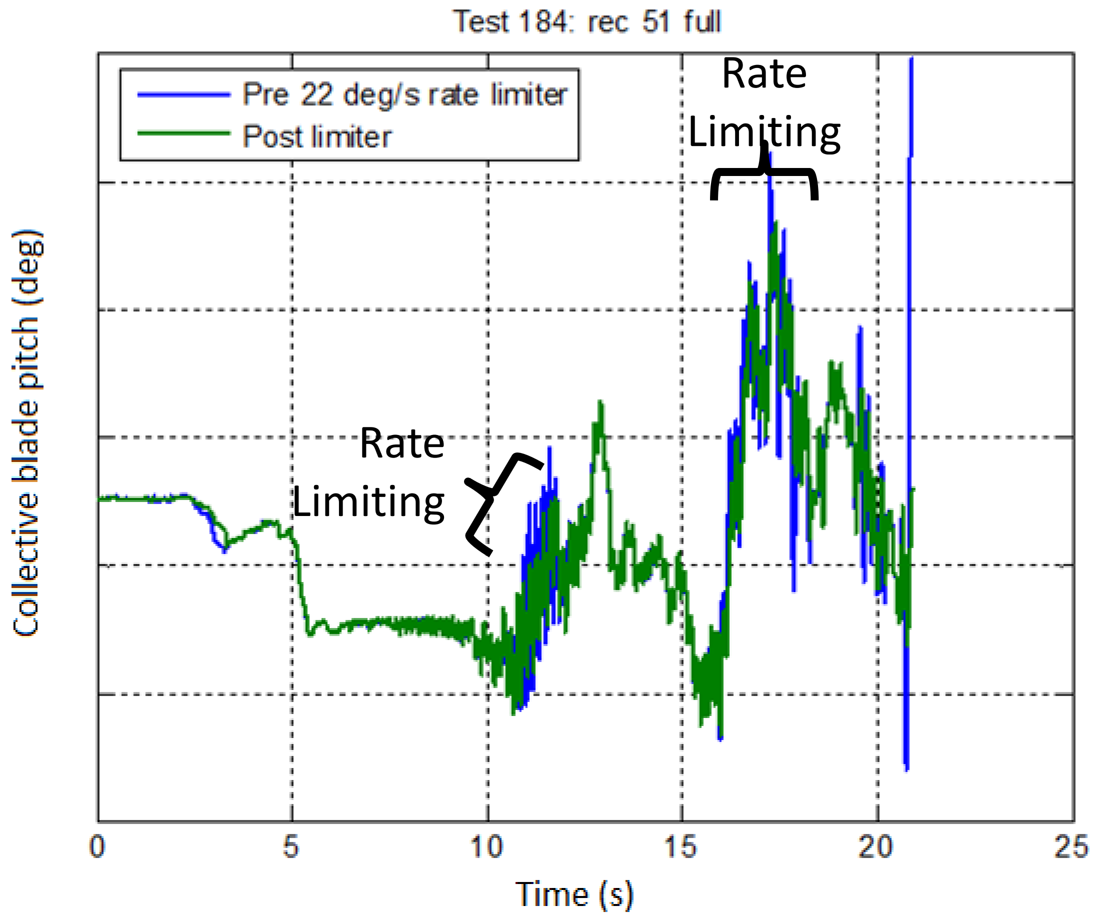

Figure 14. Rate limiting evidence in collective blade pitch angle parameter. Plot courtesy of Bell Helicopter.

The rate limiting evidence correlated with decreases in the amplitude of the 6 Hz vibration at those times throughout the aircraft, Figure 15. However, these reductions in vibration amplitude also coincided with the rotor speed moving away from 92% Nr so the relative contribution of the control law rate limiting influence on the response amplitude could not be determined.

