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

Systems Group Chairman's Factual Report

April 3, 2006

A. ACCIDENT DCA06MA009

Location:	Midway Airport, Chicago, Illinois
Date:	December 8, 2005
Time:	1914 Local Time (CST)
Aircraft:	Southwest Airlines Flight 1248, a Boeing 737-7H4, N471WN

B. GROUP

Chairman: Tom Jacky
National Transportation Safety Board
Washington, DC

Group Members, initial on-scene investigation at Chicago Midway

Member: David Borgens
The Boeing Company
Seattle, WA

Member: Bob Shelton
Federal Aviation Administration (FAA)
Schiller Park, IL

Member: Randy Reeves
Southwest Airlines
Dallas, TX

Member: Brian Winchell
Southwest Airlines
Dallas, TX

Member: Captain James Duffy
Southwest Airlines Pilots Association
Phoenix, AZ

Member: Bill Gray
Southwest Airlines Mechanics Union
Dallas, TX

Group Members, Second Activities at Chicago Midway

Member:	David Borgens The Boeing Company Seattle, WA
Member:	Bob Shelton Federal Aviation Administration (FAA) Schiller Park, IL
Member:	Brian Winchell Southwest Airlines Dallas, TX
Member:	Captain James Duffy Southwest Airlines Pilots Association Phoenix, AZ
Member:	Bill Gray Southwest Airlines Mechanics Union Dallas, TX

Group Members, Component Examination at Dallas, Texas

Member:	Randy Reeves Southwest Airlines Dallas, TX
Member:	Brian Winchell Southwest Airlines Dallas, TX
Member:	Bill Gray Southwest Airlines Mechanics Union Dallas, TX

Group Members, Third Activities at Chicago Midway

Member:	David Borgens The Boeing Company Seattle, WA
Member:	Bob Shelton Federal Aviation Administration (FAA) Schiller Park, IL

Member: Brian Winchell
Southwest Airlines
Dallas, TX

Member: Bill Gray
Southwest Airlines Mechanics Union
Dallas, TX

Group Members, Component Examination at Portland, Oregon

Member: Randy Reeves
Southwest Airlines
Dallas, TX

Member: Captain James Duffy
Southwest Airlines Pilots Association
Phoenix, AZ

Member: Bill Gray
Southwest Airlines Mechanics Union
Dallas, TX

Group Member, Component Examinations at Burbank and Santa Clarita, California

Member: Brian Winchell
Southwest Airlines
Dallas, TX

In addition, three removed components were examined at the Rockwell Collins facility in Cedar Rapids, Iowa under the witness of an inspector from the Federal Aviation Administration flight standards district office (FSDO) in Des Moines, Iowa. Finally, four components were examined in Redmond, Washington, under witness to another aerospace engineer within the Office of Aviation Safety.

C. SUMMARY

On December 8, 2005, 1914 central standard time, Southwest Airlines flight 1248, a Boeing B-737-7H4, registered as N471WN, overran runway 31C at Chicago Midway Airport in Chicago, Illinois, during the landing rollout. The airplane departed the end of the runway, rolled through a blast fence, a perimeter fence, and onto a roadway. The airplane came to a stop after impacting one automobile. There were 98 passengers and 5 crewmembers on board. There was one ground fatality. Instrument meteorological conditions prevailed at the time. The airplane was substantially damaged. The flight was conducted under 14 CFR Part 121 and had departed from the Baltimore/Washington International Thurgood Marshall Airport, Maryland.

The group met at the accident site from December 9, to December 13, 2005, to document the relevant airplane systems. Several system components were removed from the airplane and retained by the National Transportation Safety Board for further examination. The following components were retained:

1. Engine Accessory Unit (EAU)
2. Flap/Slat Electronics Unit (FSEU)
3. Auto Speed Brake Control Unit (ASBCU)
4. Integrated Flight Systems Accessory Unit (IFSAU)
5. Heads Up Guidance System (HGS) Computer
6. HGS Control Panel (HCP)
7. HGS Drive Electronics Unit (DEU)
8. Antiskid/Autobrake Control Unit (AACU)
9. Autobrake Valve Module
10. Passenger Address Unit (PAU-700)
11. Flight Control Computer Number 1
12. Flight Control Computer Number 2
13. Mark V Enhanced Ground Proximity System (EGPWS) Computer
14. Radio Altimeter Number 1
15. Radio Altimeter Number 2
16. Digital Flight Data Acquisition Unit (DFDAU)
17. PCMCIA Card Removed From DFDAU
18. Engine Vibration Monitor (EVM 280)

The group met at the American Trans Air (ATA) maintenance hangar at Midway Airport from January 18 to January 19, 2006, and again on February 15, 2006 to examine the airplane's thrust lever angle resolvers. The group developed a test procedure to measure the thrust reverser interlock latch position and used a tool to more precisely measure the interlock latch position as a function of throttle resolver angle. The procedure was performed on the accident airplane and was also performed on two additional Southwest Airlines airplanes at the Southwest maintenance hangar at Midway Airport.

The group met at the BAE Systems Dallas Service Center in Irving, Texas from January 31 to February 1, 2006 to inspect the Engine Accessory Unit (EAU), Flap/Slat Electronic Unit (FSEU), Auto Speed Brake Control Unit (ASBCU), and Integrated Flight Systems Accessory Unit (IFSAU) components removed from the accident airplane. BAE Systems (formerly Boeing Commercial Electronics) manufactured each of these components. Each component was subjected to a physical examination and to the BAE Systems acceptance test procedure. In addition, the two units with non-volatile memory (the EAU and FSEU) had the NVM downloaded and converted into engineering units.

The group met at the Rockwell Collins HGS facility in Portland, Oregon from February 22 to February 23, 2006 to inspect the three Head-Up Guidance System (HGS) line replaceable units (LRUs) removed from the accident airplane. Each component was subjected to a physical examination and then to the unit's acceptance test procedure. In addition, the non-volatile random access memory (NVRAM) in the HGS computer was downloaded, converted into engineering units, and examined. Finally, the three components were installed together on a Rockwell Collins engineering test unit (ETU) and the last, manually entered information read from the HCP.

The group met at the Crane Aerospace Hydro-Aire facility in Burbank, California on March 7, 2006 to inspect the Antiskid/Autobrake Control Unit (AACU), part number 42-935-2, serial number 1762, which was removed from the accident airplane.

The AACU was examined for physical damage and the Crane acceptance test procedure functional test was performed. The AACU passed the acceptance test procedure with no faults found. The non-volatile memory (NVM) in the AACU, including fault analysis and built-in test equipment (BITE) memory, was examined through the Crane computerized test stand. However, the fault memory associated with last flight leg did not correlate to any fault codes and was considered erroneous. No AACU faults were available for the last flight leg.

The group met at the HR Textron facility in Santa Clarita, California on March 8, 2006 to inspect the Autobrake Valve Module removed from the accident airplane.

