NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety
Washington, D.C. 20594

June 17, 2015

STRUCTURES

Group Chairman's Factual Report

DCA15MA019

A. ACCIDENT: DCA15MA019

Operator:	Scaled Composites, LLC
Location:	Koehn Dry Lake, California
Date:	October 31, 2014
Time:	1007 Pacific Daylight Time
Vehicle:	Model 339 SpaceShipTwo
Registration Number:	N339SS

B. STRUCTURES GROUP

Chairman: Clinton R. Crookshanks
National Transportation Safety Board
Denver, Colorado

Member: Paul Wilde
Federal Aviation Administration
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Member: Jonathan Carter
Scaled Composites, LLC
Mojave, California

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Mojave, California

Member: Boyd Dellwo
The Spaceship Company
Mojave, California

C. SUMMARY

On October 31, 2014, about 1007 Pacific daylight time,[1] a Scaled Composites SpaceShipTwo (SS2) reusable suborbital rocket, N339SS, experienced an in-flight anomaly during a rocket-powered flight test, resulting in loss of control of the vehicle. SS2 broke up into multiple pieces and impacted terrain over a 5-mile area near Koehn Dry Lake, California. One test pilot (the copilot) was fatally injured, and the other test pilot was seriously injured. SS2 had launched from the WhiteKnightTwo (WK2) carrier aircraft, N348MS, about 12 seconds before the loss of control. SS2 was destroyed, and WK2 made an uneventful landing. Scaled Composites was operating SS2 under an experimental permit issued by the Federal Aviation Administration's (FAA) Office of Commercial Space Transportation under the provisions of 14 *Code of Federal Regulations* (CFR) Part 437.

[1] Unless otherwise indicated, all times in this report are Pacific daylight time based on a 24-hour clock.

D. DETAILS OF THE INVESTIGATION

The Structures Group was formed on November 13, 2014, with group members from the FAA, Scaled Composites and The Spaceship Company. The goal of the group was to examine the recovered wreckage and document the major separations in an effort to identify the sequence of the vehicle break up. Three of the group members were also members of the External Imagery Group. The Structures Group convened in Mojave and examined the wreckage in the hangar December 8-10, 2014, and January 26-29, 2015.

1.0 Vehicle

Manufacturer's Serial Number (MSN): 001
Total Time: 83.03 hours (prior to accident flight)

SpaceShipTwo (SS2) was a hybrid rocket powered, suborbital vehicle designed to carry 2 crew and 6 passengers[2] into space after being carried aloft by WhiteKnightTwo (WK2). The vehicle was about 60 feet long, had a wingspan about 23 feet, a height about 15 feet at the tail, and had a fuselage diameter about 7.5 feet (Figure 1[3]). The vehicle was equipped with extension-only tricycle type landing gear that utilized two wheeled main landing gear and a nose skid. For re-entry into the Earth's atmosphere the vehicle was equipped with a unique feathering system that rotated the twin tailbooms and feather flap assembly up about 60 degrees from its normal configuration (Figure 2). The vehicle was constructed primarily of carbon fiber reinforced polymer (CFRP). Metallic components were used for fittings and fastening points throughout the vehicle. See Figure 3 for the vehicle station diagram.

The vehicle was on its 4th powered test flight when it experienced an inflight anomaly and loss of control. As a result, it broke up in flight about 46,000 feet MSL. The debris was scattered over a large area northeast of Mojave, California. The vehicle wreckage and debris was recovered and transported to a hangar at the Mojave Air and Space Port, Mojave, California, for examination[4]. Several videos of the accident sequence were recovered and examined[5]. The video evidence indicated that the structural break up initiated in the aft end of the vehicle near the junction between the wing and feather flap.

The feather flap assembly consisted of the left and right tailbooms, left and right feather flaps, and the feather flap torque tube (Figures 4 & 5). The feather flap assembly was attached to the wing rear spar at four hinges, two inboard hinges and two outboard hinges. The positon of the feather flap assembly was controlled by handles in the cockpit that drove two feather actuators installed in the aft fuselage of the vehicle. The forward ends of the actuators were connected to the left and right actuator horns on the torque tube and the aft ends were connected to metallic fittings installed between the upper and lower fuselage longerons. In the unfeathered configuration, feather locks installed on the wing forward spar engage the feather lock pins at the

[2] The accident vehicle did not have the passenger seats installed.
[3] All figures are presented in Appendix A to this report.
[4] See the NTSB Vehicle Recovery Group Factual Report in the public docket for the details of the recovery.
[5] See the External Imagery Factual Report in the public docket for the details of the video examination.

forward end of the tusks (boom structure forward of the hinge line) to lock the feather in the down position. Control of the feather locks was provided through a handle in the cockpit.

The booms were composite structures that extended aft from the wing and transitioned into the vertical stabilizers. A trimmable horizontal stabilizer with elevon was attached to the outboard side of each boom and a fixed strake (known as a megastrake in company jargon) was attached to the inboard side. A rudder was installed at the trailing edge of each boom.

The feather flap torque tube was a built up composite structure that tied the two booms and feather flaps together through the center of the aft fuselage. Outboard of the right and left inboard feather flap hinges the torque tube transitioned from a triangular tube into separate structural members, the forward torque tube and aft torque tube, that then tied into the booms. The outboard feather hinges were installed at the outboard ends of the forward torque tube.

A mid spar and trailing edge spar tied the booms to the feather flaps aft of the torque tube. Three longitudinal ribs, the outboard rib, inboard rib, and root rib, were installed on each side between the booms and the inboard edges of the feather flaps. A stub rib was installed between the boom and the outboard, forward rib aft of the forward torque tube.

The portion of the feather flap assembly aft of the aft torque tube and between the root rib and inboard rib was secondary structure known as the “lightweight taco” in company jargon. Upper and lower skins were installed over the internal structure of the feather flap. See Figure 5 for an annotated drawing of the internal feather flap assembly structure.

2.0 Accident Vehicle Examination

The aft end of the vehicle was laid out on the hanger floor to include the main oxidizer tank (MOT), the left and right wings, the left and right booms, and the feather flap assembly structure. Only the larger identifiable pieces were included in the layout. See Figure 6 for a photo of the layout of the aft portion of the vehicle.

Left Boom

One major piece of the left boom was recovered and identified (Figure 7). The section extended from the left outboard feather flap hinge aft to the rudder. A piece of the tusk about 3.5 feet long was recovered separately that included the lock pin. The lock pin was intact. This section was reportedly buried in the ground at recovery site 3. The remaining portion of the left boom tusk forward of the hinge was highly fragmented and not conclusively identified. There was a crack in the left outboard boom skin below the left horizontal stabilizer near fuselage station 575. The aft upper portion of the vertical stabilizer was missing above a line that ran from the upper leading edge at the megastrake forward spar location to the center rudder hinge. Two small pieces of the upper boom leading edge were recovered separately.

The upper portion of the rudder above the center hinge was not present. A section of left outboard rudder skin about 6 feet long and 1.5 feet wide was recovered separately. The lower portion of the rudder below the center hinge remained attached to the boom but was damaged and fractured at several locations. The center rudder hinge was intact and the lower hinge was fractured. The left rudder balance weight was recovered separately. A section of the left outboard boom skin about 2 feet by 1.5 feet that contained the outboard balance weight bubble for the rudder was recovered separately.

The left megastrake was separated from the left boom and not identified in the recovered wreckage. The left horizontal stabilizer remained attached to the boom with minimal damage. There was a crack in the upper skin of the left horizontal stabilizer along the root rib forward of the forward shear web. There was an L-shaped crack in the lower skin of the left horizontal stabilizer aft of the forward shear web near the root rib. The left elevon was fractured into three pieces. The inboard piece between the inboard hinge and the center hinge remained attached to the horizontal stabilizer. The center portion between the center hinge and outboard hinge and the piece outboard of the outboard hinge that included the balance weight were recovered separately from the boom near the left boom. The Kapton reflective tape was missing from the inboard surface of the left boom except at three access doors.

