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A SERVICE PUBLICATION OF LOCKHEED-GEORGIA COMPANY, A DIVISION OF LOCKHEED CORPORATION

Mark II Anti-Skid

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Cover: General William G. Moore, commander of the Military Airlift Command at the time this picture was taken, prepares to land a C-130 Hercules on an autobahn near Stuttgart, Germany, as part of a NATO exercise. The landing on this unconventional "airstrip" demonstrates once again the ability of the Hercules aircraft to operate from almost any kind of surface at anytime. Photo courtesy of the USAF.

Robert B. Ormsby

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Robert B. Ormsby
Lockheed-Georgia Company
Marietta, Georgia 30063

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Sincerely,

Robert B. Ormsby
President
Lockheed-Georgia Company

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MARK II Anti-Skid for Hercules Aircraft

– a reliable system that stops a skid before it starts.

by: **Harold Cook**, *Senior Functional Test Engineer*
Elbert Fields, *Service Analyst*

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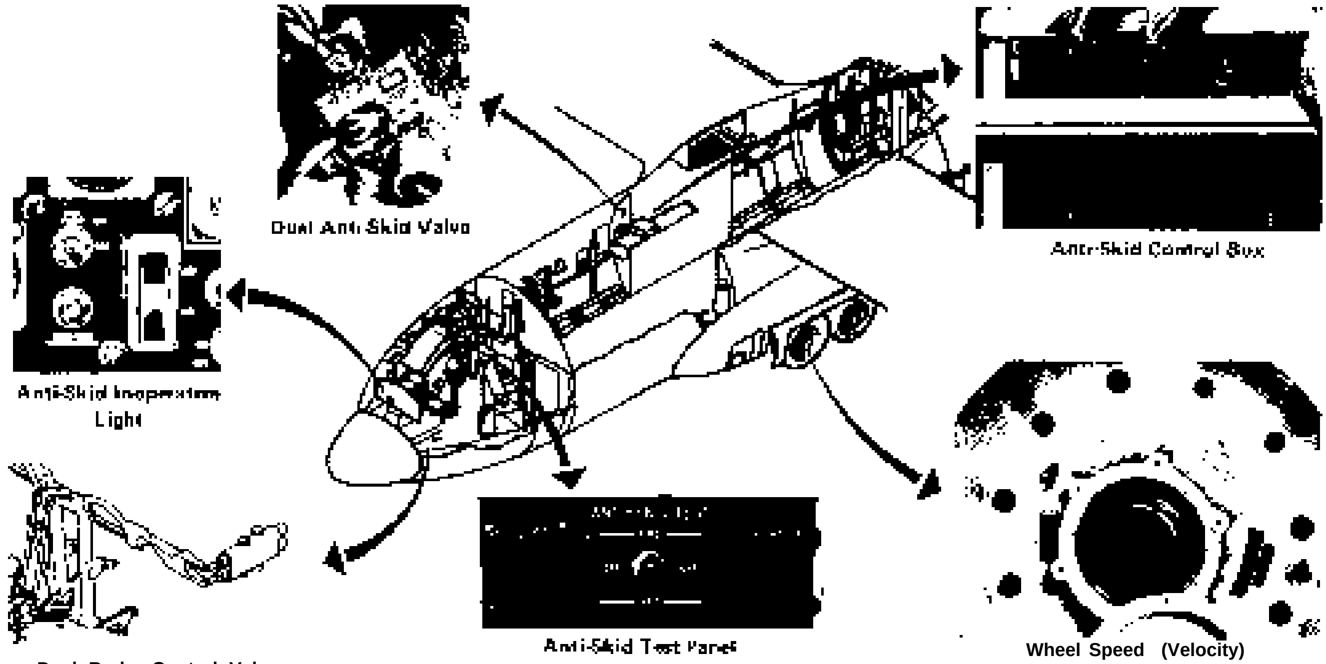

MARK II ANTI-SKID COMPONENTS LOCATION

Anytime a Hercules aircraft lands on a wet or icy runway, the flight crew has confidence that the Mark II anti-skid system will prevent a skid. The crew members know that they can depend on this reliable system to control skids by modulating brake pressures.

Because maintenance personnel can sometimes find the Mark II anti-skid rather complicated, we feel that a review of the system, its operations, features and trouble-shooting procedures would be helpful.

Originally, the Mark II anti-skid system was installed during production of late-model Hercules aircraft. Along the way, most of the early model Hercules have been retrofitted with the Mark II anti-skid in accordance with maintenance directives.

MARK II ANTISKID OPERATION

The Mark II system detects an approaching skid by measuring wheel deceleration. If a wheel slows down more than it should, and before it can begin to skid, the Mark II reduces brake pressure proportionally so that the skid is prevented. The Mark II prevents skidding by sensing the exact amount of brake pressure needed for safe braking under all runway conditions and without tire damage.

The Mark II monitors itself continually with a fail-safe system that detects power loss or open valves and displays these findings visually in the Hercules aircraft flight station.

Of the Hercules' two independent hydraulic braking systems, the Mark II controls the "normal" system only; the direct-acting "emergency" system is unaffected by the anti-skid system.

