NATIONAL TRANSPORTATION SAFETY BOARD
Office of Aviation Safety
Washington, D.C. 20594
March 31, 2015

AIRWORTHINESS GROUP CHAIRMAN'S FACTUAL REPORT

ERA14MA271

A. ACCIDENT:

Location:	Hanscom Field, Bedford, MA
Date:	May 31, 2014
Aircraft:	Gulfstream G-IV, N121JM, S/N 1399

B. AIRWORTHINESS GROUP:

Chairman:	Adam Huray National Transportation Safety Board Washington, DC
Member:	Robert Hendrickson FAA Washington, DC
Member:	Brian McDermid UK AAIB Accredited Representative Aldershot, UK
Member:	Kimberly Lascell Gulfstream Aerospace Corporation Savannah, GA
Member:	Wayne Haug Rockwell Collins Cedar Rapids, IA
Member:	Jay Eller Honeywell Phoenix, AZ
Member:	Graham Hesling Rolls-Royce Derby, UK

C. ACCIDENT SUMMARY:

On May 31, 2014, about 2140 eastern daylight time, a Gulfstream Aerospace Corporation G-IV, N121JM, operated by Arizin Ventures, LLC, crashed after a rejected takeoff and runway excursion at Laurence G. Hanscom Field (BED), Bedford, Massachusetts. The two pilots, a flight attendant, and four passengers were fatally injured. The airplane was destroyed by impact forces and a postcrash fire. The personal flight, which was destined for Atlantic City International Airport (ACY), Atlantic City, New Jersey, was conducted under the provisions of 14 *Code of Federal Regulations Part 91*. An instrument flight rules flight plan was filed. Night visual meteorological conditions prevailed at the time of the accident.

D. ACCIDENT SITE:

Figure 1: Accident Scene from End of Overrun Pavement

Figure 2: Accident Scene from Just Prior to ILS Fixture

Figure 3: Wreckage Site

Note: All measurements in this section are approximations based on a survey done by the Massachusetts State Police Collision Analysis and Reconstruction Section and the document is available in the public docket for this accident.

The airplane was departing from runway 11, a 7,011-foot-long, 150-foot-wide, asphalt runway. The runway was grooved with a smooth overrun area that was approximately 1,020 ft in length. The right main tires' skid marks started approximately 1,440 ft prior to the threshold for runway 29 and continued to just prior to the end of the pavement of the runway 11 overrun area. The left main tires' skid marks started approximately 1,375 ft prior to the threshold for runway 29 and continued to just prior to the end of the pavement of the runway 11 overrun area. There were no visible skid marks from the nose tires. The four main tire skid marks were considered heavy overall, with intermittent lighter and darker tracks consistent with anti-skid operation. The tire skid marks were measured at multiple locations and measured between 5 to 6 inches in width. A runway walk was performed and no debris was identified.

The light fixtures located at the end of the runway overrun pavement that were in the approximate path of the fuselage were knocked down. After the pavement ended the elevation sloped down to a small service road. Approximately 10 ft past the road (estimated 55 ft past the end of the overrun area pavement) were three distinct sets of ruts dug into the grass consistent with the nose, left main, and right main tires (see Figure 1). All ruts started at approximately the same distance from the runway. The main gear ruts continued for approximately 40 feet and then became abruptly shallower. The rut from the left main gear measured 16 inches deep in one of its deeper locations. The nose gear rut was approximately 85 ft long with the lower portion of the nose gear including both tires located at the end of the rut.

Shallow ground scarring remained visible from the ruts to the aircraft wreckage located in a ravine approximately 850 feet from the end of the runway overrun pavement. The debris path from the ruts to the final resting area was along a slight incline. The left main gear door, a 6 ft inboard section of the left flap, and the left main gear assembly were found along the debris path. Three light structures located along the right side of the airplane track were knocked down and broken. Approximately halfway between the first and second light structures the grass to the right of the right main gear track began to turn slightly brown. Just prior to the second light structure, the grass to the left of the left main gear track also began to turn slightly brown. The left and right brown trails of grass continued until the airplane hit the localizer antennas approximately 180 ft prior to the ravine.

The localizer antennas were in the airplane path and were broken and knocked down. Multiple localizer antenna poles were severed approximately 17 to 18 inches above their base. The grass between the localizer antennas and the ravine was blackened and charred (see Figure 2). Just prior to the ravine was a large perimeter fence that was partially knocked down. The airplane was found spanning the width of the ravine with the tail on the airport side, the engines, wings, and fuselage down in the ravine, and the cockpit on the far side of the ravine (see Figure 3). The majority of the airplane was destroyed by fire, with the greatest fire damage occurring near the wing root area of the fuselage. All major portions of the airplane were accounted for at the accident site.

E. FLIGHT DECK:

Figure 4: Forward Fuselage Structure

E.1 General Observations:

The fuselage structure remained mostly intact from the fractured nose cone aft approximately 14 feet (see Figure 4). Aft of this point the fuselage structure was destroyed by fire. The exterior paint forward of the forward entry door remained intact, and around the door and aft of the door the paint was discolored or burned away. The cockpit windows were cracked and discolored, and the left side forward windscreen was cut out by rescuers. The nose cone was broken and partially missing. The bottom of the forward fuselage was deformed at approximately a 45° angle from the nose cone attachment frame extending down and aft approximately 3 feet. Circumferential buckling around the fuselage was noted approximately 6 feet aft of the nose cone attachment frame. The main entry door was partially opened by rescuers by operation of the external door handle prior to the arrival of investigators.

Figure 5: Flight Deck

The interior of the cockpit and door entry way was heavily sooted, melted, or destroyed by fire (see Figure 5). Prior to the occupant recovery efforts, inspection of the cockpit revealed that the throttle levers were in a position between full forward and half throttle. The thrust reverser handles were down (stowed) and the gust lock handle was in the full forward position (gust locks not engaged). The left Control Display Unit (CDU) was out of its slot and laying on top of the pedestal assembly. Later analysis of the CDU found that the screws were not in the secured position.

The following observations were made from the flight deck following occupant recovery efforts but prior to wreckage recovery efforts unless otherwise noted:

- Prior to occupant recovery, the gust lock handle was in the full forward position (gust locks not engaged)
- Prior to occupant recovery, the throttle levers were between the full forward and the half throttle position, with the trailing edge of the lever approximately in line with the mid-point dust cover attachment screws. Following occupant recovery, the left throttle lever could be moved while the right throttle lever could not be moved.
- Prior to occupant recovery, the thrust reverser handles were down (stowed)
- The flight power shutoff valve handle was pulled up (hydraulic power removed from select flight controls)

- The flaps handle was in the 10 degree detent. The flap handle could move easily.
- The speed brake handle was in the full forward position (speed brakes retracted)
- The right fuel shutoff valve (fuel cock) was approximately ½ inch from the top stop (up position is "OPEN")
- The left fuel shutoff valve (fuel cock) was approximately 3/4 inch from the top stop
- Both fire handles were stowed
- The parking brake was stowed
- The nose wheel steering switch cover was down (switch on)
- The landing gear dump valve D-ring was stowed and in the normal position. The landing gear emergency extension T-handle was slightly extended, measuring approximately 2 inches from the top of the T-handle to the nut.
- The flap alternate extension handle was in the "neutral" position. The emergency flap switch was in the "normal" position.
- Both control columns were found in the full forward position. After wreckage recovery, it was discovered that the elevator control cable sector was protruding through the frame at Fuselage Station (FS) 133 causing a jam. The control linkage just forward of the jam was severed by the investigative team and the control columns were then able to move. Once free of the jam, the control columns moved together and when pulled aft would return to the full forward position with some damping action. The fracture surfaces in the structure around the jam at FS 133 were sooted.
- Both control wheels were found rotated to the right**.** Following wreckage recovery it was discovered that an aileron control link at FS 100 on the left side of the aircraft was broken and that both control wheels could be moved left and right and moved together.
- Both sets of rudder pedals were found with the left pedal forward of the right pedal. The left seat rudder pedals were adjusted to approximately the mid-adjustment range. Following wreckage recovery it was found that the rudder pedals were jammed and that the rudder controls infringed on damaged structure at FS 133.
- The heads up display was stowed
- The landing gear handle was down
- The right side oxygen mask was stowed
- The left side oxygen mask was not stowed
- Trim positions could not be determined on scene due to heat damage and/or the presence of heavy soot. Following wreckage recovery, the aileron trim setting was six units of right trim and the rudder trim setting was six units of left trim. The pitch trim setting could not be viewed. The trim control systems are fully mechanical and Gulfstream confirmed that all trim control wheels and trim actuators' positions can be changed by forces acting on the trim control cables.
- The circuit breaker panels were heavily damaged by heat. Multiple circuit breakers appeared to be open and/or deformed.
- Guarded switches on the maintenance panels of the left and right radio racks in the vestibule area were in the guarded position

E.2 Pedestal Switch Panel:

The STALL BARRIER, ANTI SKID, GROUND SPOILER, and THRUST REVERSER EMERGENCY STOW pushbuttons on the pedestal switch panel were removed from the accident aircraft and examined using x-ray to determine their contact position. The pushbuttons are guarded and will stay depressed when pushed in and will remain extended when not depressed.

The STALL BARRIER pushbutton was in the depressed position. The switch will disengage the stall barrier protection if it is in the extended position.

The ANTI SKID pushbutton was in the extended position. In the extended position the switch will disengage the anti-skid system. The tire marks on the runway and the brake pressures recorded on the FDR were consistent with anti-skid operation which would have required that the pushbutton be in the depressed position during the takeoff attempt.

The GROUND SPOILER pushbutton was in the extended position. In the extended position the switch will disengage the automatic ground spoiler system. If the switch had been extended during the takeoff attempt the autothrottles would not have been able to engage. According to the FDR, the autothrottles were engaged during the accident takeoff roll.

The THRUST REVERSER EMERGENCY STOW pushbutton was in the extended position. If the pushbutton is depressed it will initiate an emergency stow of the thrust reversers.

F. FLIGHT CONTROLS AND SYSTEMS:

F.1 Flaps:

Two single-slotted Fowler type flaps provide lift augmentation for takeoff and landing. The flaps extend spanwise along the trailing edge of the wing from just outboard of the fuselage to the ailerons. Each wing flap is of one piece construction operating on four tracks. Power is supplied by a hydraulic motor driving a central gearbox which transmits power through spanwise torque tubes driving mechanical ball-screw actuators (two in each wing) to extend and retract the flaps. Normal hydraulic power is supplied by the combined hydraulic system. The flap control handle is located on the right side of the control pedestal just outboard of the gust lock handle. The flaps have four positions: 0°, 10°, 20°, and 39°.

The left wing inboard flap actuator was measured prior to wreckage recovery. The screw measured approximately 21 5/8 inches from the body face to the base of the yoke. This measurement corresponded to a flap position of 20°.

The other actuator drive screws were measured after the wreckage was removed from the ravine. The left wing outboard flap actuator screw measured approximately 17 1/2 inches from the body face to the base of the yoke. The right wing inboard flap actuator drive screw measured approximately 21 1/2 inches from the body face to the base of the yoke. The right outboard flap actuator drive screw measured approximately 17 1/2 inches from the body face to the base of the yoke. All flap measurements approximately corresponded to a flap position of 20°.

F.2 Horizontal Stabilizer and Elevators:

The positioning of the horizontal stabilizer is a direct function of flap position. The stabilizer drive incorporates an irreversible mechanical actuator driven by torque tubes from the flap central gearbox. The horizontal stabilizer actuator was examined following wreckage recovery and measured approximately 22.8 inches from attach bolt to attach bolt. According to Gulfstream this measurement can vary due to aircraft rigging but was within an acceptable range to be consistent with a flaps 20 setting.

Aircraft pitch is controlled by means of conventional type elevators in a T-tail configuration. Each elevator is hinged to the trailing edge of a swept horizontal stabilizer. Control is provided by a dual set of conventional control columns. Movement of the control columns transmits motion through a mechanical linkage (pushrods, cranks and cables) to displace a servo valve causing the hydraulic power boost actuator to move the elevators in the desired direction. Trim is provided by trim tabs on the trailing edge of each elevator by means of a cable operated mechanical screwjack actuator. The elevator travel is 24 ± 1/2° trailing edge up and 12-13° trailing edge down. The elevator position sensor is installed on the elevator boost output crank.

During normal operations, the elevators are driven by a hydraulic power boosted servo actuator (moving body type) located in the tail compartment. The elevator actuator is a tandem arrangement of two double acting balanced cylinders supplied by two separate hydraulic systems (combined and flight hydraulic systems). If all hydraulics should fail, or be shut off via the Flight Power Shut Off Valve, the elevator control reverts to manual operation. Gulfstream stated that flight testing has demonstrated that the airplane can rotate and takeoff successfully if the elevator controls are in manual operation mode. The force required at the column would be dependent on trim input.