The feather flap mid spar remained attached to the left boom and was essentially intact but had fractured about 10 inches inboard of the boom. The inboard portion of the feather flap mid spar was deformed forward. About 30 inches of the aft torque tube remained attached to the left boom and was separated at the location of the feather flap left outboard rib. The aft torque tube lower cap fracture was consistent with tension and the upper cap fracture was consistent with compression. The stub rib, about 30 inches long, remained attached to the left boom and was essentially intact. A portion of the forward torque tube remained attached to the left boom and was fractured along a 45 degree line from about 30 inches inboard of the boom at the upper edge to about 15 inches inboard of the boom at the lower edge.

The left outboard feather flap hinge lug and carbon tow was intact on the forward torque tube attached to the left boom. The hinge pin was intact and the spherical bearing moved freely. The clevis arms from the wing side of the hinge were fractured from the hinge and remained attached to the lug. The deformation of the clevis arms was indicative of rotation of the left boom clockwise as looking forward.

Right Boom

Two major pieces and several smaller pieces of the right boom were recovered from site 1 and identified (Figure 8). The two major pieces encompassed the area from the right outboard feather flap hinge aft to the trailing edge of the vertical stabilizer. The boom was fractured in two near fuselage station 575 about 5 feet forward of the right horizontal stabilizer attach point. There was a small piece of composite honeycomb embedded in the inboard right boom skin below the megastrake. The portion of the right boom tusk forward of the hinge was highly fragmented. A piece of the tusk about 2.5 feet long was recovered separately that included the lock pin. The lock pin was intact. This section was reportedly buried in the ground at the recovery site. Three additional pieces of the right tusk were recovered separately and conclusively identified.

Most of the rudder was separated from the right vertical stabilizer except for a small piece of left skin, spar, and balance horn that was attached at the center hinge. The lower rudder hinge was intact and the upper hinge had separated from the boom. A few smaller pieces of the right rudder were identified but most of it was not.

The right megastrake remained attached to the right boom with minimal damage. The right horizontal stabilizer remained attached to the right boom but was damaged at the outboard leading edge and in the aft inboard area. There was a crack on both the upper and lower skins of the right horizontal stabilizer that ran diagonally from the root of the forward spar aft and outboard to and around the trailing edge. There was a crack on the right horizontal stabilizer lower skin at the root rib near the forward end. The right elevon was fractured into three pieces. The inboard piece between the inboard hinge and the center hinge remained attached to the horizontal stabilizer. The center portion between the center hinge and outboard hinge and the piece outboard of the outboard hinge that included the balance weight were recovered separately from the boom about 500 yards away. The Kapton reflective tape was missing from the inboard surface of the right boom except at three access doors.

A section of the feather flap trailing edge spar about 16 inches long remained attached to the right boom. The right feather flap mid spar remained attached to the right boom and was essentially intact. About 30 inches of the aft torque tube remained attached to the right boom and was separated at the location of the feather flap right outboard rib. The aft torque tube lower cap fracture was consistent with tension and the upper cap fracture was consistent with compression. The aft torque tube was also fractured about 8 inches inboard of the boom and the inboard portion was deformed forward.

The right outboard feather flap hinge and underlying structure was separated from the right boom. The lug and carbon tow around the lug was intact. The tow was fractured where the hinge area separated from the boom. The hinge pin was intact and the spherical bearing rotated freely along only the y-axis. The clevis arms from the wing side of the hinge were fractured from the hinge and remained attached to the lug.

Feather Flap

Reference Figures 9 through 12 for drawings showing the feather flap assembly structure that was recovered and conclusively identified. The yellow highlighted areas represent identified structure. The green highlighted areas represent structure with significant damage where only portions of the structure were identified. The red lines indicate fractures or cracks in the composite structure.

Several pieces of the feather flap skin were conclusively identified in the wreckage. Two pieces of the left lower feather flap skin and two pieces of the left upper feather flap skin were recovered and identified. One piece of left lower feather flap skin measured about 5 feet by 2.5 feet and was normally installed between the aft torque tube and trailing edge spar and between the left boom and outboard feather flap rib. The other piece of left lower feather flap skin measured about 7.5 feet by 2 feet and was normally installed between the forward and aft torque tubes and between the root rib and left outboard rib. One piece of left upper feather flap skin measured about 4.5 feet by 3 feet and was part of the lightweight taco normally installed between the root rib, aft torque tube, left inboard rib and trailing edge. The fuselage seal remained attached to this piece of skin. The other piece of left upper feather flap skin measured about 5 feet by 2.5 feet and contained the triangular skin bay between the forward torque tube, aft torque tube and the forward inboard rib and a portion of the skin bay between the aft torque tube, forward inboard rib and forward outboard rib.

Four pieces of the right upper feather flap skin were recovered and identified. One piece of the right upper feather flap skin measured about 4 feet by 2.5 feet and was part of the lightweight taco normally installed between the root rib, aft torque tube, right inboard rib and trailing edge. The inboard flange containing the fuselage seal was separated from this skin piece and not identified. The other three pieces of right upper feather flap skin had mating fractures and measured about 6.5 feet by 5 feet together. These three pieces were normally installed between the right inboard rib, right boom, mid spar and forward torque tube. The forward inboard section of the skin bay between the forward outboard rib, forward torque tube and aft torque tube and the skin between the mid spar and trailing edge were separated and not identified.

Some of the feather flap structure from the left side was not conclusively identified in the wreckage to include all of the left inboard and left outboard ribs, the trailing edge spar, and most of the left root rib. The aft 2 feet of the left root rib was separated. Some portions of the left feather flap internal structure remained attached to the left boom as described earlier.

Some of the feather flap structure from the right side was not conclusively identified to include all of the right inboard rib, the forward 20 inches of the right forward outboard rib, and the lightweight taco trailing edge. The right root rib was separated and essentially intact. The inboard section of right trailing edge spar, about 60 inches long, was recovered separately and had a portion of the right outboard mid rib about 15 inches long attached. The remainder of the outboard mid rib was recovered separately. Some portions of the right feather flap internal structure remained attached to the right boom as described earlier.

Most of the feather flap torque tube was recovered at site 3a and identified. The center portion, about 10 feet long between the kinks where the transition to the forward and aft torque tubes occurs, was intact with minimal damage. The forward torque tube on the left side was fractured about 4 feet outboard of the kink along a 45 degree plane. There was about 3-4 inches of left forward torque tube material not identified between the fracture on the boom side and the fracture on the feather flap side. The left aft torque tube was fractured at the forward inboard rib location about 20 inches outboard of the kink. A 28-inch section of the left aft torque tube between the left forward inboard and forward outboard rib locations was not identified. The forward torque tube on the right side was fractured about 2 feet outboard of the kink along 45 degree planes. The right outboard 4-5 feet of forward torque tube was fragmented into several pieces. There was about 4-6 inches of right forward torque tube material not identified between the fracture on the feather flap side and the remnants on the boom side. The right aft torque tube was fractured at the right forward inboard rib location about 20 inches outboard of the kink. The right aft torque tube was separated from the center portion at the kink and recovered near the main torque tube wreckage. A 28-inch section of the right aft torque tube between the right inboard and outboard rib locations was not identified.

The right and left inboard feather flap hinge clevises were intact on the torque tube. The pins were intact and the spherical bearings were free to move. The wing lugs fractured from the fitting flanges on the aft spar and lower longerons and were retained in the clevises (Figures 13 and 14). The fractures had an hourglass shape on the wing side of the fitting and were linear on the longeron side of the fitting. The aft inboard bolt hole was intact on the fractured right inboard feather flap hinge fitting. There was no evidence of over travel on the hinge fittings.

