----- SYSTEMS GROUP CHAIRMAN’S FACTUAL REPORT -------

1 ACCIDENT:

Location:	Greenville, South Carolina
Date:	September 27, 2018
Time:	About 1840 GMT (1340 LCL)
Aircraft:	Dassault Falcon 50
Serial Number:	17
Registration:	N114TD

2 SYSTEM’S GROUP:

Chairman: Michael Bauer
National Transportation Safety Board
Washington, D.C.

Member: Patrick Hempen
Federal Aviation Administration
Washington, D.C.

Accredited Rep: Gilles Romeuf
Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile
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Member: Jay Sigmann
Dassault Falcon Jet
New Castle, DE

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Meggitt
Akron, OH

3 SUMMARY

3.1 Event Summary

On September 27, 2018, about 1346 eastern daylight time, a Dassault Falcon 50 business jet, N114TD, operated by Air American Flight Services, Inc., was substantially damaged when it overran the departure end of runway 19 during landing at Greenville Downtown Airport (GMU), Greenville, South Carolina. The airline transport pilot (ATP) seated in the left cockpit seat and private pilot seated in the right cockpit seat were fatally injured, and the two passengers received serious injuries. Visual meteorological conditions prevailed, and an instrument flight rules flight plan was filed for the flight that departed St. Pete-Clearwater International Airport (PIE), St. Petersburg-Clearwater, Florida, destined for GMU. The personal flight was conducted under the provisions of Title 14 Code of Federal Regulations Part 91.

3.2 Systems Group Summary

The systems group was formed on September 28, 2018 during the on-scene phase of the investigation. The group documented the cockpit, hydraulic, landing gear, brakes and associated systems from September 28 to 30, 2018.

The systems group reconvened from November 27 through the 28, 2018 at Atlanta Air Recovery (AAR) in Griffin, Georgia, to remove some additional braking components and inspect the brake components that were inaccessible on-scene.

The systems group reconvened from March 5 through the 8, 2019 to conduct examinations of the recovered anti-skid braking system components. The group met at Meggitt Aircraft Braking Systems (MABS) facility in Akron, Ohio and at Projects Unlimited Incorporated (PUI) in Dayton, Ohio to perform the inspections and the component testing.

The systems group chairman received the main landing gear wiring harnesses from the left hand (LH) and right hand (RH) main landing gear (MLG) on April 29, 2019. The harness was removed from the aircraft wreckage at AAR in Griffin, Georgia by an National Transportation Safety Board (NTSB) field investigator and was then packaged and shipped to NTSB headquarters. The RH MLG harness was inspected by the group chairman on May 14, 2019 and May 21, 2019.

This report contains factual information related to the group’s inspection and testing of the aircraft’s braking system.

3.3 Acronyms

Ω	Ohms	kt, kts	knot, knots
AAR	Atlanta Air Recovery	L,LH	Left Hand
AFM	Aircraft Flight Manual	L/G	Landing Gear
AFU	Auxiliary Feel Unit	LHS	Left Hand Side
AMM	Aircraft Maintenance Manual	mA	milliamps
ATA	Airline Transport Association	MABS	Meggitt Aircraft Braking Systems
ATP	Airline Transport Pilot	MLG	Main Landing Gear
AWG	American Wire Gauge	MMEL	Master Minimum Equipment List
CAGE	Commercial and Government Entity	NLG	Nose Landing Gear
CCA	Circuit Card Assembly	NTSB	National Transportation Safety Board
CCW	Counter-Clockwise	OTBD	Outboard
CMM	Component Maintenance Manual	P/N	Part Number
cSt	centistokes (kinematic viscosity unit of measurement)	PIE	St. Pete-Clearwater International Airport
CT	Computed Tomography	PM	Procedures Manual
CW	Clockwise	psi	Pounds per Square Inch
DC	Direct Current	PUI	Projects Unlimited Incorporated
FSR	Full Scale Range	R,RH	Right Hand
FTI	Functional Test Instructions	RHS	Right Hand Side
g	standard gravity	RPM	Revolutions Per Minute
GMU	Greenville Downtown Airport	sec	second
HYDR	Hydraulic	ST-BY	Standby
in	inch	sys	system
INBD	Inboard	V	Volts
INDIC	Indication	VDC	Volts Direct Current
INOP	Inoperative	WDM	Wiring Diagram Manual
IR	Internal Resistance	WHL	Wheel

4 DETAILS OF THE INVESTIGATION:

4.1 Systems Descriptions

The following sections contain descriptions and operation information for the respective systems. The information is based on aircraft flight manual documentation and manufacturer documentation.

The Dassault Falcon 50 is a mid-sized long-range business jet, which features a three-engine configuration with a conventional, retractable landing gear configuration. The engines are mounted at the rear of the aircraft with the left engine identified as number 1, the center engine identified as engine 2, and the right engine identified as engine 3, ref Figure 1.

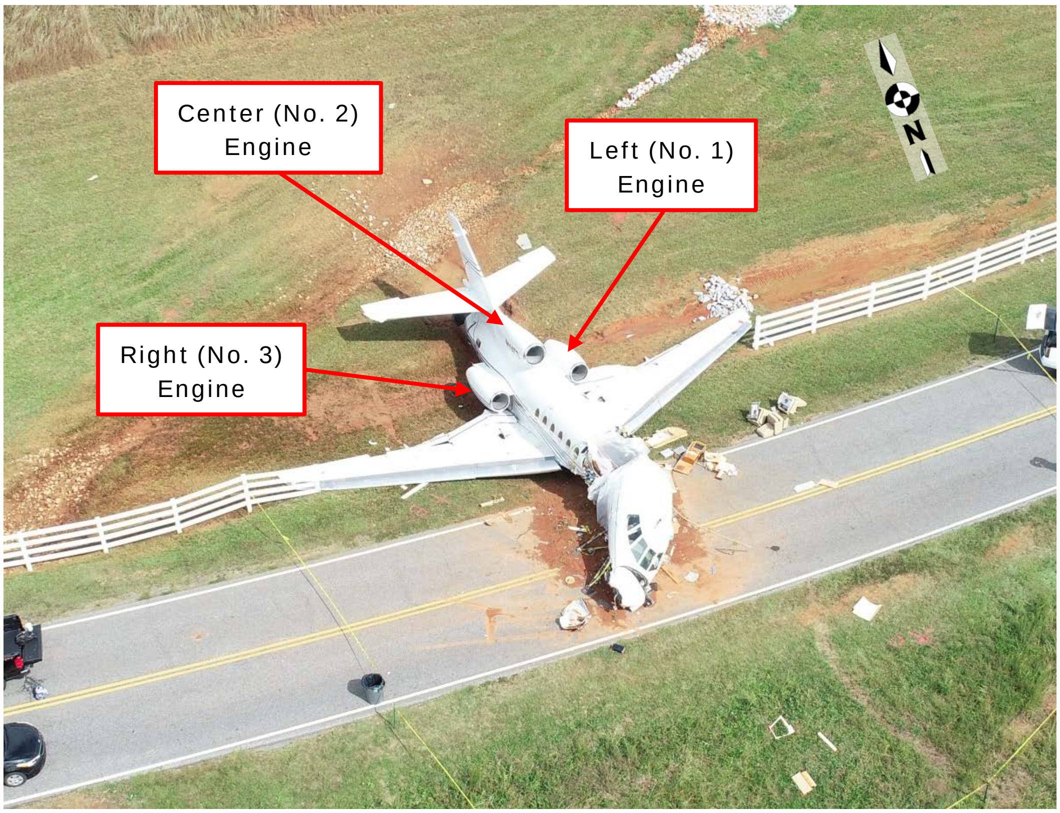


Figure 1 - N114TD, Falcon 50, accident scene (positional arrow approximate)

4.1.1 Hydraulic System Description and Operation:

The aircraft has two main hydraulic systems and one auxiliary system, ref Figure 2. The main systems, 1 and 2, are independent of each other and operate simultaneously. The number 1 and number 2 engines each drive a self-regulating pump supplying system 1 in parallel. The number 3 engine drives a self-regulating pump supplying system 2. A standby pump is located in the rear compartment and it supplies the auxiliary hydraulic system. The auxiliary system is used as a standby or additional source to supply system 2, should the number 3 engine or its pump fail. The system uses MIL-5606 hydraulic fluid. The main system operates at a system pressure of 3000 psi and the auxiliary system operates at a pressure between 1508 and 2150 psi.

System 1 provides hydraulic pressure to the following subsystems:

- one body of the power servo actuators,
- "ARTHUR-Q" units[1],
- inboard slats (normal operation),
- outboard slats (normal and automatic operation),
- normal brakes,
- thrust reverser,
- landing gear and landing gear doors.

For system 1 while on the ground, the above equipment can be supplied by the auxiliary hydraulic system for test purposes.

System 2 provides hydraulic pressure to the following subsystems:

- one body of the power servo unit,
- flaps,
- airbrakes,
- outboard slats (emergency and automatic operation),
- steering system,
- emergency brakes,
- parking brakes.

For system 2, if the engine driven pump fails, the above components are supplied in flight or on the ground by the auxiliary hydraulic system.

The auxiliary hydraulic system can also be used on the ground for testing the system 1 and system 2 user subsystems.

[1] The “ARTHUR-Q” units are part of the flight control system consisting of a variable bellcrank which controls the travel of the Auxiliary Feel Unit (AFU), and therefore its’ loads, as a function of the indicated airspeed.

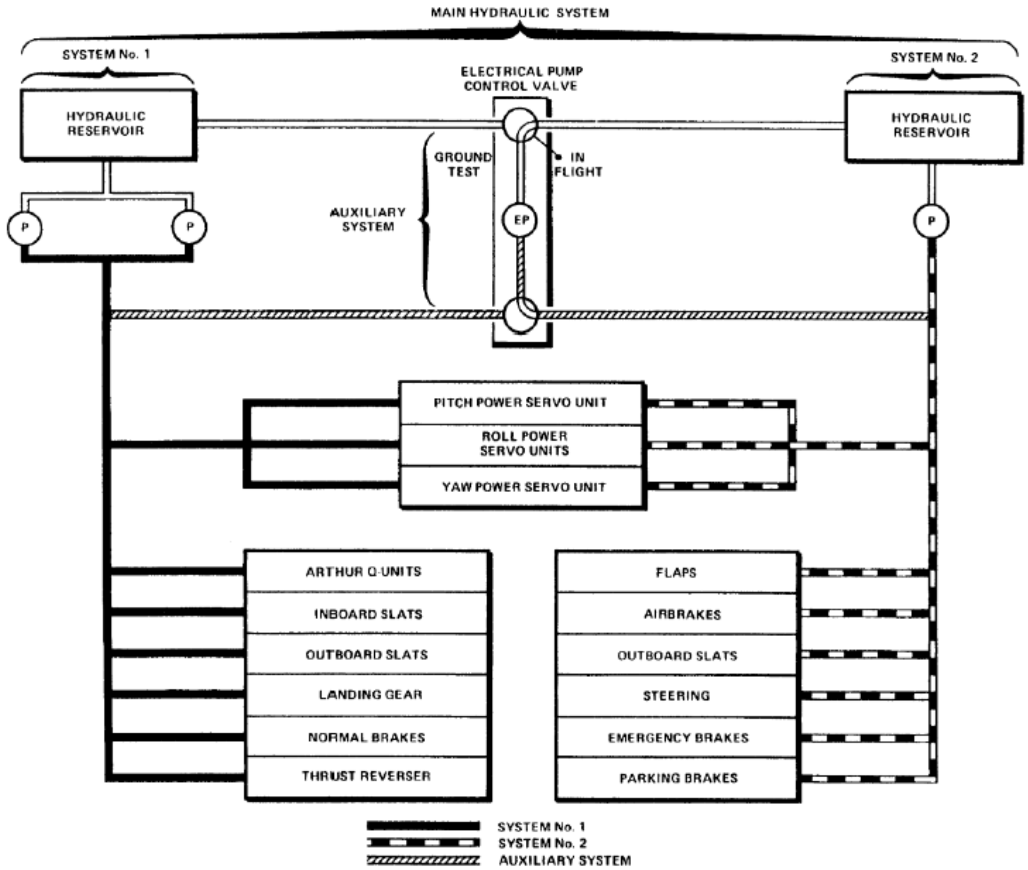


Figure 2 – Loads block diagram for Falcon 50 hydraulic system (From AMM)

4.1.2 Main Landing Gear System Description and Operation:

The main landing gear consists of a left and right landing gear leg. Each landing gear leg is fitted with an inboard and an outboard wheel assembly. Each main landing gear wheels interface with the braking system.

4.1.2.1 Wheel Brake Assembly

The brake assembly consists of a housing, a torque tube, a pressure plate, four rotating disks, three stationary disks and a back plate[2]. The assembly is held together with five bolts, five washers and five self-locking nuts. The housing contains ten cylinder sleeves, ten pistons and five return mechanisms. Each return mechanism consists of a return pin, a grip and tube subassembly[3], a spring housing, a release spring, a spring guide and a retaining ring.

[2] The description of the brake assembly was taken from the brake manufacturers component maintenance manual (CMM) revision 8 dated January 14th, 2013.
[3] For the brake assembly which was installed on the accident aircraft, the grip and tube subassembly is a swage and tube subassembly.

The brake assembly uses two hydraulic actuating hydraulic systems, A and B[4]. Systems A and B are identical and either system can be used for normal or emergency braking operations. A bleeder valve and two inlet ports are provided for each system.

4.1.3 Braking System Description and Operation:

The braking system can operate in three modes, Normal, Emergency and Parking.

The brake pedals are interlocked mechanically (by cables) and act directly on the normal/emergency brake selector valve. The selector valve, which is mounted to the forward pressure bulkhead, has two independent chambers corresponding to the LH and RH brakes. Each chamber is divided into two bodies. The first body comprises a slide valve actuated mechanically by the brake pedals or hydraulically by the landing gear retraction pressure, and a slide valve distributing hydraulic system 1 normal braking pressure. The second body contains a slide valve distributing hydraulic system 2 emergency braking pressure. This slide valve is actuated jointly with the normal braking pressure slide valve. Depending on the mode of operation, normal or emergency, hydraulic pressure from the respective system is supplied to the corresponding systems brake pistons on the wheels' brake assemblies.

4.1.3.1 Normal Braking (Anti-Skid Operational)

The normal braking system is powered by hydraulic system 1, Figure 3. There are two functions, normal braking, which is mechanically applied via the rudder/brake pedals located in the cockpit at the pilot and co-pilot locations, and automatic wheel braking applied during landing gear retraction. The application of the brakes via the brake pedals is differential and progressive and are normally operated in conjunction with the anti-skid system.

In the normal operation mode, the brake switch is in the "#1-ON" position. Brake pressure is provided by hydraulic system 1, through the normal brake selector valve, and the antiskid system is available when the landing gear control lever is in the "DOWN" position. "L" and "R" indicator lights on the brake control panel are connected to a pressure switch and will illuminate when pressure above 600 psi is sensed in the system (in both normal and emergency mode)

The anti-skid system's main components are the anti-skid control unit, two brake servo valves, one for the left MLG and one for the right MLG and four wheel speed transducers, one for each main gear wheel.

[4] The brake manufacturer identifies the hydraulic system as A and B, whereas the aircraft manufacturer hydraulic system identifies the systems as number 1 and number 2.

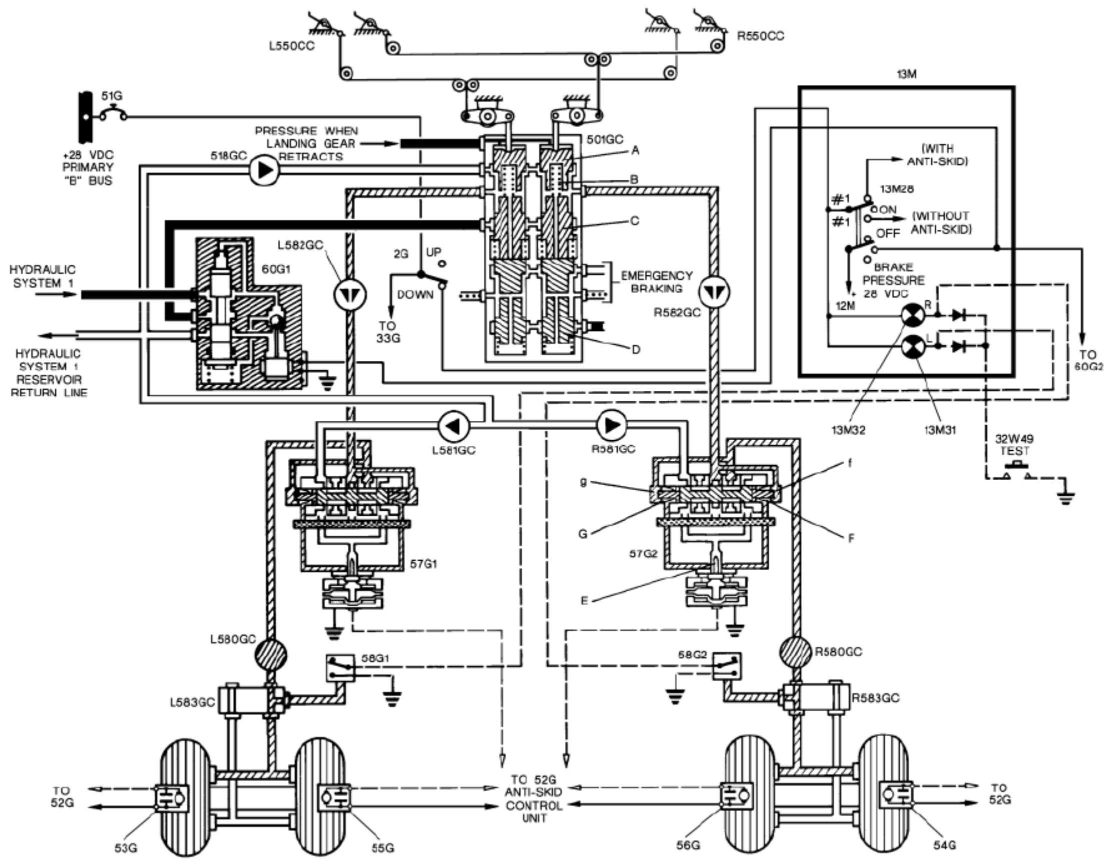


Figure 3 – Brake system diagram for normal brake operation (from AMM)

4.1.3.1.1 Anti-skid System – Anti-Skid Control Unit:

The anti-skid control unit, or control box, is located in the RH electrical cabinet of the cockpit. The unit contains five individual circuit card assemblies (CCA). Two CCAs are amplifier circuits for the skid system control, one CCA controls power and locked wheel protection, one CCA provides deceleration control and test functions and one CCA provides wheel pairing functions.