The elevator actuator was found on scene and was severely discolored by heat. The actuator was manufactured by Parker Aerospace and was labeled as P/N 87000-5013 BK, S/N 3333GT. Gulfstream and Parker Aerospace both stated that they have no history of hydraulic jamming in an elevator actuator. The elevator actuator load relief bungee forward attach bolt was installed in the correct direction as required by the Aircraft Maintenance Manual, Revision 24, dated April 30, 2013. The load relief bungee spring felt typical when applying a manual force in both directions. There was no gap between the spring end cap and jam nut when the bungee was at rest. X-ray was performed and no anomalies were identified with the internal bungee spring.

Three down springs are mounted on the forward side of the input cable sector and attached to the structure below the sector at FS 761. The purpose is to introduce an approximate 13 pound pull at the control columns. The springs are never disconnected and although the pull is always present it is required only to keep the force gradient positive during low trim speeds. During takeoff, the springs will keep the elevators full trailing edge down until speed is increased enough for aerodynamic loads to pull the elevators to a neutral position which will bring the control columns aft.

F.3 Rudder Actuator, Yaw Damper, and Rudder Limiter:

Yaw control is provided by a conventional single rudder which is hinged to the trailing edge of the vertical fin. Movement of the rudder pedals in the flight deck transmits motion through a mechanical linkage (pushrods, cranks and cables) to displace a control valve in the rudder hydraulic servo actuator causing the actuator to move the rudder in the desired direction. The rudder actuator is a tandem arrangement of two double acting pistons on a common shaft which are contained in one housing. The actuator incorporates a series mode yaw damper that does not provide feedback to the rudder pedals. The yaw damper makes computer controlled adjustments to the rudder position to provide yaw stabilization and turn coordination. Electrical input signals from the autopilot system (yaw damper function) are applied to the coils of the torque motor within the rudder electro hydraulic servo valve which controls the volume and direction of fluid flow. Yaw damper control is limited to 1 1/2° left and 1 1/2° right rudder travel. The design of the rudder system allows for mechanical inputs to be made in addition to the electrical inputs. A rudder trim system varies the position of the rudder surface left or right from neutral by providing inputs directly to the rudder actuator input sector. The rudder position sensor is installed on the rudder boost output crank.[1] Rudder trim authority is 7.5 +/-1° left and 7.5 +/-1° right of neutral. Maximum rudder travel is 22.5° left and 22.5° right from neutral. The rudder actuator label could not be read on scene. Service records recorded in Gulfstream's Computer Maintenance Program (CMP) show that the installed rudder actuator was manufactured by Parker Aerospace and identified as P/N 1159SCH253-15L, S/N 0959GT. There were no manufacturer service bulletins applicable to this serial number and Parker Aerospace had no records of repair for this unit.

A load limiter valve included in the rudder actuator is provided to protect the aircraft tail structure against overload. The load limiter valve limits actuator output load capacity to a maximum of 2100 +/-150 psi regardless of systems supply pressure. This is done by butting the two cylinder differential pressure sensing pistons together at the manifold interface so that the combined load developed by the two pistons will act against a reference spring. When the load limiter valve operates a RUDDER LIMIT light is displayed on the Engine Instrument and Crew Advisory System (EICAS) display. The RUDDER LIMIT light is blue and is considered an advisory message. The actuator load limiter is checked by bottoming the rudder against its stops (no surface load) with the combined and flight hydraulic systems operating. The RUDDER LIMIT message on EICAS will come on instantly upon pressure limiting within the actuator. The rudder limiter may also activate if the rudder is commanded to move by a manual (rudder pedals or trim) or electrical (yaw damper) input while rudder movement is restricted by an external force such as an engaged gust lock mechanism. The CVR revealed that the crew noted the occurrence of the RUDDER LIMIT light during the final turn onto the runway. The FDR revealed that at no time during the accident power cycle (prior to the airplane departing the end of the overrun pavement) did the rudder position move more than 1.05° left or more than 1.17° right of neutral.

If all hydraulics should fail or be shut off to the rudder actuator, the rudder control is designed to revert to manual operation. Without hydraulic pressure available for transfer valve operation the

[1] The rudder actuator, rudder surface control linkage, and rudder gust lock hook engagement pin are also connected to the rudder boost output crank.

yaw damper system cannot operate. The yaw damper system is designed to disengage when the actuator does not respond to yaw damper inputs. In addition, if there is no hydraulic pressure available to the rudder actuator, the system is designed so that there will be no effective rudder trim capability.

F.4 Roll Control and Ground Spoiler Systems:

Roll control about the longitudinal axis is provided by both ailerons and spoilers located on each wing. These control surfaces are positioned by operating the pilot or copilot control wheel or by the autopilot. There are three spoilers located on the upper aft area of each wing and hinged near the rear beam. The inboard spoiler is primarily a ground spoiler. It is also utilized as a speed brake in conjunction with the flight spoilers when speed brakes are deployed. The two outboard spoilers work in conjunction with the ailerons and respond to the aileron up movement only. All six spoilers act as speed brakes through use of a mechanical mixer and as ground spoilers during landing roll out. Aileron travel is 10 +/- 1° down and 10 +/- 1° up. Maximum flight spoiler travel for roll control is 26 +/-2° up from neutral. Maximum ground spoiler and flight spoiler travel for speed brake control is 26 +/-2° up from neutral. When armed by the ground spoiler arming switch located on the control pedestal, all spoilers move up to the 55° position upon aircraft touchdown. Trim is accomplished by means of a tab on the left aileron. The aileron and spoiler position sensors are located at the control surfaces.

Both the pilot and copilot control wheels are mechanically connected. Displacing either of the control wheels will move push-pull rods, bellcranks and cables to operate a control valve on the aileron servo actuators. Each aileron servo actuator is a tandem arrangement of two double acting balanced cylinders supplied by separate hydraulic systems. The actuators operate at 3000 psi. If all hydraulics should fail or be shut off, the aileron controls will revert to manual operation. The aileron actuators were manufactured by Parker Aerospace. The left aileron actuator was labeled as P/N 87000-5007 BK, S/N 3326GT; the right aileron actuator was labeled as P/N 87000-5032, S/N 1014CD. Gulfstream and Parker Aerospace both stated that they have no history of hydraulic jamming in an aileron actuator. The right aileron load relief bungee remained attached to the actuator after the actuator was removed from the aircraft. The spring felt typical when applying a manual force in both directions. There was no gap between the spring end cap and jam nut when the bungee was at rest. The left aileron load relief bungee did not remain attached to the actuator after actuator removal and therefore was not inspected.

The flight spoilers can be used as speedbrakes to slow the aircraft down or as roll spoilers to assist in controlling the aircraft. The speedbrake input originates at the speedbrake handle located on the center pedestal while spoiler roll input comes from the pilot or copilot control wheels. One flight spoiler actuator is located in each wing. Each flight spoiler actuator is a tandem arrangement of two double acting balanced cylinders supplied by separate hydraulic systems. The actuators are normally extended, which holds the flight spoiler control surfaces down flush to the wing. If all hydraulic systems fail or are shut off manually, the flight spoiler actuators become inoperative although roll control will still be maintained by the aileron system upon manual reversion. When the flight spoiler actuators lose hydraulic pressure, the flight spoiler control surfaces can be moved by external forces such as aerodynamic loads. Testing on a similar aircraft demonstrated that when hydraulics were removed from the flight spoiler actuators with flaps at 20° and at a speed of 165 kts, the flight spoiler control surfaces would "float" up approximately 3° due to aerodynamic loading.

One ground spoiler actuator is located in each wing. Each actuator is a tandem arrangement of two double acting cylinders supplied by two separate hydraulic systems. To operate the ground spoilers, the following conditions must be met:

- Power must be applied to the left main dc bus.
- Aircraft weight on wheels (nutcracker relays in ground configuration) or aircraft wheel antiskid generator spool up speed of 60 - 70 mph (signal from one wheel on each strut required).
- Both throttle levers must be at ground idle.
- Throttle lever relays energized.
- GND SPLR OFF/ARMED switch in the ARMED position.

The ground spoiler actuators are normally extended, which holds the ground spoilers in the stowed (down) position. If all hydraulics should fail or be shut off, the ground spoiler actuators become inoperative and the ground spoiler control surfaces can be moved by external forces. Testing on a similar aircraft demonstrated that when hydraulics were removed from the ground spoiler actuators with flaps at 20° and at a speed of 165 kts, the ground spoiler control surfaces would "float" up approximately 2° to 3° due to aerodynamic loading.

The CVR recorded a triple chime audible alert that occurred at approximately time 21:40:16. A ground spoiler warning is designed to alert if the ground spoilers do not operate when commanded. Review of the FDR revealed that at this time the throttles were reduced to idle and the conditions to automatically operate the ground spoilers as described above were likely met; however, the spoiler surfaces did not move beyond 3°. The warning was not recorded on the FDR. The sample rate for the warning discrete is 1 hz. The crew may select the warning inhibit feature during takeoff. The warning inhibit select switch will disable advisory aural alerts (single chime), the master caution light, and master caution aural alerts (double chime), but will not inhibit master warning aural alerts (triple chime) or the illumination of the master warning light. The associated EICAS messages are not inhibited. Inhibited alerts will not be recorded on the FDR.

F.5 Flight Power Shutoff Valve:

The flight power shutoff system provides a means to manually shut off hydraulic pressure to the flight control system actuators. This system is an added safety feature that can be used in the event of a hydraulic actuator malfunction in a primary flight control. The ailerons, elevator, and rudder are designed to revert to manual operation when hydraulics are removed.

This system is operated by the flight power shutoff T-handle, located on the aft portion of the pedestal (see Figure 6). When the T-handle is pulled up, it operates a shutoff valve which shuts off hydraulic pressure to the following actuators: ailerons, flight spoilers, ground spoilers, elevator, and rudder. Returning the flight power shutoff T-handle to its normal position will return the flight power shutoff valve to the open position.

The valve is a two section, two position (open or closed), manually operated rotary valve. The two sections of the valve are mounted in a single body. The sections of the valve are mechanically connected, but hydraulically separate. The valve is rotated by means of a crank splined to the valve shaft and is connected by a sliding cable, which is routed under the floorboards to the flight power shutoff T-handle. One section of the valve controls the application of hydraulic pressure from the combined hydraulic system to the flight control system actuators. The other section of the valve controls the application of hydraulic pressure from the flight hydraulic system to the flight control system actuators.

The flight power shutoff T-handle was found in the up position on the accident aircraft. The flight power shutoff valve was also located in the wreckage after removal from the initial wreckage location. The input lever was forward indicating the valve was in by-pass mode (hydraulic power disabled to the flight controls listed above). The lever moved with little applied force. The sliding cable was found severed in the wreckage.

Review of the FDR showed that at approximately time 21:40:05 the right and left ground spoilers and the right and left inboard flight spoilers began to "float" up to between 1° and 3°. Also at this time both left and right aileron surfaces moved in an upward direction. During normal operation the ailerons will move in opposite directions; however, Gulfstream reported that when hydraulics are removed from the ailerons that both the left and right aileron surfaces can "float" due to aerodynamic loads. The FDR data further showed that at approximately FDR time 21:40:06 the yaw damper disengaged. The flight power shutoff valve can affect the operation of each of these systems.

F.6 TLA/PLA/EPR:

The throttle lever angle (TLA) is measured directly at the throttle levers and is not recorded on the FDR. The TLA can be measured by using a protractor fitted to the throttle control assembly. The power lever angle (PLA) is measured at the engine by a position transducer and is recorded by the FDR. It records the input lever position at the engine fuel control. While these values are similar near idle, they diverge as power is increased. The PLA and TLA relationship may also vary slightly from aircraft to aircraft. From an informal sample of 4 aircraft, with the throttles placed between about 18° and 23°, the PLA was on average about 4° lower than the corresponding TLA. The maximum difference among the test articles was 6.1° (PLA lower) and the minimum difference was 2.8° (PLA lower).

PLA commands engine power. The higher the PLA the more thrust the engine will produce. One way to measure engine performance is by monitoring the engine pressure ratio (EPR). The PLA/EPR relationship will vary slightly from engine to engine. EPR is ambient air temperature dependent in that the colder the temperature, the greater the EPR will be for a given PLA. According to Gulfstream, a G-IV can typically achieve 1.59 EPR (minimum EPR for a Flex takeoff) with a PLA of 20° if the ambient air temperature is approximately -5°C or with a PLA of 25° if the ambient air temperature is approximately 15°C.