The right actuator horn remained attached to the torque tube with minimal damage. The glass layers on the upper surface were fractured and the two bobbins were pulled from the lugs. The unidirectional carbon tow around the bobbins was all fractured. The pin and bobbins remained attached to the right feather actuator. The left actuator horn remained attached to the torque tube but had significant damage. The horn was displaced down and the lower cap was fractured at its junction with the torque tube. The upper cap unidirectional carbon was damaged and much of it was separated but recovered adjacent to the torque tube wreckage. Both bobbins were pulled from the lugs. The unidirectional carbon tow around the bobbins was all fractured. The left pin with one bobbin installed was recovered separately with some tow material trapped between the bobbin flanges.

The left and right inboard feather hinges were disassembled. The nested hinge pins were removed easily and didn't have any obvious signs of damage. There was evidence of red grease on the hinge components. There was some surface rust on the bushing in the right hinge fitting. Each of the hinge lugs had some light scoring on the outboard face. There was some evidence of impressions on the inboard and outboard torque tube hinge fitting clevises where the spacers abut the face of the lug. The damage on the inboard lugs was more pronounced. There was some plastic type debris of unknown origin in the left hinge when it was disassembled. There were indications of overstrain in the glass layer overwrap around the lugs on the torque tube.

All of the fractures in the metal fittings were examined and had a dull grainy appearance consistent with overstress separation.

Wings

The right and left wings were separated from the fuselage at impact site 2 with significant damage. The one-piece rear spar was essentially intact but separated from the wings. Portions of the left and right outboard feather flap hinge fittings remained attached to the rear spar. The clevis ears on both were fractured and retained in the boom side of the hinge as described earlier. All of the carbon tow material around the clevis ears was fractured. There was no obvious direction of failure noted on the outboard hinge remnants on the rear spar for both the right and left outboard hinges.

The left and right inboard feather flap hinge fitting flanges remained attached to the rear spar. The lugs on both were fractured from the fittings and retained in the torque tube side of the hinges as described earlier. The forward end of the lower longerons remained in place forward of the fittings. The right inboard hinge fitting mount was rotated counter clockwise (as viewed looking forward) and the outboard leg of the fitting was slightly pulled away from the mount at the lower edge.

Aft fuselage

Two sections of aft fuselage skin were identified from the right and left sides in the area where the upper and lower longerons were attached. The left aft fuselage skin section measured about

4.5 feet longitudinally by 4 feet circumferentially. The upper longeron, forward gusset, and lower longeron were separated consistent with a cohesive failure of the bond. The aft right fuselage skin section measured about 5 feet longitudinally by 4 feet circumferentially. The upper longeron, forward gusset, and lower longeron were separated consistent with a cohesive failure of the bond. No other significant pieces of the aft fuselage skin were identified.

See Figure 15 for a drawing showing the internal aft fuselage structure recovered and conclusively identified. The right feather actuator was recovered but is not shown in the drawing. The mid bulkhead was not identified with the exception of the upper left gusset and two small sections attached to the inboard feather flap hinge fittings.

A section of the aft bulkhead about 2 feet long from the left side was identified that included the feather actuator fitting, the left case-throat-nozzle (CTN) support fitting, the upper left longeron and a 1-foot section of the aft end of the left lower longeron. The outboard rod end remained installed in the CTN support fitting with the bolt intact but had fractured in the threaded area. A section of the aft bulkhead about 4.5 feet long from the upper portion was identified that included the upper CTN support fitting. The outboard rod end remained installed in the upper CTN support fitting with the bolt intact and the support link was attached. The inboard rod end was fractured in the threaded area. A section of the aft bulkhead about 2 feet long from the right side remained attached to the right aft feather actuator fitting. Portions of the aft ends of the right upper and lower longerons, each about 1 foot long, remained attached to the actuator fitting. The right CTN support fitting was attached to this section of aft bulkhead and the right support link was attached. The link was fractured in the threaded portion of the inboard rod end. The remainder of the aft bulkhead was not identified.

The right lower longeron and the forward gusset were identified and had separated along the fuselage bond line. The longeron was fractured at the forward and aft ends of the feather flap hinge fitting. The right feather flap hinge fitting flanges remained attached to the forward portion of the right lower longeron. The aft portion of the right lower longeron in the area of the feather actuator fitting, about 7 inches long, remained attached to the actuator fitting. Most of the right upper longeron was not identified with the exception of the section that remained attached to the right feather actuator fitting.

The forward and aft ends of the left lower longeron in the areas of the left inboard hinge fitting and left actuator fitting, respectively, were identified. The left inboard feather flap hinge fitting flanges remained attached to the forward portion of the left lower longeron. The center portion and two gussets on the left lower longeron were not identified. The left upper longeron was attached to a portion of the aft bulkhead as described earlier and was separated along the bond lines with the fuselage skin. The left upper longeron was fractured and deformed downward about 12 inches forward of the aft bulkhead.

The left and right feather actuators were both recovered[6]. The left feather actuator was separated from the feather horn at the forward end and from the left actuator fitting at the aft end. The left feather actuator forward rod end was rotated about 10 degrees from the aft attach lug. The outboard clevis ear of the left aft feather actuator fitting was fractured from the fitting. There was

[6] See the Systems Group Factual Report in the public docket for detailed information on the actuators.

mechanical damage and deformation to both ears of the clevis fitting that was consistent with the left feather actuator departing with the forward end unconstrained and moving inboard. The bushing remained in the inboard clevis ear. The outboard bushing and pin assembly were not identified. The left feather actuator aft lug had mechanical damage consistent with the damage on the fitting clevis ears. The actuator lug spherical bearing was not identified. The forward rod end on the left feather actuator was intact and the spherical bearing moved freely. There was mechanical damage on the inboard aft and outboard forward portion of the rod end.

The right feather actuator was recovered with both the forward and aft pins intact. The forward pin remained installed in the forward rod end with both bobbins installed and the hardware intact. The forward rod end was rotated about 90 degrees counter-clockwise as looking forward from the aft attach lug. The inboard bobbin left flange was bent inboard about 45 degrees. The right feather actuator remained attached to the clevis fitting at the aft end. There was gouging and damage to the clevis fitting and feather actuator lug consistent with motion between the lug and clevis. The clevis fitting was fractured along the lower inboard clevis flange inboard of the bolt holes that attach the fitting to the lower longeron. The outboard clevis flange was buckled below the pin location. The snap ring on the aft feather actuator pin was missing but the cotter pin was installed. The cotter pin was removed so the aft feather actuator pin could be removed and inspected. There was no evidence of damage to the actuator pin.

The left wing to body fairing was separated along the bond line from the aft fuselage and sustained minimal damage.

About 90% of the aft tailcone closeout was identified and fractured into 3 pieces. The missing portion was around the 6 o'clock position as viewed looking forward.

3.0 Maintenance Records

The group reviewed the Aircraft Log, inspection logs, and discrepancy logs for the accident vehicle, a Scaled Composites, LLC Model 339, S/N 001, N339SS. The vehicle had a manufacturing date of June 26, 2010. The first conditional inspection of the vehicle per the Model 339 Inspection Plan was completed on June 26, 2010, and a Special Airworthiness Certificate was issued by the FAA on June 30, 2010. The vehicle underwent 4 captive carry flights before its first glide flight on October 10, 2010.

The second conditional inspection of the vehicle per the Model 339 Inspection Plan was completed on September 10, 2011, with 52.4 hours time in service (TIS) after 11 captive carry flights and 15 glide flights. A new Special Airworthiness Certificate was issued by the FAA on September 12, 2011.

The third conditional inspection of the vehicle per the Model 339 Inspection Plan was completed on December 11, 2012, with 69.8 hours TIS after 5 additional captive carry flights and 7 additional glide flights. A new Special Airworthiness Certificate was issued by the FAA on December 11, 2012. On October 29, 2013, an error in the flight time was found in the logbook entry for February 14, 2011, that over reported the total TIS by 3 hours and was corrected.