During an approach to landing, touchdown protection is activated by the squat switch located on the nose landing gear indicating an extended (in-air) position. The system will send a “full dump” signal to the servo valves to port all hydraulic/brake pressure to the return system and remove all brake pressure from the brakes. The intent of touchdown protection is to avoid a condition at touchdown with brake pressure applied causing the MLG wheels to be locked upon touchdown. Brake pressure is withheld from the system until the wheel speed sensors spin up to approximately 12 knots.

Upon reaching 12-knots, via wheel speed spin-up, and the squat switch indicates a compressed nose landing gear (NLG), on ground condition, the locked wheel arming circuit is activated. While the speed remains above 12 knots, touchdown protection is removed, and full brake pressure can be applied. While aircraft speed remains above 25 knots, if one of the wheels falls below the 12-knot threshold, the locked wheel detection circuit will provide commands to the servo valve on the side with the locked wheel to remove brake pressure. When the aircraft speed falls below 25 knots, the locked wheel protection circuit will deactivate to prevent interference with normal braking and anti-skid operations.

The operation of the anti-skid control circuits is designed such that a sensed skid, via the wheel speed transducers, on either the two left wheels or two right wheels will cause the system to modulate the brake pressure via the control valves on the side where the skid is sensed to remove the skidding condition. The anti-skid system ceases to function when the aircraft speed is less than or equal to 10 knots.

Deceleration control circuits are inactive at speeds above 40 knots, but below 40 knots, the circuits sense aircraft deceleration above 0.1g (3 ft/sec^2) and operate the anti-skid system similar to the operation above 40 knots when a skid is sensed via the wheel speed transducers.

The system is designed such that the control unit limits maximum pressure of 800 psi to the brakes and aircraft deceleration to 0.3g (10 ft/sec^2).

4.1.3.1.2 Anti-skid System - Servo Valve:

The servo valve, also referred to as a control valve, is a conventional two-stage flapper-nozzle pressure control valve. Two valves, a left and right valve, are located in their respective main landing gear bays. The servo valve consists of a polarized electrical torque motor and two stages of hydraulic power amplification. The first stage contains the flapper and the second stage contains a sliding spool. As current from the anti-skid control unit is applied to the torque motor, a flapper moves and regulates the orifice size between two nozzles, which results in an unbalance between the orifices and in turn causes the spool to displace. The displacement of the spool results in a pressure differential between the supply and brake pressure lines. The spool will continue movement until the force unbalance returns to a balanced state. The current supplied to the torque motor has a proportional relationship to the pressure differential.

4.1.3.1.3 Anti-skid System – Wheel Speed Transducer:

The wheel speed transducer, also referred to as a wheel drive unit, is a DC tachometer generator. Each transducer is mounted in the MLG axle and is mechanically coupled to the wheel assembly drive caps. During operation, as the wheel rotates, an output voltage is generated which is proportional to the aircraft ground speed. The voltage is used by the anti-skid control box to control braking pressure to the monitored wheel during landing operations. The control unit input expects a positive voltage during wheel rotation experienced during landing and braking operations.

4.1.3.2 Emergency Braking (Anti-Skid Not Operational)

The emergency braking system is powered by hydraulic system 2, Figure 4. The brake control switch is placed in the “#2-OFF” position to enable emergency braking mode. The application of the brakes via the brake pedals will provide differential and progressive brake pressure to the wheels’ brake assemblies. In this mode of operation, there are no protections (touchdown, locked wheel and anti-skid) provided.

An additional brake switch position on the brake control panel allows for operation of the brakes using hydraulic system 1 without anti-skid protection. This switch position is used during ground testing only and is a gated position required the crew to pull the switch out and move it to the “#1-OFF” position.

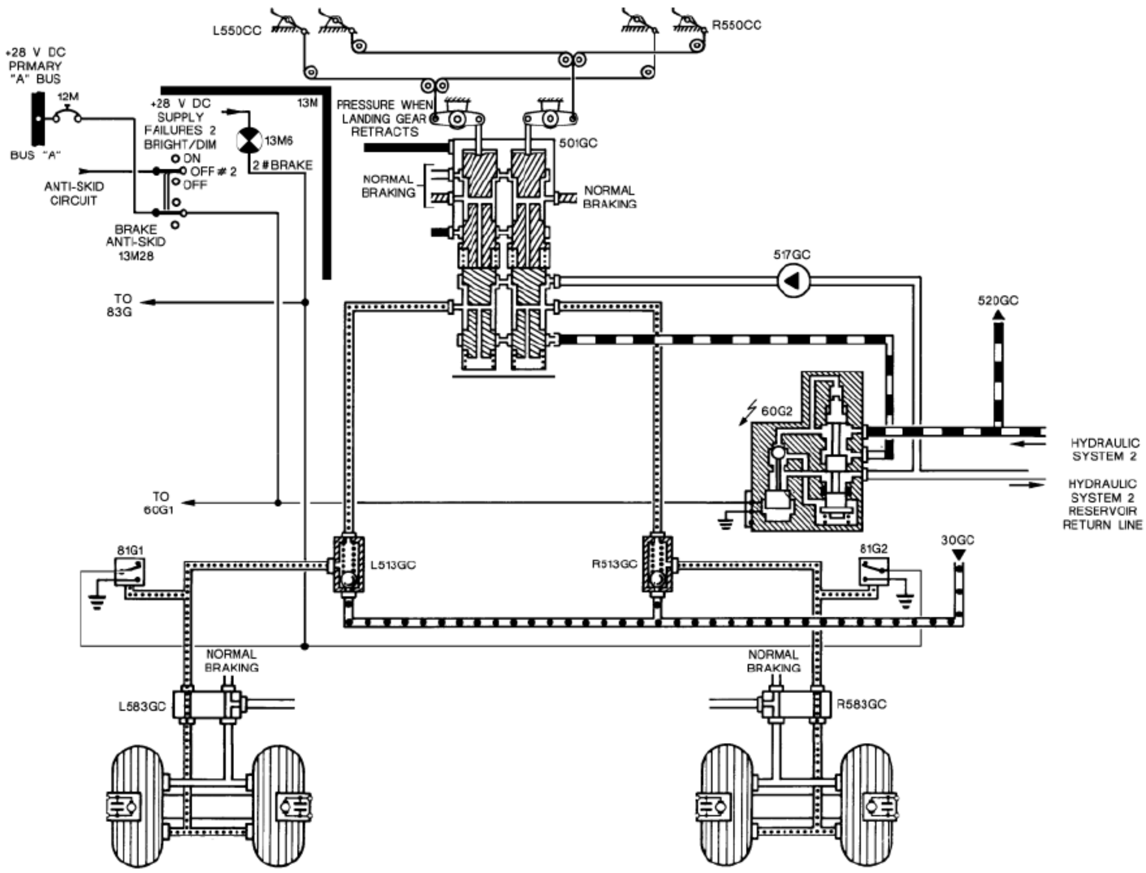


Figure 4 - Brake system diagram for emergency brake operation (from AMM)

4.1.3.3 Parking Brake

The parking brake system is powered by hydraulic system 2, or when hydraulic system 2 is unavailable, a parking brake accumulator. The parking brake accumulator provides a reserve of pressurized hydraulic fluid and contains a check valve to protect the system during system 2 pressure loss. The parking brake system is designed to allow operation in the event of a failure of both hydraulic system 1 and hydraulic system 2.

The system is actuated via a handle mounted on the glareshield in the cockpit. The handle has two latches. To operate the parking brake, the crew pulls the handle. An intermediate latch position corresponds to a light braking force of approximately 400 psi. When the handle is pulled to the end of travel latch, full braking force of approximately 1560 psi is applied to the brake assemblies. The brake pressure is applied through the same set of brake pistons as the emergency system and is non-differential (applied to all four brake assemblies simultaneously).

4.2 Aircraft Examination:

4.2.1 Cockpit

The following switch/control positions were the positions as found when the cockpit was examined on September 28th, 2018. Images were taken after first responders extricated the crewmembers.

Systems related to the landing gear/brakes/hydraulics are noted by the green boxes in Figure 5 and Figure 6.

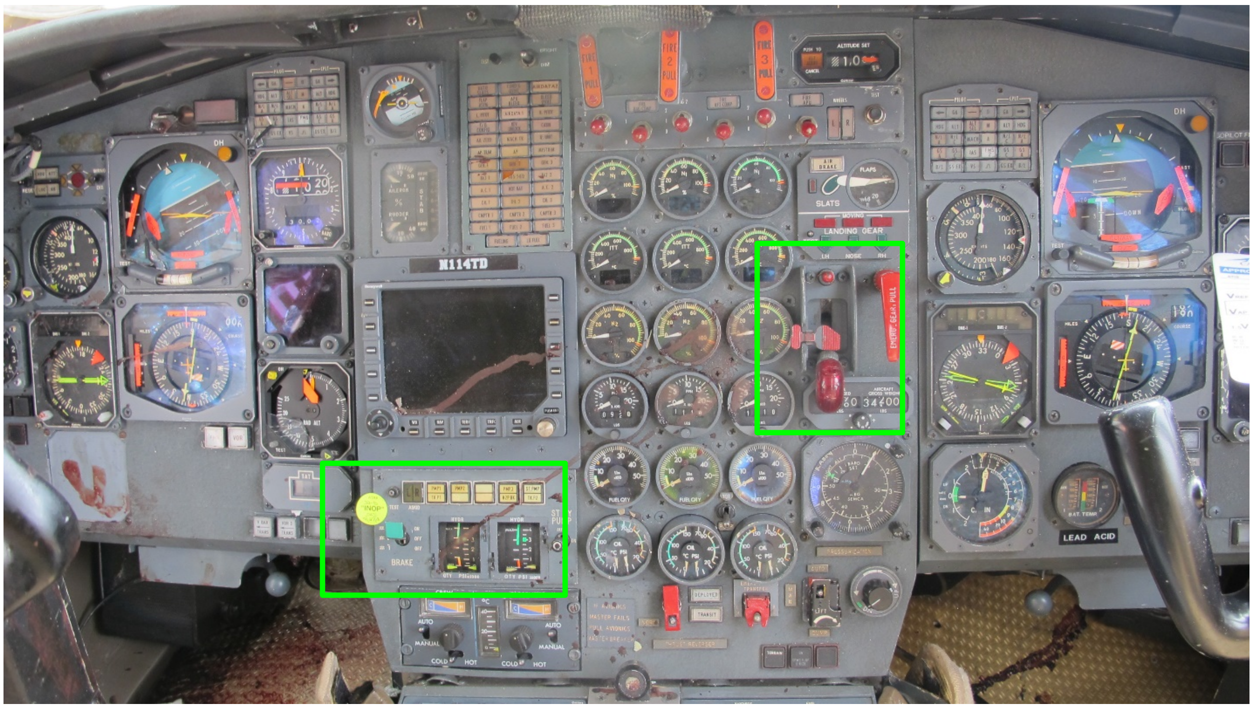


Figure 5 - Cockpit with gear handle and brake control/hydraulics panel highlighted

Figure 6 – Glareshield with Parking Brake Handle highlighted

4.2.1.1 Main Instrument Panel (ref. Figure 5, Figure 7):

The landing gear handle was in the down position.

The “Brake” switch was found in the “#1-ON” position. Next to the switch, a green sticker was affixed to the panel with the label “ATA # 32-5 “INOP” DATE: 9/27/18”, ref Figure 7.

The Standby pump switch was in the “ON” position.

Figure 7 – Brake switch showing "INOP" sticker closeup

4.2.1.2 Glareshield (ref. Figure 6):

The parking brake handle was in the stowed detent.

4.2.1.3 Circuit Breaker Panel

The overhead circuit breaker panels were inspected for any circuit breakers in a tripped/open/collared state. Relevant circuit breakers for the hydraulic, "HYDR", were noted in the following positions as stated in Table 1 and a portion of the panel is shown in Figure 8.

Table 1 - Noted Circuit Breakers

Name	Panel Ref.	Condition
HYDR 1	HYDR	Closed
HYDR 2	HYDR	Closed
NOSE WHL	HYDR	Closed
ANTISKID	HYDR	Closed
L/G INDIC	HYDR	Closed
L/G CONTROL	HYDR	Closed
ST-BY PUMP	HYDR	Open/Trip

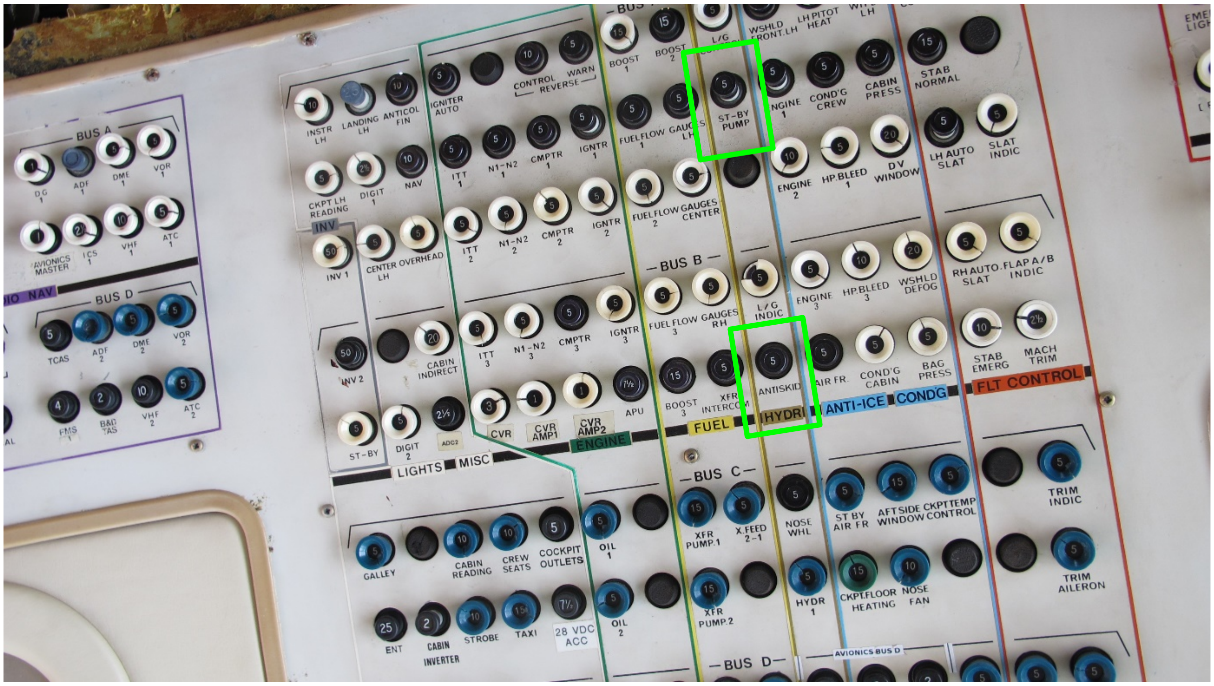


Figure 8 – Overhead C/B Panel with antiskid and standby pump breaker highlighted

4.2.2 Anti-Skid System Components and Wiring

Prior to testing and after removal from the aircraft, the anti-skid system components were subjected to computed tomography (CT) scanning to document the internal condition of the components and electrical connections. The results from the CT scanning are documented in the "Computed Tomography Specialist's Factual Report".

4.2.2.1 Wheel Speed Transducers

All four wheel speed transducers were recovered from the accident aircraft. Prior to removal, engagement of the wheel hub to the each wheel speed transducer shaft was confirmed during the removal of the wheel hub and transducers, ref Figure 9 and Figure 10. No anomalies were noted on all four hub assemblies, ref Figure 11. Each transducer was removed following AMM procedures, except the cannon plugs were not disconnected, but the wiring was cut upstream of the plug to retain the condition of the connector/component interface.

Figure 9 - Left inboard wheel/tire assembly prior to hub removal

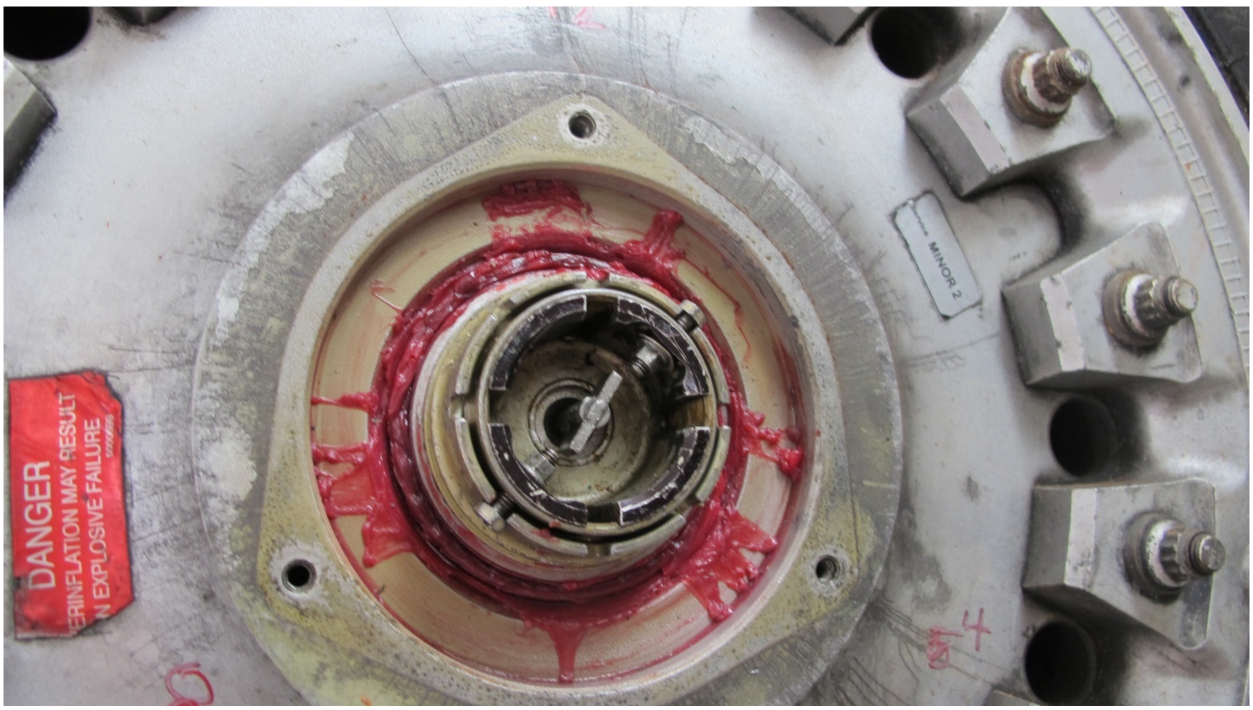


Figure 10 - Left inboard wheel speed transducer mounted in axle

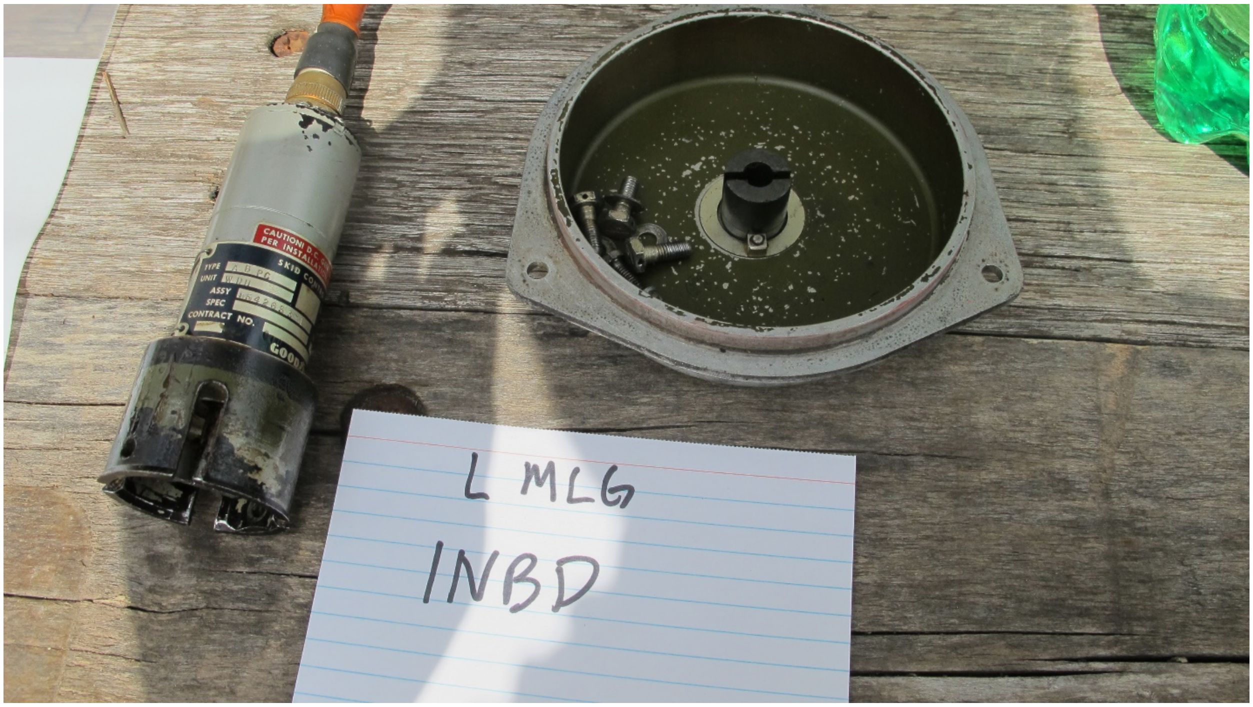


Figure 11 - Typical wheel speed transducer and wheel hub after removal from aircraft