The Autobrake Valve Module was examined for physical damage, with only minor, superficial damage noted. The HR Textron acceptance test procedure was performed on the unit. The unit failed the Steady State Pressure Gain element of the acceptance test procedure. Since the module failed an element of the test, the unit was considered to have failed the entire acceptance test procedure.

The Passenger Address Unit (PAU) and the airplane's two Flight Control Computers (FCC) were examined at the Rockwell Collins facility in Decorah, Iowa and Cedar Rapids, Iowa, respectively on February 15 and 16, 2006. The examinations were conducted under witness of an FAA principal avionics inspector (PAI) from the Des Moines Flight Standards District Office.

None of the evidence gathered on-scene or examinations of the removed components revealed a failure of an airplane system.

In addition, the group found no evidence of a jammed or binding thrust reverser levers that would have prevented the actuation of the thrust reversers.

D. <u>DETAILS OF INVESTIGATION</u>

The group identified and documented the following relevant systems of the accident airplane:

1. <u>Communications</u>

The Passenger Address Unit (PAU-700), Rockwell Collins part number 622-5342-101, serial number 1FW1P, was identified on the E1-3 shelf in the forward electronic equipment (EE) bay. The E1-3 shelf was damaged and displaced upwards in the area of the PAU. No attempt was made to determine the continuity between the PAU and the E1-3 connection.

The unit's "as-is" condition was confirmed by subjecting the PAU to the Production Test Requirements Rockwell Collins, Inc. document number 671-1988-001, Production Test Requirements for PAU-700 Passenger Address Amplifier (CPN 622-5342-001/-002/-101). The PAU passed the production test requirement with no faults found. Although the physical examination revealed some minor cosmetic damage to the unit, the damage did not affect the unit's operation during the production test.

The passenger address unit controls the communication and announcements from the cabin crew to the passengers and from the flight deck to the cabin.

Both hand-held megaphones were operational.

2. Equipment & Furnishings

The displaced nose gear assembly impacted the forward Electronic Equipment (EE) bay. The nose gear pushed the bay hatch door up and into the forward EE bay; the hatch door was found on top of the J23 Box, with the door side notation "AFT" facing the nose and the external side of the door facing up. Several racks and many components were splattered with mud and snow.

The E1 equipment racks were dislodged and pushed up by the displaced nose gear (see Figure 1) entering the forward EE bay. Each of the E1 racks were damaged and/or bent. Components located on the racks were wedged underneath the above shelf, dislodged from the shelf, and/or broken open. No attempt was made to determine whether the components on these shelves were still connected to their respective shelf connector.

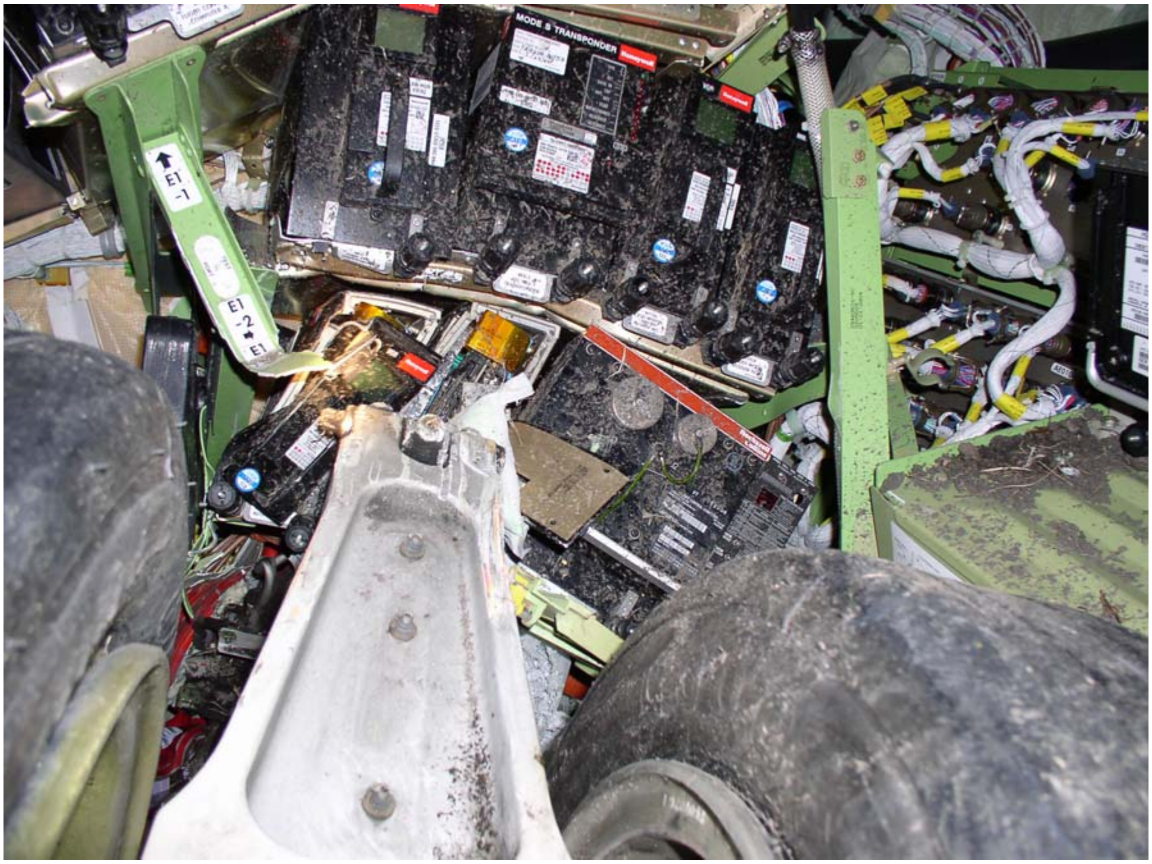

Figure 1 - Damage to Forward EE Bay. Note nosegear assembly.

Components were removed from the E1-1, E1-3, E1-4, E3-1, E3-2, E4-1, and E4-2 racks for further investigation.

3. Fire Protection

The fire bottle quantities were noted from the gauges in the airplane’s wheel wells. The forward fire bottle indicated zero, while the aft fire bottle was noted as 0.6.

The left engine, right engine, and auxiliary power unit (APU) fire handles, located on the center pedestal control stand in the flight deck, were found pulled and turned counter clockwise.

4. Flight Controls

A. Primary Flight Control Systems

The primary flight control systems were examined. No damage was noted to any of the primary flight control system power control units (PCU), associated components, or cables.

B. Secondary Flight Control Systems

1) Wing Leading Edge Slats

The leading edge slats on each wing were examined and documented. Each wing has four leading edge slats, with the actuator located near the midpoint of each slat. For the left wing, slat number 1 appeared undamaged, slat number 2 was broken and damaged, slat number 3 exhibited impact damage, and slat number 4 appeared undamaged. For the right wing, slat numbers 5 and 6 exhibited severe impact damage, while slat number 7 exhibited slight damage on its inboard side, and slat number 8 appeared undamaged. A visual examination of the two-stage slat actuators determined that the inner rod of the slat number 2 actuator was bent while all other slat actuators appeared undamaged.