A new Special Airworthiness Certificate was issued by the FAA on November 12, 2013. The fourth conditional inspection of the vehicle per the Model 339 Inspection Plan was completed on November 13, 2013, with 76.81 hours TIS after 1 cold flow flight, 2 powered flights, 1 additional captive carry flight and 4 additional glide flights.

The fifth conditional inspection of the vehicle per the Model 339 Inspection Plan was completed on October 1, 2014, with 82.03 hours TIS after 1 additional cold flow flight, 1 additional powered flight, and 3 additional glide flights. A new Special Airworthiness Certificate was issued by the FAA on October 1, 2014. Glide flight 30 was conducted on October 7, 2014, after which the total aircraft time was reported as 83.03 hours TIS. The vehicle had performed 17 captive carry flights, 30 glide flights, 2 cold flow flights, and 3 powered flights prior to the accident flight.

The most recent ATC Transponder test per 14 CFR 91.413 and altimeter test per 14 CFR 91.411 were complied with on May 5, 2014.

The group reviewed the discrepancy log for the inspections performed prior to glide flight 30 and noted the following.

- Discrepancy 40, the left outboard feather hinge pin was found to be scratched/scored after removal for scheduled maintenance. The pin was replaced.
- Discrepancy 42, the feather hinge pins were dye penetrant inspected with no signs of damage.

The group reviewed the discrepancy log for the inspections performed prior to powered flight 4 and noted the following.

- Discrepancy 10, Some propagation of delamination on the left feather horn was noted. No repair was performed but the damage area was marked for monitoring.

No other significant items were noted. The conditional inspection, special inspection and preflight inspection paper work was all signed off with no discrepancies.

Before the FAA issues an experimental permit, the vehicle must be made available for inspection by the FAA per 14 CFR 437.21(d). The FAA performed a first inspection of SS2 on April 11, 2012, and they determined that all the vehicle components were installed as represented in Scaled's permit application except for the propulsion system components that had yet to be installed.

A second inspection occurred on April 15, 2013, after the propulsion system was installed. The FAA determined that the propulsion system was installed as represented in the permit application with the exception of the CTN and igniters since they were not installed. The FAA determined that the vehicle inspection was complete at this time.

4.0 Tests and Research

Recorded Data

A wealth of flight data was being recorded on the vehicle and telemetered to a ground station during the accident flight. Early in the investigation a discrepancy was noted between the time stamp on the onboard recorded data and the telemetered data. In addition, each of the video recordings utilized a different time standard. For a discussion of the various time standards and the time correlation of information, see the Electronic Recording Devices and Flight Data Factual Report and the External Imagery Factual Report. The structures group was interested in plotting several recorded parameters for use in correlating the breakup sequence and establishing the hinge reaction loads experienced during the accident flight. To aid in comparing the data a zero time was established at the instant that the SS2_RMC_carrier_separated discrete signal changed state to indicate that SS2 was separated from WK2 to begin the powered flight. The time of separation was 10:07:19.10 PDT. All of the plots presented utilize the elapsed time from separation along the y-axis. The following parameters were selected by the group.

Feather Position – This parameter recorded the feather percent travel measured using a string potentiometer installed on the feather flap torque tube at the location of the right feather actuator horn. The maximum feather angle is 60º so the feather position in degrees was calculated by multiplying the feather position in percent by 0.6.

Feather Rate – This parameter was calculated by the group as the time rate of change of the feather position data in degrees/second. This was not a recorded parameter. The data was truncated starting at 13.34 seconds for ease of viewing due to excessive noise.

R Feather Actuator DN Pressure – This parameter recorded the pressure using a transducer in the right feather actuator tubing on the down side of the piston in psig.

L Feather Actuator DN Pressure – This parameter recorded the pressure using a transducer in the left feather actuator tubing on the down side of the piston in psig.

SS2_Cabin_Pressure – This parameter recorded the absolute cabin pressure measured using a transducer installed on an avionics rack in the aft cabin in psia.

SS2_Cabin_Altitude – This parameter recorded the derived cabin altitude based on the cabin pressure parameter above in feet MSL.

L & R Feather Hook Strain – These parameters recorded strain gage output values in millivolts from strain gages installed on the feather hooks. These strain gages have not been calibrated and are not temperature compensated but the data trends should be valid.

L_Bucket_Rib_UPR_OUTBD – This parameter recorded the strain gage output value in millivolts from a strain gage installed on the bucket rib in the left tailboom where the left horizontal stabilizer is installed. This strain gage is calibrated and can be used to derive the horizontal stabilizer load.

X, Y, Z Acceleration – These parameters recorded the accelerations processed in g’s measured by accelerometers in the INS unit in the cabin.

Left_gyro_y_proc – This parameter recorded the gyroscope measured rotation rate about the y-axis from the left inertial navigation unit located under the cabin floor in degrees per second.

L & R Feather Hook OPEN – These discrete parameters record the state of the feather lock open positon switches. Both the left and right feather locks began opening 10.40 seconds after the separation (10:07:29.50 PDT).

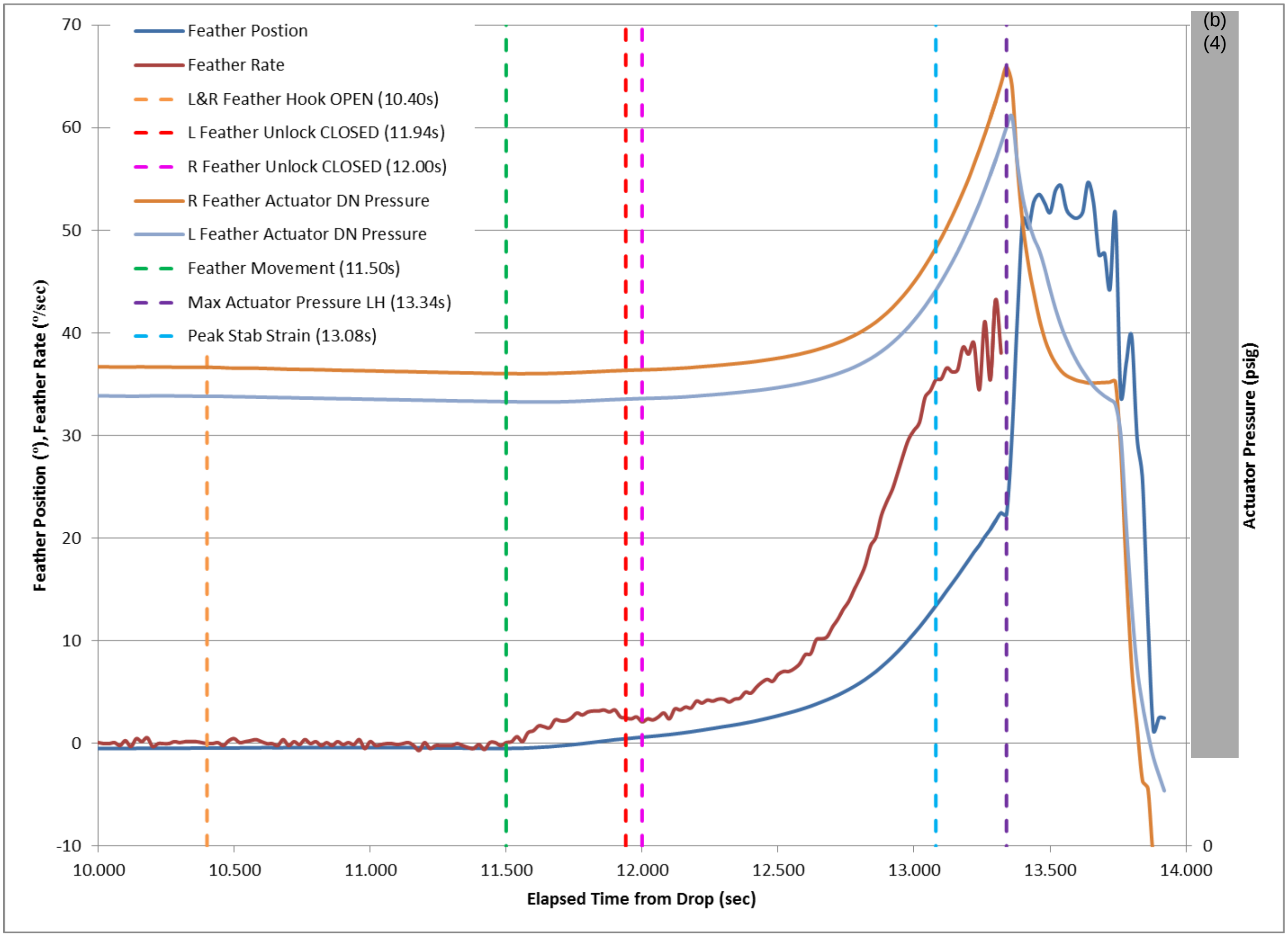
L & R Feather Unlock CLOSED – These discrete parameters record the state of the feather unlock position switches. The left feather lock is fully unlocked 11.94 seconds after the separation (10:07:31.04 PDT) and the right feather lock is fully unlocked 12.00 seconds after the separation (10:07:31.10 PDT).