Each transducer was tested at the MABS facility under supervision of the NTSB using MABS personnel and procedures. Prior to testing the accident components Dassault-Falcon Jet provided an exemplar unit to check the testing setup. All tests for the exemplar unit were within expected results. Testing was conducted using the Functional Test Instructions (FTI) for the Wheel Driven Unit Assembly P/N 9542653-1 Rev G dated 07/14/1980 (Doc. Number 5D1-3694) and the manufacturer's test fixture, ref Figure 12. An additional test was added to record the output voltage at 900 RPM in both the clockwise (CW) and counterclockwise (CCW) direction. Additionally, the resistance measured and recorded between pins A and B using a procedure developed by the Dassault Field Tech Rep[5]. The FTI was performed twice, once with the aircraft harness installed, and once with the harness removed. All FTI tests were performed except for unit weight. The following sections will describe any issues specific to each respective unit.

[5] Expected resistance was 331 Ω and an acceptable range between 250 Ω and 375 Ω.

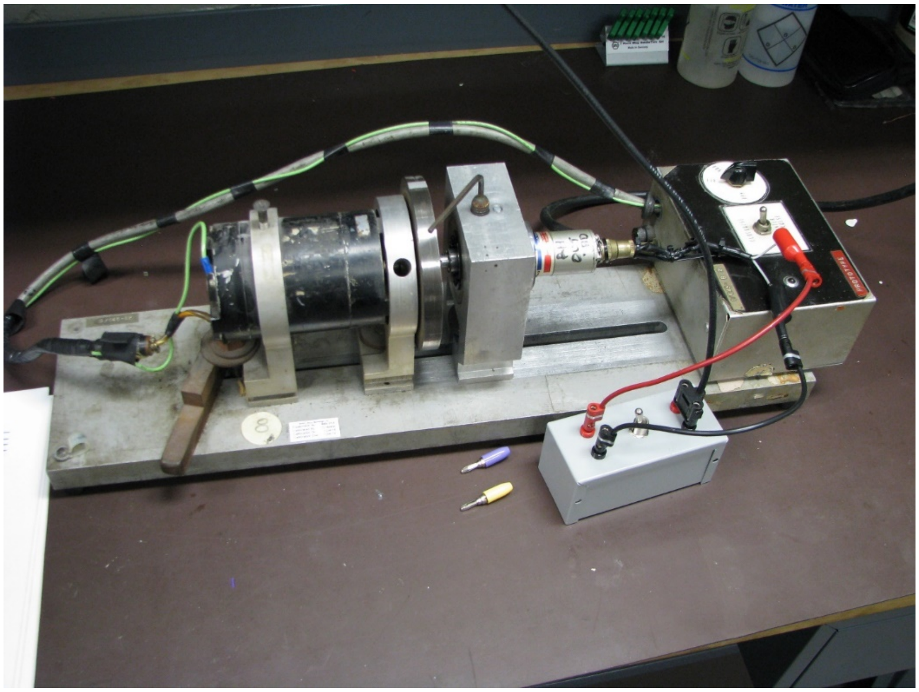


Figure 12 - Wheel speed transducer on test setup

4.2.2.1.1 RH Outboard Transducer:

Part Number: 9542653-1
Serial Number: APR79-1234

Figure 13 - RH Outboard transducer wiring

The unit was manufactured in April of 1979.

No notable signs of visible damage were noted, ref Figure 13. The aircraft wiring harness had indications of a field splice on the wiring which was covered using vinyl electrical tape. The electrical tape was not removed and remained in place for the CT scans.

The unit passed all tests within expected results, with and without the aircraft wiring harness installed.

With the aircraft harness installed, the pin to pin resistance was 337.8 Ω. The output voltage at 900 RPM in the CW direction was measured to be 6.96 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -6.96 VDC.

Without the aircraft harness installed, the pin to pin resistance was 343.9 Ω. The output voltage at 900 RPM in the CW direction was measured to be 6.97 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -6.97 VDC.

After the testing the electrical tape was removed to inspect the wire splices, ref Figure 14. It was noted that the wiring between the aircraft and the connector were of two different wire gages. Dassault informed the group that the use of splices in the wiring is not a recommended practice. The removal and replacement of the electrical connectors of the wheel speed transducers is specified in AMM Task 32-61-01-900-801 “Removal/Installation of the MLG Harness.” This task also includes electrical tests after reinstallation of the electrical connectors.

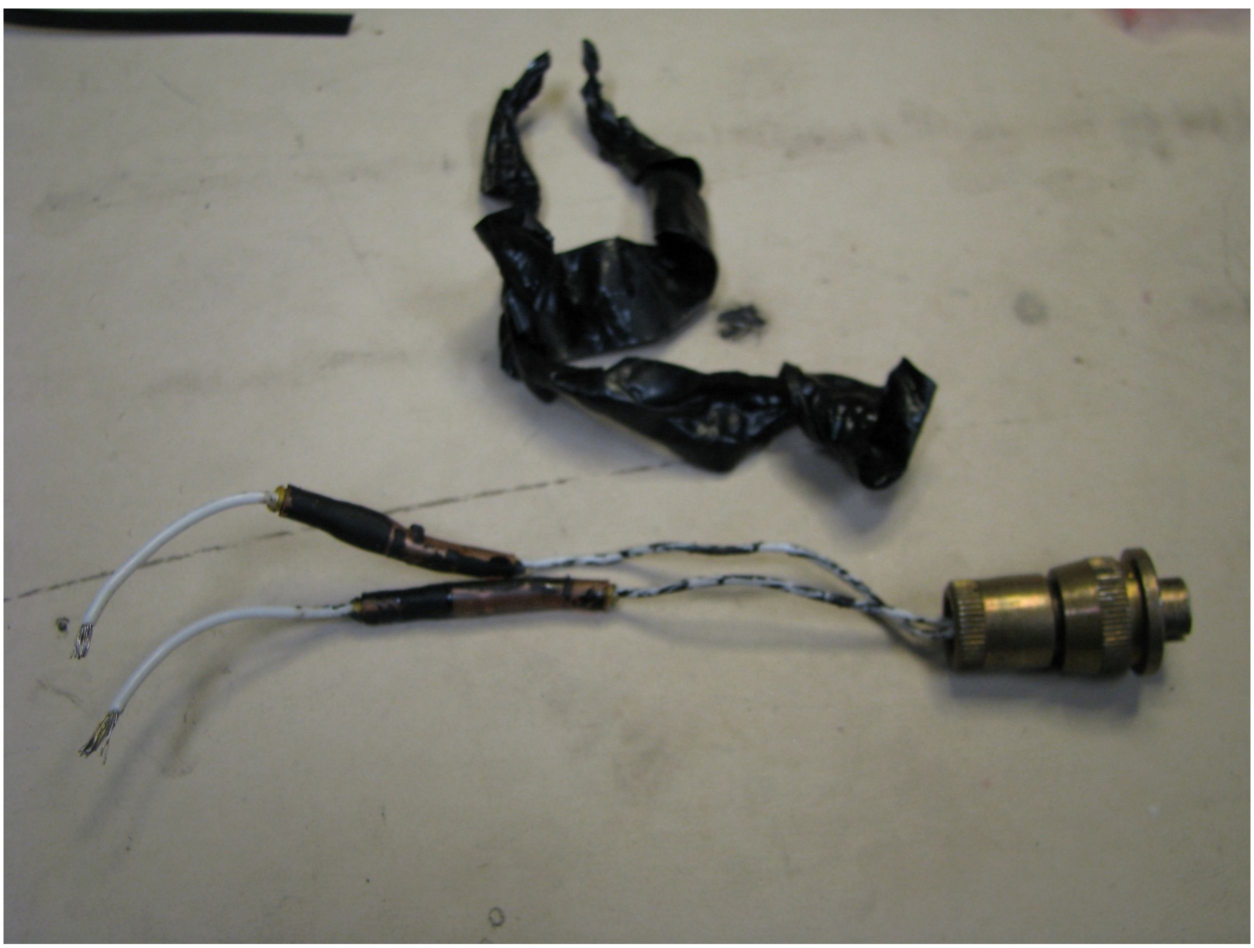


Figure 14 - RH outboard transducer wiring with electrical tape removed.

4.2.2.1.2 LH Outboard Transducer:

Part Number: 9542653-1
Serial Number: NOV77-959

The unit was manufactured in November of 1977.

No notable signs of visible damage were noted, ref Figure 15. The aircraft wiring harness had no indications of any field splice on the wiring.

Figure 15 – LH outboard transducer

The unit passed all tests within expected results, with and without the aircraft wiring harness installed.

With the aircraft harness installed, the pin to pin resistance was 372.8 Ω. The output voltage at 900 RPM in the CW direction was measured to be 6.99 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -6.98 VDC.

Without the aircraft harness installed, the pin to pin resistance was 333.6 Ω. The output voltage at 900 RPM in the CW direction was measured to be 6.99 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -6.99 VDC.

After the testing the insulation was removed from the wiring harness, Figure 16. The original wiring identifying tags were located close to the connector.

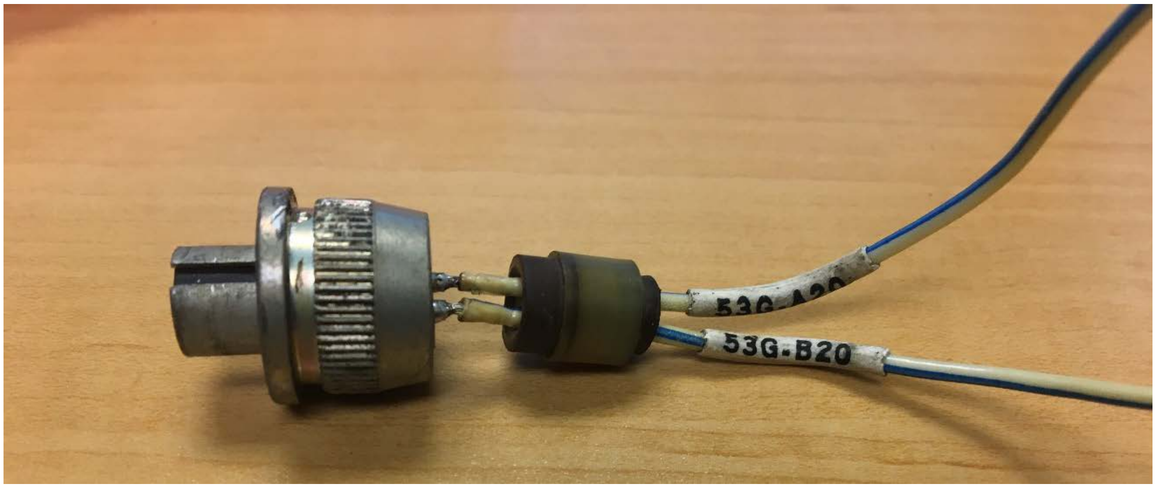


Figure 16 - LH outboard wiring with original wire markings

4.2.2.1.3 LH Inboard (55G):

Part Number: 9542653-1
Serial Number: SEP78-1082

The unit was manufactured in September of 1978.

No notable signs of visible damage were noted, Figure 17. The aircraft wiring harness had no indications of any field splice on the wiring. There was some additional orange seal tape on the wiring harness.

Figure 17 – LH inboard transducer

With the aircraft harness installed, the unit passed the insulation resistance and dielectric test points. All remaining tests failed. If the harness was lightly moved, an intermittent voltage could be read from the device, but could not be sustained and resulted in a failed test.

With the aircraft harness installed, the pin to pin resistance was reading an open circuit. Rotating the sensor would intermittently flash a resistance but would then fall back to an open circuit. The output voltage at 900 RPM in the CW and CCW direction was not measured.

With the aircraft harness removed. The unit passed all tests within expected results.

Without the aircraft harness installed, the pin to pin resistance was 308.1 Ω. The output voltage at 900 RPM in the CW direction was measured to be 7.05 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -7.09 VDC.

After the testing the insulation was removed from the wiring harness, ref Figure 18. The solder connection at Pin B was found to be fractured and not making solid contact with the pin.

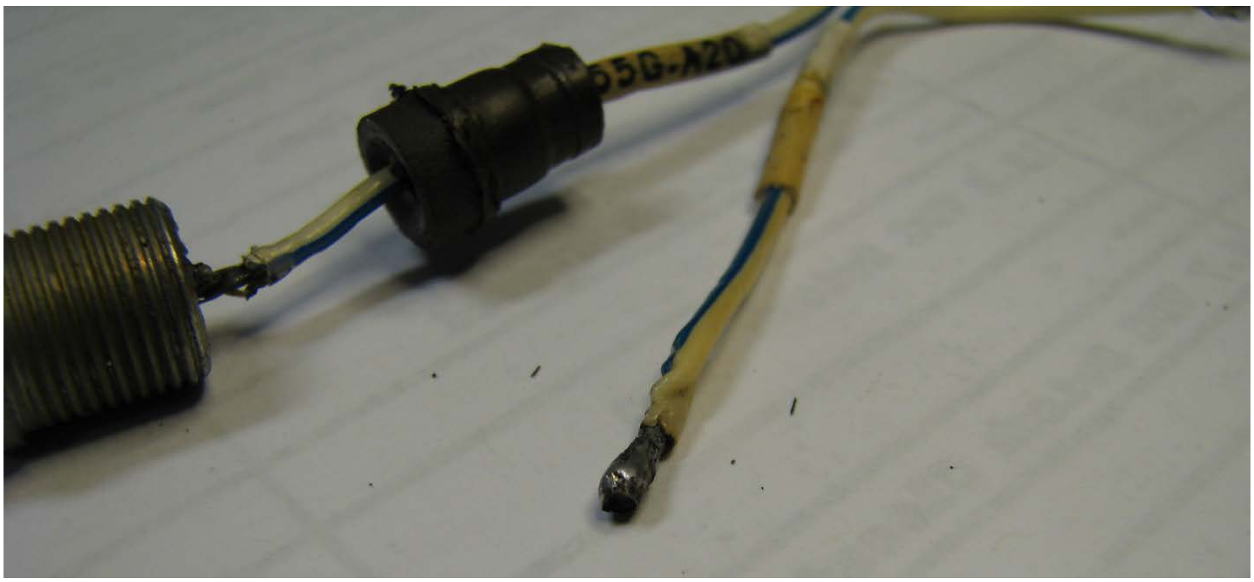


Figure 18 - Broken wire to pin on wheel speed transducer connector

4.2.2.1.4 RH Inboard transducer:

Part Number: 9542653-1
Serial Number: JAN79-1138

The unit was manufactured in January of 1979.

No notable signs of visible damage were noted, ref Figure 19. The aircraft wiring harness had indications of a field splice on the wiring and was covered using vinyl electrical tape. The electrical tape was not removed and remained in place for the CT scans.

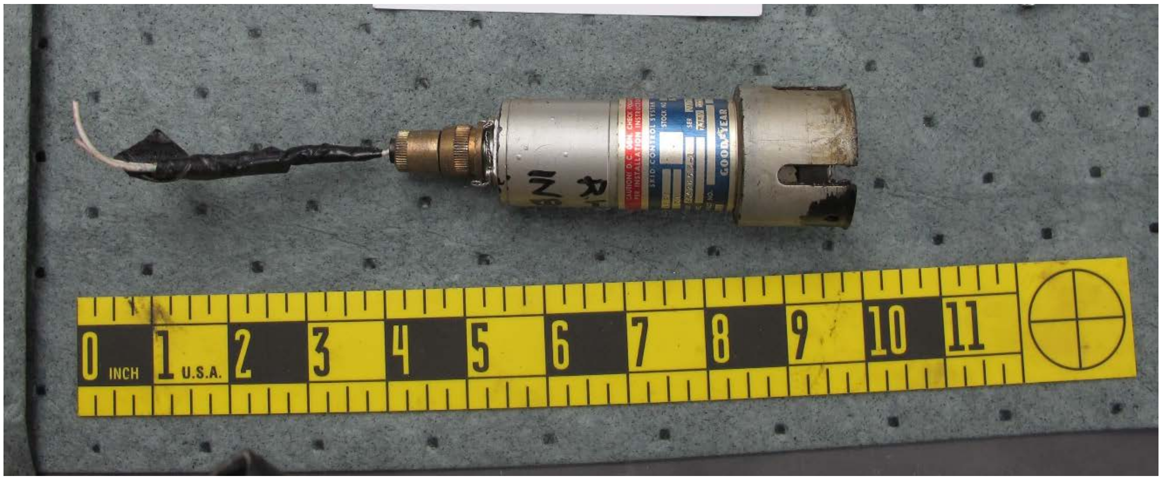


Figure 19 – RH inboard transducer

The unit passed all tests within expected results, with and without the aircraft wiring harness installed.