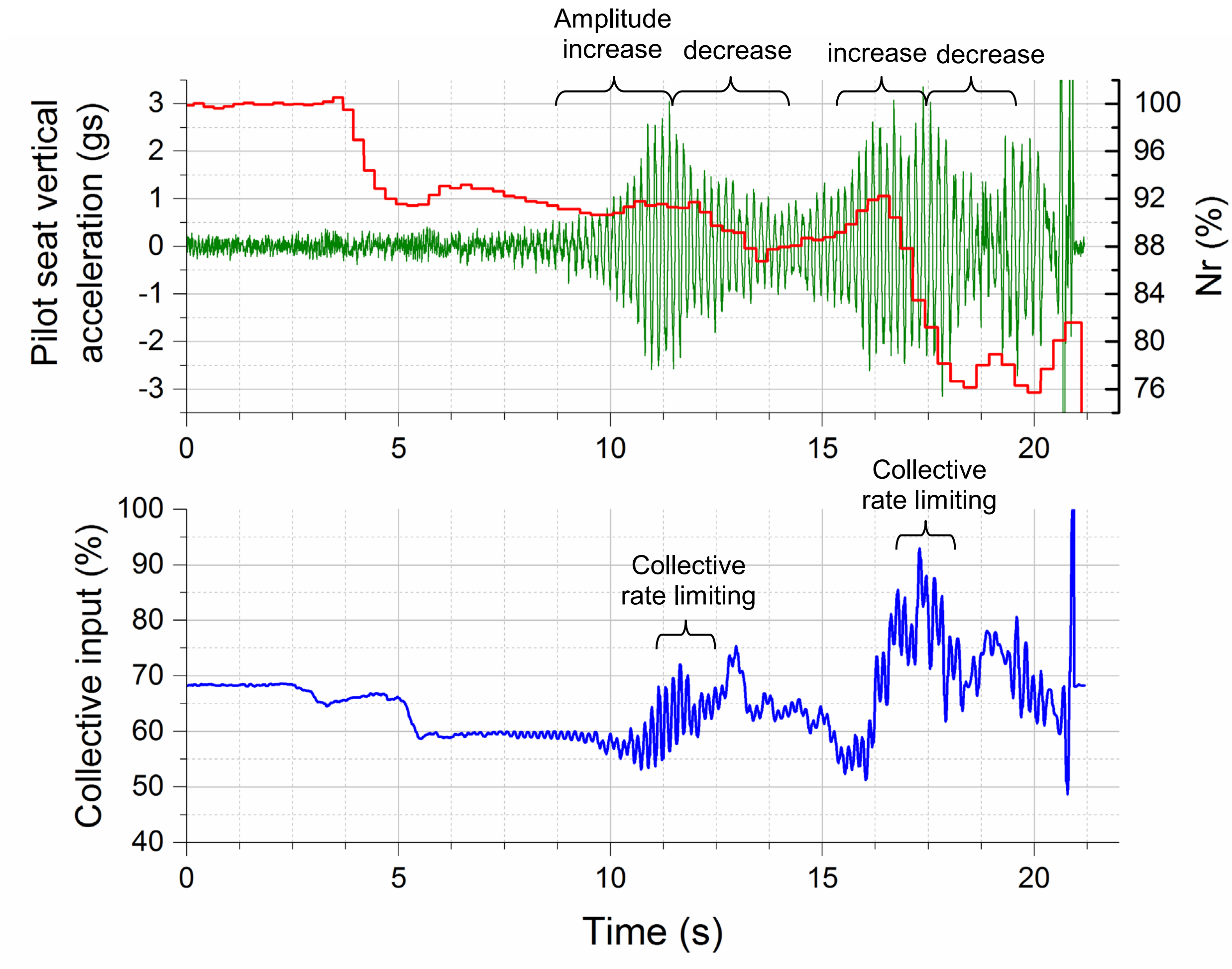

Figure 15. Vertical acceleration and collective input rate limiting.