Hooks Unlocked – This time was selected based on the recorded hook strains and represents the time when both hooks have unloaded and the strain remains essentially constant at 10.74 seconds after the separation (10:07:29.84 PDT).

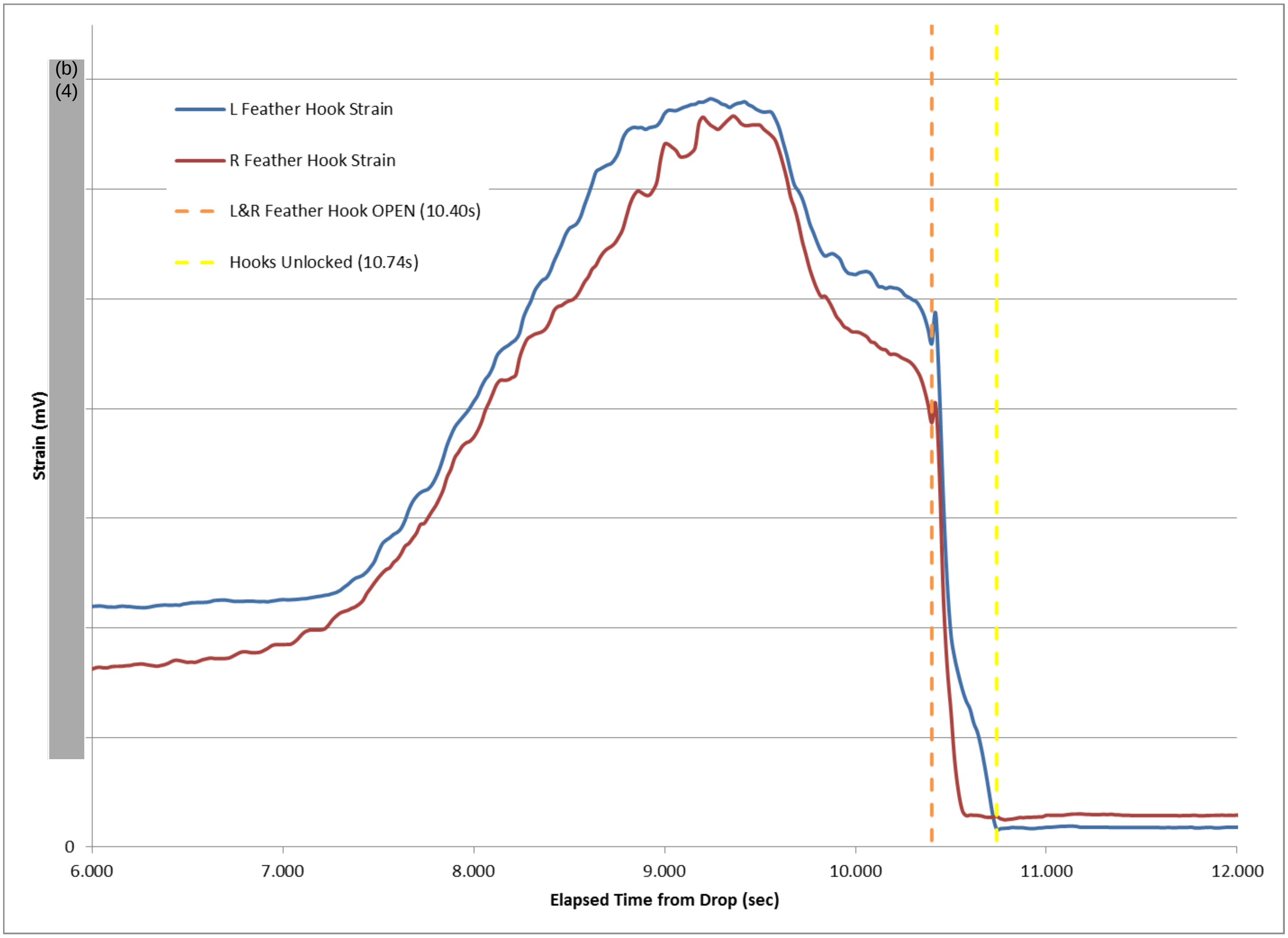
Feather Movement – This time was selected based on the calculated rate of feather movement and represents the time when the rate starts increasing after oscillating around zero for the previous portion of the flight. The feather began moving 11.50 seconds after the separation (10:07:30.60 PDT).

Max Actuator Pressure – This time was selected based on the recorded feather actuator pressures and represents the time when the right feather actuator pressure peaked 13.34 seconds after the separation (10:07:32.44 PDT). The left feather actuator pressure peaked 20 milliseconds later.

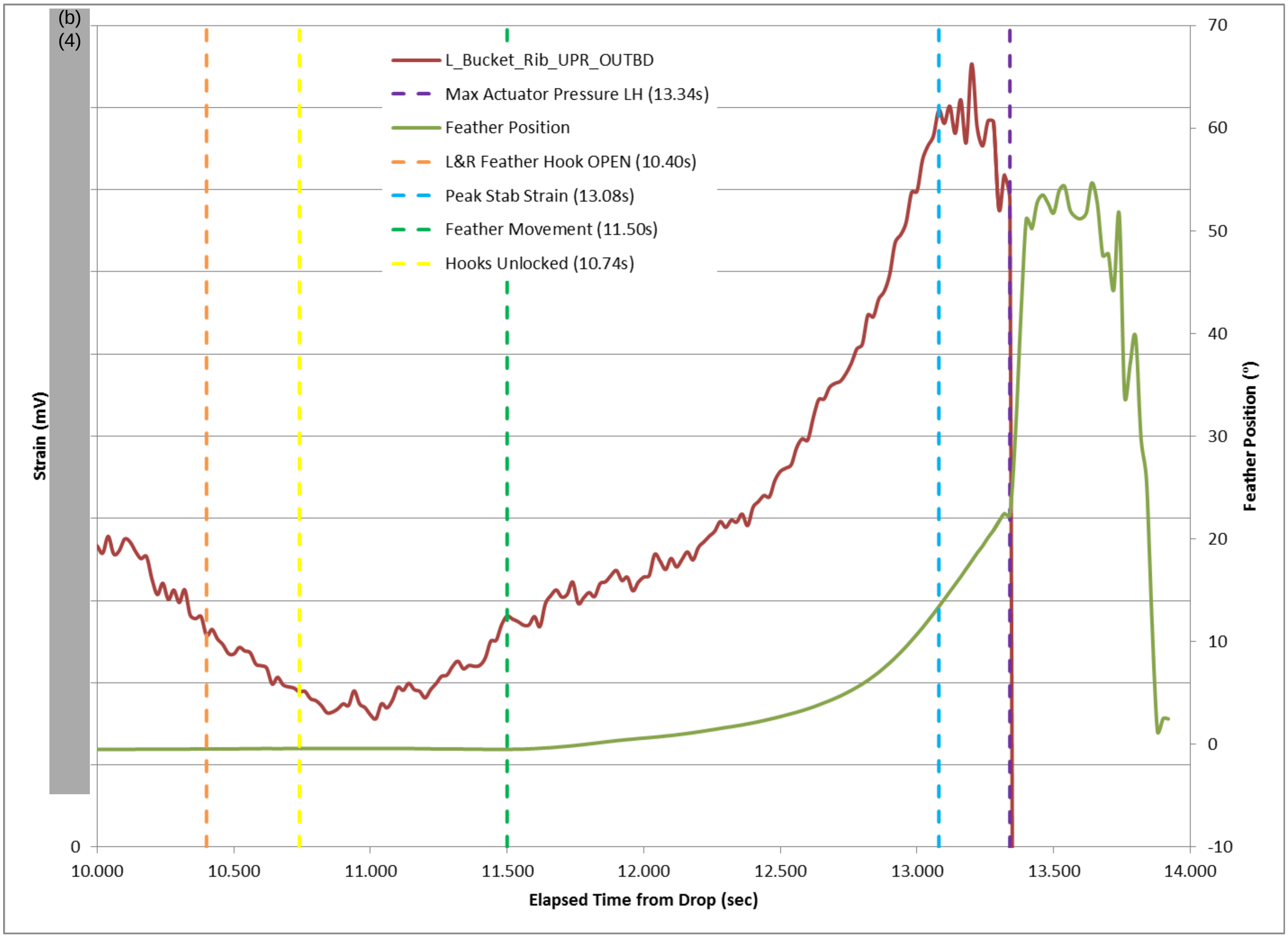
Peak Stab Strain – This time was selected based on the recorded bucket rib strain and represents the peak strain value recorded before the signal levels off and begins oscillating. The peak stab strain occurred 13.08 seconds after the separation (10:07:32.18 PDT).