During the CT scans, Pin B on the transducer was found to be bent and appeared to be in contact with the side of the connector insulator (Refer to CT factual report for images of fouling condition). Prior to removing the aircraft harness, the insulation resistance and dielectric testing was repeated, but a portion of the paint on the wheel adapter was removed to provide a metal to metal contact. Both tests failed. In order to replicate an aircraft installation and possible grounding path, the probe lead was connected to the locking bolt hole. Both the IR and dielectric tests failed at voltages exceeding 350 V (the test required 500 V). The test was repeated using an exemplar unit and the unit passed the testing.

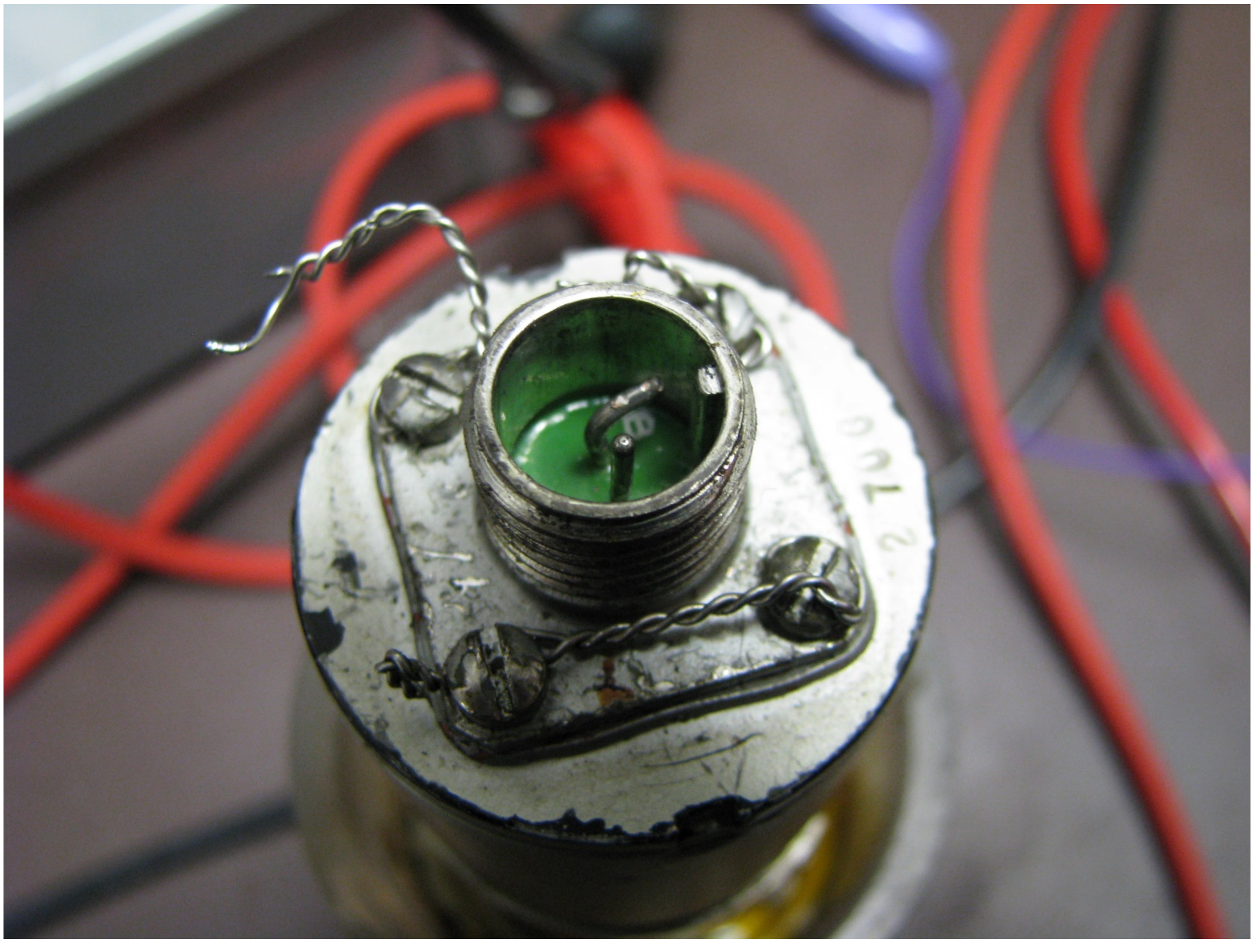


Figure 20 - RH inboard transducer bent pin B after connector removal

The harness was removed, and the bent pin examined, ref Figure 20. When the connector was removed, no obvious signs of the pin contacting the connector side were visible. The insulation resistance and dielectric testing was repeated, and the unit passed per the FTI requirements.

With the aircraft harness installed, the pin to pin resistance was 330 Ω. The output voltage at 900 RPM in the CW direction was measured to be 7.00 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -7.02 VDC.

Without the aircraft harness installed, the pin to pin resistance was 326 Ω. The output voltage at 900 RPM in the CW direction was measured to be 6.99 VDC. The output voltage at 900 RPM in the CCW direction was measured to be -6.99 VDC.

After the testing the electrical tape was removed to inspect the wire splices. It was noted that the wiring between the aircraft and the connector were of two different wire gages. Refer to the RH outboard sensor, section 4.2.2.1.1 regarding wire splice and maintenance practices.

4.2.2.2 Servo Valves

The servo-valves were mounted in the left and right main landing gear bays, ref Figure 21 and Figure 22. Due to the damage to the landing gear attachment points, some hydraulic lines were damaged in the local area. Each servo valve was removed along with a portion of the hydraulic line to each of the ports. The hydraulic lines were cut upstream of check valves within the system. The flex hoses from the distribution blocks to manufacturing breaks on the main landing gear yokes were disconnected. Electrical connections were cut, upstream of the connector and the connectors were not removed.

Figure 21 - RH Servo-Valve as installed

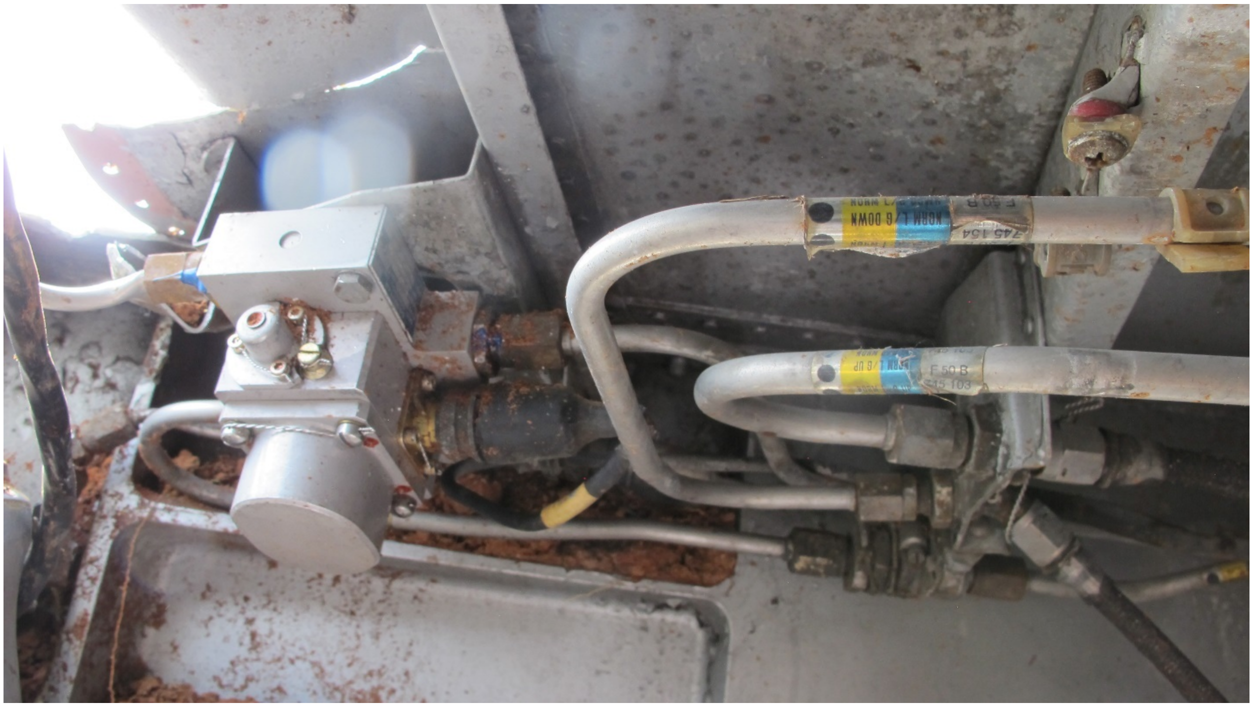


Figure 22 - LH Servo valve installed

4.2.2.2.1 LH Servo Valve

Part Number: 9542732-2
Serial Number: AUG80-1677D

Based on serial number the valve was originally manufactured in August 1980.

The valve passed the coil resistance and insulation resistance, leakage and proof pressure tests. The valve failed the pressure control checks and the pressure drop tests.

The valve was then tested using a supply pressure of 1500 psi and varying the input current to the flapper valve motor from 0 to 8.5 mA while monitoring brake pressure. The output was plotted on an X-Y plotter and compared to expected results. The brake pressure did not fall within the expected results (see attached data sheet). When the maximum current was applied, the valve should output 0 psi to the brake pressure port and all pressure to the return port. Throughout the test, the valve would produce between 1500 and 700 psi at the brake pressure outlet.

The supply pressure was reduced to 300 psi, per the FTI and the brake pressure continued to remain outside of the expected test ranges.

The unit was then disassembled to visually inspect the internal components of the valve, ref Figure 23.

The visual inspection of the filter assembly did not show any obvious signs of blockage or debris. The filter was rinsed using an alcohol solution and one black speck of material was visible.

The spool and sleeve were removed. The sleeve and packing material were inspected under a microscope. The seals visually appeared to be intact and installed correctly. Areas of discoloration and corrosion were noted on some of the spool surfaces. The spool would move freely in the spool when acted upon by gravity. No obvious signs of friction were noted.

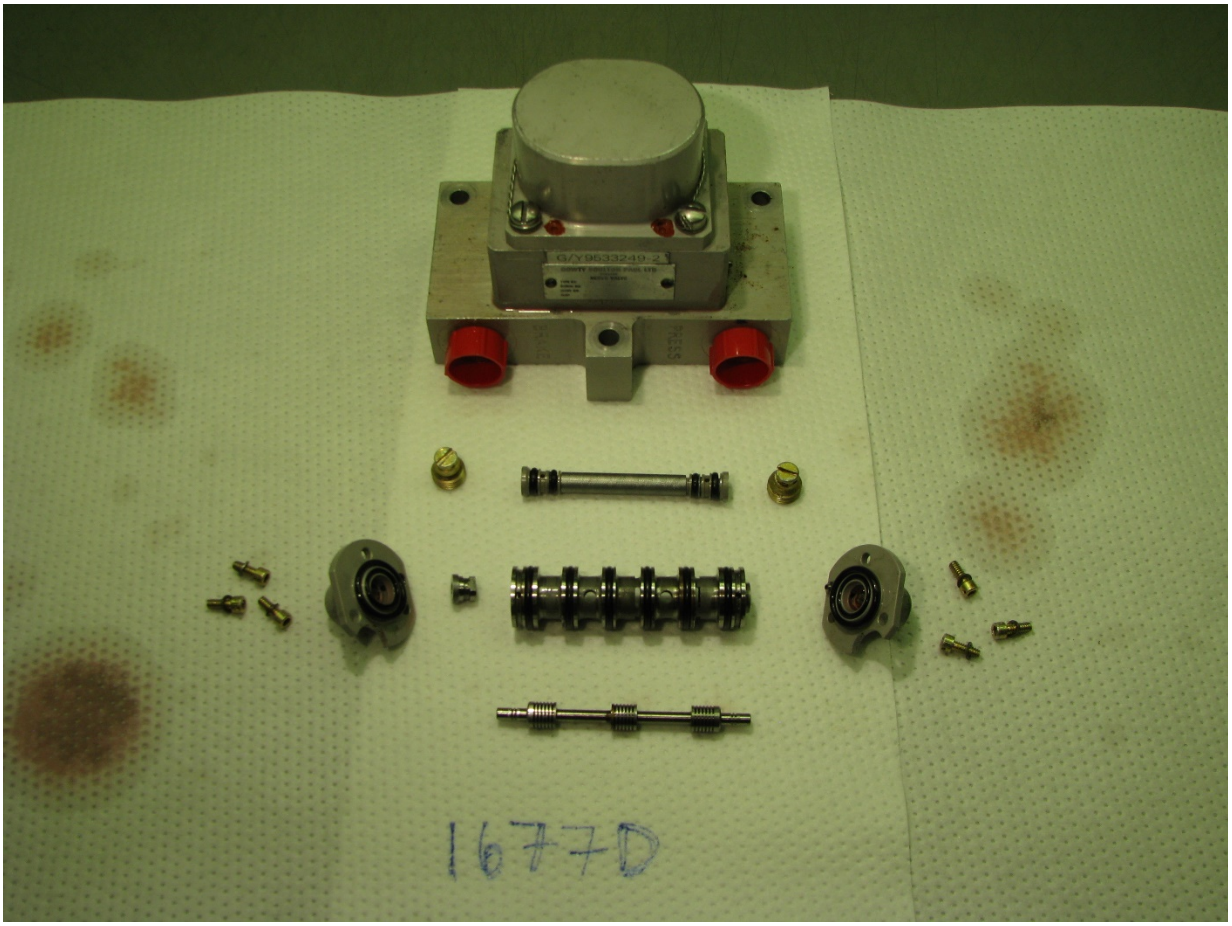


Figure 23 - LH disassembled control valve

4.2.2.2.2 RH Servo Valve

Part Number: 9542732-2
Serial Number: JULY67-343D

Based on serial number the valve was originally manufactured in July 1967.

The valve passed the coil resistance and insulation resistance, leakage and proof pressure tests. The valve failed the pressure control checks and the pressure drop tests.

The valve was then tested using a supply pressure of 1500 psi and varying the input current to the flapper valve motor from 0 to 8.5 mA while monitoring brake pressure. The brake pressure was plotted on an X-Y plotter and compared to expected results. The brake pressure did not fall within the expected results (see attached data sheet). When the maximum current was applied, the valve should output 0 psi to the brake pressure port and all pressure to the return port. Throughout the test, the valve would also produce between 1500 and 600 psi at the brake pressure outlet.

The supply pressure was reduced to 300 psi, per the FTI and brake pressure continued to remain outside of the expected test ranges.

The unit was then disassembled to visually inspect the internal components of the valve, ref Figure 24.

The visual inspection of the filter assembly did not show any obvious signs of blockage or debris. The filter was rinsed using an alcohol solution no foreign material was visible.

The spool and sleeve were removed. The sleeve and packing material were inspected under a microscope. The seals visually appeared to be intact and installed correctly. Areas of discoloration and corrosion were noted on some of the spool surfaces. The spool would move freely in the spool when acted upon by gravity. No obvious signs of friction were noted.

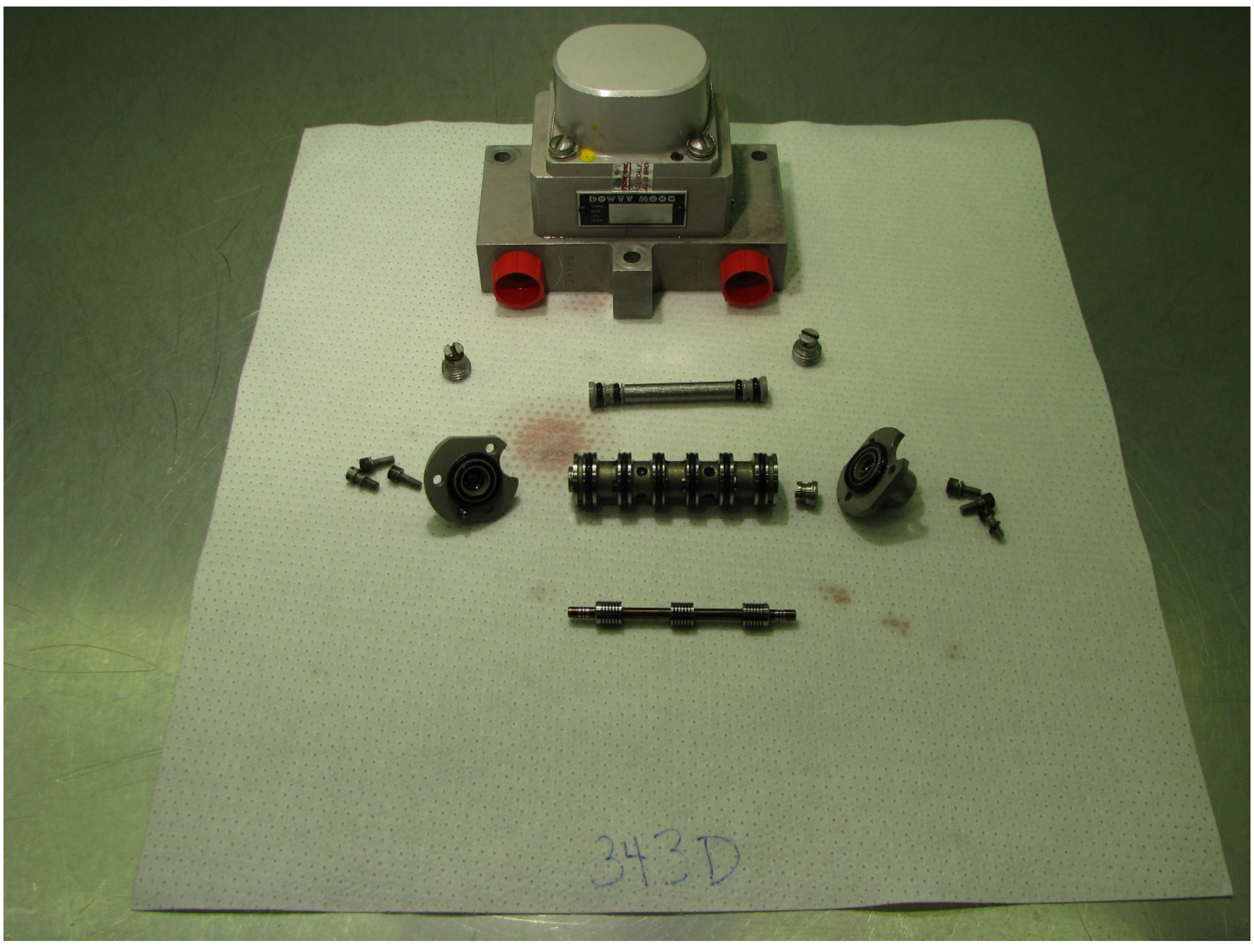


Figure 24 - RH disassembled control valve

4.2.2.2.3 Servo Valve Test Setup Follow-up

Since both servo valves failed pressure control checks and the pressure drop tests, the group requested that MABS test an exemplar valve. Dassault provided a flight approved spare servo valve and the test was conducted by MABS using the same equipment and procedure as the accident aircraft servo valves. The valve passed all of the tests per the FTI.