The slat actuators were measured for actuation. Each measurement was taken from the end of the actuator body to the centerline of the rod end bolt. The measurements were as follows:

Slat Number 1:	15½ inches (in.)
Slat Number 2:	15½ in.
Slat Number 3:	15½ in.
Slat Number 4:	15½ in.
Slat Number 5:	15½ in.
Slat Number 6:	15½ in.
Slat Number 7:	15½ in.
Slat Number 8:	15½ in.

All of slat measurements correspond to the fully extended (gapped) position.

2) Wing Leading Edge Flaps

Each wing has one leading edge flap (also known as a Krueger flap) on the inboard leading edge section of each wing. Each leading edge flap has two actuators.

The damage to the leading edge flaps was documented. The left leading edge flap was found damaged and had a portion of the instrument landing system (ILS) antenna wedged underneath the flap. The right leading edge flap was severely bent and damaged, especially the inboard portion of the surface.

The leading edge flap actuators were measured from the edge of the actuator body to the centerline of the rod end bolt. The actuator measurements were documented as follows:

Left Leading Edge Flap, Outboard Actuator Measurement:	13 in.
Left Leading Edge Flap, Inboard Actuator Measurement:	13 in.
Right Leading Edge Flap, Inboard Actuator Measurement:	13 in.

Right Leading Edge Flap, Outboard Actuator Measurement: 13 in.

According to Boeing the measurement for each leading edge flap was consistent with the fully extended position.

3) Trailing Edge Flaps

The trailing edge flap lever in the flight deck was found in the 40º detent.

The trailing edge flap jackscrews were measured to verify the extended position. The measured values were:

Left Wing Outboard Flap

Outboard actuator: 9½ in.
Inboard actuator: 11 in.

Left Wing Inboard Flap

Outboard actuator: 12 in.
Inboard actuator: 12 in.

Right Wing Inboard Flap

Inboard actuator: 12 in.
Outboard actuator: 12 in.

Right Wing Outboard Flap

Inboard actuator: 11 in.
Outboard actuator: 9½ in.

According to Boeing the measurement for each actuator was consistent with a 40º flap setting.

4) Flight Spoilers

There are 12 spoiler panels on the airplane. Spoilers 1 through 6 are on the left wing and spoilers 7 through 12 are on the right wing. Spoilers 2 through 5 and spoilers 8 through 11 are flight spoilers. All flight spoilers have one actuator.

All flight spoilers were noted in down or retracted position. The inboard trailing edge of the number 5 spoiler was noted as bent slightly downward; otherwise, no damage was noted to any of the flight spoilers.

The actuator of each spoiler was measured. The measurement was taken from the actuator body to the centerline of the rod end bolt. The flight spoiler actuator measurements were documented as follows:

Spoiler Number 2:	2⅛ in.
Spoiler Number 3:	2⅛ in.
Spoiler Number 4:	2⅛ in.
Spoiler Number 5:	2⅛ in.
Spoiler Number 8:	2⅛ in.
Spoiler Number 9:	2¼ in.
Spoiler Number 10:	2⅛ in.
Spoiler Number 11:	2⅛ in.

For each flight spoiler, the noted positions appear faired and within ¼ inch of the fixed trailing edge, except for the bent portion of the trailing edge of spoiler number 5.

5) Ground Spoilers

Spoilers 1, 6, 7, and 12 (of twelve total spoiler panels on the airplane) are ground spoilers. Spoilers 1 and 6 are located on the left wing and spoilers 7 and 12 are located on the right wing. Spoilers 6 and 7 have two actuators, while Spoilers 1 and 12 have one actuator.

All ground spoilers were found in the down or retracted position (Note: the speed brake lever in the flight deck was found in the down detent position). No damage was noted to any of the ground spoilers.

Each spoiler's actuation was measured. The measurement was taken from the actuator body to the centerline of the rod end bolt. The spoiler actuator measurements were:

Spoiler Number 1 Actuator:	1¼ in.
Spoiler Number 6, Outboard Actuator:	$3^{7}/_{16}$ in.
Spoiler Number 6, Inboard Actuator:	$3^{7}/_{16}$ in.
Spoiler Number 7, Inboard Actuator:	$3^{7}/_{16}$ in.
Spoiler Number 7, Outboard Actuator:	3⅜ in.
Spoiler Number 12 Actuator:	1¼ in.

Each ground spoiler appeared well faired and within ¼-inch of the fixed trailing edge.

The ground spoiler interlock valve was checked for rigging. While the airplane was lifted off the ground on jacks, the rig pin was inserted into the interlock valve. The pin fit was noted as a bit tight. However, no damage was noted to the interlock valve or the connecting cable to the right main landing gear.

Several electronic components from the secondary flight control systems were removed from their respective shelves in the forward EE bay for further examination. The flap/slat electronic unit (FSEU) and Integrated Flight Systems Accessory Unit (IFSAU) were removed from the E1-1 shelf, and the auto speed brake accessory unit was removed from the E4-2 shelf.

C. Examination of the Flap/Slat Electronic Unit (FSEU)

The Flap/Slats Electronic Unit (FSEU), identified as part number 285A1200-1, serial number D01687, was examined at the BAE Systems Dallas Service Center. As part of the examination, the FSEU was placed onto an engineering test bench and power applied to the unit. The unit's NVM was queried and extracted. The non-volatile memory information was converted into fault history information.

The recorded faults related to the last and prior flight leg in memory were identified and examined. These flight leg faults are listed in Attachment 1.

The FSEU was placed onto a BAE systems ATS-182a test station. The BAE Systems acceptance test procedure was performed on the FSEU. The FSEU passed the procedure with no faults found.

D. Examination of the Auto Speed Brake Control Unit (ABSCU)

The Auto Speed Brake Control Unit (ASBCU), part number 65-84209-21, serial number D02319, was examined at the BAE Systems Dallas Service Center. As part of the examination, the group noted that the ASBCU had tamper seals indicating that a company other than BAE Systems had previously serviced the unit. The ASBCU had not been previously serviced at the BAE Systems Dallas Service Center.

The ASBCU was placed onto a BAE Systems test bench and the unit was tested in accordance with the Component Maintenance Manual (CMM). The unit passed the bench test with no faults found.

The ASBCU was then placed onto a BAE systems ATS-182a test station and the BAE Systems factory test procedure was performed on the unit. The ASBCU passed the procedure with no faults found.

E. Examination of the Integrated Flight Systems Accessory Unit (IFSAU)

The Integrated Flight Systems Accessory Unit (IFSAU), part number 65-52820-2, serial number D02782, was placed onto a BAE Systems ATS-182a test station and the BAE Systems acceptance test procedure was performed on the unit. The BAE Systems acceptance test procedure was performed on the IFSAU. The unit passed the procedure, with one minor fault identified – one current was identified as 0.1 milli-amp below the test specification range of 30-34 milli-amps. However, BAE Systems indicated that another exemplar unit exhibited the same minor fault, and that the low reading was the result of an IFSAU test setup anomaly for that measurement.