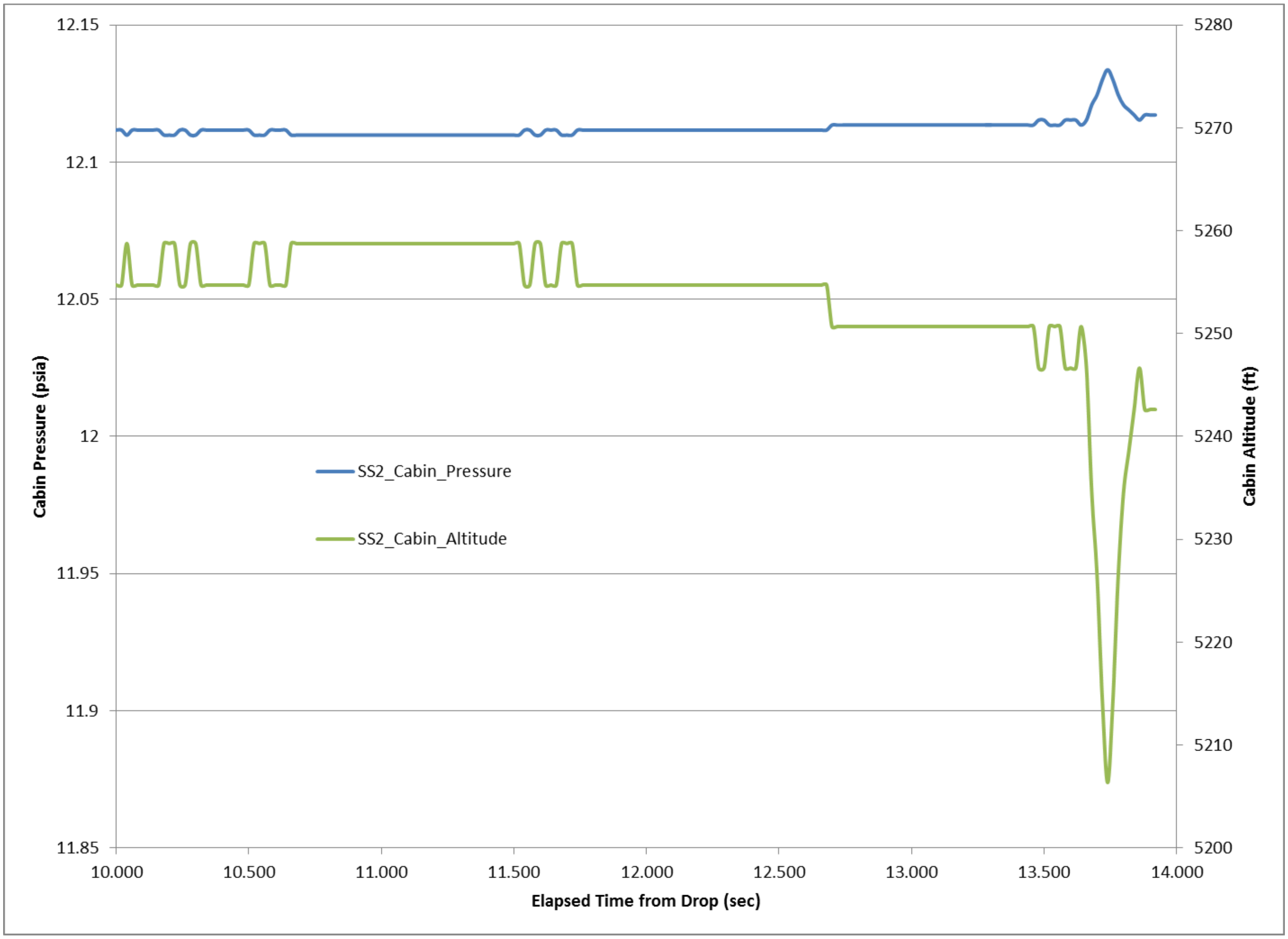
Plot 1 shows the feather actuator pressures, feather position, and feather rate for the period between 10.00 and 14.00 seconds after the separation with relevant discrete time events.