4.2.2.3 Control Unit

The antiskid control unit was recovered from the accident aircraft.

Part Number:	6002614-3 Rev C
Serial Number:	NOV80-087

Based on the serial number the unit was manufactured in November 1980.

PUI noted that they did not have any records of the unit being sent to their facility for any repairs or overhauls.

The unit was tested at the PUI facility under supervision of the NTSB using PUI personnel and procedures. Testing was conducted using the Functional Test Instructions (FTI) for Control Box Assembly P/N 6002614-3 Rev Basic dated 07/23/85 (Doc. Number 5D1-5016).

The unit was connected to the test unit, which tests one cannon plug at a time. The J1 connector (left) passed all the FTI steps[6].

[6] Initially test step 3.5.2 failed with an expected voltage outside of the acceptable range. The test was repeated, and the unit was within acceptable limits.

Figure 25 - Control Unit and test bench setup.

The J2 connector (right) passed all tests except for the outboard pressure control rate validation test, inboard pressure control rate validation test and the valve control signal validation test.

During the outboard pressure control rate validation test and the inboard pressure control rate validation test, the output current to the control valve was expected to drop to 0 mA within a certain voltage. The current would not drop to zero at the maximum voltage value. The current would then steadily increase over time without any input from the test bench. After approximately 10-15 minutes the current increased from roughly 0.5 mA to 5.5 mA. It should be noted that when the control valve output current is zero, the control valve would port full hydraulic pressure to the brakes (as modulated by the brake pedals) as current increases, the pressure balance will tend to drive towards the return port, effectively removing pressure from the brakes.

During the valve control signal validation test, both parts A and B the voltage response did not fall within the acceptable range. Figure 26 and Figure 27 highlight the voltage response from the unit during the test and shows the area that that voltage response is expected to fall into during an acceptable test.

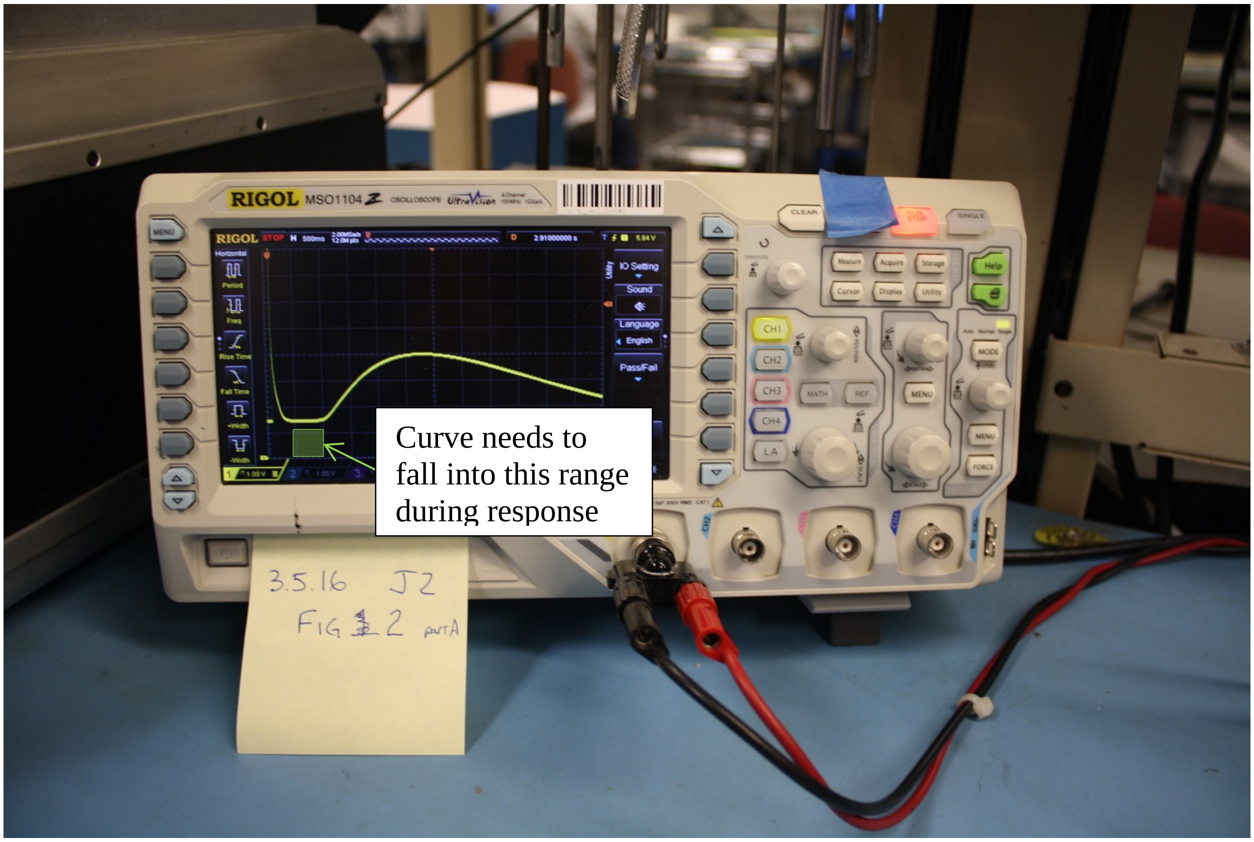


Figure 26 - Voltage Response for valve control signal validation test part A

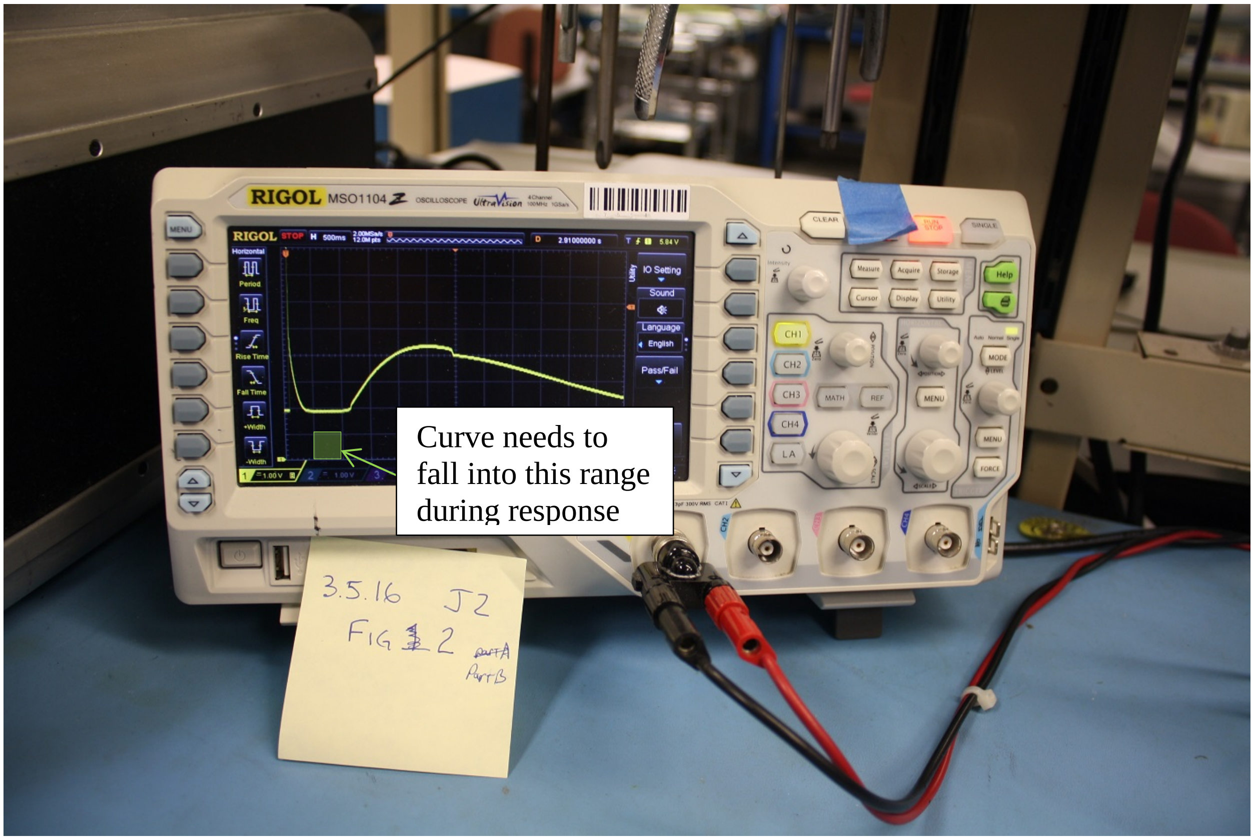


Figure 27 - Voltage response for valve control signal validation test part B

After the testing per the FTI was completed, some additional tests were completed to simulate the response of the control unit from both the J1 and J2 connectors and check the output current to both the control valves during simulated wheel speed transducers failures.

Wheel speed transducers were simulated using a resistor (365 Ω) and a power supply. The power supply voltage could be regulated to simulate wheel speeds and skid conditions. The voltage could also be reversed to simulate a reverse polarity. The test condition was setup using the air/ground switch and the control valve currents were monitored.

During the testing, when the control valve current was in the “full-scale” position (approximately 21.00 mA measured on a Fluke meter), on the aircraft this would open the valve and port hydraulic fluid to the return system and not to the brake assembly. If the control valve current is at 0 mA, full hydraulic pressure is applied to the brakes and is regulated by pedal pressure.

4.2.2.3.1 Simulated Test with one side (RH) operational and one failed transducer on opposite side (LH)

The test setup simulated an open circuit on one-wheel speed transducer on the left side, one functioning wheel speed transducer on the left side and two functioning wheel speed transducers on the right-hand side. The test set was placed in air mode and both control valve currents were driven full scale. This was designed to show the unit simulating the locked wheel protection and hydraulic pressure was ported to the return hydraulic system. Two seconds after the air/ground switch was placed in Ground mode, the valve current on the RHS (with two operating transducers) would reduce to 0 mA (full hydraulic pressure to the brake) and the LHS (with one operating transducer) would remain at the full-scale range value (no hydraulic pressure to the brakes). If one of the wheel speed transducers on the RHS was reduced to simulate a skid or locked wheel condition, the current to the servo would go full scale, which would be an attempt to control the skid or protect from a locked wheel condition[7].

4.2.2.3.2 Simulated Test with one side (RH) operational and one reverse polarity transducer on opposite side (LH)

The test setup simulated a reverse polarity on one-wheel speed transducer on the left side and a proper signal to the other left-hand transducer and two functioning transducers on the right-hand side. The test set was placed in air mode and both control valve currents were driven full scale. This test was designed to show the unit simulating the locked wheel protection. With the air/ground switch remaining in Air, the voltage on both transducers (normal and reverse polarity) would be increased. At no time during the test did the control valve current reduce to 0 mA and would remain in the FSR. This test was repeated on both the inboard and outboard wheel speed transducers.

4.2.2.3.3 Simulated Test with one side (RH) with one failed transducer and on opposite side (LH) one reverse polarity transducer.

The test simulated one failed wheel speed transducer on the right-hand side, and a reverse polarity sensor on the left-hand side. During this test the current to the control valves remained in the full-scale range position (no hydraulic pressure to the brakes) where no braking would be available if the anti-skid system was selected on.

4.2.2.4 Wheel Speed Transducer Wiring Harness on RH MLG

The RH MLG harness was received in a protective bag and contained two separate sections of the harnesses, ref Figure 28 after being removed from the aircraft. During the aircraft wreckage removal process, the harness was cut was cut approximately 1-2 feet from the MLG disconnect (P182), which is located in the RH gear wheel well, to allow for gear removal, ref Figure 29. Prior to removal from the MLG, each wheel speed transducer location was marked on the two-wire set that was a part of the transducer system wiring. The transducers wires were cut from the aircraft during the initial inspection by the systems group.

[7] During a skid condition, the system would cycle between applying and removing brake pressure at a high rate from the effected wheel, based on the fluctuating wheel speed transducer reading.

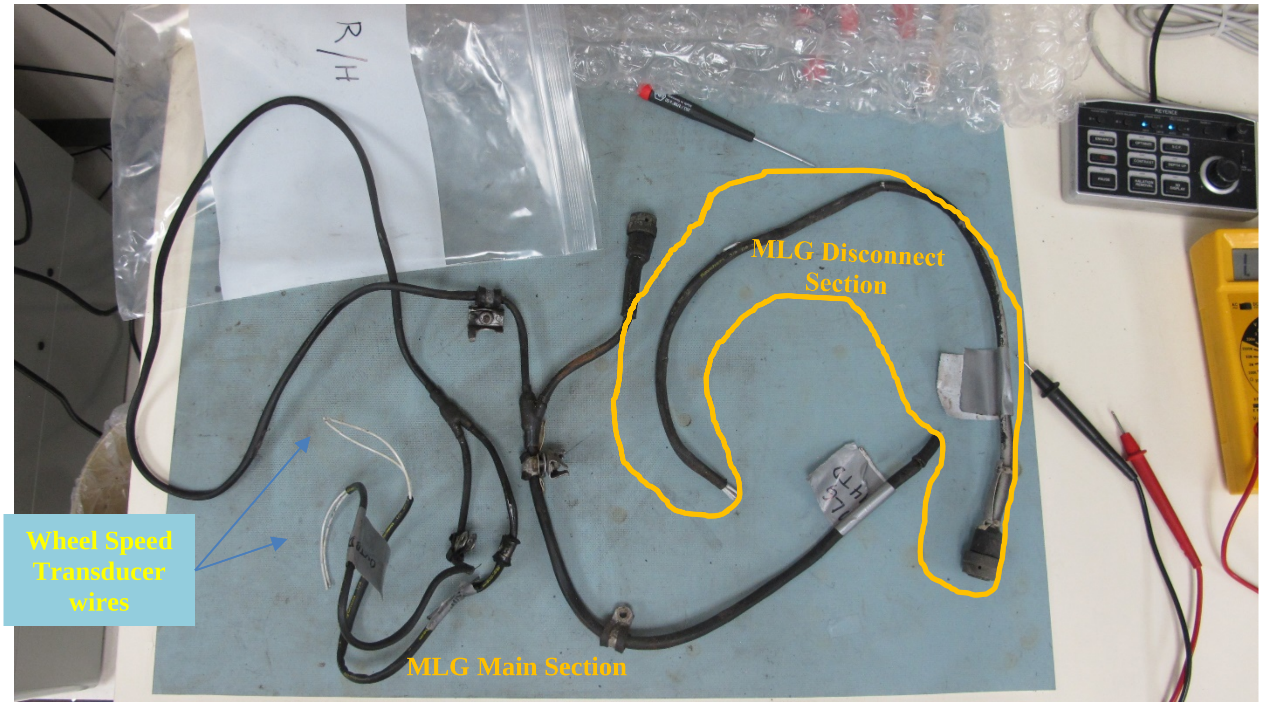


Figure 28 - RH MLG harness as received

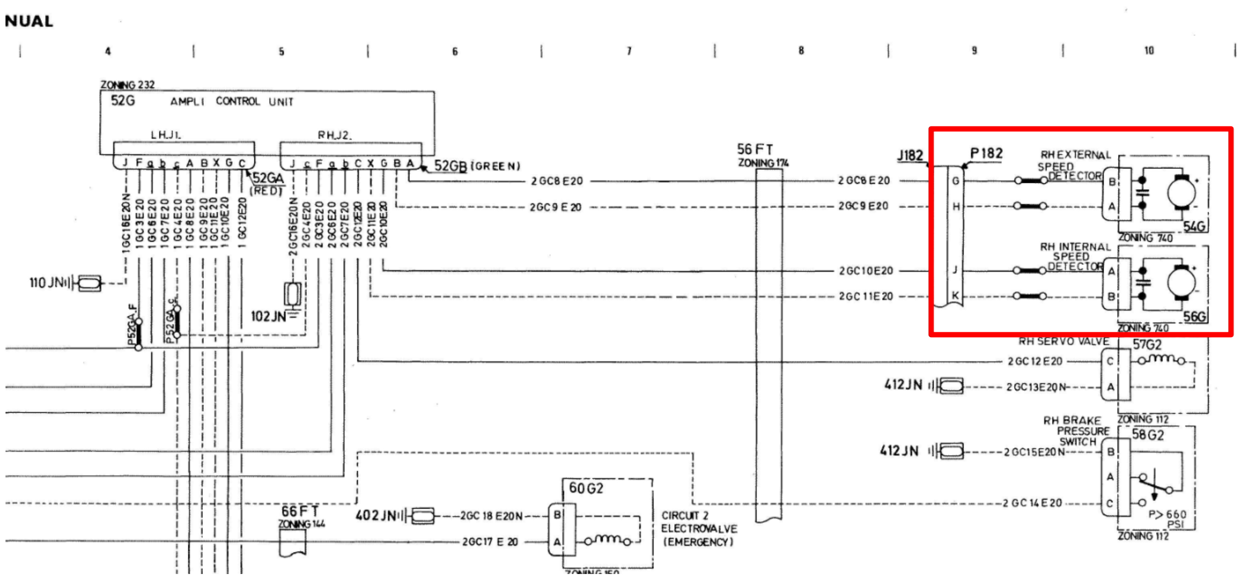


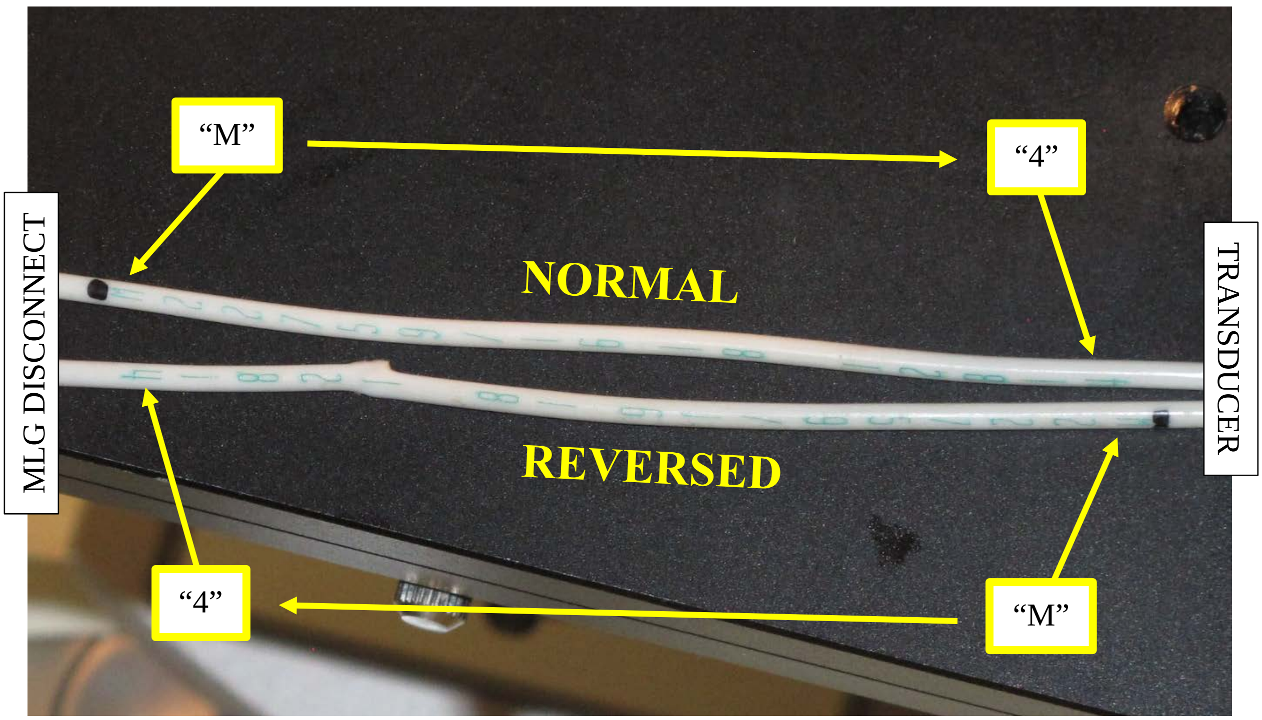
Figure 29 - Portion from Falcon 50 wiring diagram manual with harness section highlighted

The wiring outer insulation was removed to identify the individual wires. None of the wires contained individual markings. The only marks noted on each individual wire, was a manufacturer wiring identifier. The wiring was marked with “M22759/16-18 12814”, in green print. The marking was broken down to the following components:

- “M22759” - Mil-Spec MIL-W-22759

- "16" - Conductor Material: Stranded Tin-Plated Copper
- "18" - Wire Size: 18 AWG
- "12814" - CAGE Code[8]

The wiring marks were in a constant direction along each individual wire. In order to assist in wiring identification, the wire marking was identified to be "Normal" when the first letter in the marking ("M"), was in the direction closest to the MLG disconnect (P182). "Reversed" marking was when the last letter in the marking ("4"), was in the marked in a direction closest to the MLG disconnect. The marking repeats approximately every 11 inches.