5. Fuel

The left engine fuel fire shutoff valve (spar valve) was found in the closed position. The valve and housing appeared undamaged.

The right engine fuel fire shutoff valve (spar valve) was noted in the closed position. The valve housing, located in the right wing root forward spar, was damaged from impact. An interview with a Southwest mechanic who responded to the accident site stated that he had manually closed the right spar valve after noticing fuel spilling from the right wing onto the ground.

6. Hydraulic Power

The hydraulic brake accumulator in the wheel well indicated 1,000 pounds per square inch (psi).

The hydraulic system indicators in the wheel well indicated the following:

A System - ½ reservoir quantity, zero pressure.

B System - Zero pressure and zero reservoir quantity.

The hydraulic fill selector valve was noted in the neutral position.

7. Indicating/Recording Systems

A. Flight Deck Indications

The flight deck was examined for switch positions and indication. The following information was noted:

1) P1 Instrument Panel

Windshield Air - On
Alt Nose Wheel Steering - Normal
All other switches normal

2) P2 Center Instrument Panel

Auto Brake Rotary Switch - MAX
Left Flap Indicator - 40º
Right Flap Indicator - 15º
Landing Gear Handle - Down

All other positions normal

3) P3 First Officer Instrument Panel

Windshield Air - On
All other switches normal

4) Control Stand

Throttles at IDLE stop
Reverse Thrust Levers Stowed
Flap Handle in 40º detent
Speed brake handle down in forward detent
Parking brake not set
Stabilizer Trim set to approximately 8 units
Fuel Handles set to cut off
Engine 1 and 2 Overheat Detect Normal
All 3 fire handles pulled up and selected left
Stabilizer Trim Cutout Switches - Normal
Radar WX position, Tilt 7º up
TCAS Switch TA/RA

Note: The Southwest mechanics that replaced the tires on the main landing gears indicated that the parking brake had been released to replace the tires.

5) P5 Forward Overhead and Aft Overhead Panels

Left and Right Main #1 and #2 Boost Pumps – On
Left and Right Center Boost Pumps – Off
Crossfeed Valve - Closed
Navigation Panel Normal or Auto
Alternate Flap Switch – Off, guard down, all other flight control and spoiler switches all on, with guards down
Yaw Damper - Off
Electrical Panel
Battery Switch – Off
Galley Power - On
STBY Power Switch – Battery
Both Engine Start Switches – Continuous
Ignition Switch – Left Ignition
Window and Probe Heat - On
Wing and Engine Anti-Ice Switches - OFF
Hydraulic Panel – All 4 Pumps On
Bleed Air/Air Conditioning Panel
L & R Pack – Auto
Isolation Valve Switch - Auto

APU Bleed - Off
Pressurization Control Valve – Auto
Number 1 and Number 2 Bleed Valves – On
Gasper Fan – Off
Recirc Fan – Auto

Pressurization Panel

Pressurization Mode Select Switch – Manual
Cabin Outflow Valve Indicator – Full Open

Lighting Panel - Logo, Strobe, Position, Anticollision, and Wing Lights ON
Left and Right IRS Switches - NAV
Both Engine EEC Push Buttons – On

Heads Up Guidance System - Stowed

6) P7 Glareshield Panel

Captain's EFIS Control Panel set to Map Mode, 5 NM scale
First Officer's EFIS Control Panel set to Map Mode, 5 NM scale

The flight deck indications and settings were indicative of the emergency checklist completion.

The airplane's digital flight data acquisition unit (DFDAU) was removed from the E3-2 shelf of the forward EE bay for further examination.

B. Examination of the Digital Flight Data Acquisition Unit

The DFDAU, part number 967-0212-002, serial number 1887, and mod status 5, was visually inspected, with no damage noted. The DFDAU was connected to a DFDAU engineering test bench and the aircraft condition monitoring system (ACMS) software part number 998-2372-507 (version SW7028) was verified as installed.

All of the stored ACMS reports (stored in non-volatile memory) were downloaded onto a PCMCIA card installed in the DFDAU. There were a total of 114 reports stored on the unit. Of the 114 reports, eight reports were from the accident flight. None of the eight reports resulted from an aircraft or DFDAU fault or exceedence; the eight reports represented the automatically generated ACMS flight reports.

The DFDAU was tested per Honeywell's Acceptance Test Plan (ATP) 967-0212-701. During the ATP, shunt discrete number 100 failed. The shunt relates to the FDR discrete parameter Main/Alt Brake Select (FDR word 169, Bit 2). Further testing of the unit revealed that the shunt signal was stuck at the low state "0" (i.e. a voltage of less than 3.0 Volts), as troubleshooting found that the voltage of the discrete input was stuck at 2.0 volts with a varying load. The low state of the discrete corresponds to the "ALTN" (alternate) state of the Main/Alt Brake Select parameter. The troubleshooting converged onto three components that could have failed: diode D1A, capacitor C5A or the FMC input selector U4. No further effort was taken to determine the source of the stuck low state.

8. Landing Gear

A. Left Main Landing Gear

The left main landing gear was documented as:

Boeing Company Model 737NG
Boeing Part Number: 161A1100-31
Serial Number: MAL03113Y1535
Manufacturer: Goodrich

The gear components were visually inspected with no appreciable impact damage noted. No visible fractures were noted. The lower assembly and gear truck exhibited some minor scrapes, and the guide flange for gear wiring was broken away. The aluminum tubing leading from the J00028 junction box was bent. No obstructions were found in either of the gaps for the air/ground proximity sensors.

B. Right Main Landing Gear

The right main landing gear was documented as:

Boeing Company Model 737NG
Boeing Part Number: 161A1100-32
Serial Number: MAL03114Y1535
Manufacturer: Goodrich

The gear components were visually inspected and only minor impact damage to the gear strut and attachment structure were noted. There were several scrapes and paint transfers noted on the main structure. The guide vane/flange for wiring on the front of the gear trunion was bent but not fractured. The J00032 junction box cover was dented inwards and the aluminum tubing leading into the J00033 junction box was bent and crumpled. The hydraulic brake pressure and wheel hydraulic line for the inboard wheel was found bent and kinked, but no leaks or fractures noted. No obstructions were found in either of the gaps for the air/ground proximity sensors.

C. Nose Landing Gear

The nose gear assembly was documented as:

Left Nose Wheel Assembly Number: 2607825-3
Part Number: 26 12798

The nose landing gear structure was found imbedded in the E&E bay, pushed back and up into the compartment. The gear was broken from its support structure at the pin of the lower drag link, with the summing mechanism and lower structure found in the E&E bay. The gear assembly was rotated 180º from its normal orientation. The gear wheels were found in the hatch opening, with the bottom of the lower wheel slightly below the floor level of the E&E bay. The gear wheel that was found lower in the compartment (left wheel) had a tire pressure of 185 psi, measured following the accident. The right nose gear tire was imbedded further up into the compartment and was pinched against the surrounding structure.