According to the GIV-SP[2] Airplane Flight Manual (AFM) performance data, the target EPRs for a FLEX takeoff and a maximum thrust (MIN EPR) takeoff for the accident aircraft configuration, runway length, and environmental conditions would have been 1.59 and 1.70 respectively. According to the FDR, at time 21:39:46, a maximum EPR value of 1.617 for the left engine and 1.614 for the right engine was achieved with autothrottle engaged and the aircraft at 63 kts calibrated airspeed, but the EPR then reduced. At around 70kts, the actual EPR was 1.54 for the left engine and 1.53 for the right engine. The Performance Computer will not calculate any EPR target that is below 1.59; however, it is possible for the crew to input a target EPR manually below the 1.59 minimum. The AFM states that "To ensure that takeoff configuration warnings are not inhibited, Flex power setting must not reduce EPR below 1.56." When either TLA is greater than 14° from idle (1.56 EPR was determined to be a conservative value to ensure the TLA is above 14°), a switch in the pedestal assembly enables the takeoff configuration warning system for flap and speedbrake handle positions. According to the FDR, the flaps setting was appropriate for takeoff and the spoiler surfaces were stowed for the initial part of the takeoff roll, and therefore a configuration warning would not be expected regardless of the TLA.

F.7 Gust Lock System:

F.7.1 General Gust Lock Overview and Findings:

The gust lock system is a ground safety system that locks the ailerons, elevators and rudder to protect them against wind gust loads. The locks are mechanical hooks located on or near the control surface, except in the aileron system which is located in the forward fuselage under the floor. They are operated by a two-position red painted gust lock handle on the right side of the control pedestal (see Figure 6). Locked in this manner, the control surfaces are capable of withstanding wind gusts of 60 mph. The gust lock system is interconnected with the throttle levers to prevent advancing the throttle levers beyond a minimum power setting with the gust lock engaged.

[2] With aircraft serial number 1214, GAC introduced modifications into production aircraft that provided for improved braking capability and increased landing weight. Airplanes with these modifications (such as N121JM) are designated "Special Performance" or GIV-SP airplanes, as a marketing term or descriptor. The modifications are available to aircraft prior to #1214 via Aircraft Service Change (ASC) 190.

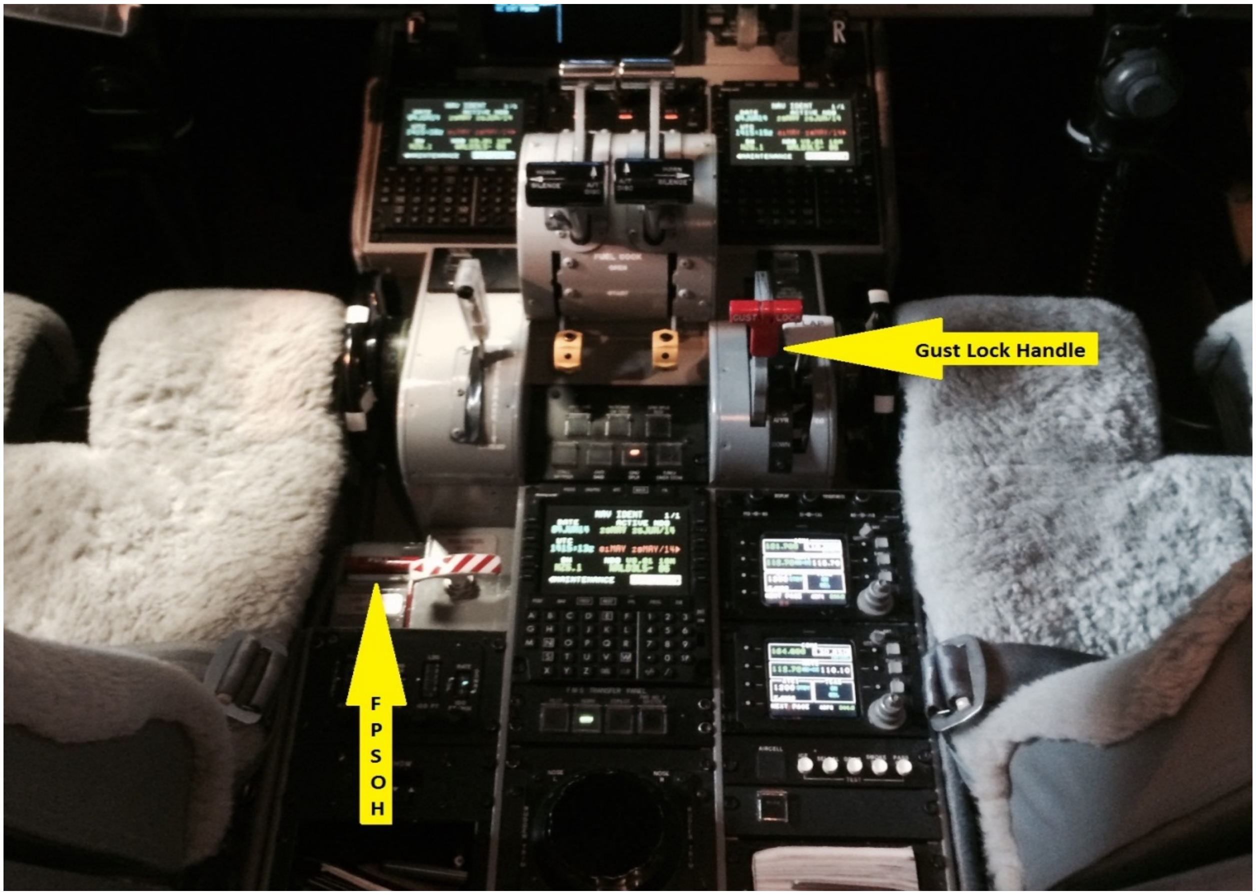

Figure 6: Cockpit Example with Arrows Referencing the Flight Power Shutoff Handle (FPSOH) in the Stowed Position and the Gust Lock Handle in the Up and Latched Position

The gust lock system is completely mechanical, consisting of pushrods[3], cables, cranks, and pulleys. The ailerons are locked in the neutral position (0 +/- 1.5°), the rudder is locked in the neutral position (0 +/- 1.5° but revised to 0 +/- 0.25° per gust lock installation drawing 1159C20005, Rev G, EO H9 dated 8-28-02), and the elevators are locked in the trailing edge down position (13 + 0 /- 1° trailing edge down). The "GUST LOCK SYSTEM – RIGGING PROCEDURE" in the Airplane Maintenance Manual (AMM) (Section 27-05-00, Step 1.B.(25)) states that "With gust lock system engaged, a small amount of movement can be made by hand at all control surfaces. Also, the controls in cockpit can be moved a small amount with a spongy feeling in lateral control wheel." The "GUST LOCK SYSTEM – OPERATIONAL TEST" in the AMM (Section 27-05-00, Step 2.A.(5)) has two notes that state "Note: The aileron surfaces will have approximately 0.75 inches of spongy freeplay with the gust lock engaged.[4] This is because the gust lock latch is "upstream" of the aileron actuators" and "Note: Although the rudder surface gust lock may be engaged and holding the rudder surface fixed, it will still be possible to move the rudder pedals several inches under force against the rudder artificial-feel bungee. This is normal behavior and is due to the clearance between the input and output sectors at the rudder actuator."

[3] With respect to the gust lock and throttle systems, "pushrod" is synonymous with "control rod".
[4] Gulfstream stated that this note was intended to read "The aileron surfaces will have approximately +/- 0.75 inches of spongy freeplay with the gust lock engaged."

Each gust lock consists of a mechanical hook, springs, and a bungee rod. The aileron gust lock mechanism is located at FS 283 below the cabin flooring. The elevator and rudder gust lock mechanism is located at FS 775 in the tail compartment. The elevator and rudder mechanism consists of a single torque tube with one input but independent outputs to each gust lock hook. Moving the gust lock handle transmits motion to the gust lock hooks through a cable.

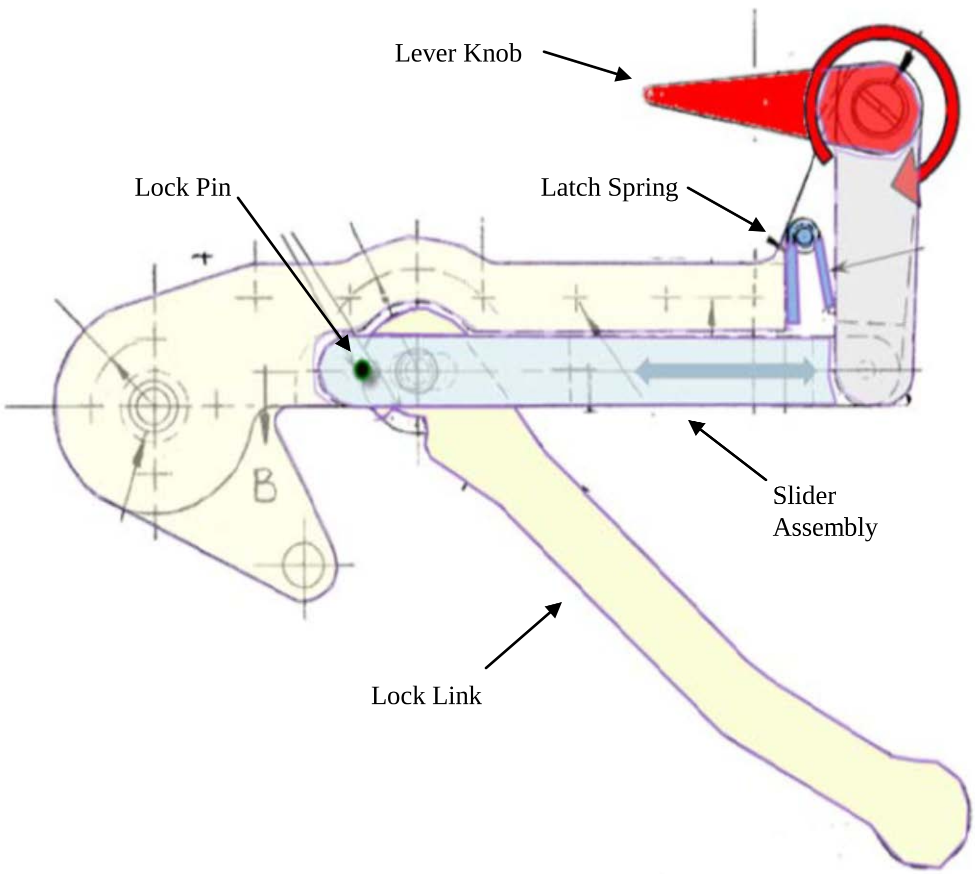

Figure 7: Gust Lock Handle Diagram

The gust lock handle is a two-position lever located on the right hand side of the control pedestal that can be in either an up/aft position (gust lock system locked) or a down/forward position (gust lock system unlocked) (see Figures 6 and 7). Moving the handle aft to the up position locks the control surfaces. Moving the handle forward to the down position releases the gust lock hooks and unlocks the control surfaces. The gust lock handle latches internally in both the down and up positions. A spring latch operated by the lever knob must be unlocked before the handle can be moved out of either latched position. The gust lock handle has 87 º of rotation from the down position to the up position (lock pin in center of the lock pin slot).

Two return springs are located on the aileron mechanism and two return springs are located on the elevator and rudder mechanism. The return springs provide a constant force acting to unlock each mechanism and pull the handle to the down position. The gust lock handle is used to overcome the force of the springs to move the gust lock hooks into the locked position and the gust lock handle latching feature is designed to counter this force when in the up and latched position. The gust lock handle latch consists of a spring loaded slider containing a lock pin that engages a detent in the lock link in both the up and down positions.

The gust lock handle was initially found in the down position. The handle was sooted and the primer was charred. The spring action of the latch was not smooth and the spring was not consistently returning the latch mechanism to the lock position when the lever knob was released. X-ray analysis later showed that the latch spring was intact.

Figure 8: Lock Pin Stub from Right Side of Slider Assembly

Following wreckage recovery, the gust lock handle was moved full up and then manually moved down until a resistance was met (the forces normally provided by the return springs were absent due to the severed control cables). This resistance was thought to be the lock pin slot engaging with the lock pin. It was noted that with increased force the handle could be pushed past the lock pin, and it was later discovered that the pin was severed and only stubs remained on both sides of the slider assembly. NTSB lab analysis revealed that the pin material remaining on the right side of the slider assembly as installed on the aircraft was significantly deformed with smearing damage having pushed the pin material in the direction consistent with the lever going from the up position to the down position (see Figure 8). The material remaining on the left side of the slider assembly was small enough to be inside the pin hole. The middle section of the pin was never found. By design, the pin will either free float in the restraint holes in the slider assembly or have a slight interference fit. During lab analysis, the stub from the left side fell out of the slider assembly.