On the MLG section of the wiring harness, based on the direction of the wiring markings, the individual wheel speed transducer positive (+) and negative (-) wires were identified on the main section of the harness. The wires were then traced to the end nearest the MLG disconnect (P182) using a digital multi-meter.

On the MLG disconnect section of wiring containing the circular connector at the disconnect point, the wires corresponding to each pin location were identified using a digital multi-meter. Based on the aircraft wiring diagram manual (WDM) Pins "G", "H", "J". and "K" all correspond to the wheel speed transducers, ref Figure 29.

Due to the absence of wire identifications, in order to determine the wiring continuity at the location of the wiring cut during the recovery operations, measurements were taken from the "M" or "4" character in the wire labels to the cut and from the cut to the "M" or "4" on both wire sections, the

[8] The CAGE (Commercial And Government Entity) code is used to identify a component supplier.

MLG disconnect and MLG main wiring sections. The direction of the labelling based on the established convention was also noted. Table 2 and Table 3 contain the observed measurements of the wiring marks.

Table 2 - Wiring Marking Measurements on RH MLG Disconnect Section

Connector	Pin	Label Direction	Distance (inches)
54G	G	Normal	7
	H	Reversed	2 ½
56G	J	Normal	7*
	K	Reversed	2 5/8
* = Measurement was slightly above the measurement but less that than the next 1/8” tape measure increment			

Table 3 - Wiring Marking Measurements on RH MLG Main Section

Wire	Label Direction	Distance (inches)
Inboard +	Reversed	8*
Inboard -	Normal	3 ¾
Outboard +	Reversed	8 ¼
Outboard -	Normal	3 ¾*
* = Measurement was slightly above the measurement but less that than the next 1/8” tape measure increment		

The measurements and wiring label directions the determination were consistent with the wires being routed as stated in Table 4:

Table 4 - Wiring Routing on RH MLG Main Section

Connector	Pin	Transducer Connection	WDM Expected Connection
54G	G	**Outboard -**	***Outboard +***
	H	**Outboard +**	***Outboard -***
56G	J	**Inboard -**	***Inboard +***
	K	**Inboard +**	***Inboard -***

4.2.3 MLG

The main landing gear on each side was detached from the trunnion mounts and found folded under the aircraft wing, ref Figure 30 and Figure 31. The inspection of the tires and brakes were conducted after the aircraft was relocated to the secured facility.

Figure 30 - LH Main gear as found

Figure 31 - RH Main gear as found

Each of the main landing gear tires was inspected. The tires appeared to be in good condition and appeared to be recent installations on the aircraft. The grease around the wheel bearings was bright red in color and generously applied. Only the right inboard tire had a single area of a flat-spot on the outermost tread area and included a cut on the side of the tire. Tire pressures along with tread depth measurements were recorded in Table 5. The tread depth was checked in multiple places and the value recorded is a representative depth around the tire circumference.

Table 5 - MLG tire pressure and tread measurements

	L OTBD	L INBD	R INBD	ROTBD
Tire Pressure (psi)	208	209	210	210
Tread depth (in 32^{nd} of inch)	3-4	3	3-4	4-5

4.2.4 Brake Assemblies

The left and right main landing gear were removed from the aircraft during the recovery process. The brake assemblies (part number and serial numbers documented in Table 6) from each wheel were inspected by the group.

Table 6 - Brake assembly part numbers and installed locations

Brake	L OTBD	L INBD	R INBD	ROTBD
Part Number	5003280-5	5003280-5	5003280-5	5003280-5
Serial Number	JAN 90-1254	JUL 82-726	JUL 82-744	JAN 84-938

To check for brake wear levels, each brake system, A and B were pressurized individually to 1,500 psi using a red oil hand pump[9]. The system was left in a pressurized state for 1-2 minutes to observe any pressure loss/leaks. When one side was pressurized, the other inlet port remained open to observe if any cross-system leaks occurred. While pressure was applied to one of the sides (A or B), a brake wear gage provided by the OEM was used to measure the brake wear, ref Figure 32. The measurement was checked in accordance with manufacturer specifications. For the brakes to be considered within serviceable limits, the gage would have to be in contact with the brake wear pins and the wear gage not be in contact with the spring housing. If the wear gage would be in contact with the spring housing and not in contact with the return pin, the brake would be considered outside of serviceable limits and require removal and overhaul.

[9] Per the brake assembly CMM, the hydraulic ports are identified as A and B, whereas the aircraft hydraulic systems are identified as 1 and 2.

Table 7 summarizes the results of the brake wear limit checks and pressure checks.

Table 7 - Results from brake assembly inspections

Brake	1,500 psi Pressure Check		Wear Gage Check	Distance from Wear Gage and Spring Housing (System Pressurized)
	System A	System B		
LH Inboard	Pass	Pass	Serviceable	~4 mm (Sys A)
LH Outboard	Pass	Pass	Serviceable	~3 mm (Sys B)
RH Inboard	Pass	Pass	Serviceable	~2 mm (Sys B)
RH Outboard[10]	Pass	Pass	Serviceable	~3 mm (Sys A)

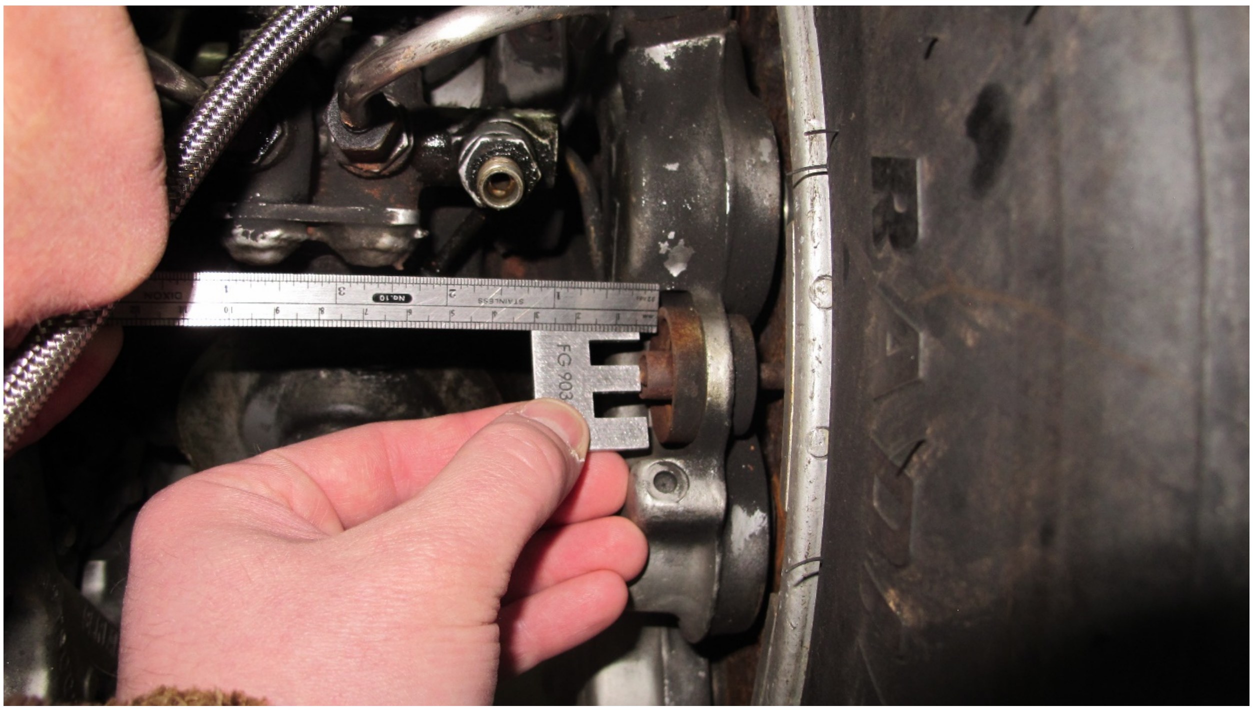


Figure 32 - LH Inboard Brake Wear Check

It was noted during the inspection that the RH Inboard brake assembly system B inlet fitting was able to be moved by hand (loose), however, no leaks were noted during the pressure testing.

During the setup of the RH Outboard brake assembly check, both system A and B inlet fittings were loose. The system A inlet leaked when initial pressure was applied, and pressure would not hold. The fittings were tightened, and system A was rechecked. The system retained pressure. The system B inlet was checked without tightening the inlet fitting and the system held pressure with no signs of leaking.

The group performed an additional visual inspection of the RH inboard brake assembly with the wheel removed, ref. Figure 33. The brake disk surface contained areas of visible rust. The brake rotors freely rotated by hand. Normal wear pattern of the stator pads was also noted (visual from the

[10] See write-up for additional details.

outside diameter). Measurements of the individual rotor and stators were taken and were consistent with a brake assembly within normal serviceable limits.

Figure 33 - RH Inboard brake assembly, wheel removed

4.2.5 Brake Pedal Linkage and Selector Valve

Access to visually inspect the brake pedal linkages was gained through an access panel on the forward pressure bulkhead. Figure 34 contains a view of the pilot side brake pedals from an exemplar aircraft. The area where the brake pedals were located sustained crushing damage which precluded any operation of the individual brake pedals and limited the visual inspection, ref Figure 35. Of the area that could be viewed, all the brake connections were per specification and all attaching hardware was present. The brake pedal connecting rods were bent in directions consistent with the crush damage to the surrounding area.

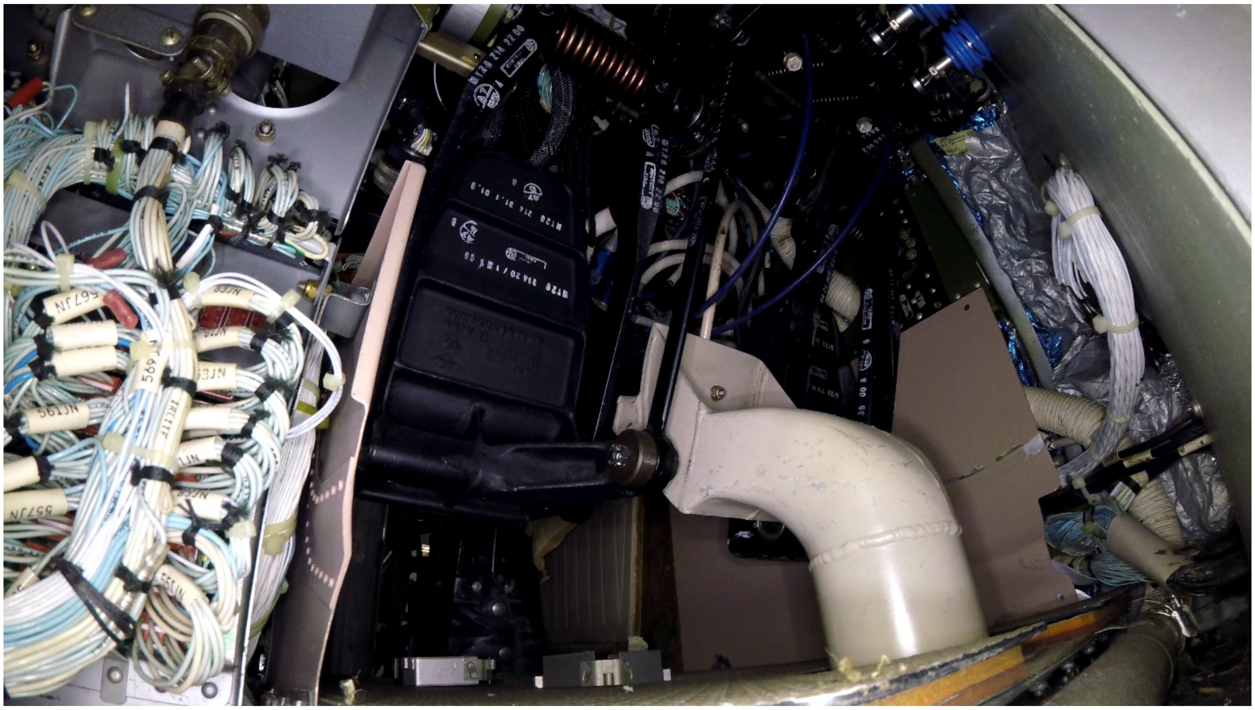


Figure 34 - View of pilot brake pedal connections in exemplar aircraft

Figure 35 - View of pilot brake pedal area on accident aircraft

The normal/emergency brake selector valve which is mounted to the forward pressure bulkhead was visually inspected, ref Figure 36. The cover was separated from the pressure bulkhead and the selector valve was fractured just forward of the mounting flange, ref Figure 37. The fracture was consistent with overload. There were no visible signs of hydraulic fluid leakage around the hydraulic fittings, ref Figure 38.

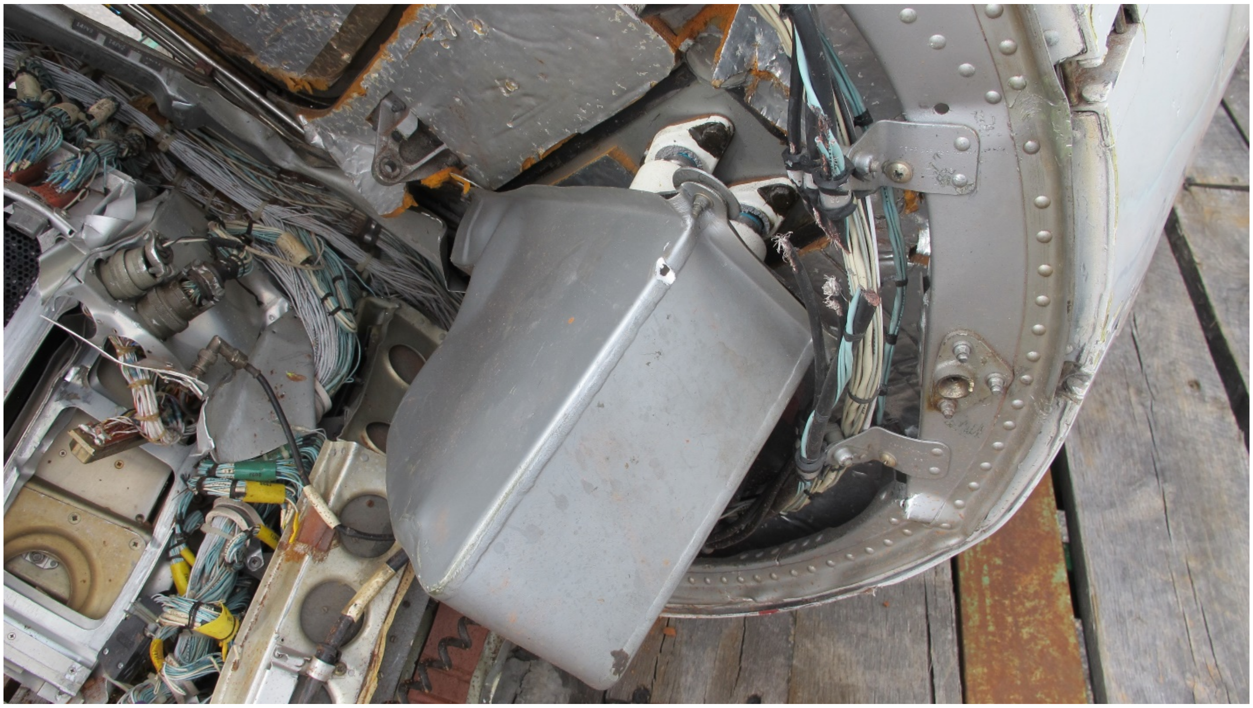


Figure 36 - Normal/Emergency Brake Selector Valve with damage to the cover.