D. Main Landing Gear Tires

Before the airplane was moved to a maintenance hangar, the main landing gear tires were removed and replaced by Southwest maintenance. The four tires were examined after they were removed from the airplane. The examination included a measurement of the tire tread wear or depth. Each tire's tread depth was measured in three points using a mechanic's tread depth gauge. However, since the tread depth was measured with all tires depressured, the measurements do not reflect the measurements that Southwest takes during routine maintenance[1]. The tread depth measurements were converted into decimal numbers. The tread depth measurements are included in Table 1.

Tire	Tread Depth Measurement (In Inches)		
	1	2	3
Left Outboard	4/32	4/32	5/32
Left Inboard	12/32	13/32	13/32
Right Inboard	9/32	10/32	9/32
Right Outboard	9/32	9/32	9/32

Table 1 – Tire Tread Depth Measurements

Duct tape was placed on the left gear outboard tire to indicate the 12 o'clock position as found on the airplane. The tire was broken and was not pressurized. There was a slash in the tire on the outer rim wall at the 10 o'clock position and a slash in the main tread at the 2 o'clock position. The main tread wear did not exhibit any evidence of flat spots or chevrons.

Duct tape was placed on the left gear inboard tire to indicate the 12 o'clock position as found on the airplane. The tire had a cut in the main tread at the 5 o'clock position, and the inner wheel hub wall was punctured through at the same position. The main tread wear did not exhibit any evidence of flat spots or chevrons.

[1] Southwest Airlines Maintenance Procedures Manual, Section 08-13 requires that any main tire be changed if the center tread is less than 1/8".

Duct tape was placed on the right gear inboard tire to indicate the 12 o'clock position as found on the airplane. The tire was still pressurized and not deflated. Cuts were found in the inner sidewall at the 1:30 o'clock position, in the outer sidewall at the 4 o'clock position, and on the main tread at the 11 o'clock position. The main tread wear did not exhibit any evidence of flat spots or chevrons.

Duct tape was placed on the right gear outboard tire to indicate the 12 o'clock position as found on the airplane. The tire was still pressurized and not deflated. No cuts were noted on the tire. The main tread wear did not exhibit any evidence of flat spots or chevrons.

E. Main Landing Gear Brakes

The airplane was jacked off the ground and each of the four wheel brakes were examined. In each case, the brake pack, brake assembly, and AMS 6302 steel brake pads showed no indications of overheating, binding, or leaking. In addition, each brake showed little wear. All rotating brake components, including the gear axle antiskid sensors, were undamaged and rotated freely. Finally, the brake wear indicator pins, two for each wheel, were measured.

Southwest maintenance procedures (for a Boeing 737-700 Maintenance Visit 2 task, MT# 712-00—02) require the brakes to be replaced when the brake wear indicator pin length is less than 0.090 inches. The indicator pin length is measured when hydraulic pressure is available and the brake pack is compressed. Damage to the airplane prevented hydraulic pressure from being applied to the brake assemblies; however, the group noted that, without hydraulic pressure, an additional shiny length of indicator pin was visible. The overall length and shiny length of the pin was measured to determine a calculated wear indicator pin measurement. The measurements and calculated wear indicator pin length are presented in Table 2.

	Forward Brake Pin Measurements (Inches)			Aft Brake Pin Measurements (Inches)		
Tire Position	Polish	Overall	Calculated Pin Length	Polish	Overall	Calculated Pin Length
Left Outboard Brake (No. 1)	3/32	31/32	28/32	3/32	31/32	28/32
Left Inboard Brake (No. 2)	3/32	½	13/32	⅛	17/32	13/32
Right Inboard Brake (No. 3)	3/32	1 9/32	1 3/16	3/32	1 5/32	1 1/16
Right Outboard Brake (No. 4)	⅛	25/32	21/32	⅛	25/32	21/32

Table 2 – Brake Pin Measurements

The antiskid/autobrake control unit was removed from the E1-3 shelf, and the brake control module was removed from the wheel well for further examination.

F. Examination of the Antiskid/Autobrake Control Unit

The Antiskid/Autobrake Control Unit (AACU), part number 42-935-2, serial number 1762, was examined for physical damage. Although several small impact and scrape marks were noted, no appreciable damage was noted. None of the tamper seals on the AACU were compromised; according to their records, Crane Aerospace had not previously serviced the AACU.

1) Download of the AACU Non-Volatile Memory

The AACU was placed onto a Crane Aerospace Hydro-Aire computerized test set and power applied to the unit. The unit's non-volatile memory (NVM), or fault code data, was queried by use of the display on the front panel of the unit. However, the displayed information was non-sensical and did not relate to any of the fault codes for the AACU. When the pertinent memory locations were accessed via the test set, the values did not represent any fault code. The memory block locations related to the last flight leg all appeared to be written as "1's".

To determine whether the memory block was considered a bad memory block, the AACU BITE card was removed from the unit and placed into a test stand. The entire contents of the memory were then written over with a test pattern and then read back. The test was accomplished twice, each with a separate pattern, and in both cases, the area of the last flight leg memory location, the test pattern was written and read back without error.

2) Functional Testing of the AACU

The AACU functional test procedure TP42-935-2, Revision C (dated October 20, 1999) was performed on the AACU. The unit passed the test with no faults found.

G. Examination of the Autobrake Valve Module

The Autobrake Valve Module, part number 20102060-103, serial number 2158, was placed onto the HR Textron Autobrake Test Station and connected to a hydraulic fluid supply. The Autobrake Valve Module ATP HR72700405, Revision F (dated May 12, 1994) was performed on the unit.

The unit failed one sub-test of the ATP and was therefore considered to have failed the ATP. The unit passed all elements of the ATP except Test 5.12, Steady State Pressure Gain. The Steady State Pressure Gain test is a measurement of the Autobrake Valve Module's input current during a controlled, pressure ramp-up from zero to 3,000 psi. The module's measured input current was about 0.075 milliamps[2] lower than the minimum limit from approximately 2 milliamps to 4.0 milliamps. At approximately 4.0 milliamps, the module's input current reached the minimum limit and stayed within limits until reaching 3,000 psi. The pressure results were within limits from 2,000 psi to 3,000 psi. However, since the unit was out of tolerance at some point during the test, the unit was considered to have failed the entire acceptance test procedure.

9. Lights

The airplane's exterior lights were examined and documented for damage. The right inboard landing light and right runway turnoff light was found broken. The nose taxi light exhibited minor damage. In addition, the left and right retractable white landing lights located on the underside of the airplane belly were both noted as extended and exhibited no visible damage. No other exterior lights exhibited visible damage.