During earlier flight tests, cyclic stirs of the swashplate were purposefully used to excite the scissors mode in the main rotor to measure and verify system damping[9]. During the accident record, excitation of the airframe's first bending modes (lateral and vertical) induced the AHRS to respond with inputs intended to stabilize the aircraft. The corresponding inputs to the main rotor pitch angle can been seen in Figure 16. The AHRS was intended to work with the control laws as though the fuselage was a rigid body responding to wind gusts or similar low frequency inputs and was not intended for handling a 6 Hz vibration. Although the AHRS included filters on the signal outputs, the filters did not specifically target the 6-Hz stirring commands to the swashplate. The stirring actions of the AHRS system at this (~6 Hz) frequency were considered to be a driver of the scissors mode amplification of the main rotor.

[9] These tests are not the two prior flights, discussed on pages 14 and 15, where the main rotor scissors mode was encountered.

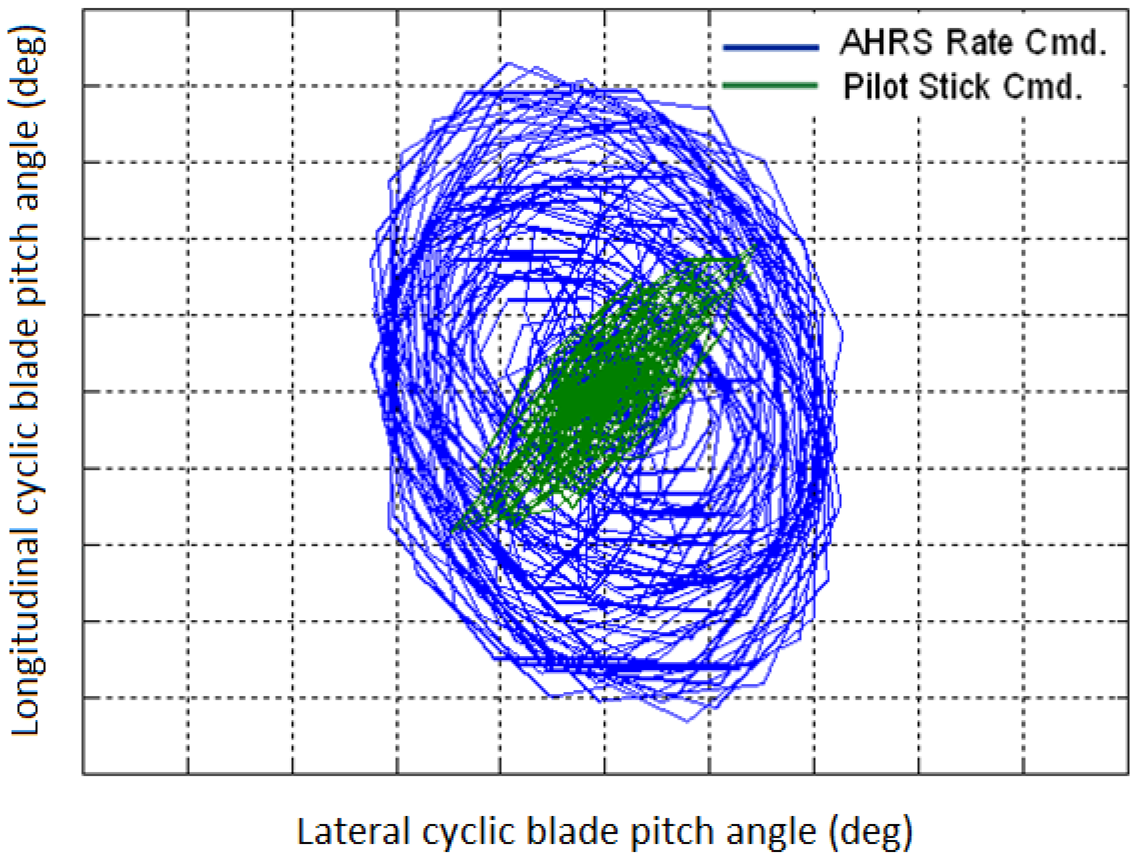

Figure 16. Lateral versus longitudinal blade pitch angle due to AHRS and pilot input during the accident test flight. Plot courtesy of Bell Helicopter.