The individual control of each main landing gear wheel is a special feature of the Mark II anti-skid system. To accomplish this, the Hercules uses two dual brake control valves, one for the right main wheels and one for the left. In addition, an anti-skid valve housing two individual control valves is incorporated for the forward and aft brake on each side of the airplane. The Mark II anti-skid system therefore limits excessive brake pressure on each main wheel to protect the wheels from skids, while still applying the brakes to the best advantage.

One velocity transducer in each main wheel axle provides wheel speed information to a control box located over-head on the aft cargo equipment rack, between fuselage stations 417 and 437. The information is in the form of frequency-modulated signals that continuously and instantaneously transmit the velocity of each wheel.

The control box receives these speed measurements and does several things. It provides fully modulated brake pressure control, including four adaptive control loops: the velocity threshold, rate threshold, pressure bias modulation (PBM), and proportioning control. The control box also monitors the valves and their wiring continuity, and the loss of system power (partial or complete). Lights on the anti-skid test panel located on the aft end of the flight station overhead panel

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monitor the system and indicate any malfunctions. The brake control panel located in front of the copilot position also has an anti-skid inoperative light.

The Mark II anti-skid system can be tested on the ground or in flight. An anti-skid test switch gives a complete check of all four main wheels through indicator light displays.

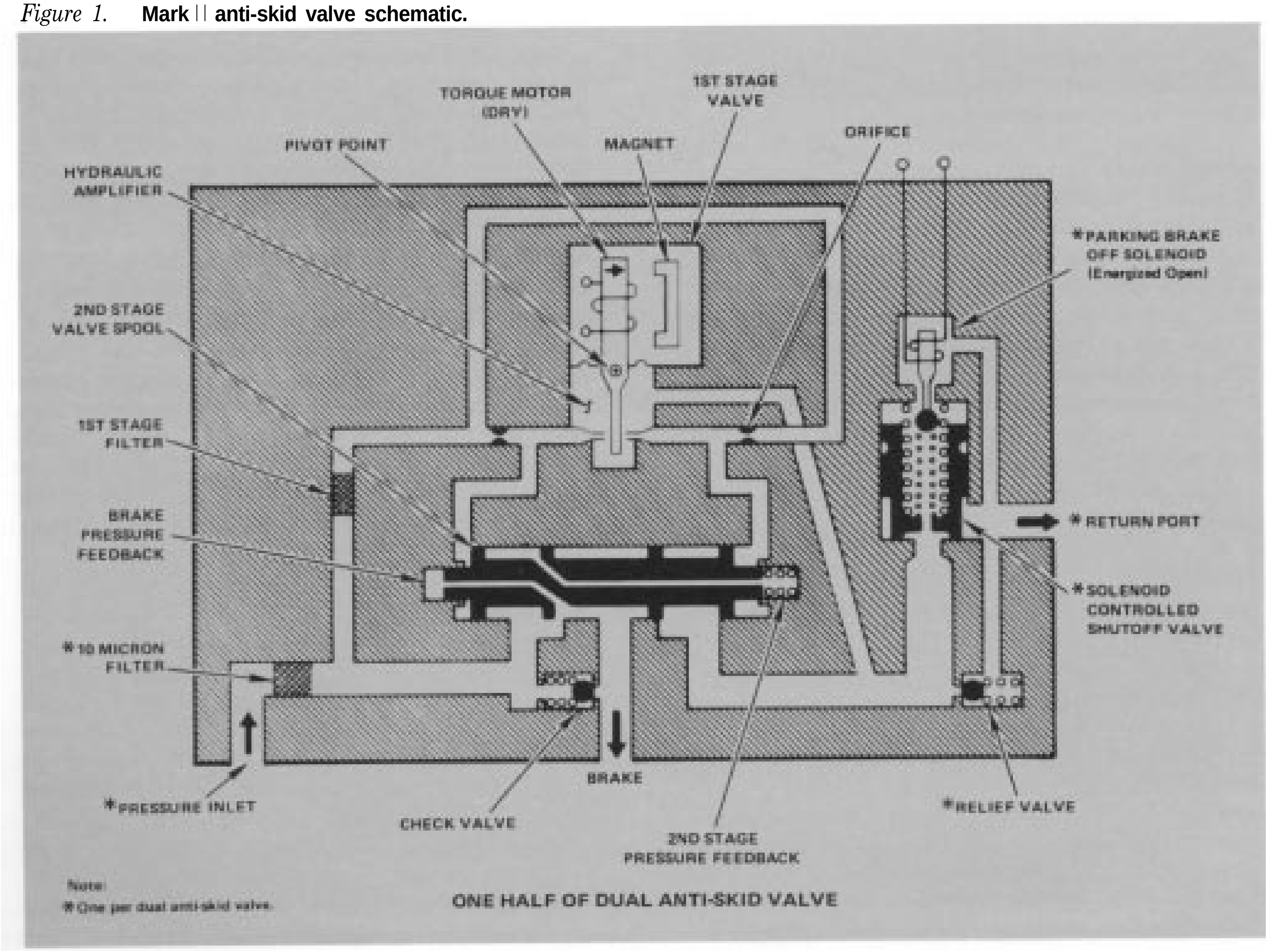
One half of each dual anti-skid valve controls a single wheel (Figure 1). Each half has an electromagnet-controlled first stage and a pressure-balanced second stage for fully modulating the hydraulic brake pressure. A mechanically driven hydraulic amplifier first stage provides a differential output pressure on signal from a DC torque motor which accepts the control box pressure relief commands. Final brake pressure is set through pressure feedback with the second stage power spool and sleeve.