10. Navigation

The airplane's HGS computer was removed from the E4-1 shelf of the forward EE bay, the HGS drive electronics unit (DEU) was removed from the airplane's E1-3 shelf in the forward EE bay, the Mark V Enhanced Ground Proximity System (EGPWS) Computer was removed from the E1-1 shelf of the forward EE bay, the number 1 and number 2 Radio Altimeters were removed from the E3-1 and E3-2 shelves in the forward EE bay, and the channel A and channel B Flight Control Computers (FCC) were removed from the E1-1 and E1-4 shelves in the forward EE bay, respectively. All these units were removed for further examination.

A. Description of Head-up Guidance System

The head-up guidance system (HGS) is a combined electronic and optical system that displays aircraft position and guidance symbols on a combiner screen placed directly in the captain's field of view. The information is presented such that the navigational information is accurately overlaid onto the view of the outside world. To use the system, the flightcrew enters runway-specific information into the HGS Control Panel (e.g., length, elevation, and glideslope angle) and selects the mode of HGS operation, depending on the weather minimums at the destination runway.

One of the HCP-selectable modes is the AIII Approach Mode, which is intended for manual, instrument landing system (ILS) approaches to FAA Category IIIa minimums. In addition to the flightpath guidance cues, the HGS, once the airplane has descended below 500' radio altitude, monitors the airplanes performance along the ILS flightpath, and, if the airplane exceeds a pre-determined performance standard, the HGS annunciates the exceedance to the flightcrew by illuminating the first officer's "APPROACH WARNING" light on the HGS annunciator panel and indicating "APCH WARN" on the captain's

[2] The 0.075 milliamp deviation was equivalent to approximately 27 psi.

combiner, in the field of view. While in the AIII mode, the HGS Computer also records all exceedences in non-volatile memory.

The HGS system includes six LRU's – the HGS Computer, HGS Control Panel, Drive Electronics Unit (DEU), OverHead Unit, Combiner, and HGS Annunciator Panel. The HCP, OverHead unit, Combiner, and HGS annunciator panel are located in the flightdeck, while the HGS Computer and DEU are located in the forward EE bay. The combiner, and OHU are on the captain's side only, while the HGS Annunciator Panel is located in the first officer's panel. The combiner is lowered into the captain's field of view prior to use and then rotated back when not needed.

B. Examination of the HGS Computer

The HGS Computer, part number 1500-1730-004, serial number 8769, was placed onto a Rockwell Collins HGS computerized test station (HCTS) and power applied to the unit. The contents of the unit's non-volatile random access memory (NVRAM) were queried and downloaded. The NVRAM information was converted into ASCII characters and placed into an electronic file.

The HGS computer records performance monitoring exceedances and internal faults into NVRAM. The information is identified by flight leg (with no time stamp), with the most recent flight leg as number 1. The NVRAM included the results for the most recent 119 flight legs. The faults recorded during flight leg 1, (the most recent flight leg) were identified as follows:

Fault #1:	APCH WARN	AIRSPEED EXCEED
Fault #2:	APCH WARN	LAT POS EXCEED
Fault #3:	APCH WARN	GLIDESLP EXCEED
Fault #4:	APCH WARN	LOC EXCEED

The Model 2350 HGS computer acceptance test procedure 9851-1265, revision C, using scripts 9801-2591-804, was performed on the unit. Since the group determined that the contents of NVRAM should be preserved and not erased, the failure to clear the NVRAM resulted in an error in the ATP, and the HGS computer was considered to have failed the ATP. However, no additional failures were identified during the ATP.

C. Examination of the HGS Control Panel

The HCP, part number 1500-0500-002, serial number 15965, was placed onto a Rockwell Collins HGS computerized test station and power applied to the unit. The HCP Acceptance Test procedure, 9851-1313, revision B, using acceptance test script 9801-2571-802, was performed on the HCP. The HCP passed the procedure with no faults found.

D. Testing of the HGS Drive Electronics Unit

During the physical inspection of the HGS Drive Electronics Unit (DEU), part number 1500-0490-002, serial number 9298, a small impact mark on the top of the front panel was noted. In addition, the three circuit boards were loose in their respective card slots.

The HGS DEU was placed onto a Rockwell Collins HCTS and power applied to the unit. The DEU was tested in accordance with ATP procedure 9851-1333, revision D, and using script number 9801-2572-804. The unit passed the test procedure with no faults found.

E. Readout of Manually Entered Information into the HCP

The three HGS components were placed into a Rockwell Collins engineering test unit (ETU). The ETU was equipped with the entire contents of the HGS system. When power was applied to the system, the following information was noted on the HCP:

MODE Line: PRI
STBY Line: VMC
RWY Line: L 6522'
G/S Line: -3.00º
RWY EL: 613'

F. Examination of Enhanced Ground Proximity Warning System Computer

The Mark V Enhanced Ground Proximity System (EGPWS) Computer, part number 965-0976-003-217-217, serial number 9476, mod status 9, and configuration 217, was visually inspected. No visible damage was noted and all external electrical connectors were found in serviceable condition. After the unit was moved to a test bench and power applied, the EGPWS computer OK LED was verified. On the front panel of the unit, the following LED's were extinguished:

1) IN PROG LED
2) CARD CHNG LED
3) XFER COMP LED
4) XFER FAIL LED

A PCMCIA card was inserted into the DFDAU card slot and the flight history data from the unit's non-volatile memory downloaded onto the PCMCIA card. The flight history data on the card was converted in ASCII characters and examined; there were no EGPWS faults or warnings recorded during the accident flight approach. One element of the EGPWS-recorded flight history was an automatic recording of the airplane's latitude/longitude position as the airplane passes through 50 feet radio altitude. For the accident flight, the recorded position was:

Latitude: 41.78015º North
Longitude: 87.74506º West.

The unit was tested per Honeywell's ATP number 076-0879-002 Rev. L. The EGPWS passed all ATP tests with no faults found.

G. Examination of Radio Altimeters

1) Radio Altimeter Serial Number 04420

The Radio Altimeter part number ALA-52B, serial number 04420, software mod 01/01, and hardware mod 1 –2, 4-8, was visually inspected with no visible damage noted to the unit.

The unit was tested per Honeywell's Component Maintenance Manual ATP 34-42-35, revision 4, dated September 1, 2004 (publication number 1.B.1152A-2). The ATP revealed one flight history fault and one failed sub-test. The fault history download found one fault, the "SG MON_Compare_Fault". The description of the fault indicated that the " the computed altitude or status from the MON disagrees with that of the digital processor as reported by the MON through shared memory." To ensure this fault message was not erased, the ATP test procedure was aborted so the fault history data would not be cleared. During task 13, test 005 "sensitivity" test failed. No other faults were noted.

2) Radio Altimeter Serial Number 04382

The Radio Altimeter part number ALA-52B, serial number 04382, software mod 01/01, and hardware mod 1 –2, 4-8, was visually inspected with no visible damage noted to the unit.