Materials analysis discovered that the pin material had a hardness value of 30 HRC. Gulfstream converted the hardness value to an equivalent tensile strength of 138ksi. Using this value and assuming a 20 lb load acting on the pin from the gust lock return springs, they determined that a 68 lb load applied to the throttle lever, a 119 lb load applied to the gust lock lever, or a cable load of 211 lb applied to the sector assembly control cable sheaves (from the gust lock cables, throttle cables, or combination of both) would have resulted in a failure of the lock pin. The gust lock handle assembly drawing calls out for the pin to be made from "Drill Rod". See the Materials Laboratory Factual Report located in the public docket for this accident for detailed information regarding the examination of the gust lock handle and lock pin.

A mechanical interlock incorporated in the sector assembly below the pedestal is a feature intended to limit throttle movement to prevent aircraft takeoff with the gust lock handle in the up position. The sector assembly consists of control cable sheaves mounted on a common axis, which transmit incoming motion of the throttle levers, fuel cocks, gust lock, and speed brake controls via pushrods to the appropriate control system cables. Per Gulfstream Specification 1159SCF451, Rev K, "An interlocking device operated by the gust lock in the locked position shall prevent advancing of either throttle beyond 6 +/- 1° from the idle position." Force applied to advance either or both throttle levers cannot override the interlocks. The interlock mechanism consists of stops on the gust lock sheave which interfaces with corresponding stops on the left and right throttle lever sheaves. These stops are positioned on the sheaves to allow full throttle movement with the gust lock handle down, and restricted rotation above idle at the throttle sheaves with the gust lock handle in the up and latched position. See Figures 9 and 10.[5]

[5] When referencing the figures, note that the term "power lever" is synonymous with "throttle lever".

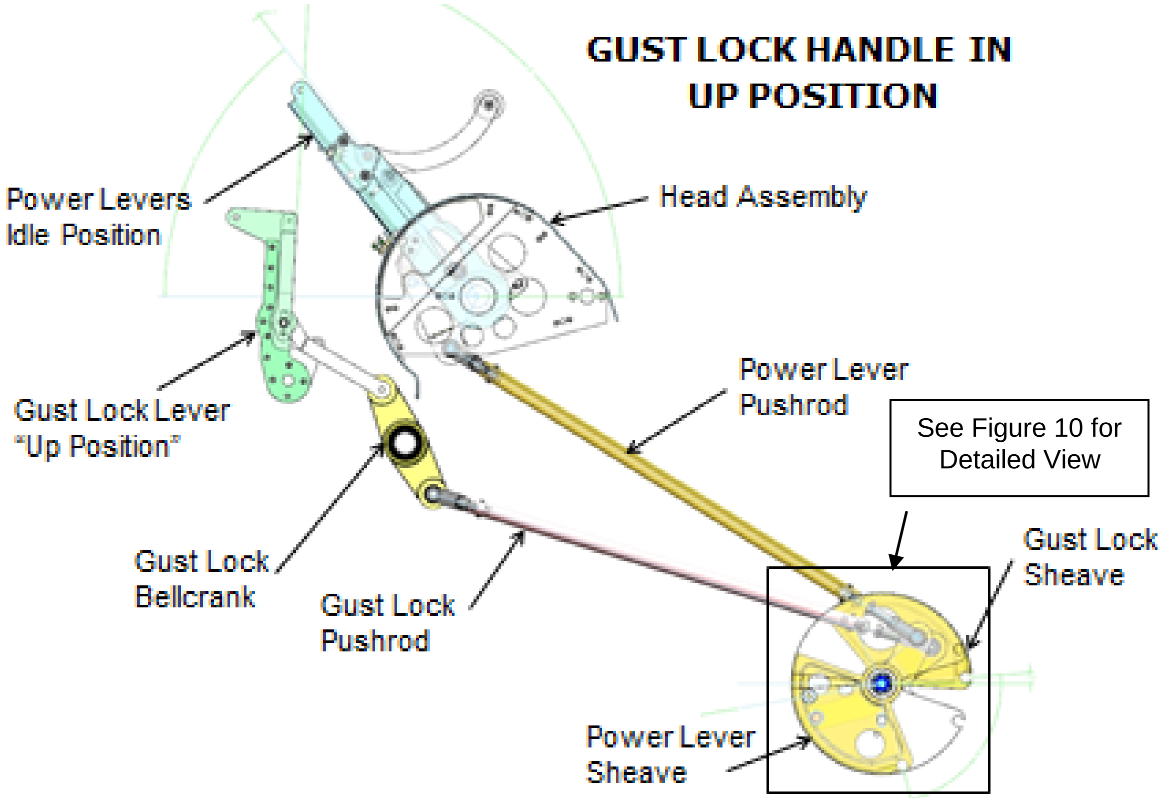

Figure 9: Gust Lock/Throttle Lever Interlock Mechanism

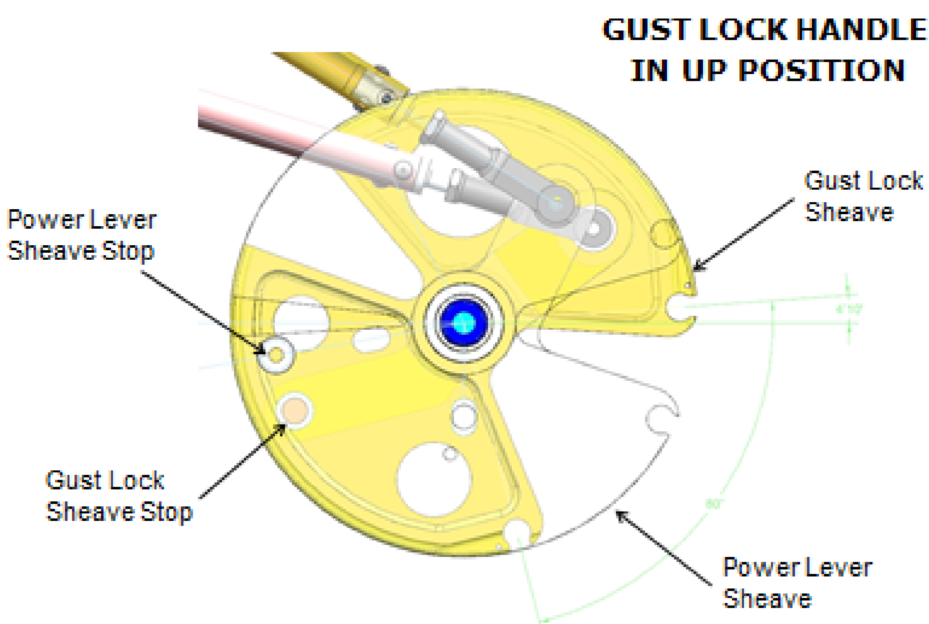

Figure 10: Gust Lock and Throttle Lever Sheave Assemblies

The G-IV pedestal assembly includes the interlock mechanism and is a modified version of the G-II/III pedestal assembly. Due to some aspects of the modification, the G-IV interlock design could allow for as much as approximately 5 additional degrees of throttle lever movement with the gust lock lever up and latched as compared to its predecessor (See System Safety and Certification Group Chairman's Factual Report in the public docket for this accident for further details). Rockwell Collins (the pedestal manufacturer) stated that the interlock mechanism provides for approximately 6.9° of throttle lever movement nominally when the gust lock lever is in the up position; however, using the worst case manufacturing tolerance stack up and kinematic effects the thrust levers could move as much as 23°**.** If incorrect rigging procedures were followed (see Section F.7.3) in addition to the worst case manufacturing tolerance stackup and kinematic effects, Rockwell found that the thrust levers could move as much as approximately 26° when the gust lock handle is in the up and latched position. Gulfstream performed an informal technical evaluation of 9 in-service G-IV's and found that with the gust lock handle in the up and latched position, the forward throttle lever movement varied from 18.2° to 24.2° from the levers' full aft positions. Gulfstream also stated that they have not received any service reports of excessive throttle movement from the G-IV fleet. On the accident airplane, the gust lock handle was moved full aft and then manually moved forward until a resistance was met (the forces normally provided by the return springs were absent due to the severed control cables). This resistance was thought to be the lock pin slot engaging with the lock pin. At this position, the throttle levers could be moved forward approximately 27° from their full aft positions (at the throttle levers' internal stops) before the interlock mechanism restricted further movement**.** As previously stated, it was later discovered that the internal lock pin in the gust lock handle was severed and only stubs remained on both sides of the slider assembly. These stubs provided enough contact force to prevent gust lock handle travel to the down position using light to medium force. As greater force was used the handle could move further forward and eventually pushed through the pin stubs to the down position. During subsequent testing on the accident aircraft, a similar gust lock handle with an intact lock pin was installed. The throttle levers could then move forward approximately 22° from their full aft positions (at the throttle levers' internal stops) with the gust lock handle in the up and latched position. Although not exactly linear, it can be roughly estimated that one degree of forward gust lock lever travel will result in approximately one more degree of throttle lever movement allowed by the gust lock/throttle lever interlock mechanism.

It is possible for the gust lock handle to be moved to a position that is between the up and down positions with the gust lock mechanisms remaining engaged. This intermediate handle condition can occur if the gust lock handle transitions from the up position towards the down position but the gust lock hooks remain engaged due to contact forces at the gust lock hook/pin interface. Contact forces at the gust lock hook/pin interface can be caused by forces acting on a flight control system while gust locks are engaged. Examples of possible forces include hydraulic loading from a flight control actuator, manual loading from pilot input, or aerodynamic loading on a flight control surface. Hydraulic loading can occur if the gust lock handle is not released prior to engine start as required by procedures in the AFM. The interlock will allow greater throttle lever movement the further forward the gust lock handle is from the up and latched position. As a single reference point, testing demonstrated that on a similar aircraft all gust locks disengaged when the gust lock handle was approximately 14° forward of the up and latched position. The aileron gust lock hook may disengage just prior to the rudder/elevator gust lock hook as the gust lock handle moves in the forward direction.

Testing on another aircraft showed that with the rudder gust lock hook preloaded using rudder trim input, the gust lock handle could be retained in an intermediate position with all gust lock hooks remaining engaged. With the gust lock handle in this position, the throttle levers could move 41° with a PLA reading of 35° before the interlock would restrict movement. It was further demonstrated on a similar aircraft that 2.5 units of rudder trim input with hydraulics on could create enough contact force to keep the hooks engaged with only return springs acting to pull down the gust lock handle. Gulfstream analysis estimated that at an airspeed of 150 kts with the elevator surface positioned 13° trailing edge down, aerodynamic loading could result in an elevator gust lock hook (if engaged) loading of 785 lb. Assuming a friction for dry steel on steel at the hook pin interface, a crew input force of approximately 189 lbs on a gust lock handle in the up position would be required to disengage the elevator hook at that airspeed.

The "GUST LOCK SYSTEM – OPERATIONAL TEST" procedure in the AMM contains instructions for checking the throttle lever travel while the gust lock handle is in the up and latched position. Section 27-05-00, Step 2.A.(6), requires the maintainer to "Move both levers forward. They should not move more than a minimal amount."

F.7.2 Gust Lock Component Condition:

A section of structure from the FS 775 area was recovered from the wreckage and contained the elevator and rudder gust lock components. The structure was deformed, fractured, sooted, and showed signs of thermal damage. A portion of the structure was later separated to facilitate examination and contained the elevator actuator and the following gust lock components: elevator and rudder gust lock torque tube, two gust lock system return springs, torque tube input crank, elevator gust lock hook bungee, elevator gust lock hook, elevator gust lock engagement pin, and a section of the rudder gust lock hook bungee. The remaining section of structure contained the rudder actuator and the following gust lock components: remaining gust lock rudder input bungee, rudder gust lock crank, rudder gust lock torque tube, rudder gust lock hook, and rudder gust lock engagement pin. The aileron gust lock components were not identified on scene; however, the aileron gust lock sector and gust lock return springs were later found as separated components amongst the relocated wreckage.