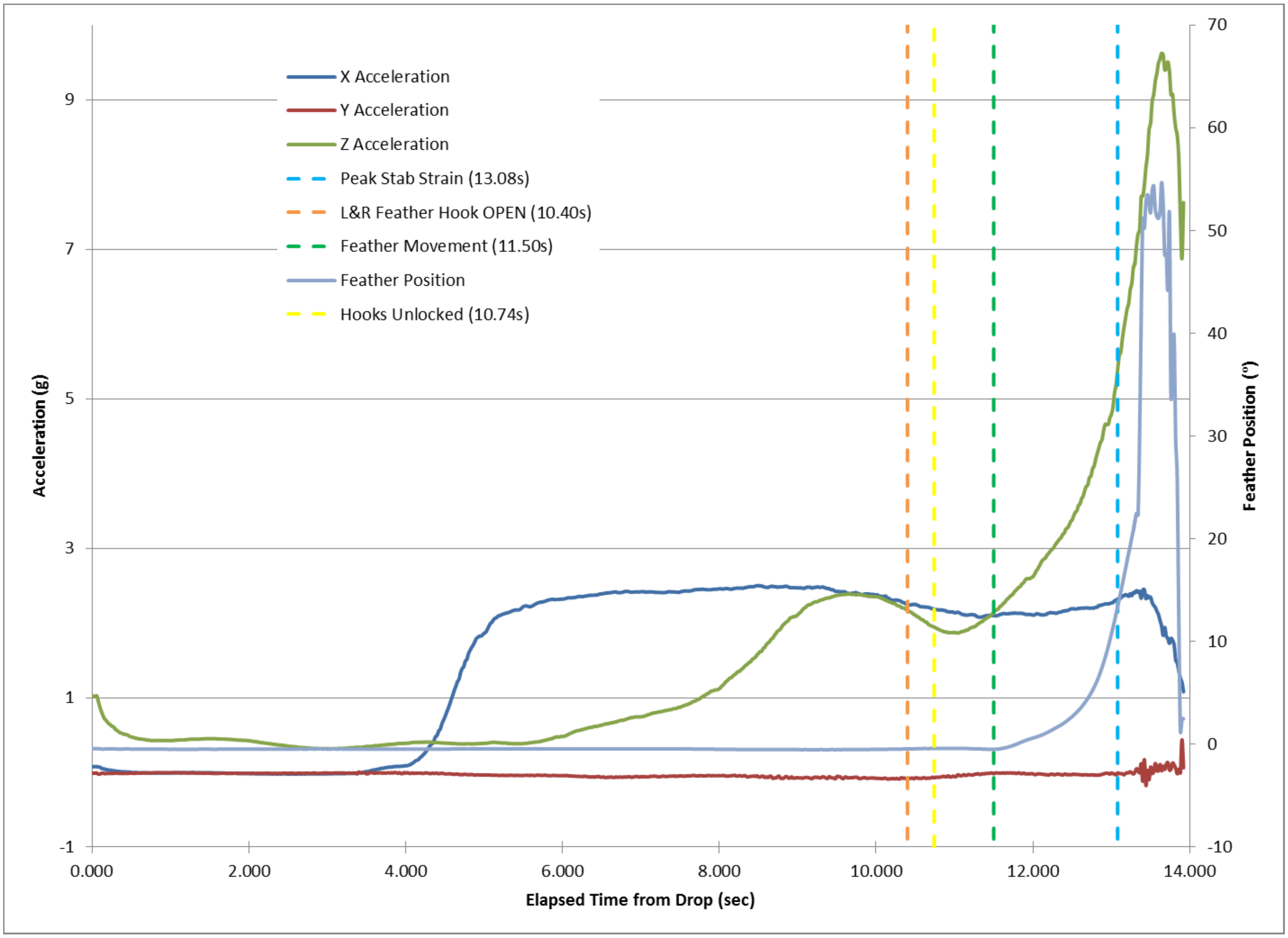
Plot 2 shows the strain in the feather lock hooks for the period between 6.00 and 12.00 seconds after the separation with the discrete times when the feather locks began opening and when both hooks had unlocked.



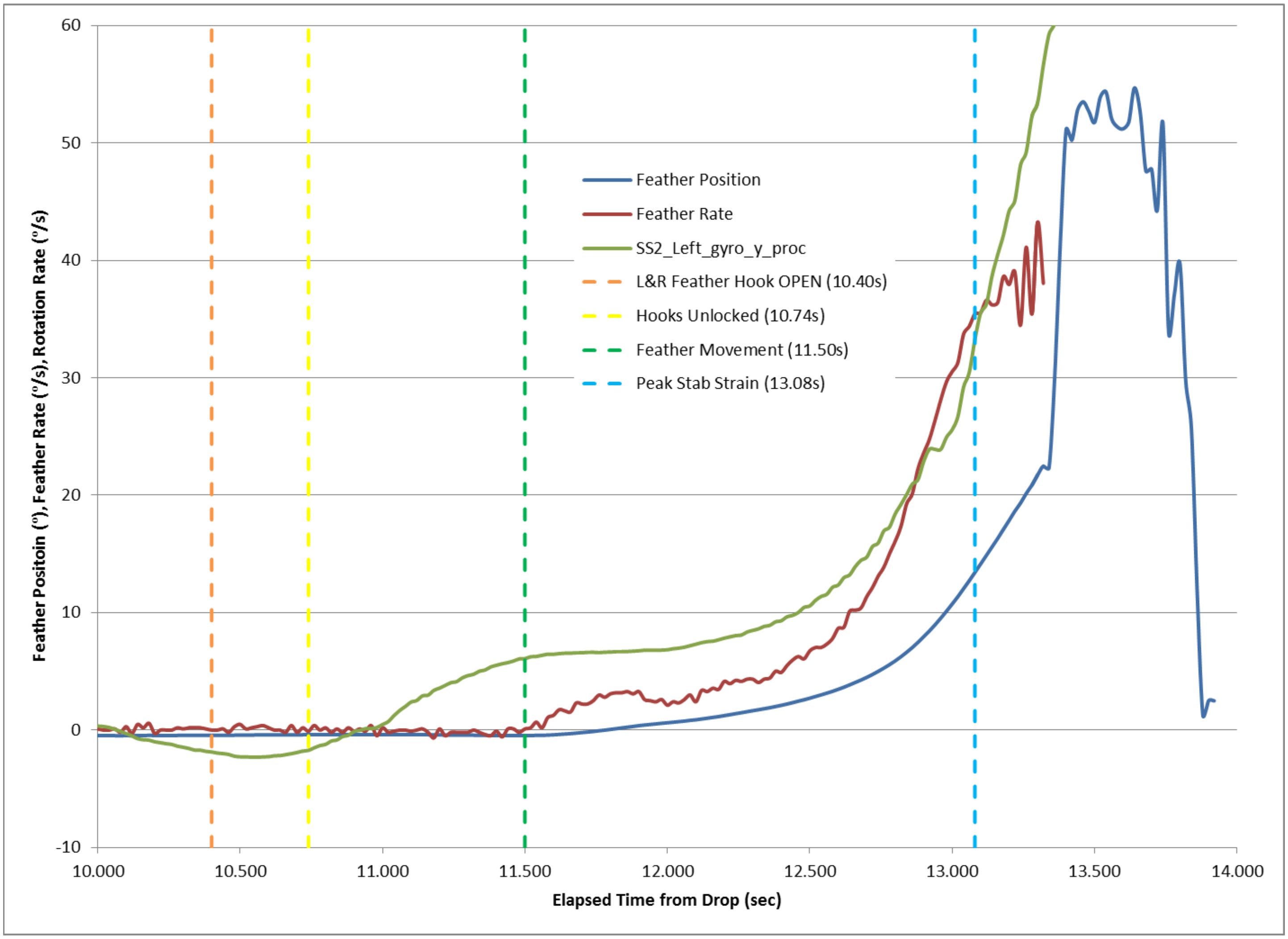
Plot 3 shows the bucket rib strain and feather position for the period between 10.00 and 14.00 seconds after separation with relevant discrete time events.



Plot 4 shows the cabin pressure and altitude for the period between 10.00 and 14.00 seconds after the separation. The last data point was recorded 13.92 seconds after the separation. These parameters were plotted for correlation with some of the information from the recovered videos.



Plot 5 shows the acceleration in the x, y, and z directions and the feather position for the period between 0.00 and 14.00 seconds after the separation with relevant discrete time events.



Plot 6 shows the feather position, feather rate, and vehicle pitch rate for the period between 10.00 and 14.00 seconds after the separation with relevant discrete time events.

Feather Flap Inboard Hinges

The inboard feather flap hinges are manufactured in accordance with Scaled Composites drawing SS2-10D502 (Attachment 1). The drawing specifies the material as 7050-T7451 aluminum alloy plate per AMS4050. The right and left inboard feather flap hinge fittings were fractured in a similar manner as shown schematically in Figure 16 and described earlier. The fractures generally were located in the flange area of the fittings between the bolt holes and the transition radii to the lugs. The aft, inboard fracture on the right fitting intersected the second aft-most bolt hole on the inboard upper flange and progressed to the aft edge of the fitting outboard of the aft-most bolt hole. The fractures in the vertical flanges of both fittings curved into machined areas on the inboard and outboard faces of the lugs. The fracture faces had damage and deformation consistent with shear overload indicative of the lugs pulling aft from the flanges attached to the rear spar and lower longerons.

The material specifications for 7050-T7451 (Attachment 2) list the ultimate tensile strength of plate material with thicknesses between 5.001 – 6.000 inches as 70 ksi. From MIL-HDBK-5 the "theoretical ratio between shear and tensile stress for homogeneous, isotropic materials is 0.577." Therefore, the ultimate shear strength of the material can be calculated to be about 40.4 ksi.