The wheel velocity transducers are bidirectional, variable reluctance pulse generators. Since the transducers are bidirectional, there are no difficulties with reversed polarity in the airplane's wiring. A fixed, wound coil provides the magnetic field. Rotation of a toothed rotor (attached to the wheel) inside a mating toothed stator produces variations in the magnetic air gap at fifty Hertz per wheel revolution. This induces an AC component on a DC input line, and the frequency of this AC component is sensed as wheel speed by the anti-skid control unit.

Each control unit has four wheel control circuit cards (one for each main wheel), and a test and failure logic circuit card, also identified as the auxiliary card. The control box uses all solid-state circuitry.

Two control box configurations are used on the Hercules aircraft. All USAF, U.S. Coast Guard and a few export Hercules equipped with the Mark II system use Lockheed P/N 697398-3 (Hydro-Aire P/N 42-109-1A) control boxes and have 2030 psi brake pressure. The U.S. Navy, all commercial, and most export Hercules are equipped with Lockheed P/N 697398-l (Hydro-Aire P/N 42-109 or 42-109A) control boxes and have 3000 psi brake pressure. The function of both control boxes is identical, but there are small circuit differences such as resistance values to compensate for the two different brake pressures.

Figure 1. **Mark II anti-skid valve schematic.**



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Left: The left dual anti-skid valve, located on the utility hydraulic valve panel. *Right:* The right dual anti-skid valve, located on the booster hydraulic valve panel.

FEATURES OF THE MARK II ANTI-SKID SYSTEM

The following features of the Mark II system should be familiar to the maintenance technician before an attempt is made to troubleshoot the system.

A fully modulated brake pressure control maintains the best braking for the existing runway conditions. This is achieved by high-response modulated control of brake pressure on skid command (pressure decrease) and increasing pressure reapplication. Pressure release and initial reapplication are controlled by the proportioning control; this allows the braked wheel to recover from a skid while pressure bias modulation sets the reapplication at an average level to hold maximum braking without an immediate recurrent skid. Such control of pressure removal with the high response of the anti-skid valve reduces efficiency losses associated with over-dumping of on-off type systems.

The Mark II provides positive locked-wheel protection in case the wheel should go into a degenerative skid, as could happen if the wheel left the ground on a bounce. Touch-down protection prevents the pilot from landing with the wheel brakes applied.

As discussed previously, individual wheel velocities are sensed as a function of signal frequency from the wheel transducer. This frequency-modulated signal is squared and converted to a DC voltage proportional to the respective wheel speed by the velocity amplifier. A rate amplifier senses changes in the DC voltage to provide the appropriate reaction.

A slight brake pressure increase will cause wheel velocity to slip beyond the point of maximum retarding force and the wheel will begin a full skid. As the wheel starts slowing faster than a pre-established maximum normal deceleration, a rate change of the DC signal is sensed. If wheel deceleration is greater than the rate threshold and continues long enough to produce a net wheel change, an output will result from the rate detection circuit. This output is applied to the anti-skid valve through the rate amplifier. The established thresholds preclude false skid signals and rapid skid cycling caused by runway noise and oscillations of the landing gear.

A key part of the Mark II system is the adaptive pressure bias modulation (PBM). As the wheel recovers from a skid, the rate signal disappears and brake pressure increases. At this point, PBM goes to work. Without PBM, brake pressure applied after a skid signal would return to the original pressure (which caused the original skid); consequently, the wheels would skid again.

To explain the features of the anti-skid systems operation further, the electronic-bias PBM system sends an additive electric signal to the skid control valve in proportion to the amplitude and duration of the last control signal. In other words, a short, low-amplitude signal will cause a small residual skid control valve current after skid correction, reducing brake pressure slightly below the pressure which produced the skid signal. Likewise, if a large amplitude signal of a long duration is sensed – and this may occur on an icy or wet runway – the PBM will add a larger bias current to the skid control valve, resulting in much less brake pressure. Bias increases

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with each skid signal proportionally to a time characteristic for the airplane. Therefore, severe skids set up a large bias and successive small skids are integrated to maintain a pressure bias level less than that which will cause a skid to happen.

A somewhat more complicated part of the Mark II's operation is a proportionate system response control. This control reduces the rate threshold to allow the skid to be sensed earlier if surface conditions cause excessive or prolonged skids leading to loss of braking efficiency. The proportionate control also provides a means of partially correcting for hydraulic lag by shaping the skid control valve signal during the recovery part of the skid cycle. This unique feature is especially valuable on slick runways where the skid signal causes a complete pressure release, and hydraulic lag will delay reapplication of the brakes.

Positive locked-wheel protection shuts off brake pressure if a wheel goes into a lock-up skid. This additional system is superimposed on the rate detection circuit, and it sends a full-pressure release signal to the anti-skid valve for the locked wheel.

The instant the wheel is unlocked, normal valve control with PBM resumes, releasing and reapplying pressure smoothly for normal braking.

Should one wheel lock while the others are still turning, the PBM circuit supplies an anti-skid valve signal higher than a normal control signal. If the cause of the locked wheel is second-stage anti-skid valve fouling, the higher control signal will clear the valve. Locked-wheel action is effective only when the airplane is moving at a speed above 15 knots. This will allow normal brake pressure application during low-speed turning and parking.