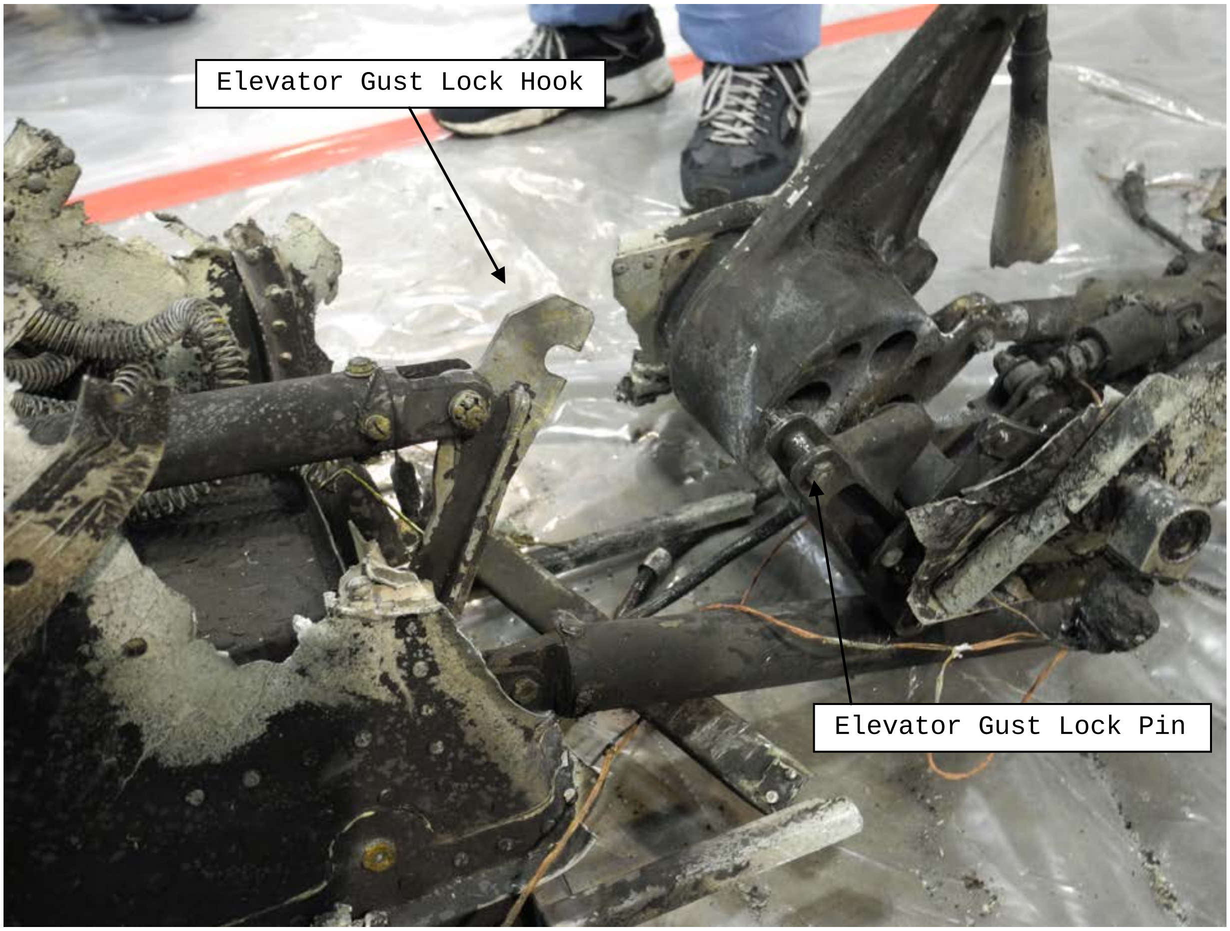

Figure 11: Elevator Gust Lock Mechanism Showing Gust Lock Hook and Gust Lock Pin

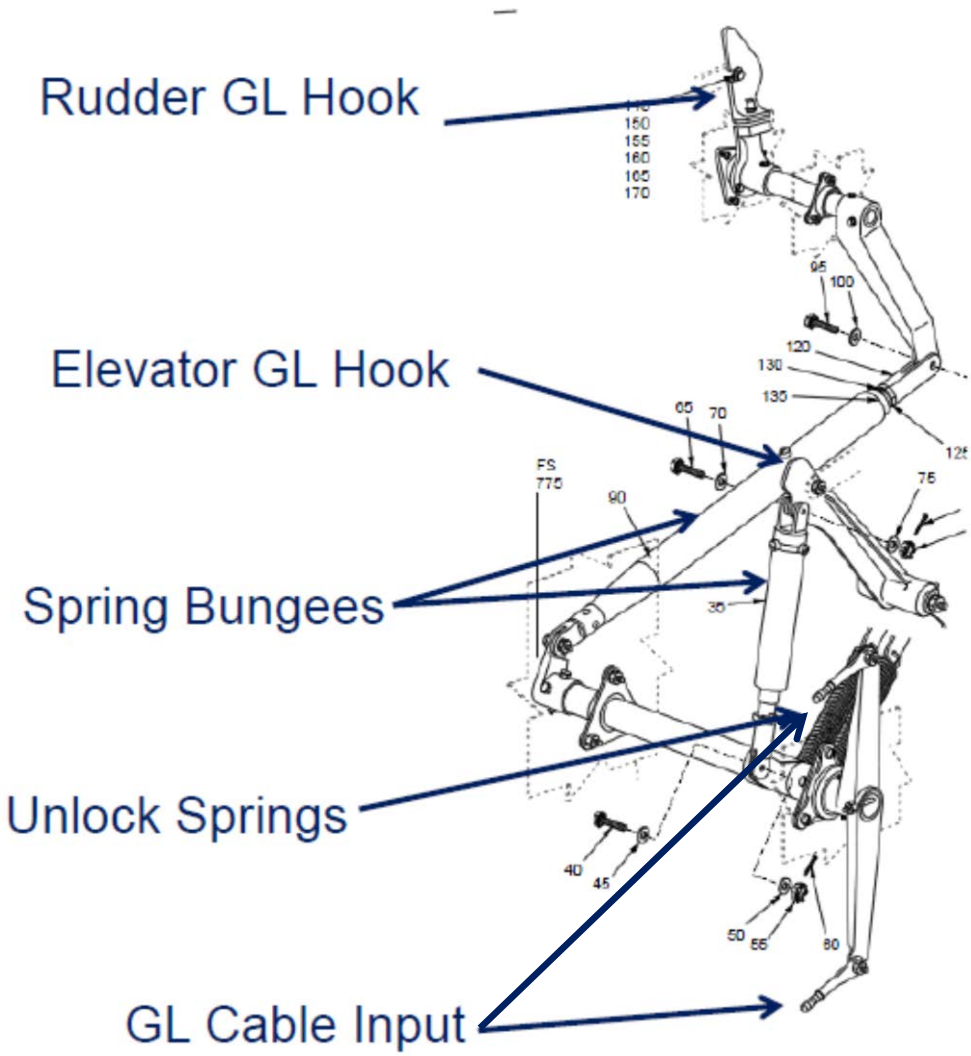

Figure 12: Diagram of the Elevator and Rudder Gust Lock Hook Mechanism

All observations regarding the gust lock components in the FS 775 area were made after the structure was removed from the ravine. The rudder and elevator gust lock hooks were not engaged with the gust lock pins when found. The two rudder/ elevator gust lock return springs were intact and provided return force to the torque tube in the direction to disengage the gust lock hooks. The cable input crank was fractured at both cable attachment lugs and both fracture surfaces contained soot. The elevator gust lock hook bungee moved freely back and forth but without any resistance from the internal spring. The elevator hook remained attached to the bungee and was straight. The inside surface of the elevator lock hook capture slot appeared sooty and/or corroded. The elevator gust lock locking pin attach bracket was intact. The locking pin was mostly sooted but had a small shiny area that spanned the length of the roller on one side. The shape could not be correlated to the shape of the gust lock hook, and it could not be determined if the area was caused by shielding during the fire or by contact after the fire. The soot could be removed easily from the roller.

The rudder gust lock bungee was severed aft of the bungee spring retention screws with one section remaining attached to the torque tube and one attached to the rudder hook crank. The fracture surface appeared shiny and clean. The remaining rudder hook bungee had some return spring action. The rudder lock hook was intact and straight. The rudder locking roller and rudder gust lock hook capture slot surfaces were heavily sooted. The hook lever could be moved and the hook and lock roller did not appear to be aligned. No damage was identified to the structure adjacent to the gust lock hook that could be positively attributed to the gust lock hook being engaged under load.

Gulfstream performed an analysis to determine the weakest links in the gust lock system assuming nominal design values and an ultimate load of 162 lbs applied to an unlatched gust lock handle (lock pin not engaged in detent slot). In the pedestal the weakest link was the handle link (P/N 43083-1015). For the aileron system the gust lock sector P/N 1159CM20506 (root of lug to bungee) was the weakest link. For the elevator system the torque tube P/N 1159C20505-11 (tube that bungee attaches to) was the weakest link. For the rudder system the input crank P/N 1159CM40501 was the weakest link. All components were found intact except for the aileron gust lock sector. This component was found separated from structure, deformed, and the lug to bungee connection was broken.

<u>F.7.3 Gust Lock/Throttle Lever Interlock Rigging:</u>

The gust lock/throttle lever interlock was examined in detail at the wreckage storage facility. Covers and panels were removed from the pedestal. A yellow marker with burn patterns was found between the elevator trim chain and the structure resting above the trim sensor. An orange marker was found on the floor to the right of the sector assembly. A charred piece of folded paper was found on the floor and dated JULY 29, 2012**.** Sun glasses were found on the floor near the aft side of the sector assembly sheaves in the direct vicinity of the left throttle lever sheave and the gust lock sheave. The sheaves rotated freely with the exception of the interference caused by the sun glasses (see section F.7.4). The control rods connected to the sector assembly were straight and secured. A maintenance protractor was installed between the throttle levers. With the gust lock handle in the down position, the throttle levers (as measured from the aft side of the lever) could move freely from approximately 46° (internal aft stop) to 104° (forward stop). By design, the throttle levers have a 58° range of rotation from idle**.**

The throttle control rigging in the pedestal assembly on the accident aircraft was examined using the Gulfstream IV Maintenance Manual, Section 76-00-00 dated Apr 30/11, "Engine Controls – Adjustment/Test", paragraph 1.B.(14), as a guide. There are two procedures in the AMM, one for throttle control head assembly P/N 43087-003/4 and one for P/N 43087-005. The throttle control head assembly data plate for the accident aircraft was labeled as P/N 43087-005, S/N 418, and therefore the -005 procedures were used as reference. The -005 throttle control head assembly includes adjustable aft idle stops consisting of screws with plastic caps covering the screw heads. For reference, a new plastic cap was measured and the spacer portion of the plastic cap stuck out 0.059 inch beyond the screw head. There were no plastic caps or positive evidence of the plastic caps on the throttle lever aft hard stops on the accident airplane. Rockwell Collins reported that the caps are made from a plastic with a temperature rating of 180° F. Per the instruction, a rig pin is inserted through the throttle levers and then the aft stop screws with a plastic cap installed are adjusted to achieve a gap between 0.125 and 0.200 inches from the aft surface of the throttle levers to the forward contact point on the hard stops. Once the aft stop screws are properly adjusted, the throttle levers must be moved aft so that the gap between the levers and the aft stop screws with a plastic cap installed is between 0.000 and 0.030 inches. The throttle lever control rods' lengths are then set to accommodate the throttle levers in this position while a rig pin is inserted through the sector assembly.

A gust lock handle with an intact lock pin was removed from an out of service aircraft and installed on the accident airplane's pedestal to perform the testing. The "up and latched" lock pin slot width on the test handle was measured by Gulfstream at 0.190 inch while the same lock pin slot on the accident gust lock handle was measured by the NTSB at 0.188 inch. Both measurements were within drawing specifications. The larger the slot width the more throttle lever travel the gust lock/throttle lever interlock will allow when the gust lock handle is up and latched.

A rig pin was placed in the throttle head assembly through the throttle levers. The gap between the screw heads without a plastic cap installed and the aft edge of the throttle levers measured approximately 0.25 inch using available tooling (ruler with increments of 1/64 inch).[6] A rig pin could be installed through all cable sheaves in the sector assembly at the same time that a rig pin was installed through the throttle control head assembly.[7] The hard stop adjustment screws slotted ends on the accident throttle control head were flush with the aft surface of the hard stop fitting and met the requirements of Figure 501 referenced in the AMM procedure. The gust lock control rod was later removed from the aircraft without adjusting the length of the control rod and a detailed inspection was performed at Gulfstream. The control rod measured 18.25 inches long (end to end) and 17.750 inches long (center of eye end to center of eye end).