Figure 37 - Normal/Emergency Brake Selector Valve with fractures at the pressure bulkhead

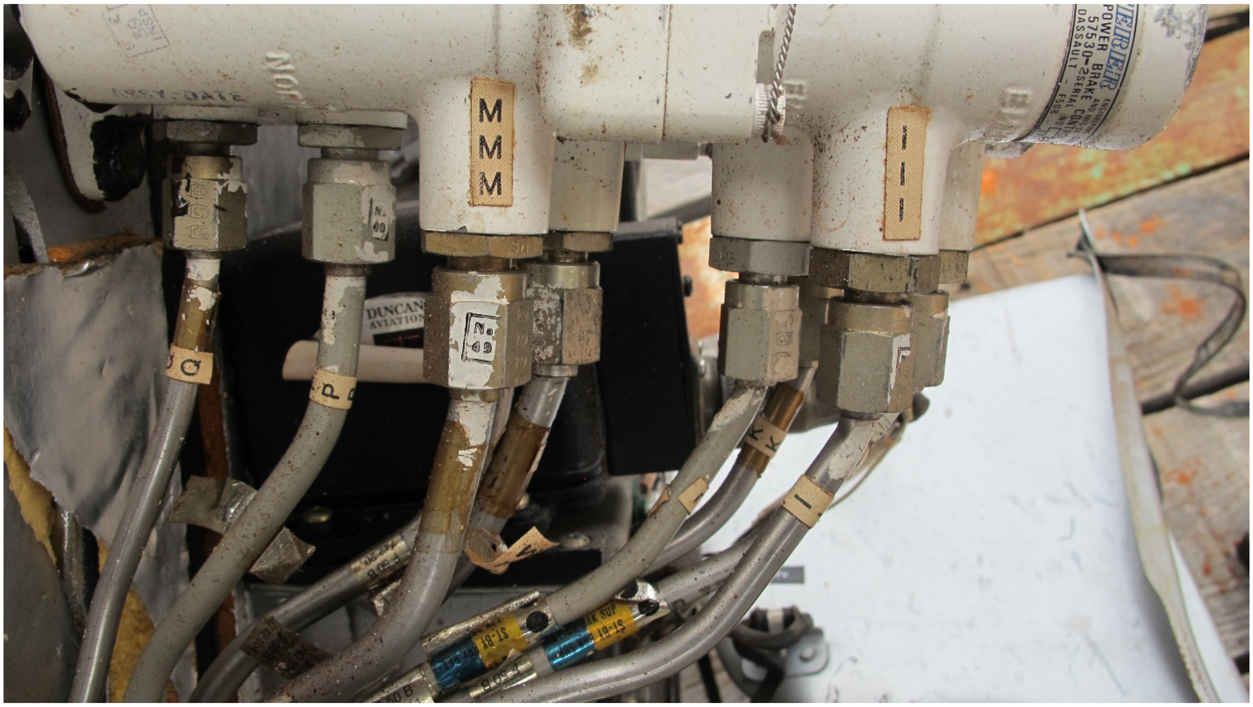


Figure 38 - Hydraulic lines from the selector valve

4.2.6 Parking Brake

The cockpit was partially separated from the fuselage in the accident and completely severed during the recovery process.

The parking brake cable end at the cockpit section break was located and the handle was exercised in the cockpit. Continuity of the control cable was confirmed from the parking brake handle to the cockpit section break.

The cable end in the fuselage section was also located and it was attempted to be moved. The lower fuselage area where the control cable ran had areas of crushing damage and required portions of the skin and structural frames to be pried away from each other to gain access to the area. Due to the crushing damage of the fuselage, the control cable would not move and additionally, the control cable end at the parking brake selector valve was broken at the lever arm.

The hardware to attach the parking brake control cable to the selector valve lever arm were present and secure, except for the cotter pin. The fracture surface on the lever arm was consistent with an overload failure. The parking brake selector valve was also displaced from its normal mounting location and rotated by approximately 90 degrees and impacted against the frame assembly. Figure 39 contains an image of the parking brake selector valve installed in an exemplar aircraft. Figure 40 and Figure 41 show the parking brake selector valve as found in the accident aircraft and includes the fractured end of the selector valve lever arm. The hydraulic lines in the area had multiple areas of crushing damage and breaches in the line runs.

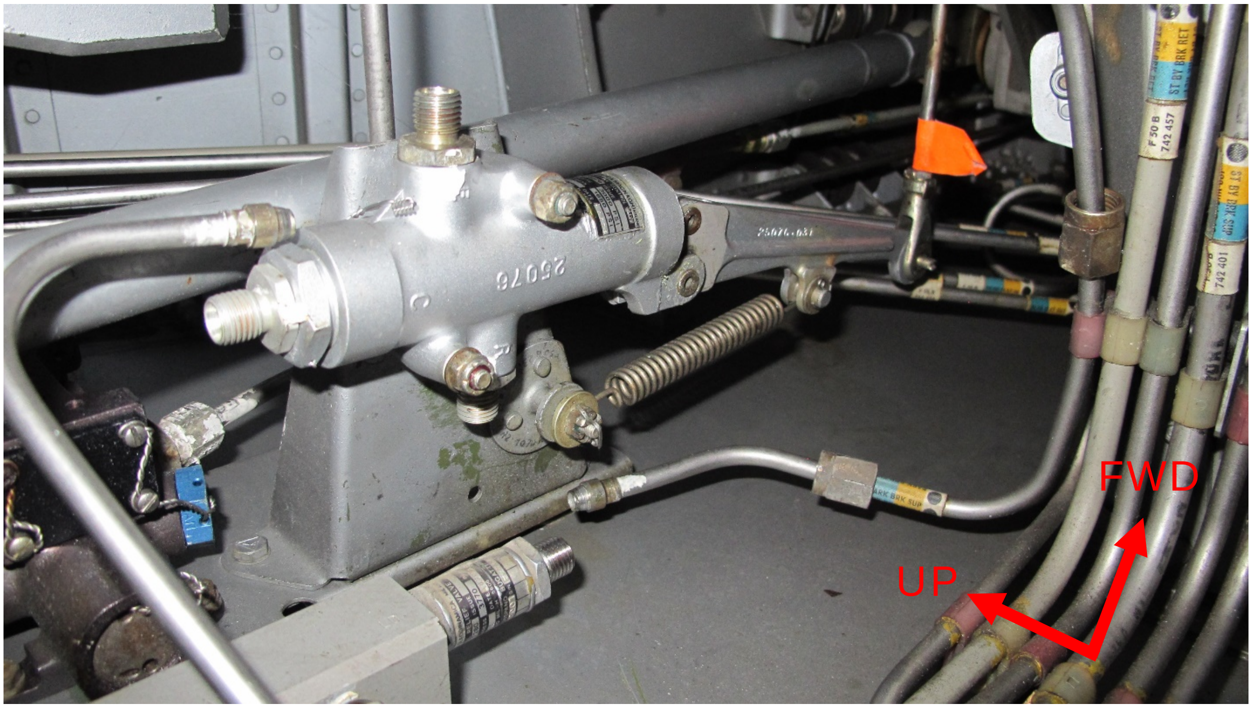


Figure 39 – Exemplar installation of parking brake selector valve

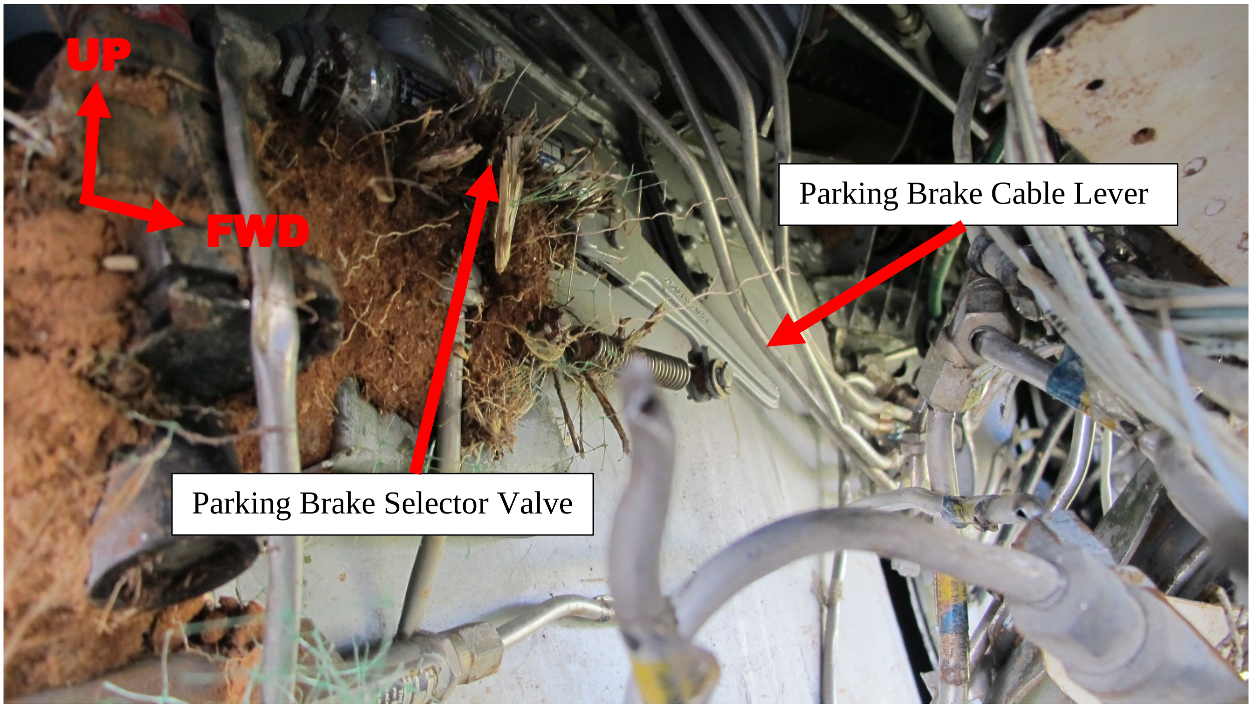


Figure 40 - Accident aircraft location of parking brake selector valve

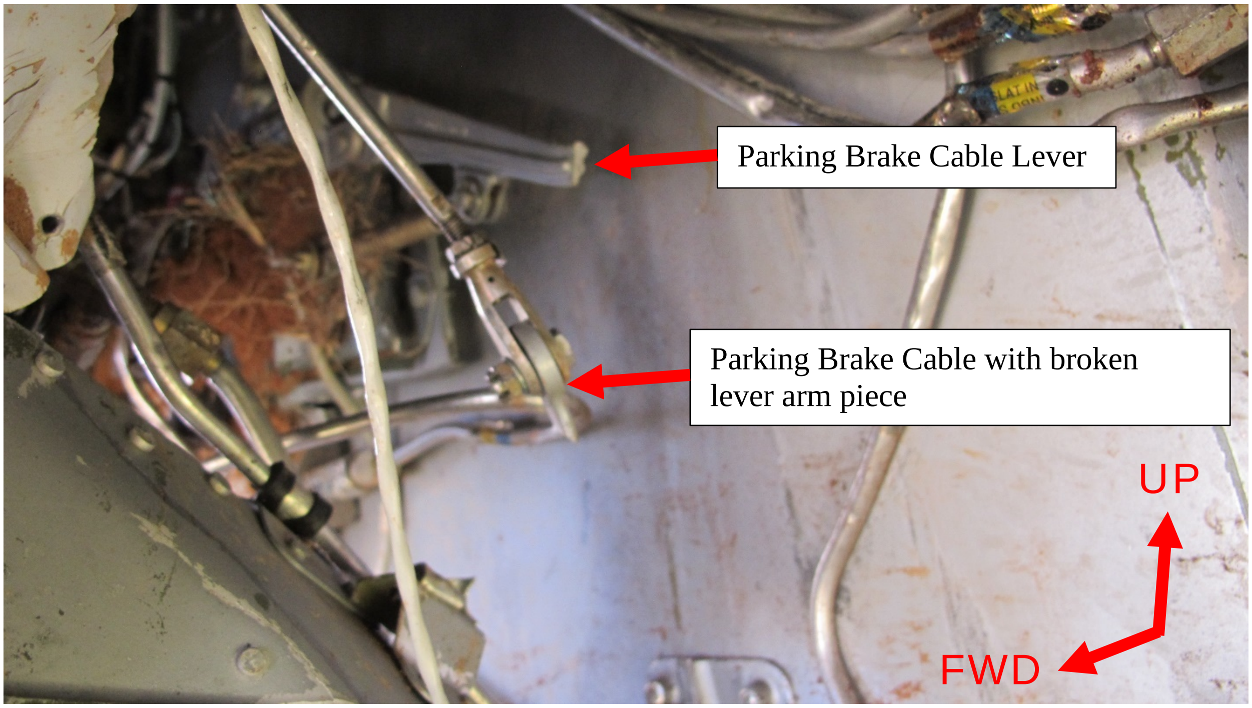


Figure 41 - Parking brake cable and selector valve severed connection

4.2.7 Hydraulic Samples

Five hydraulics samples were taken from the accident aircraft at various points on the #1 and #2 hydraulic systems after the aircraft had been moved from the accident site but prior to being transported to the secure wreckage storage site. They were collected in glass jars and sealed.

Hydraulic system 1 reservoir fluid quantity was found to be below the sight glass. A fluid sample was drawn from the reservoir and collected in a glass jar and sealed.

Hydraulic system 2 reservoir fluid quantity was found to be below the sight glass. A fluid sample was attempted to be drawn from the reservoir, but no fluid could be drained from the system.

Hydraulic system 1 and 2 fluid samples were also taken from the hydraulic lines at the wheel brake assembly connections for the left and right brake system. Multiple areas of breeched hydraulic lines were present on both the left and right MLG brake hydraulics lines. Small amounts of fluid were collected from the #1 system side on both left and right brakes, slightly more amounts of fluid were collected from the #2 system.

All hydraulic fluid samples were tested at the MABS facility under supervision of the NTSB using MABS personnel and procedures. All of the samples were characterized as extremely dirty and the viscosity was lower than specification. Fluid cleanliness and viscosity readings are outlined in Table 8.

Table 8 - Hydraulic sample results

	System 1	System 2
Cleanliness[11]	Class 11	Class 12
Viscosity (measured at 100° F)[12]	12.2 cSt	11.7 cSt

As previously stated, the fluid samples were dirty. During the testing, the samples were noted to have a tacky residue left on the filter screens. The samples were sent to the NTSB materials laboratory for further testing in order to determine the origin of the tacky residue, however due to test equipment failures, the substance could not be identified. Based on available evidence related to the hydraulic system and brake system, further testing of the sample to determine the substance was discontinued.

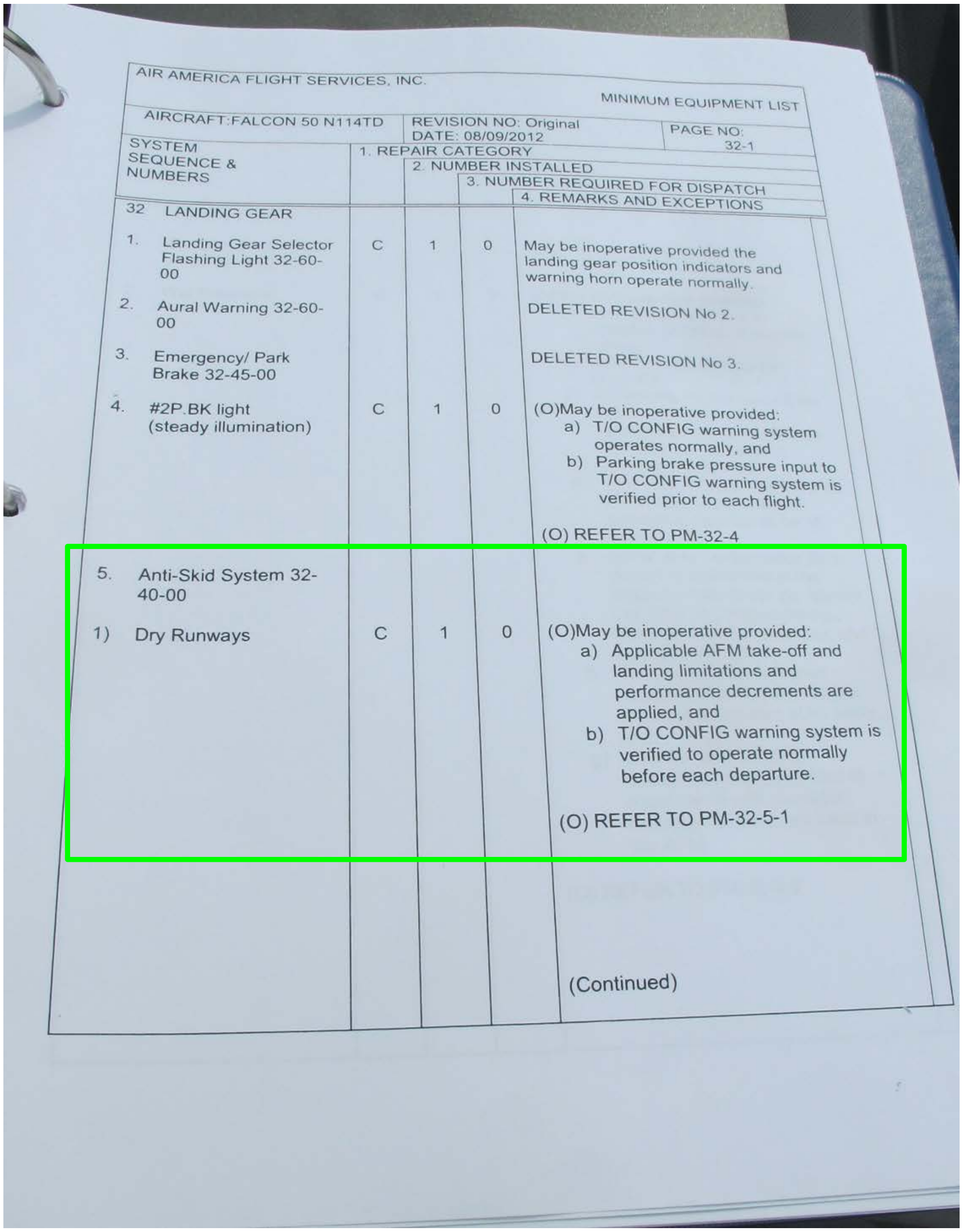
4.2.8 Aircraft Flight Manuals

As discussed in section 4.2.1.1, the brake control panel had a Master Minimum Equipment List (MMEL) sticker placed on the unit stating "ATA# 32-5 INOP" effective September 27, 2018.

Figure 42 and Figure 43 are pictures from the accident aircraft MMEL related to item 32-5. For the dry runway condition, the procedure refers to the Procedures Manual (PM) section PM-32-5-1, ref. Figure 44.

An attempt was made to locate in the aircraft AFM Procedures section 3, subsection 151 which was identified in the PM during landing. The referenced page(s) could not be located in the documentation found on the aircraft. Figure 45 and Figure 46 contain the pages from the AFM abnormal flight procedures related to brake system failures, that was found in the AFM on the aircraft.

[11] The fluid was tested to specification AS4059 Aerospace Fluid Power – Cleanliness Classification for Hydraulic Fluids
[12] Per specification for MIL-PRF-5606, viscosity at 100° F is minimum 13.8 cSt.