Reaction Loads

Utilizing the data plotted above and the failure information learned from examining the wreckage, the group decided to estimate the reaction loads at the feather hinges, feather locks, and feather actuators during specific points in the flight. The loads were calculated by Scaled Composites using their FEMAP/Nastran NX Finite Element Analysis model (FEM). The input loads to the FEM were derived from the inertial loads (accelerations) and horizontal stabilizer strain recorded during the accident flight. The recorded parameters were used to scale a known and validated limit load design case to obtain the inputs to the FEM. The finite element model of the feather flap assembly is shown in Figure 17. The horizontal stabilizers are not included since the derived loads (from strain readings) are input as point loads at the horizontal stabilizer attach point. The input loads were assumed to be symmetrical. All reaction results are for the left side of the vehicle only.

The group identified six discrete load cases for examination. The first two cases occur at 10.40 seconds after separation when the left and right feather hooks start to open. The first of these cases looks at the nominal reaction loads with the hooks fully engaged and reacting a portion of the feather opening moment.

The second case, at the same instant in time, looks at the nominal reaction loads with the hooks fully opened and not contributing to the reaction loads. These two were selected to show the load redistribution between the locked and unlocked conditions at the same flight condition. It should be noted that based on the data in plot 2 above, the hooks were reacting load at this time.

The third case occurs at 10.74 seconds after separation when the hook strain in plot 2 shows that the hooks have fully opened and are no longer reacting any of the opening moment. Plot 3 shows a decrease in the horizontal stabilizer strain between 10.40 seconds and 10.74 seconds.

The fourth case occurs at 11.50 seconds after separation when the first evidence of feather movement occurs. This time was selected based on when the calculated rate of change of the feather position is positive and starts to increase from its nominal value around zero.

The fifth case occurs at 13.08 seconds after separation when the peak strain on the horizontal stabilizer occurs as shown in plot 3. From this time forward the average strain levels off and begins oscillating. While there is strain data recorded after this time, the group is not confident that the data is accurate or realistic.

The sixth case represents a limit load case used for the design of the vehicle. This case was a static case run with the feather in the down and locked position and the vehicle in a trimmed condition.

Two different methods were used to generate the input loads for the FEM. The first method was used for cases 1 through 4. In all of these cases the feather was in the down position. The method assumes that the pressure loads on the feather assembly are linear so that the pressure loads from a design limit load case could be scaled using the ratio of the PF04 (accident flight) horizontal stabilizer aerodynamic lift load and the design limit horizontal stabilizer aerodynamic lift load at each particular flight condition. The horizontal stabilizer drag loads were scaled in the same manner. For each load case the inertial acceleration corresponding to that point in time as shown in plot 5 above was also input to the model. The rotational inertia effects were neglected for these cases since the data shows little rotational acceleration during the first part of the event.

The second method was used for case 5 because the linear assumption was thought to be invalid after the feather moved significantly and there was no aerodynamic data available. At this point (13.08 seconds) the feather was deployed 13.43 degrees. Scaled Composites utilized a new, full-vehicle computational fluid dynamics (CFD) run using STAR-CCM to determine a more realistic pressure distribution for input to the FEM. The data in plot 6 above shows that the feather rate is 35.4 deg/s at this point. The corresponding vehicle pitch rate at this time is 33.3 deg/s at this time indicating that the vehicle is pitching up at this point while the feather is essentially stationary with respect to the Earth. The CFD was run with a full vehicle pitch rate of 33.3 deg/s and predicted a horizontal stabilizer lift load of about (b) (4) pounds. Correlation of the horizontal stabilizer strain at this point yielded a lift load of about (b) (4) pounds. The over-prediction in horizontal stabilizer lift load is likely a result of the difficulty in modeling the motion of the feather relative to the forward fuselage. To bound the problem, the full vehicle pitch rate was iterated until the predicted horizontal stabilizer lift load more closely matched the load calculated from the measured strain. A full vehicle pitch rate of 5 deg/s resulted in a horizontal stabilizer lift load of about (b) (4) pounds. Both full pressure distributions were imported onto the FEM in order to compare the reaction loads. The horizontal stabilizer lift and drag point loads and inertial loads were also input for this case.

Table 1 below shows a summary of the input loads to the FEM that were applied symmetrically.

Condition	Nx	Nz	H-Stab Lift	Aero Scaling factor	Feather Moment
	g	g	lbs	-	ft-lbs
10.40s	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]
10.74s	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]
11.50s	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]
13.08s - 33deg/s	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]
13.08 - 5deg/s	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]
Design Limit Load	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]

Table 1 – Input Loads to FEM

Table 2 below shows the reaction loads for cases 1 and 2 at 10.40 seconds after the separation with the feather hooks locked and unlocked for the left side of the vehicle. With the feather locks engaged, the feather opening moment is shared among the inboard and outboard hinges and the feather locks. With the feather locks disengaged, the feather opening moment causes a redistribution of load so that the actuators and inboard hinges must react more load in the longitudinal direction.

Condition	Feather Actuator	Inboard Hinge X	Inboard Hinge Y	Inboard Hinge Z	Outboard Hinge X	Outboard Hinge Z	Feather Lock
	lbs	lbs	lbs	lbs	lbs	lbs	lbs
10.40s locked	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]
10.40s unlocked	(b) (4)	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]

Table 2 – Reaction Loads for Locked and Unlocked Cases

Table 3 below shows the reaction loads for cases 1 and 3 through 6 for the left side of the vehicle. These cases represent the actual flight conditions during the accident flight. The underlying assumption on all of these results is that the feather opening moment must be reacted by the hinges and feather actuators. The results below also are for the static cases at each instant in time. The feather extension during the flight starts slowly and grows almost exponentially which would introduce dynamic effects that cannot be taken into account by the method especially for case 5. The design limit load case has a vertical acceleration of (b) (4) g which produces some amount of predicted wing bending. Case 5 at 13.08 seconds has a recorded vertical acceleration of 5.12 g which would increase the amount of wing bending above the prediction. Any amount of wing bending would deflect the feather flap hinge line causing the actual reaction loads to be different than those calculated. The results shown do not account for any structural deflection of the hinge line.

Condition	Feather Actuator	Inboard Hinge X	Inboard Hinge Y	Inboard Hinge Z	Outboard Hinge X	Outboard Hinge Z	Feather Lock
	lbs	lbs	lbs	lbs	lbs	lbs	lbs
10.40s	(b) (4) [redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]
10.74s	(b) (4) [redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]
11.50s	(b) (4) [redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]
13.08s 33deg/s	(b) (4) [redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]
13.08 5deg/s	(b) (4) [redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]
Design Limit Load	(b) (4) [redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]	[redacted]

Table 3 – Reaction Loads for Accident Flight Cases

At 11.5 seconds the first feather movement is evident in the data. Utilizing the recorded actuator pressures during the accident flight, the actuator capabilities are calculated to be (b) (4) [redacted] pounds and (b) (4) [redacted] pounds for the left and right actuators respectively. The reaction calculated with the FEM is (b) (4) [redacted] pounds at this time which correlates very well with the capabilities and gives some confidence in the method. The actuator reaction at 10.74 seconds shows that the reaction is well within the capability and feather movement should not be expected at this time.

The results show that there is a large difference in the hinge reaction loads at 13.08 seconds between the different vehicle pitch rates. The group was not able to refine the results even more without substantially more testing and research but felt that the results presented likely bounded the actual loads.

Scaled input the reaction loads at the inboard feather flap hinges for the two pitch rates into the Catia model of the hinges to determine the stress state in the hinges. Figures 18-21 show the Von Mises stress distribution in the hinges for these cases. The maximum stress (red area) has been clipped at 72 ksi for all the plots. This stress level was chosen by Scaled based on the ultimate tensile stress for 7050-T7451 from MIL-HDBK-5, 5-6 inches thick, B-basis.

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