A touchdown control ensures full anti-skid valve current (zero brake pressure) before touchdown. This prevents brake application until the wheels have spun up above 15 knots, even if the pilot should land with the brake pedals depressed.

TROUBLESHOOTING

The following troubleshooting procedures are intended primarily as an aid for isolating a malfunction to a component of the Mark II anti-skid system or eliminating the anti-skid system components as the source of brake system malfunctions.

The following circuit breakers must be closed for proper operation of the system and test/failure logic:

Circuit Breaker	Location	Bus
EMER BRAKE VALVE	Copilot's lower circuit breaker panel	ESSENTIAL DC BUS
ANTI-SKID CONTROL	Copilot's lower circuit breaker panel	MAIN DC BUS
ANTI-SKID FAIL-SAFE LIGHT	Copilot's lower circuit breaker panel	MAIN DC BUS
ANTI-SKID TEST 26 VAC	Pilot's lower circuit breaker panel	AC INST & ENG FUEL CONT BUS

System Test Logic

Let us first discuss the system's self-test capabilities, or test logic. The ANTI-SKID TEST switch and four indicator lights are located on the anti-skid test control panel (Figure 2). The lights are identified as LEFT FWD, RIGHT FWD, LEFT AFT, and RIGHT AFT.

Figure 2. **Anti-skid test panel.**

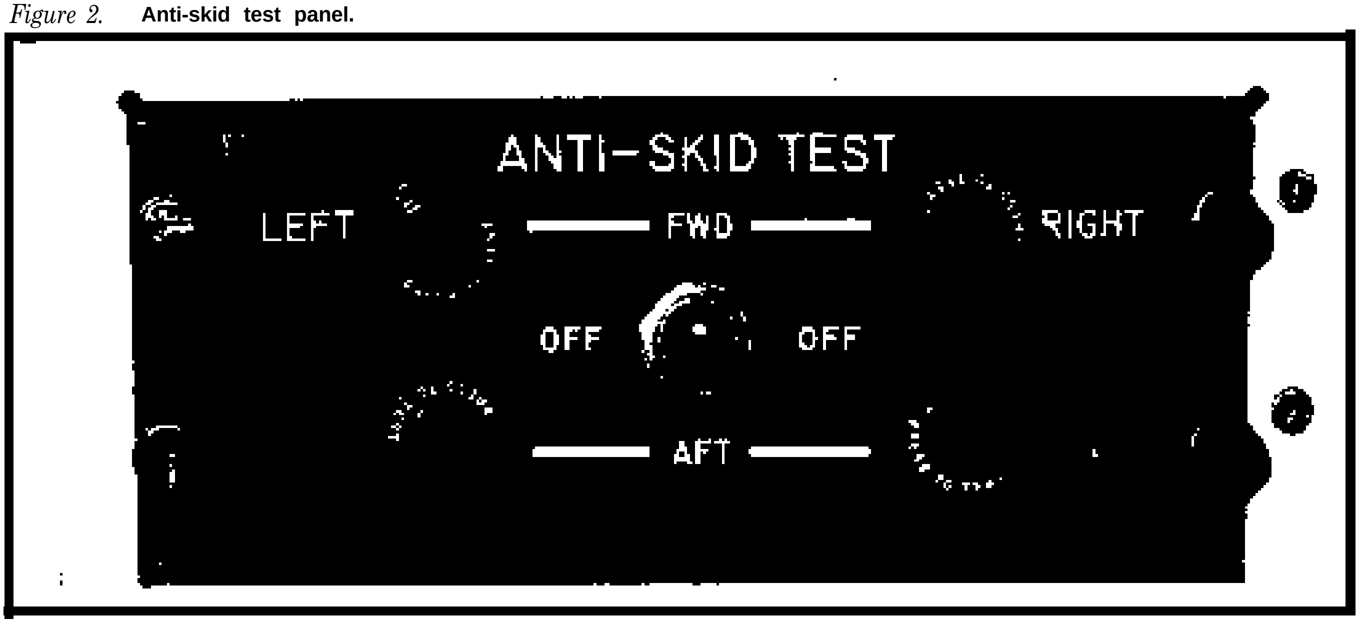
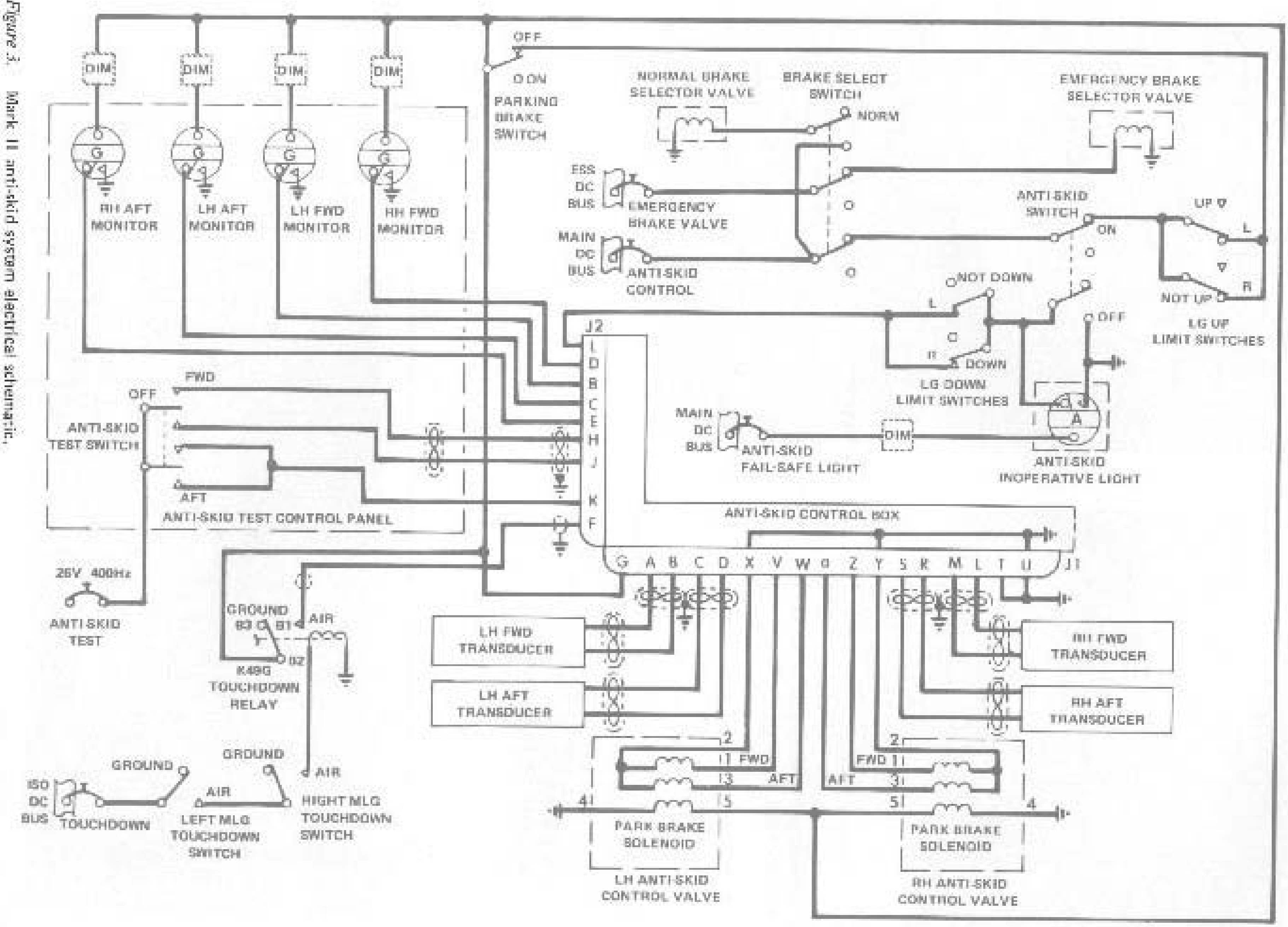