Additional testing was performed to estimate the available travel of the throttle levers while the gust lock handle with an intact lock pin was in the gust lock engaged position. With all rig pins removed and the gust lock handle up and latched and against the lock pin, the throttle levers could freely move forward approximately 22° from their full aft position (contacting the internal stop). The left thrust reverser handle could be operated without difficulty when the left throttle lever was pulled to its full aft position. Operation of the right thrust reverser was not attempted when the right throttle lever was pulled to its full aft position. A track is incorporated in the throttle levers that will only allow the thrust reversers to operate when the levers have reached an internal aft stop.

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[6] The throttle head assembly was later removed and placed on a 43083-003 test pedestal assembly at Gulfstream for further observation. This measurement was retaken using digital calipers and measured approximately 0.29 inch. Rockwell Collins confirmed that this dimension should not be affected by moving the throttle head assembly to a new pedestal assembly.

[7] According to Rockwell Collins, based on the most extreme scenarios for manufacturing and backlash tolerance stack up, it is possible to simultaneously insert both the sector and the throttle lever rig pins in a properly rigged -005 throttle control head if the throttle levers are pushed forward to take up all potential backlash. Outside of the extreme tolerance stack up scenarios, both rig pins should not be able to be inserted at the same time if the -005 pedestal assembly rig procedures in the AMM are followed.

The rigging procedures in the Pedestal Installation drawing 1159F40300 that were applicable to aircraft S/N 1399 differ from the AMM procedures referenced above for a -005 throttle control head assembly. Note F2 of the pedestal installation drawing valid at the time of build for S/N 1399 stated "Prior to adjusting and attaching push rods to sectors, position throttle arms to "idle", fuel cock to "start" and gust lock to "off". Place .312 dia rigging pin thru rigging hole in sector housing and all sectors." Engineer Order J5 was added in October 2001 (after the build date for S/N 1399) and stated:

> "F2 = For apl 1000-1472 adjust and attach power levers, fuel cock and gust lock pushrods as follows:
> 1) Position power sectors in the rig position (idle).
> 2) Position fuel cock sectors in the rig position (start).
> 3) Position gust lock sector in the rig position (off).
> 4) Insert .312 diameter sector rigging pin thru rigging hole in sector housing and all sectors.
> 5) Position power levers in idle, reverse levers in stow, fuel cock levers in start, and gust lock lever in off positon.
> 6) Insert .312 diameter head assy rig pin thru the control head assy.
> 7) Insert .312 diameter gust lock rig pin thru pedestal and gust lock lever.
> 8) Adjust length of pushrods and attach to sectors.
> 9) Remove all rig pins and check controls for proper operation."

When the aircraft is properly rigged, the lower sector assembly rig pin sets the engine idle position for PLA and the throttle pushrods are adjusted to set the throttle lever position to idle. Because the relationship between the throttle sheaves' interlock stops and the gust lock sheave interlock stops remains the same when the lower sector assembly rig pin is installed, the maximum PLA that the gust lock interlock will allow does not change regardless of where the throttle levers are rigged to idle (aft stop or throttle lever rig point). The maximum TLA that can be achieved will change dependent on where the throttle levers are rigged to idle. The further forward the throttle levers are when rigged to idle, the greater the TLA will be when the gust lock interlock engages, assuming a constant TLA measurement origin. The extra aft throttle lever travel will result in a negative PLA value from the rigged idle position aft to the position where the internal throttle lever stop or external hard stops are contacted. The FDR revealed that at various times during the accident power cycle when the throttles were expected to be pulled back to a full aft position, the left PLA was approximately -3° and the right PLA was approximately 0°.

<u>F.7.4 Sun Glasses - Foreign Object Debris Testing:</u>

During initial pedestal rigging tests at the wreckage storage facility, a pair of sunglasses was found inside the accident airplane's pedestal. The sunglasses impeded the gust lock sheave movement, and the gust lock handle travel was stopped before reaching the full down position. The glasses were charred. The left lens was found separated from the frame and broken prior to rigging tests at the storage facility being performed. Pieces of the lens were found in various parts of the pedestal floor and the edges of the broken pieces were sooted.

Figure 13: Sunglasses as Found on the Pedestal Assembly Floor

Additional testing using the sunglasses that were found and a new pair of similar sunglasses was performed to determine the effects of the sunglasses in that area. The accident sunglasses were placed in a similar position as initially found in the wreckage (see Figure 13). The left throttle lever was moved to idle and the gust lock handle was in the up and latched position. The left lens of the glasses was missing, and the lens frame was around the left throttle sheave with the frame extending towards the adjacent gust lock sheave. When the gust lock lever was moved down the gust lock sheave would be impeded by the frame of the glasses. Following multiple tests, the greatest restriction observed with this configuration allowed the gust lock handle to move down approximately 15° from the up and latched position. Any forward movement of the left throttle lever rotated the throttle sheave such that the glasses were moved to a lower position. This allowed the gust lock sheave to have more range of travel before contacting the glasses. With the left throttle lever moved 25° forward of idle, the most restrictive test case observed allowed for the gust lock handle to have 62° of downward movement from the up and latched position.

Using the accident sunglasses as well as a representative new pair of sunglasses multiple scenarios were run to try to restrict the gust lock sheave in various ways. The testing revealed that there were multiple ways that a pair of sunglasses could partially restrict the movement of the gust lock sheave. For example, the glasses were placed on top of the gust lock sheave and wedged between the sheave and surrounding structure. In this configuration the gust lock handle could move down approximately 8° from the up and latched position before becoming restricted.

F.7.5 Gust Lock Handle Reliability:

Gulfstream conducted a historical search for gust lock handle failures on G-IV aircraft. Of the ten records that were identified, the failure descriptions of four were labeled as "Broken", two as "Failure To Operate", two as "Binding Or Stuck", one as "Does Not Meet Specification", and one was unlabeled. The gust lock handle lock pin condition was not specifically mentioned in any of the report descriptions.

In March/April of 1991, Gulfstream IV Service News bulletin No. 91-3 was issued after a G-IV operator reported that during the ground checkout of their autothrottle servo, a loud bang was heard inside the pedestal. It further stated that, "The checkout followed completion of a Phase II modification and was being accomplished without the engines operating and with the gust lock engaged. A visual check of the pedestal area revealed that the gust lock handle had released and could not be reset. Further inspection revealed that the shear pin had broken."[8] The bulletin recommended that the gust lock be released anytime the autothrottle is being operated.

F.8 Autothrottle:

Each throttle is driven independently by a separate SM-810 servo that is connected to the throttle control cables at location FS 257.5 – 287.5. Each servo consists of a servo motor, gear train, electrical engage clutch, mechanical slip clutch, and a drum grooved for aircraft cable. When the autothrottle is engaged, the PZ-800 Performance Computer outputs analog rate commands to the servos to control thrust. The maximum load that each autothrottle servo can input into the throttle cable is 60 lbs.

The autothrottle can be engaged during takeoff after a minimum 1.17 EPR has been achieved. When the autothrottle engages, the throttles are automatically advanced to a required takeoff engine setting based on a full-thrust (MIN EPR) takeoff or a reduced-thrust (FLEX EPR) takeoff. A target EPR input requires either an operating performance function and a CDU selection by the pilot during performance initialization, or a pilot input of EPR command on the manual thrust reference page of the display controller. During takeoff the autothrottle servos are automatically depowered at 60 kts with the clutches remaining engaged (autothrottle hold mode). The autothrottle servos will remain depowered until the hold mode cancels automatically once the aircraft has climbed to 400 feet above ground level.

Section 22-00-00 of the AMM states that the autothrottle can be overridden with a force of 15-32 lbs at the throttle levers during functional checks of the autothrottle system. Honeywell (the autothrottle manufacturer) stated that the torque required to slip the autothrottle clutch is between 57.2 to 85.8 in-lbs, which is equivalent to 40 to 60 lbs of pull at the control cable. The torque required to back drive the geartrain and servo motor are higher than that to slip the slip clutch when the autothrottle is in autothrottle hold mode. The system measures an output on the aircraft side of the autothrottle slip clutch. If the servo command and the tach output differ by more than

[8] Gulfstream stated that "shear pin" is not a correct term as the pin is not designed to shear. "Shear pin" in this statement is a reference to what is labeled as "lock pin" in this report.

8 degree/sec the autothrottle will disconnect. The Honeywell Engineering Specification for autothrottle servo installation states that "The mechanical ratio between the throttle handle and the engine power lever should be 1.159 x 1".

Flight testing on a similar aircraft (Reference Gulfstream report GIV-GER-9980) showed that for the test aircraft, when the gust lock lever was in the up and latched position and throttle levers were moved manually until the interlock was contacted, the throttle levers could move 20° from idle and the corresponding maximum PLA reading was 18.75°. On the same aircraft with autothrottles moving the levers, a maximum PLA reading of 21.4° for the left throttle and 22.8° for the right throttle could be achieved before the autothrottle system would disengage and the PLA moved back to 18.7°. Flight testing on a different test aircraft (Reference Gulfstream report GIV-GER-0017) revealed that with the gust lock handle in the up and latched position and when moving the throttles forward with firm force until the interlock is contacted, the autothrottle system achieved an EPR that was 0.05 higher than what was achieved manually and a PLA that was 1.25° higher than what was achieved manually before the autothrottle disconnected.

F.9 Landing Gear:

All three landing gear assemblies were found. A portion of the nose landing gear assembly was found shortly after the ruts in the grass along the debris field. The piston was severed approximately 29 ½ inch from the axle. The tires still had pressure but a pressure measurement was not recorded. The tires were identified as Goodyear 21 x 7.25-10, P/N 217K22-1. The tread appeared normal with no signs of flat spotting or rubber reversion identified. Both wheels spun freely.

The left main gear assembly was located past the nose gear along the debris path. The gear separated from the wing and was still attached to the sponson box. The gear was identified as P/N P500626, S/N MK0303. The wheels spun freely and the strut pressure was released for safety purposes prior to inspection. The inboard brake wear pin measured approximately 1/4 inch and the outboard wear pin measured approximately 3/8 inch extension. The brakes were not pressurized for this reading. The tires were Goodyear H34 x 9.25-18, P/N 349K82-3. The tread appeared normal with no signs of flat spotting or rubber reversion. Both tires were deflated and the inboard tire was split at an approximate 45° angle that spanned the width of the tread. The split measured approximately 9 inches in length.

The right main gear assembly was located underneath the wing root area of the right side of the fuselage. The gear remained partially attached to this location. The gear P/N and S/N could not be identified from the label. Both brake wear pins measured 0.5 inch extension. The brakes were not pressurized for this reading. The tires were Goodyear H34 x 9.25-18, P/N 349K82-3. The tread on both tires appeared normal except for approximately 1/4 of the diameters. The rubber in these sections was reverted and consistent with heat exposure.

F.10 Avionics:

The two Flight Guidance Computers, the two Navigation Flight Management Computers, and the two Performance Flight Management Computers were removed from the aircraft and sent to Honeywell for a download of their Static Random Access Memory (SRAM). The Left Flight Guidance Computer label was partially readable with a part number that ended in -908 and a serial number of 93053609. The labels on the other computers were missing.

All computers were sooted and thermally damaged both externally and internally. Due to the excessive heat damage of the circuit card components and memory holdup batteries during the post-crash fire, no data was available nor able to be extracted from any of the units.

SRAM is a type of memory that uses bistable latching circuitry to store each bit. SRAM is still volatile in that data is eventually lost when the memory is not powered. The design of the units is such that a properly charged holdup battery can maintain the memory for 6 months without depleting. A volt meter was used to determine that in no case was power from the memory holdup batteries reaching the SRAM.

G. ENGINES:

The aircraft had two Rolls-Royce Tay 611-8 engines installed. The Number 1 engine (left side) was S/N 16917. According to the aircraft logbook entry dated May 20, 2014, the engine had acquired 4832.8 hours and 2653 cycles since new. A Mid-Life and 10 Year inspection was performed in Oct 2007. The Number 2 engine (right side) was S/N 16918. According to the aircraft logbook entry dated May 20, 2014, the engine had acquired 4863.6 hours and 2693 cycles since new. A Mid-Life and 10 Year inspection was performed in Nov 2009.

Both engines remained attached to the fuselage. The thrust reverser unit (TRU) buckets on both of the engines were in the closed (stowed) position. The outer nacelles of both engines were severely fire damaged and there was considerable delamination of the composite structure. The exhaust "pastry cutter" mixer nozzles appeared to be relatively clean and undamaged.

The left hand side (outboard) TRU actuator for the Number 1 engine was visible, the cover plate having been consumed in the post-crash fire. The actuator rod position was consistent with the TRU "buckets" being in the stowed position. The cover plate of the right hand side (inboard) TRU actuator for the Number 1 engine was only partially damaged by the post-crash fire and the actuator was not visible. The nacelle intake was destroyed by fire and the anti-ice manifold ring was visible. There was some visible impact damage to the anti-ice manifold towards its right hand side.