The unit was tested per Honeywell's Component Maintenance Manual 34-42-35; revision 4 dated September 1, 2004 (Publication # 1.B.1152A-2). There were no flight history faults recorded.

H. Examination of Flight Control Computers

1) Flight Control Computer Serial Number 1G1VH

The Flight Control Computer (FCC-730), part number 822-1604-101, serial number 1G1VH, was examined for visible damage, with only minor cosmetic damage to the unit covers and evidence of mud and water splashing on the FCC. The unit was then placed onto a Rockwell Collins production test station and power was applied to the unit. The unit's NVM was downloaded into an electronic file and the contents converted to ASCII characters for evaluation.

The unit was tested per Rockwell Collins FCC-730 acceptance test procedure. The ATP was accomplished with no faults found.

2) Flight Control Computer Serial Number 1G1WC

The second FCC-730, part number 822-1604-101, serial number 1G1WC, was examined for visible damage, with only minor cosmetic damage to the unit covers and evidence of mud and water splashing on the FCC. The unit was then placed onto a Rockwell Collins test bench and power applied to the unit. The unit's NVM was downloaded into an electronic file and the contents converted to ASCII characters for evaluation.

The unit was tested per Rockwell Collins FCC-730 ATP. During the test, a fault was logged that indicated a ground return measurement in the unit. Further investigation by Rockwell Collins determined that a misaligned adapter pin on the rear electrical plug connector caused the fault. Measurement of the resistance between the pins on the unit indicated it was not failed. No other faults were found.

11. Engine Controls

An inspection of the maintenance logbook located in the airplane flight deck was examined for the accident flight and previous flights. There were no notations or write-ups regarding the engine controls on any of the logbook pages, including the accident flight.

The engine accessory unit (EAU) was removed from the E3-2 shelf of the forward EE bay for further examination.

A. Examination of Engine Controls

The area of the throttle quadrant (see Figure 2) was examined and documented for thrust reverser operation in both the flight deck area and the linkages below the cockpit floor level. The area of the throttle quadrant linkages, push-pull rods, and microswitch package were examined. Portions of the Reverse Thrust Lever (Difficult to Move In Reverse Thrust) – Fault Isolation procedure (76-05-00-810-805-F00) were performed on the airplane, without disconnecting any components during the procedure.

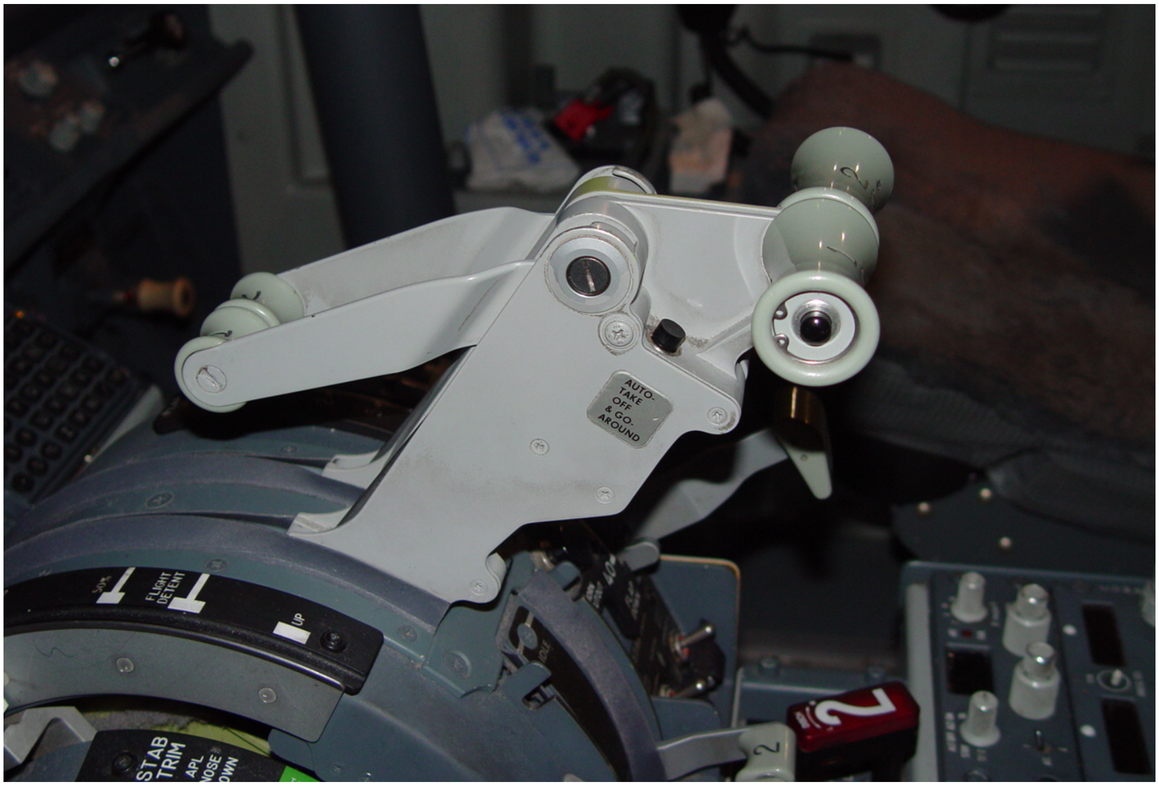

Figure 2 - Thrust Levers Against Idle Stops.

The blue torque stripe material in the area of the throttle quadrant linkage, push-pull rods and fasteners were unbroken and undisturbed. The blue torque stripe materials on the push-pull rod attachment hardware and adjustment nuts were undisturbed. The area looked clean, and the throttles and associated hardware moved smoothly and without binding. There were no materials on the white insulation blanket material beneath the throttle quadrant that appeared to have been jammed or stuck in the throttle linkage area.

All the cams for the thrust reverser resolver micro switches appeared clean and shiny. The wire bundles leading from the reverser resolver and microswitches appeared clean and did not exhibit any chafing, breaks, or kinks. There were no visible areas of wire burns or shorts in the wiring.

B. Measurement of the TRA Thrust Reverser Lockout Position

The group developed a procedure to measure the position of the reverse thrust interlock latch as a function of thrust resolver angle. The procedure measured the minimum TRA at which the thrust reverser levers were locked out. The procedure also measured the TRA at the flight idle stop. The tests were accomplished on the accident airplane, as well as additional Southwest 737-700 airplanes.

The first iteration of the procedure, performed on the accident airplane, used aluminum shims to build up a buffer to prevent the throttle lever from moving while engaging the thrust reverser levers. Using this method, the number 1 and number 2 throttles could be moved 0.1875 inch forward of the throttle idle stop before the reverser lockout mechanism would engage. In addition, this procedure was used on another Southwest 737-700, N483WN, which had been positioned overnight at Midway Airport. The procedure and the results from the examination of N483WN are included in Attachment 2.