Figure 3. Mark II anti-skid system electrical schematic.



MARK II ANTI-SKID SYSTEM ELECTRICAL SCHEMATIC

With the BRAKE SELECT switch positioned to NORMAL, the ANTI-SKID switch to ON, the gear down limit switch closed and the parking brake released, the ANTI-SKID INOPERATIVE light should extinguish.

Test Logic Requirements: Airplane on ground, static, hydraulic pressure not required.

With the ANTI-SKID switch set to ON, the BRAKE SELECT switch set to NORMAL, and the parking brake released momentarily, position the ANTI-SKID TEST switch to FWD. While the switch is in the FWD position, none of the lights should illuminate. Release the switch and the LEFT FWD and RIGHT FWD lights should illuminate momentarily, then extinguish.

If either or both of the test lights fail to illuminate, check first to determine that 26 VAC is available. Next, see that the associated transducer and wiring circuit is complete. The 26 VAC, 400-Hertz test signal is applied to the anti-skid control box, travels through a resistor in the control box to the transducer return wire, then through the transducer and back into the control box on the transducer high wire. This will verify the integrity of the transducer and associated wiring with respect to continuity.

Verify that the transducer and associated wiring circuit is functional and that the AC test signal is available to the anti-skid control box. If problems still exist, then the fault should be located in either the associated individual wheel circuit card or the auxiliary (logic) card in the control box.

Repeat the preceding tests for the AFT switch position by simply substituting the AFT switch position in the place of the FWD switch position in the test description.

If the preceding tests are satisfactory, you have accomplished the following:

- The integrity of the transducers and associated wiring with respect to continuity was verified.
- The integrity of the dynamic (rate of deceleration) sensing capability of the control box was verified.
- The skid control valve output signal available from the control box was verified.

When the ANTI-SKID TEST switch is positioned to FWD, a 26 VAC, 400-Hertz signal is input to the left forward and right forward wheel cards in the control box, simulating a rotating wheel speed equivalent to 50 or 60 knots. At the same time, this reduces the sensitivity of the PBM and inhibits the locked-wheel sensing capabilities of the AFT wheel circuits.

Upon releasing the test switch, the test signal is removed from the forward control section. The removal of the test signal simulates a forward wheel skid to the anti-skid control box, which results in a valve signal to operate the FWD lights and forward valves.