On the Number 2 engine, the right hand side (outboard) TRU actuator was visible, the cover plate having been consumed in the post-crash fire. The actuator rod position was consistent with the TRU "buckets" being in the stowed position. The left hand side (inboard) TRU actuator was not visible as it was obscured by the partially consumed cover plate.

Both engines were recovered from the accident site and subjected to further visual examination. The majority of the fire-damaged composite material of the nacelles was cut away and removed. This enabled access to the location of the high pressure shut-off valve (HPSOV) and fuel flow regulator (FFR) on both of the engines.

A visual inspection of the fan assembly on the Number 1 engine was performed. Some minor damage to the leading edge of some of the fan blades was observed. The stage 3 turbine blades were in good condition with no visible physical damage. The HPSOV lever was angled towards the rear of the engine, consistent with the fuel being on. The geared quadrants of the FFR control were observed to be in a position consistent with a high power setting, but not at the full power end stop.

For the Number 2 engine, some of the fan blades were observed to have leading edge damage consistent with 'hard-body' impact. There were also blades that exhibited damage consistent with 'soft-body' impact, having a much more rounded indentation and distortion of the leading edge, but without tearing of the metal. The stage 3 turbine blades appeared in good condition with no visible physical damage. The HPSOV lever angle was towards the rear of the engine, which is consistent with the HPSOV being open and fuel on. The geared quadrants of the FFR control were observed to be in a position consistent with a high power setting, but not at the full power end stop.

No evidence was found that would suggest that either of the engines had experienced a significant or uncontained mechanical event or control malfunction. Review of the engine parameters on the FDR did not reveal any concerns regarding engine performance in response to commands.

H. ACTUATOR EXAMINATIONS:

Representatives from the NTSB, FAA, Gulfstream, and Rockwell Collins met at the Parker Aerospace facility in Irvine, CA on January 13-15, 2015 for the examination of the elevator and both aileron actuators removed from the accident airplane. The following observations were made during the examinations:

H.1 Elevator Actuator:

P/N: 87000-5013 BK
S/N: 3333GT
Assembly Date: 4Q99

Figure 14: Elevator Actuator

The box containing the actuator was removed from a secure locker. The unit was sooted and melted aluminum was adhered to the surface in some locations. The unit measured 15.75 inches from rod end bearing center to tail stock bearing center. A more exact measurement was taken from the face of the body to the end of the piston rod (before the rod end nut) which measured 2.072 inches. According to Parker Aerospace, the latter measurement corresponds to an elevator position on the G-IV of approximately 7° TE down.

The molten aluminum was chipped away and the tubes were removed from the body. The inside of the system 2 pressure and return plugs revealed a substance similar to a solidified fluid. The actuator was put on a test stand. The input lever was stuck in the retract position and 15 lbs of force could not move it. The center of the rig pin hole on the input lever was 0.161 inches from the center of the rig pin hole in the body. The piston would not move with 80 lbs force applied. The control valve cap assembly was removed and the bearing underneath the cap was dry, seized, and corroded. A substance resembling tar was noted on internal components. The system 1 filter assembly was opened and contamination similar to solidified fluid was found inside the housing. This filter also had a black melted substance adhered to a portion of the outer diameter of the mesh. The system 2 filter assembly appeared much cleaner.

Hydraulic pressure was slowly applied to the pressure ports. The piston would not move with both systems pressurized to 500psi. The system 2 bypass valve cap leaked externally with 100 psi of pressure applied. System 1 had no leakage to return while system 2 appeared to be free flowing to return. The actuator was allowed to sit overnight with hydraulic fluid contained within.

The next morning the piston and input lever initially could not be moved. As pressure was applied to system 1, the return dripped fluid at a steady rate until approximately 95 psi, then the drip significantly reduced. While reducing pressure, the drip significantly increased again when the pressure reached approximately 60 psi. For system 2, a steady stream of fluid was going to return at low hydraulic pressure. The flow significantly reduced when the pressure increased to between 110-140 psi. As pressure was decreased, the flow again increased when approximately 50 psi was reached. The increase and decrease in fluid to return is consistent with bypass valve operation.[9] During the flow tests, it was also noticed that the piston moved to the retract position but the input lever remained stuck. Due to the damaged condition of the unit, further functional testing was not performed.

A teardown of the unit was performed. Due to the internal contamination and the stuck state of the input lever, the control valve slide was cut adjacent to the input crank eye bolt to facilitate disassembly. The sleeve was removed and all seals and backup rings appeared intact. The manifold housing contained some seal residue after the sleeve was removed. A substance similar to a solidified fluid was identified on the outer diameter of the sleeve and appeared to be in a greater concentration on the system 1 side. The spool would not move inside the sleeve.

The system 1 and 2 bypass valves were removed. The system 2 bypass valve cap seals were deformed. Both bypass valves appeared to have a substance similar to solidified fluid inside the housing. All valve components including the springs were intact. The damper shaft was removed and both sides of the damper piston assembly were stuck together and could not be separated. Light was visible when looking through the main orifice of the two halves. All damper components appeared intact including the damper shaft. The piston was also removed with no anomalies noted other than the presence of a substance similar to solidified fluid. The manifold body was scoped and the control valve and piston scallop cuts contained a black residue in them. The system 1 bypass valve bore was also heavily contaminated with a solidified fluid and a white residue was present.

The control valve (slide and sleeve together) was cut in half longitudinally using wire EDM. Considerable corrosion and solidified fluid buildup was present on the system 1 side of both components. The system 2 side of both components appeared less contaminated but corrosion was still present. The slide lap left a mirrored image of the slide location on the sleeve inner diameter revealing that it was in a piston retract position. There was a 0.035 inch slide opening in the retract direction by measuring the system 2 return port. The elevator trailing edge moves up as the piston retracts. Many of the flow passage edges in the sleeve were corroded/pitted/or built up with solidified fluid material making inspection difficult. No evidence of chip shear was found on either the spool or slide metering edges.

Parker Aerospace performed a maintenance records search for this serial number. They had no records of return for this unit. AD 2004-21-03, which the FAA issued in 2004 to detect and correct broken damper shafts in some P/N 87000 actuators, was not applicable to this unit.

[9] The bypass valves port fluid directly to return when the hydraulic pressure in the pressure port drops below approximately 60 PSI. This will release any hydraulic lock in the actuator and allow the piston to move freely when the flight control surfaces are operated manually or when the actuator is operating on a single hydraulic system.

H.2 Left Aileron Actuator:

P/N: 87000-5007 BK
S/N: 3326GT
Assembly Date: 4Q99

The box containing the actuator was removed from a secure locker. The unit was sooted and dirty. The unit measured 16 inches from rod end bearing to tail stock bearing. The piston scraper was noted to be cracked.

The rig pin fit through the input lever indicating it was in a neutral position. The input lever could move with a damped feel. The actuator lug on the fixed end was bent and the unit was placed in a vise and straightened. The external tubing was removed and rusty water was noted inside the cavities. The unit was connected to the test bench and hydraulic pressure was applied.

Select tests from the manufacturer's ATP were performed on the actuator to obtain an understanding of the units operation in its post-accident condition. The results of the performed tests can be found in Appendix A. During testing it was observed that at both low and high pressures hydraulic fluid would always freely flow to return on the system 1 side. One possible cause for this behavior is a bypass valve that is stuck in the bypass position. A bypass valve in bypass mode will port hydraulic pressure directly from the pressure port to return. The design of the actuator is such that if one system's bypass valve is stuck in bypass mode, the actuator can still operate as commanded due to the independent function of the system 2 side.

A teardown was performed on the unit. The system 1 cap exposing the input lever was removed and water and sludge was found inside. The system 1 and 2 bypass valves were removed. The system 1 bypass valve contained some contamination while the system 2 bypass valve appeared relatively clean. All components of the bypass valves appeared intact including the springs. The control valve sleeve was removed. The slide was removed from the sleeve and the base metal appeared to have a dirty film that resembled tar or rust. It was noted that the system 2 side appeared to have greater concentrations of the film than the system 1 side. The piston was removed and no anomalies were noted other than some slight contamination. The number 1 filter assembly was removed and was found to be bulging in an outward direction. The filter length measured 1.522 inch. The system 2 filter was removed and no anomalies were identified. The filter length measured 1.532 inch. The damper assembly was removed and all components appeared intact including the damper shaft. The main damping orifice was free of debris. All seals and backup rings were intact. In general, the housing demonstrated some small bits of particulate contamination.

The sleeve inner diameter was cleaned and scoped. Significant corrosion was found inside the sleeve. The sleeve was inspected for evidence of chip shear on all fluid passages and no evidence was found. The slide was inspected for evidence of chip shear on the metering edges of the slide and no evidence was found. The system 1 bypass valve bore in the manifold assembly was scoped. The mechanical condition of the bore looked typical but particulate debris was found.

Parker Aerospace performed a maintenance records search for this serial number. There were no records of return for this unit. AD 2004-21-03, which the FAA issued in 2004 to detect and correct broken damper shafts in some P/N 87000 actuators, was not applicable to this unit.

H.3 Right Aileron Actuator:

P/N: 87000-5032
S/N: 1014CD
Assembly Date: 3Q81

The box containing the actuator was removed from a secure locker. The unit was sooted and dirty. The unit measured 14.25 inches from rod end bearing to tail stock bearing.

The input lever was in an extend position and the rig pin could not fit. The input lever could move with a damped feel. The external tubing was removed and the unit was connected to the test bench. Select tests from the manufacturer's ATP were performed on the actuator to obtain an understanding of the units operation in its post-accident condition. The results of the performed tests can be found in Appendix A.

A teardown was performed on the unit. The system bypass valves and piston were removed with no anomalies noted other than contamination. The control valve sleeve was removed and the slide was separated. The slide input crank eye hole was darkened and a black colored residue was observed. The damper was removed and all components including the damper shaft were intact. The main orifice of the damper was free of debris. All seals and backup rings were intact.

The sleeve inner diameter was cleaned and scoped. A scratch was noted that began 1.261 inches into the bore on the system 1 side in the pressure area. The scratch went towards the system 2 side and then back to the system 1 side to the end of the bore. The scratch also changed positions circumferentially. This would be consistent with damage caused during removal of the spool from the sleeve.

Some evidence of corrosion (little to no depth) was found around the system 1 cylinder extend flow hole, but overall the bore was relatively free of corrosion. The sleeve had many burnishing marks that were not straight in the longitudinal direction. The sleeve was inspected for evidence of chip shear on all fluid passages and no evidence was found. The slide was inspected for evidence of chip shear on the metering edges of the slide and no evidence was found.

Parker Aerospace performed a maintenance records search for this serial number. The unit was returned to Parker Aerospace in September of 2007 for a seal upgrade. Parker Aerospace had no other records of return for this unit. AD 2004-21-03, which the FAA issued in 2004 to detect and correct broken damper shafts in some P/N 87000 actuators, was not applicable to this unit.

I. MAINTENANCE RECORDS:

Aircraft Registration Number: N121JM
Aircraft Serial Number: 1399
Airworthiness Certification: Issued 1/27/2000
Aircraft Registered to: SK Travel LLC
Aircraft Operated by: Arizin Ventures LLC
Airframe Total Time: 4944.7 (per logbook entry dated 05/31/2014)
Airframe Total Cycles: 2745 (per logbook entry dated 05/31/2014)

A log book was found in the aircraft. The log book contained pages from the accident flight back to January 1, 2014. The log sheets were being used to track aircraft and engine time, aircraft fuel load, and VOR checks. The log sheets did not record maintenance defects and no record log of aircraft maintenance was found on the airplane. A hand written maintenance record on notebook paper was identified in the wreckage and was dated May 12, 2014. The paper was a record of the specifics for replacing the #2 Air Data Computer.

SK Travel LLC utilized the Gulfstream personalized Computer Maintenance Program (MyCMP) with scheduled maintenance and component replacement tracking. This database contains only those records that were reported by the operator or other entities on the operator's behalf. All service records that were reported to MyCMP were reviewed. Select Federal Aviation Administration records, Gulfstream service facilities' records, and engine service records were also reviewed. Most record sets were reviewed for the year prior to the accident. The aircraft was placed on the Manufacturer's Recommended Inspection Program on June 20, 2007.