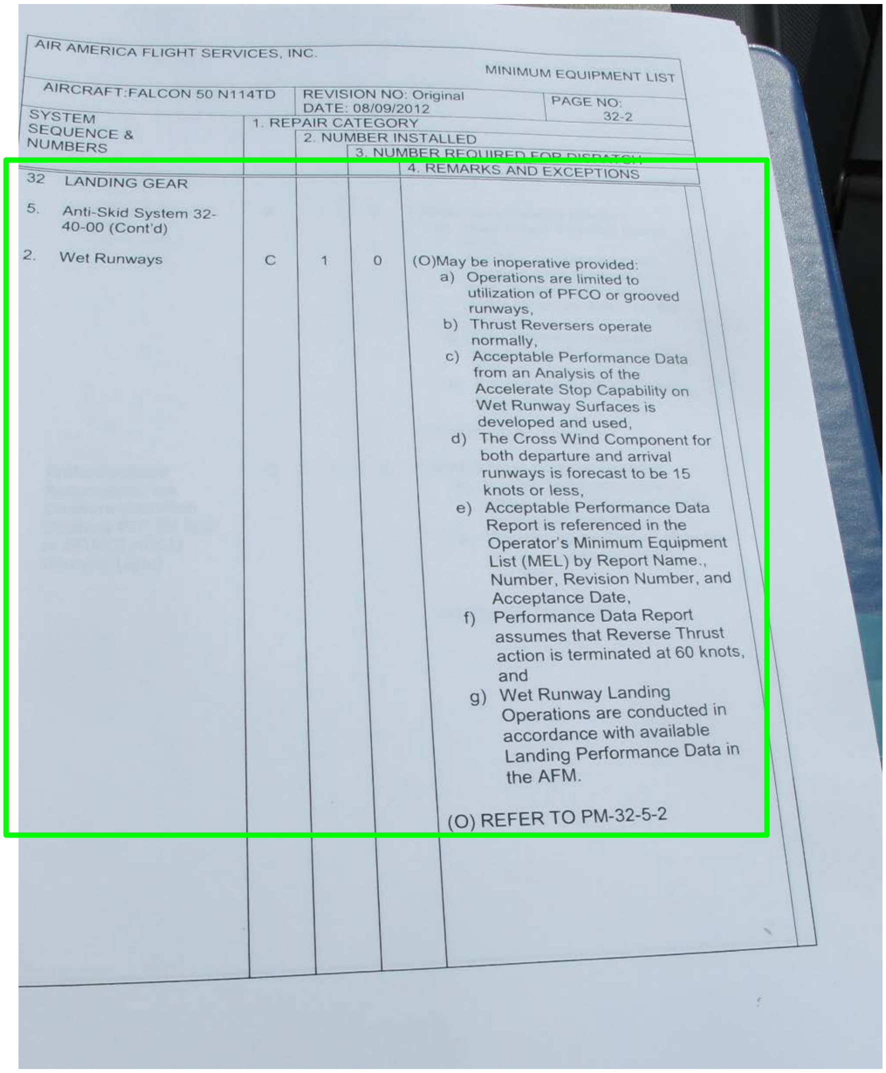


AIR AMERICA FLIGHT SERVICES, INC.

MINIMUM EQUIPMENT LIST

AIRCRAFT:FALCON 50 N114TD	REVISION NO: Original DATE: 08/09/2012			PAGE NO: 32-1
SYSTEM SEQUENCE & NUMBERS	1. REPAIR CATEGORY	2. NUMBER INSTALLED	3. NUMBER REQUIRED FOR DISPATCH	4. REMARKS AND EXCEPTIONS
32 LANDING GEAR				
1. Landing Gear Selector Flashing Light 32-60-00	C	1	0	May be inoperative provided the landing gear position indicators and warning horn operate normally.
2. Aural Warning 32-60-00				DELETED REVISION No 2.
3. Emergency/ Park Brake 32-45-00				DELETED REVISION No 3.
4. #2P.BK light (steady illumination)	C	1	0	(O)May be inoperative provided: a) T/O CONFIG warning system operates normally, and b) Parking brake pressure input to T/O CONFIG warning system is verified prior to each flight. (O) REFER TO PM-32-4
5. Anti-Skid System 32-40-00				
1) Dry Runways	C	1	0	(O)May be inoperative provided: a) Applicable AFM take-off and landing limitations and performance decrements are applied, and b) T/O CONFIG warning system is verified to operate normally before each departure. (O) REFER TO PM-32-5-1
				(Continued)

Figure 42 - N114TD MMEL ATA 32-5 page 1 for dry runways

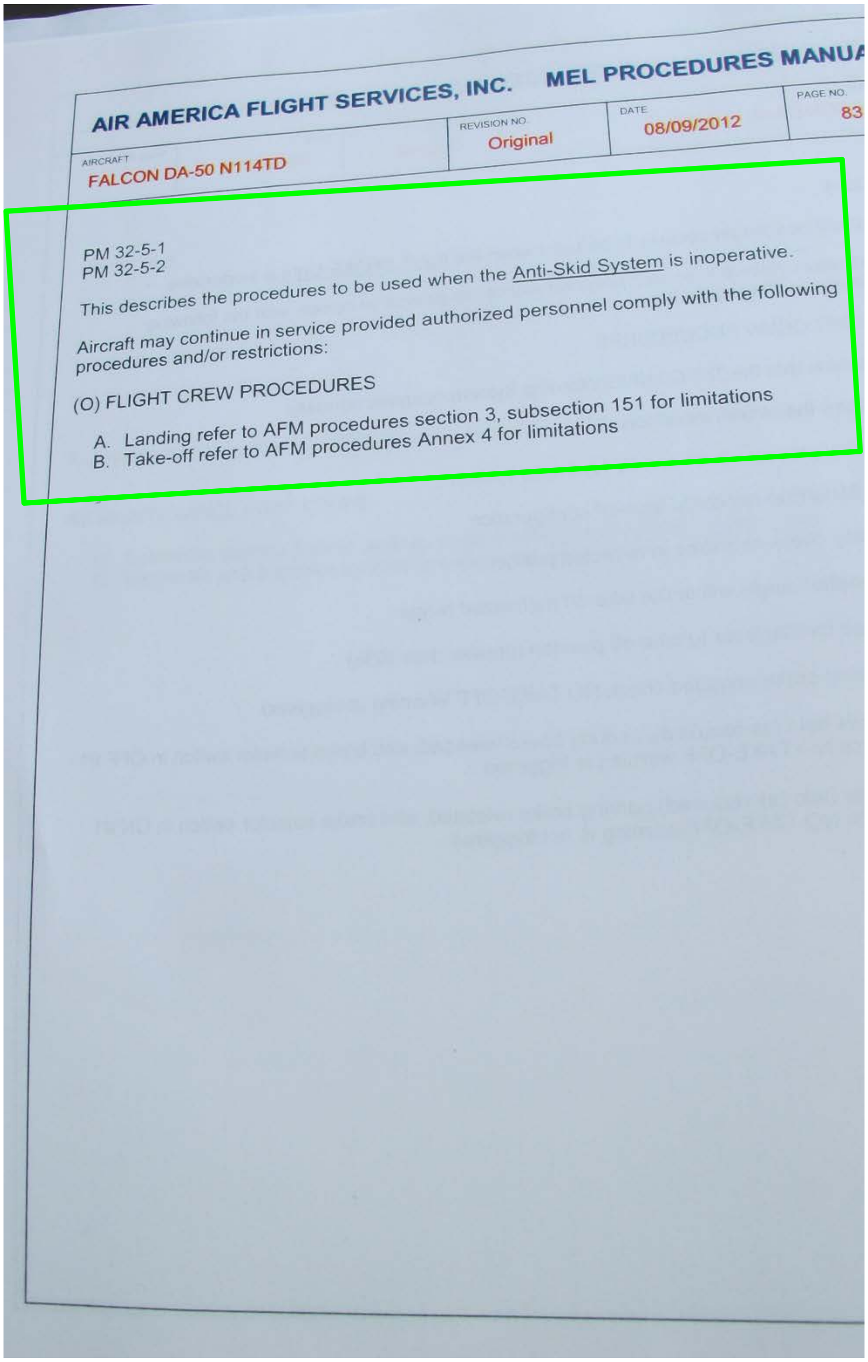


AIR AMERICA FLIGHT SERVICES, INC.

MINIMUM EQUIPMENT LIST

AIRCRAFT:FALCON 50 N114TD	REVISION NO: Original DATE: 08/09/2012			PAGE NO: 32-2
SYSTEM SEQUENCE & NUMBERS	1. REPAIR CATEGORY	2. NUMBER INSTALLED	3. NUMBER REQUIRED FOR DISPATCH	4. REMARKS AND EXCEPTIONS
32 LANDING GEAR				
5. Anti-Skid System 32-40-00 (Cont'd)				
2. Wet Runways	C	1	0	(O)May be inoperative provided: a) Operations are limited to utilization of PFCO or grooved runways, b) Thrust Reversers operate normally, c) Acceptable Performance Data from an Analysis of the Accelerate Stop Capability on Wet Runway Surfaces is developed and used, d) The Cross Wind Component for both departure and arrival runways is forecast to be 15 knots or less, e) Acceptable Performance Data Report is referenced in the Operator's Minimum Equipment List (MEL) by Report Name., Number, Revision Number, and Acceptance Date, f) Performance Data Report assumes that Reverse Thrust action is terminated at 60 knots, and g) Wet Runway Landing Operations are conducted in accordance with available Landing Performance Data in the AFM. (O) REFER TO PM-32-5-2

Figure 43 - N114TD MMEL ATA 32-5 page 2 for wet runways



AIR AMERICA FLIGHT SERVICES, INC. MEL PROCEDURES MANUA

AIRCRAFT	REVISION NO.	DATE	PAGE NO.
FALCON DA-50 N114TD	Original	08/09/2012	83

PM 32-5-1
PM 32-5-2

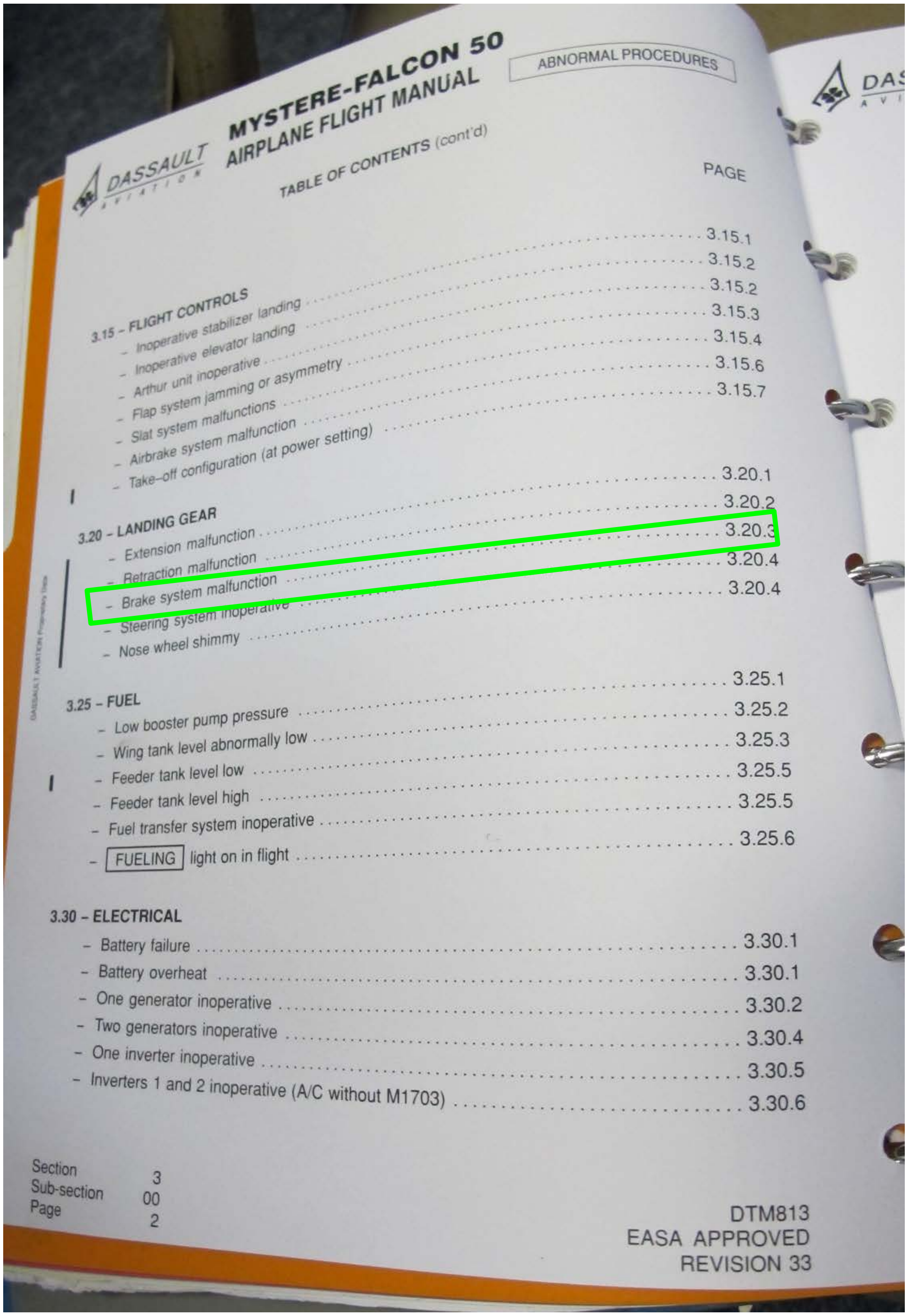
This describes the procedures to be used when the Anti-Skid System is inoperative.

Aircraft may continue in service provided authorized personnel comply with the following procedures and/or restrictions:

(O) FLIGHT CREW PROCEDURES

A. Landing refer to AFM procedures section 3, subsection 151 for limitations
B. Take-off refer to AFM procedures Annex 4 for limitations

Figure 44 - N114TD PM PM-32-5-1



DASSAULT AVIATION MYSTERE-FALCON 50 AIRPLANE FLIGHT MANUAL

ABNORMAL PROCEDURES

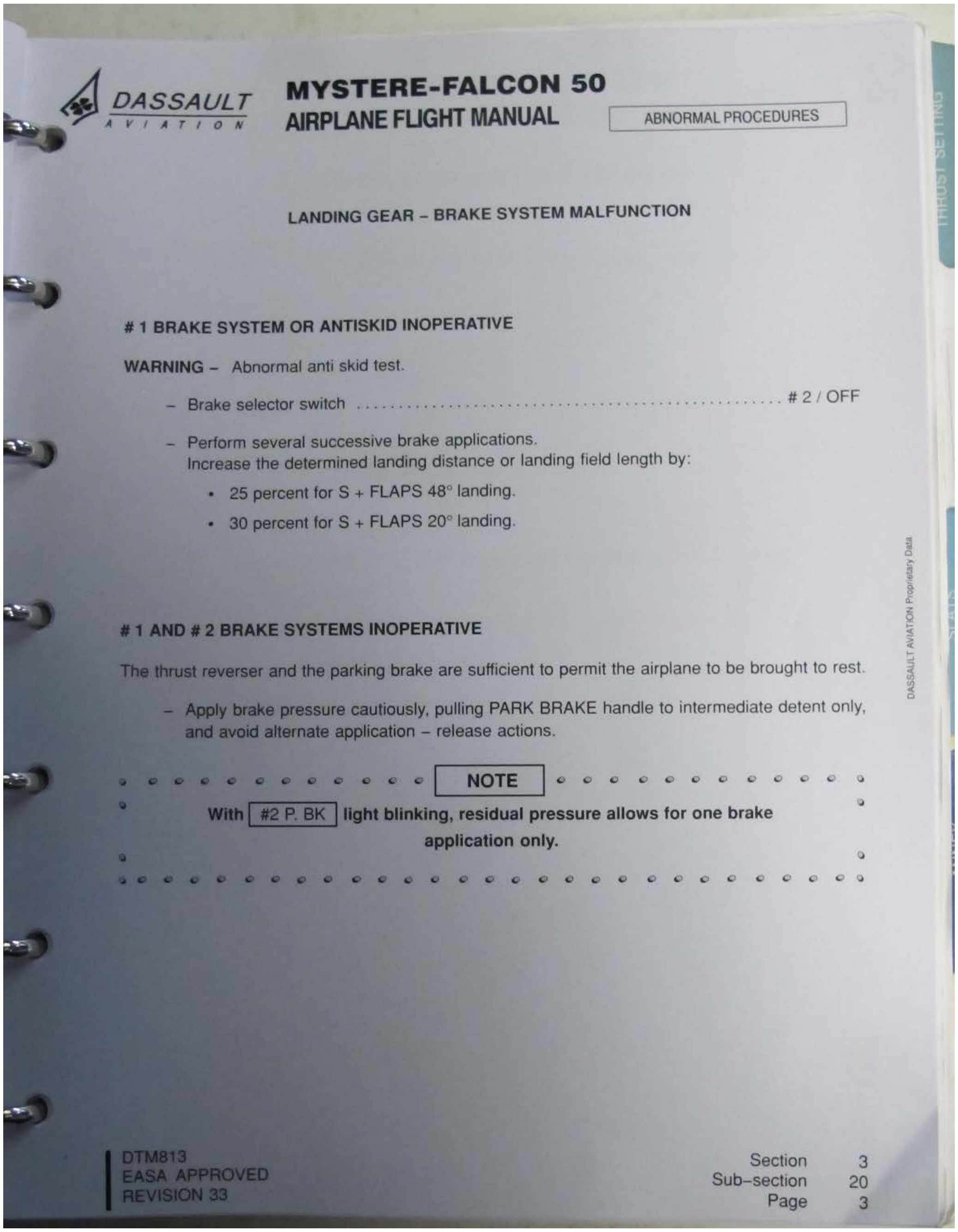
TABLE OF CONTENTS (cont'd)

PAGE

Section 3
Sub-section 00
Page 2

DTM813
EASA APPROVED
REVISION 33

Figure 45 - N114TD AFM abnormal flight procedures table of contents, brake system highlighted



DASSAULT AVIATION

MYSTERE-FALCON 50
AIRPLANE FLIGHT MANUAL

ABNORMAL PROCEDURES

LANDING GEAR – BRAKE SYSTEM MALFUNCTION

 # 1 BRAKE SYSTEM OR ANTISKID INOPERATIVE

WARNING – Abnormal anti skid test.

– Brake selector switch .... # 2 / OFF

– Perform several successive brake applications.
Increase the determined landing distance or landing field length by:

- 25 percent for S + FLAPS 48° landing.
- 30 percent for S + FLAPS 20° landing.

 # 1 AND # 2 BRAKE SYSTEMS INOPERATIVE

The thrust reverser and the parking brake are sufficient to permit the airplane to be brought to rest.

– Apply brake pressure cautiously, pulling PARK BRAKE handle to intermediate detent only, and avoid alternate application – release actions.

NOTE

With #2 P. BK **light blinking, residual pressure allows for one brake application only.**

DASSAULT AVIATION Proprietary Data.

DTM813
EASA APPROVED
REVISION 33

Section 3
Sub–section 20
Page 3

Figure 46 - N114TD AFM Abnormal procedures for brake system malfunction

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