Test Logic Requirements: Airplane on ground, static, in-flight condition simulated, hydraulic pressure not required.

Simulate in-flight conditions by placing a jumper between Bl and B2 on touchdown relay No. 1 (Figure 3).

When the touchdown relay (normally energized in flight) is jumpered to supply 28 VDC to pin “F” on J2 of the anti-skid control box, all four of the test lights on the anti-skid test control panel should illuminate.

Momentarily position the ANTI-SKID TEST switch to FWD. All four test lights should extinguish. Shortly after release of the test switch, the two FWD lights should illuminate momentarily. After 2 or 3 seconds all four lights should illuminate and stay illuminated.

Repeat the preceding test for AFT switch position by simply substituting the AFT switch position in the place of the FWD switch position.

If the above checks are satisfactory, the following was accomplished :

- The integrity of the locked- wheel circuits in the control box was verified.
- The spin-up override of the 28 VDC squat signal was verified.

Test Logic Requirements: Preflight, normal hydraulic pressure required.

Depress and hold the airplane pedals with the airplane stopped and the ANTI-SKID INOPERATIVE light not illuminated.

Check to see that all four anti-skid test lights are extinguished.

Momentarily position the ANTI-SKID TEST switch to the FWD position. Upon the release of the test switch the LEFT and RIGHT FWD indicator lights should illuminate for approximately half a second before extinguishing. Have an observer check the brake pistons and the brake pressure plates for “brake-off’ and “brake-on” conditions. The brake-off condition (brake pressure plates relaxed) should coincide with the illumination of the test lights for the associated wheels only. A brake-on condition (brake pressure plates compressed) should occur when the test lights extinguish for the associated wheels.

Repeat the preceding test for the aft test switch position by simply substituting the AFT switch position in the

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place of the FWD switch position in the test description. Observe corresponding results for the associated lights and brakes.

If the functional integrity of the anti-skid system is in question, proceed as follows:

Connect a brake pressure gage to the brake bleed valve on each wheel and repeat the immediately preceding test. The brake pressure gages should indicate the following:

After releasing the test switch from the FWD position the LEFT and RIGHT FWD indicator lights should illuminate and the indication on the pressure gages attached to the forward wheels should drop to less than 100 psig. The FWD indicator lights should extinguish in approximately one-half second and the brake pressure will be reapplied to the forward wheels as indicated by the forward brake pressure gages. The pressure gages for the aft wheels will show a momentary drop in brake pressure concurrent with the reapplication of brake pressure to the forward wheels. Repeat the test in the AFT test switch position and look for corresponding results at the aft wheels.

System Failure Logic

System failure logic will detect the following system malfunction 100 percent of the time if the system is energized:

- Open valve – The associated test light will illuminate.
- Open valve wiring – The associated test light will illuminate.
- Loss of 28 VDC control box input power -- The ANTI-SKID INOPERATIVE light will illuminate.
- Parking brake set – ANTI-SKID INOPERATIVE light will illuminate.
- BRAKE SELECT switch in emergency – ANTI-SKID INOPERATIVE light will illuminate.

A control box power loss detector is connected to the system input power. When the system input power is absent or below the voltage level of the indicator circuit, the power failure detector supplies the necessary ground path for the ANTI-SKID INOPERATIVE light.

So far we have verified the actual operational integrity of the control box and, with respect to the continuity, the integrity of the transducers and associated wiring and the valve and associated wiring.

If the last test did not isolate the cause of the reported malfunction, perform these additional tests:

Wheel Transducer Operational Check

Remove the individual wheel covers. Drive the wheel speed transducers one at a time at 200 rpm or higher, using a variable-speed electric drill motor and adapter (Figure 4).

Observe the lights on the anti-skid test control panel while rotating the wheel speed transducers. The light associated with the transducer being rotated should extinguish. The other three panel lights remain illuminated.

NOTE: Fluctuations in drill speed can simulate a skid, causing the light on the anti-skid test control panel for that wheel to illuminate. Therefore, be careful to maintain a smooth drill motor speed during this test.

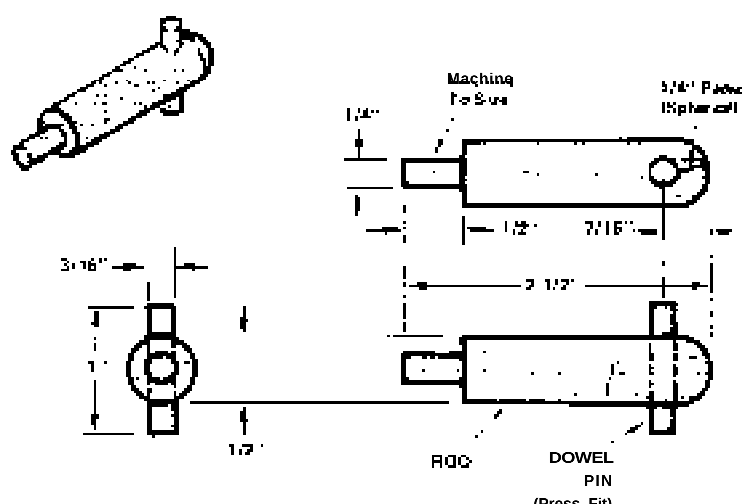
The same transducer test should be performed for the other three transducers with corresponding results.