The main rotor scissors mode had been encountered at 100% Nr in two previous tests at lower airspeed, but in high load factor banked turns, where the rotor blades are highly loaded but the scissors mode behavior is well damped. The high forward speed of the accident test produces a similar highly loaded aerodynamic environment for the rotor blades. One aspect of the main rotor's aerodynamic environment can be described by examining the aircraft's advance ratio (true airspeed/blade tip speed) in relation to the blade loading. High blade loading and high advance ratios produce a more complex aerodynamic environment. In all tests, the scissors mode response was only encountered on the outer edge of the blade-loading/advance ratio environment and indicate that a complex aerodynamic environment was needed to excite this response.

While the scissors mode was quickly damped out in the earlier tests, it grew in amplitude during the accident test record until the whole aircraft was vibrating at that frequency. It was determined that biomechanical feedback into the controls and the response of the AHRS system through the control laws increased the amplitude of the scissors mode response. While evidence of each can be found in the flight data, the relative contribution of each cannot be precisely determined.

Oscillations within a complex system like a helicopter can create feedback loops where some or all of the contributions increase the amplitude of the response of the whole system.

The manufacturer will focus on mitigating the biomechanical feedback and the AHRS induced swashplate stirring via control law filtering to prevent the amplification of the scissors mode response. This approach suggests that a damped system could be achieved by control law filtering of high frequency (~6 Hz) actuator commands.

E. CONCLUSIONS

On the day of the accident, the test crew were performing one engine inoperative (OEI) tests at increasing airspeeds for the heavy, forward cg configuration. On the final high speed point a full recovery of rotor rotational speed was not made after entering the OEI condition. Unlike earlier tests, the collective input was not decreased sufficiently to allow full recovery of the rotor speed. After encountering a significant and unexpected vibration, the collective was increased and Nr dropped below 80%, coinciding with the main rotor blades beginning to flap significantly out of plane. Sometime after 20 s, a near full-aft cyclic control input coincided with the main rotor blades flapping high enough over the front of the airframe to damage its pitch link rod end and low enough over the aft portion of the airframe to impact the tail boom and sever it from the rest of the aircraft.

Throughout the accident sequence, the aircraft responded as expected to pilot inputs in attitude, speed, altitude, and rotor speed. However, as the rotor speed approached 92% the growth in amplitude of a 6 Hz vibration was felt throughout the aircraft. At its peak, the vertical acceleration at the pilot's seat was greater than ± 2 g's.

At 5 s, the rotor speed was reaching 92%, where the first airframe vertical bending mode and the tail rotor cyclic regressing mode frequencies coalesced at 5.4 Hz. At 6.5 s, the main rotor scissors regressing mode became the dominant vibratory signature throughout the airframe, shaking the fuselage and crew at a frequency of about 6 Hz. As the test record progressed, biomechanical feedback into the controls and cyclic stir input by the AHRS contributed to the increase in the amplitude of the vibration. Pilot collective control increases led to further rotor rotational speed decay.

This vibration and excitation of the main rotor scissors mode had been seen in two earlier tests although at significantly lower amplitude and at a slightly higher frequency due to higher (100%) Nr, but the mode self-damped within seconds. The excitation of the scissors mode appears to require a complex aerodynamic environment including high blade loading and/or high advance ratio. Growth of the amplitude of the scissors mode also appears to require additional input sources such as biomechanical feedback into the controls and/or a cyclic stir input.

To prevent the scissors mode amplification, the manufacturer plans to modify the control laws such that a biomechanically induced 6 Hz control input would be filtered so that the excitation is not passed to the actuators. The AHRS signal, which already included filters, will also be redesigned to not respond to 'high' frequency perturbations of the aircraft.

Marie Moler
Specialist – Airplane Performance
National Transportation Safety Board

F. REFERENCES

1. Flight Data Recorder Factual Report, DCA16FA199, National Transportation Safety Board.

2. Airworthiness Factual Report, DCA16FA199, National Transportation Safety Board

3. Human Performance Report, DCA16FA199, National Transportation Safety Board