The group determined that shims were difficult to use for the test. When several were placed together, the shims would slide relative to each other and make the test difficult to complete. In addition, the group found that finer fidelity than offered by the shims was needed to better measure the throttle position.

At the group's request and coordination, Boeing created a tool to more precisely measure the resolver position by use of adjustment screws. The tool (see Figure 3) attached to the thrust idle stop by the center idle stop plate screw and allowed the user to rotate the adjustment screw to determine a repeatable (and more precise) TRA for the position of the reverse thrust interlock latch.

Figure 3 - Tool, shown as installed, used to assist TRA measurement.

The group re-measured the accident airplane using the tool (see Figure 4). In addition, a revised test procedure (revised from previous tests) was used for the test (see Attachment 3). Power could not be applied to the airplane, so the thrust resolver angle was measured by use of a Boeing Position Transmitter Test Set (part number FD1104-26.53, unit number 5, serial number 30-144385). The test set applied ground power to the thrust resolver and provided a digital readout of the TRA and in-phase voltage of the S3-S1 (y-x) winding.

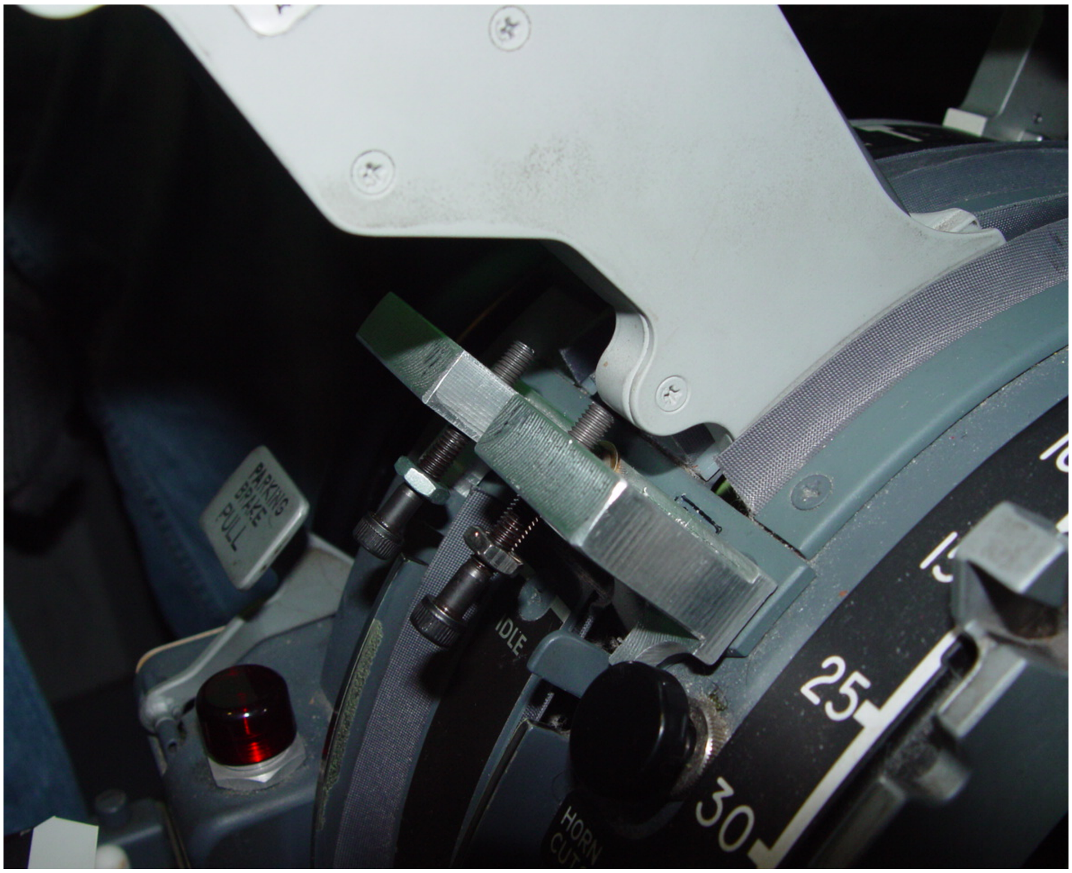

Figure 4 - Measurement tool and Number 2 Throttle.

The test procedure was also used to measure the TRA on two additional Southwest airplanes – N794SW and N441WN. These airplanes were stationed overnight in the Southwest maintenance hangar at Midway Airport. For these tests, ship power was available, so the thrust resolver angle was read from the airplane FMCS CDU; the Boeing test box was not used for these tests. For each airplane the procedure was accomplished twice. During the test procedure on N794SW some slight movement was noted in the test measurement tool about the center idle stop plate screw while manipulating the throttles.

Southwest Airlines downloaded the FDR from the two additional airplanes after the test. Copies of these raw binary FDR data files were provided to the NTSB and converted to engineering units. The measured TRA and the resultant FDR-recorded TRA data (with all values described in degrees) were entered into a spreadsheet for comparison against the accident airplane and the other Southwest 737-700's. The spreadsheet of TRA values is included in Attachment 4.

C. Examination of the Engine Accessory Unit (EAU)

The EAU, part number 285A1300-1, serial number D01700, was placed onto an engineering test bench and power applied to the unit. The contents of the unit's 40 bytes of non-volatile memory (NVM) queried and copied. The values were extracted in ASCII characters, as follows:

Engine 1: x{{{{{{{{{{{{{{_^^^^^___________AY[A@@b@

Engine 2: x{{{{{{{{{{{{{{ONNNOOOOOOOONNOOOQYJC@@bP

The extracted fault information was converted into individual fault codes using a spreadsheet. A copy of the resultant spreadsheet was provided in an electronic format.

In addition to the NVM, the EAU front panel has a press-button feature that annunciates the current thrust reverser deploy faults and thrust reverser stow faults by use of red light-emitting diodes (LED). Each engine has a button to show either the Deploy or Stow faults.

For engine 1, the following fault lights were illuminated when the T/R Stow Fault button was pushed:

S831 – L SLEEVE STOW SENSOR
S835 – L SLEEVE LOCK SENSOR
S833 – HYD ISO VALVE SENSOR
S832 – R SLEEVE STOW SENSOR
S836 – R SLEEVE LOCK SENSOR

For engine 1, when the T/R Deploy Faults button was pushed, no fault lights were illuminated.

For engine 2, the following fault lights were illuminated when the T/R Stow Fault button was pushed:

S831 – L SLEEVE STOW SENSOR
S835 – L SLEEVE LOCK SENSOR
S832 – R SLEEVE STOW SENSOR

For engine 2, the following fault light were illuminated when the T/R Deploy Fault button was pushed:

S830 – HYD ISO VALVE SENSOR
S836 – R SLEEVE LOCK SENSOR

The EAU was placed onto a BAE systems ATS-182a test station and the BAE Systems acceptance test procedure was performed on the EAU. The EAU passed the procedure with no faults found.

Thomas R. Jacky
Aerospace Engineer