If the test proves satisfactory, you have accomplished the following:

- The operational integrity of each wheel speed transducer and associated wiring was verified.
- The proper operation of the control box locked-wheel sensing and memory circuits with an actual rotating transducer was verified

Figure 4. Wheel spin-up adapter and speed transducer.

Left: Construction details of a wheel spin-up adapter. *Right:* Whe adapter engages these slots for the operational check.



Materials List:

1 ea. steel or alum. rod 2-1/2" x 1/2" dia.
(machine end to 1/4" to fit drill chuck)

1 ea. steel dowel 1" x 3/16" dia.

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Brake Pressure Check (Normal hydraulic pressure required)

Install a pressure gage at each brake bleed valve.

Position the ANTI-SKID switch to ON, the BRAKE SELECT to NORMAL, and the park brake to RELEASED.

The pressure gages for all four wheels should indicate 100 psi or less with hydraulic system pressure available and brake pedals released. Depress the brake pedals against the "brake-on" stopbolt and determine that the four pressure gages indicate 1700 psi minimum for the 2030 psi brake system and 2800 psi minimum for the 3000 psi brake system. (Determine which brake system your airplane contains.)

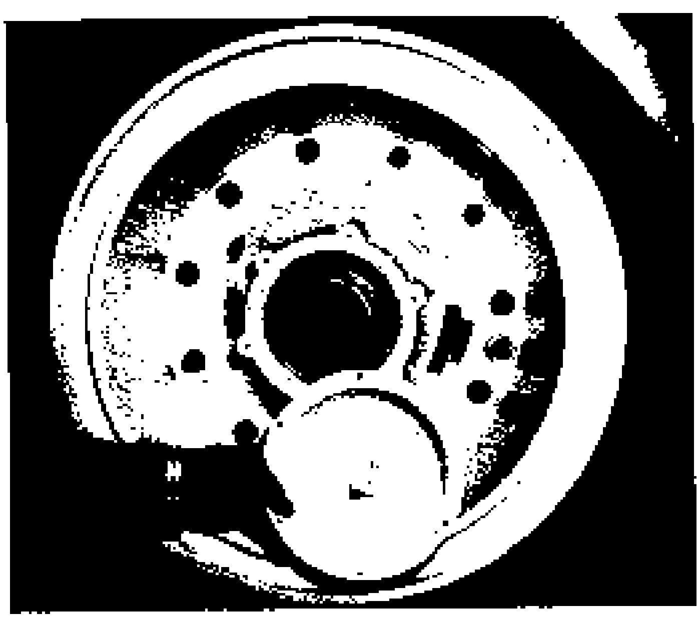
NOTE: The four test lights on the anti-skid test control panel should not illuminate.

Operate the ANTI-SKID TEST switch momentarily in the FWD position. The brake pressure should not change and test lights should not illuminate.

Release the test switch and observe the following:

- The LEFT FWD and RIGHT FWD test lights should illuminate and then extinguish after a short time delay.

over removed to show slots in wheel speed transducer. The spin-up



- The brake pressure on the two FWD pressure gages should drop to 100 psi or less. This will coincide with illumination of the FWD test lights.

- As the FWD test lights extinguish, pressure will be reapplied to the forward wheels and the forward pressure gages should indicate 1700 psi minimum (2030 psi system) or 2800 psi minimum (3000 psi system).

- Upon reapplication of pressure to the forward wheels, the pressure at the aft wheels of the airplane should fluctuate.

Repeat the procedure for the AFT switch position.

Additional Troubleshooting Tips

. If only one of the test lights illuminates when the ANTI-SKID TEST switch is released, the most probable causes are as follows:

A. An open transducer or transducer wiring.

B. A defective anti-skid control box.

C, A defective test light or defective wiring

- If none of the test lights illuminates for FWD and AFT tests, the most probable causes are:

A. 26 VAC, 400-Hertz ANTI-SKID TEST circuit breaker open (pilot's lower circuit breaker panel).

B. No power on the 26 VAC bus (AC INST & ENG FUEL CONT BUS).

C. Anti-skid control box.

D. Defective test switch or wiring.

E. Open transducers or transducer wiring.

. If one or more of the test lights are illuminated prior to operation of the test switch, the most probable causes are :

A. Open valve wiring,

B. Open valve.

C. Anti-skid control box,

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. The actual voltage available at the valve connector can be determined as follows:

A. Disconnect the valve connector and connect a 200 ohm, 1/4-watt resistor between connector pins 1 (pus) and 2 (neg) to check the forward brake circuit, and between pins 3 (pos) and 2 (neg) to check the aft brake circuit (Figure 3).

B. Check the voltage across the resistor during the FWD and AFT test switch operations for 4.5 VDC minimum (2030 psi system) or 5.5 VDC minimum (3000 psi s&stem); and during the sensor operational check fur 7.5 VDC minimum (either system) using a volt-ohm meter.

C. 28 VDC should be present at the connector between pin 5 and ground pin 4 to energize the parking brake shutoff valve solenoid (Figure 3).

GENERAL INFORMATION TO REMEMBER

- All test lights should be illuminated when the airplane is in flight, gear down, and the wheel rpm is less than 15 knots.

- All test lights should extinguish upon touchdown (wheel spin up above 15 knots).