The following relevant records were identified:

- The last inspection was completed on 9/20/2013 by Gulfstream Savannah FAA CRS #GR4R216M. The inspection included hourly, 12 month, 24 month, and 72 month maintenance inspections.
- The last gust lock system operational test was performed on 09/13/2012 (reference MyCMP logbook position 277011) by Gulfstream Savannah FAA CRS #GR4R216M. The gust lock system operational test is not a time or cycle based inspection.
- The last Stall Barrier / Angle of Attack system operational check was performed on 9/13/2012 (reference MyCMP logbook position 270101) by Gulfstream Savannah FAA CRS #GR4R216M.
- The last engine control operational test for the No. 1 and No. 2 engines occurred on 9/10/2012 by Gulfstream Savannah FAA CRS #GR4R216M.
- The last engine control functional rig for the No. 1 engine occurred on 12/12/2007 by a third party maintenance facility.
- The last engine control functional rig for the No. 2 engine occurred on 12/9/2009 by a third party maintenance facility.

- The last record for the removal of the left and right Control Display Units occurred on 11/29/2013 and 09/15/2012 respectively.

Adam Huray
Mechanical Engineer

APPENDIX A
ACTUATOR TEST RESULTS

Parker

Parker	CAGE CODE 82106	87000T		REV BY
ACCEPTANCE TEST DATA SHEET			SIZE A	PG 17 OF 24
SERIAL NO 3326 GT	PART NAME ACTUATOR-ELEVATOR/AILERON	V/N		

PART NO & DASH NO (CIRCLE DASH NO.)
87000-5001, -5002, -5005, -5006, -5007, -5008, -5009, -5011, -5013, -5017, -5018, -5019, -5020, -5021, -5023, -5024, -5025, -5026, -5027, -5029, -5030, -5031, -5032, -5033, -5035, -5036, -5037, -5038, -5039, -5043, -5045, -5046, -5047, -5048, -5121, -5122, -5125, -5126, -5131, -5132, -5137, -5138

PAGE 1A OF 2A	ACCEPTANCE TEST DATA SHEETS		PROJECT GULFSTREAM		
SPEC NO	CONTR NO	JOB NO R802193	DATE 1/13/15	INSPECTION	

	TEST	REQUIREMENTS		RESULTS	ACPT	REJ
1.	RETURN PROOF	NO DEFORMATION OR EXTERNAL LEAKAGE		3 - 5 PSI____ 3000 PSI____	NOT PERFORMED	
2.	PRESSURE PROOF		4500 PSI	+ STOP____ - STOP____	NOT PERFORMED	
3.	DYNAMIC LEAKAGE 17	1 DROP/50 CYCLES ENDS 1 DROP/25 CYCLES VENT. 1 DROP/100 CYCLES SHAFT		HEAD END____ ROD END____ VENT____ SHAFT____	NOT PERFORMED	
4.	CLEARANCE (REDUCED PRESSURE)	NO BINDING		+ STOP NONE - STOP NONE		
5.	INPUT STROKE	POINT "A" STROKE .395 - .402 IN.		STROKE .417 (SYSTEM 1) .398 (SYSTEM 2)		
6.	INPUT LOAD	16 OZ. MAXIMUM		+ DIRECTION 90 OZ 6 LB (SYSTEM 1) - DIRECTION____OZ 16 OZ (SYSTEM 2)		
7.	RIG NEUTRAL	PISTON TRAVEL WITH RIG PIN INSTALLED CAN BE STOPPED AT ANY POINT.		PISTON TRAVEL YES		
8.	CYL. FRICTION	20 LBS. MAXIMUM		BREAKOUT FRICTION EXT. 10 LBS. RETR. 10 LBS. RUNNING FRICTION EXT. 8 LBS. RETR. 8 LBS.		
9.	LEAKAGE	85 CC/MIN MAXIMUM 4		+ STOP R1 FREE CC R2 FLOW CC - STOP R1 FREE CC R2 FLOW CC		
10. A	CYL ROD EXT POSITION	ROD BRG ADJUST ± 3-3/4 MIN TURNS FROM EXT RIG. TORQUE ROD BRG NUT. RIG PIN MUST INDICATE ROD FULL EXT.		OUT____TURNS IN____TURNS ____IN. LBS. GO____NO GO____	NOT PERFORMED	
10. B	CYLINDER ROD STROKE	2.248 – 2.234		STROKE 2.241 IN		

Parker	CAGE CODE 82106	87000T	REV BY

PAGE 2A OF 2A	ACCEPTANCE TEST DATA SHEETS		SIZE	PG 18 OF 24
SERIAL NO 3326GT	PART NAME ACTUATOR/ELEVATOR/AILERON	V/N	A	

PART NO & DASH NO (CIRCLE DASH NO.)
87000-5001, -5002, -5005, -5006, -5007, -5008, -5009, -5011, -5013, -5017, -5018, -5019, -5020, -5021, -5023, -5024, -5025, -5026, -5027, -5029, -5030, -5031, -5032, -5033, -5035, -5036, -5037, -5038, -5039, -5043, -5045, -5046, -5047, -5048, -5121, -5122, -5125, -5126, -5131, -5132, -5137, -5138

	TEST	REQUIREMENTS	RESULTS	ACPT	REJ
11.	BYPASS VALVE FUNCTION	PISTON OSCILLATION STARTS 125 PSI MAX. PISTON OSCILLATION STOPS 60 PSI MIN.	P1 PSI / P2 PSI* OSC. START * / 120 OSC. STOP * / 85	*COULD NOT TEST. PRESS. FREE FLOWING TO RTN SYSTEM!	
12.	INTER-SYSTEM LEAKAGE	0.2 CC/MIN. MAXIMUM	+ STOP Ø CC - STOP Ø CC		
13.	NEUTRAL FLOW	150 CC/MIN. MAXIMUM	R1 320 CC/MIN R2 35 CC/MIN		
14.	THRESHOLD	0.006" MAXIMUM DIFFERENCE	RETRACT .025 IN. EXTEND .025 IN. DIFFERENCE .050 IN.		
15.	PRESSURE SWITCH ACTIVATION	N/A	N/A	N/A	N/A
16.	CYCLE (500A) 17	1 DROP/50 CYCLES ENDS (20 DROPS/1000 CYCLES - ENDS) 2 DROPS/50 CYCLES VENT (40 DROPS/1000 CYCLES - VENT) 1 DROP/100 CYCLES SHAFT (10 DROPS/1000 CYCLES - SHAFT)	HEAD END___ DROPS ROD END___ DROPS INTERFACE___ DROPS SHAFT___ DROPS	NOT PERFORMED	
		N/A	N/A	N/A	N/A
17.	DELETED	N/A	N/A	N/A	N/A
18.	SERVO VERIFICATION	-5013, -5021, -5027, -5033, -5043 ONLY 11.5 .IN/SEC MAX ALL OTHER DASH NUMBERS 12.5 IN/SEC MIN.	PEAK PISTON VELOCITY (AVG.) SYS. 1___ IN/SEC SYS. 2___ IN/SEC	NOT PERFORMED	
19.	DYNAMIC RESPONSE VERIFICATION	N/A	N/A	N/A	N/A
20.	FREQUENCY RESPONSE	N/A	N/A	N/A	N/A

Parker		CAGE CODE 82106	87000T		REV BY
ACCEPTANCE TEST DATA SHEET				SIZE A	PG 17 OF 24
SERIAL NO 1014CD	PART NAME ACTUATOR-ELEVATOR/AILERON (2ND)		V/N		

PART NO & DASH NO (CIRCLE DASH NO.)
87000-5001, -5002, -5005, -5006, -5007, -5008, -5009, -5011, -5013, -5017, -5018, -5019, -5020, -5021, -5023, -5024, -5025, -5026, -5027, -5029, -5030, -5031, -5032, -5033, -5035, -5036, -5037, -5038, -5039, -5043, -5045, -5046, -5047, -5048, -5121, -5122, -5125, -5126, -5131, -5132, -5137, -5138

PAGE 1A OF 2A	ACCEPTANCE TEST DATA SHEETS		PROJECT GULFSTREAM	
SPEC NO	CONTR NO	JOB NO	DATE	INSPECTION

	TEST	REQUIREMENTS		RESULTS	ACPT	REJ
1.	RETURN PROOF	NO DEFORMATION OR EXTERNAL LEAKAGE		3 - 5 PSI____ 3000 PSI____	NOT PERFORMED	
2.	PRESSURE PROOF		4500 PSI	+ STOP____ - STOP____	NOT PERFORMED	
3.	DYNAMIC LEAKAGE 17	1 DROP/50 CYCLES ENDS 1 DROP/25 CYCLES VENT. 1 DROP/100 CYCLES SHAFT		HEAD END____ ROD END____ VENT____ SHAFT____	NOT PERFORMED	
4.	CLEARANCE	NO BINDING		+ STOP NONE - STOP NONE		
5.	INPUT STROKE	POINT "A" STROKE .395 - .402 IN.		STROKE .398		
6.	INPUT LOAD	16 OZ. MAXIMUM		+ DIRECTION 12 OZ - DIRECTION 16 OZ	S1 12 15	S2 14 9
7.	RIG NEUTRAL	PISTON TRAVEL WITH RIG PIN INSTALLED CAN BE STOPPED AT ANY POINT.		PISTON TRAVEL PASS		
8.	CYL. FRICTION	20 LBS. MAXIMUM		BREAKOUT FRICTION EXT. 14 LBS. RETR. 14 LBS. RUNNING FRICTION EXT. 10 LBS. RETR. 16/10 LBS.		
9.	LEAKAGE	85 CC/MIN MAXIMUM 4		+ STOP R1____CC R2____CC - STOP R1____CC R2____CC	NOT PERFORMED	
10. A	CYL ROD EXT POSITION	ROD BRG ADJUST ± 3-3/4 MIN TURNS FROM EXT RIG. TORQUE ROD BRG NUT. RIG PIN MUST INDICATE ROD FULL EXT.		OUT____TURNS IN____TURNS ____IN. LBS. GO____NO GO____	NOT PERFORMED	
10. B	CYLINDER ROD STROKE	2.248 – 2.234		STROKE 2.240 IN		

Parker	CAGE CODE 82106	87000T	REV BY

PAGE 2A OF 2A	ACCEPTANCE TEST DATA SHEETS		SIZE	PG 18 OF 24
SERIAL NO D14CD	PART NAME ACTUATOR/ELEVATOR/AILERON 2nd	V/N	A	

PART NO & DASH NO (CIRCLE DASH NO.)
87000-5001, -5002, -5005, -5006, -5007, -5008, -5009, -5011, -5013, -5017, -5018, -5019, -5020, -5021, -5023, -5024, -5025, -5026, -5027, -5029, -5030, -5031, -5032, -5033, -5035, -5036, -5037, -5038, -5039, -5043, -5045, -5046, -5047, -5048, -5121, -5122, -5125, -5126, -5131, -5132, -5137, -5138

	TEST	REQUIREMENTS	RESULTS	ACPT	REJ
11.	BYPASS VALVE FUNCTION	PISTON OSCILLATION STARTS 125 PSI MAX. PISTON OSCILLATION STOPS 60 PSI MIN.	P1 PSI P2 PSI OSC. START 100 100 OSC. STOP 80 80		
12.	INTER-SYSTEM LEAKAGE	0.2 CC/MIN. MAXIMUM	+ STOP .05 CC - STOP .05 CC		
13.	NEUTRAL FLOW	150 CC/MIN. MAXIMUM	R1 68 CC/MIN R2 48 CC/MIN		
14.	THRESHOLD	0.006" MAXIMUM DIFFERENCE	RETRACT .001 IN. EXTEND .001 IN. DIFFERENCE .002 IN.		
15.	PRESSURE SWITCH ACTIVATION	N/A	N/A	N/A	N/A
16.	CYCLE (500A) 17	1 DROP/50 CYCLES ENDS (20 DROPS/1000 CYCLES - ENDS) 2 DROPS/50 CYCLES VENT (40 DROPS/1000 CYCLES - VENT) 1 DROP/100 CYCLES SHAFT (10 DROPS/1000 CYCLES - SHAFT)	HEAD END___ DROPS ROD END___ DROPS INTERFACE___ DROPS SHAFT___ DROPS	NOT PERFORMED	
		N/A	N/A	N/A	N/A
17.	DELETED	N/A	N/A	N/A	N/A
18.	SERVO VERIFICATION	-5013, -5021, -5027, -5033, -5043 ONLY 11.5 .IN/SEC MAX ALL OTHER DASH NUMBERS 12.5 IN/SEC MIN.	PEAK PISTON VELOCITY (AVG.) SYS. 1___ IN/SEC SYS. 2___ IN/SEC	NOT PERFORMED	
19.	DYNAMIC RESPONSE VERIFICATION	N/A	N/A	N/A	N/A
20.	FREQUENCY RESPONSE	N/A	N/A	N/A	N/A