- It is perfectly normal for the test lights to illuminate and extinguish during the landing roll as the brake pressure is varied by the anti-skid system.

- The test lights indicate anti-skid control (valve voltage).

- Performing the test switch operation rapidly and repeatedly to either position or between positions can sometimes cause an apparent system malfunction.

- The rapid operation between FWD and AFT test switch positions can result in the loss of brake pressure to all wheels (during and possibly a few seconds after this type of operation).

- Air in the hydraulic system can cause an apparent electrical malfunction.

- At least one of the main wheels must be rotating above 1.5 knots before the anti-skid system locked-wheel protection is available.

- The anti-skid valve voltage should rise to a minimum of 4.5 VDC with control box (P/N 42-109-1A) or 5.5 VDC control box (P/N 42-109), when the ANTI-SKID TEST switch is released from either the FWD or AFT position.

- The shield for the transducer wiring should be grounded only at the control box.

- The anti-skid valve voltage for a locked wheel should be a minimum of 7.5 volts (check during sensor operational check).

- The brake system pressure is 2030 psi or 3000 psi, depending on the pilot's brake metering valves used. Lockheed P/N 695015 or 695893-( ) brake metering valves provide 2030 psi, while Lockheed P/N 697395-l valves provide 3000 psi pressure. The 2030 psi and 3000 psi brake metering valves must not be interchanged, since the proper function of the anti-skid system will be impaired, with possible serious consequences.

The Hercules MARK II anti-skid system is one of the safest and most dependable anti-skid systems to be used on any airplane. Like any system, however, a certain amount of care and attention is necessary to maintain the highest level of performance. We hope this article will enhance your understanding of the MARK II system and help keep maintenance problems to a minimum.

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StarTip

TROOP SEAT INSTALLATION TOOL

by **Jack McHaney**, *Aircraft Mechanic, General*

After washing, the nylon in the Hercules aircraft troop seats tends to shrink when drying. This shrinkage sometimes makes reinstallation of the troop seats a back-breaking job.

Attaching the troop seats to the aircraft's sidewall is easy; it is securing the seat legs to the aircraft's floor that can be difficult. This is when a troop seat installation tool can come in handy.

To use the troop seat installation tool described below, attach it to the troop seat leg as shown in Figure 1. The tool gives the necessary leverage and control to assist in stretching the nylon in the seat, aligning the leg, and attaching it to the aircraft floor.

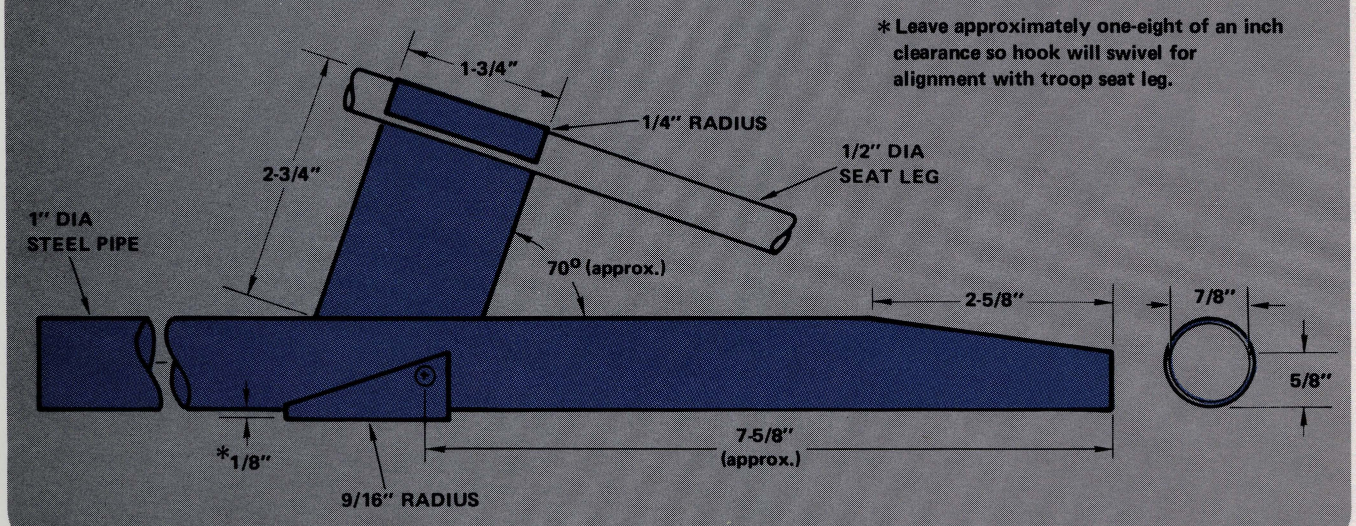
Note: Avoid applying excessive leverage; damage to seat leg could result.

The troop seat installation tool is relatively easy to make, and it can save a lot of time and effort. For your convenience, we have included the following drawing and a list of materials to aid you in constructing the tool.

Figure 1. Troop seat installation tool shown in position for assisting in seat leg alignment. *Photo courtesy of Jack McHaney.*

MATERIALS LIST

1 ea. Steel plate 1/8" x 1-3/4" x 6-1/2"
1 ea. Steel pipe 1" diameter x 28"
1 ea. NAS623-3-20 screw
1 ea. MS21042-3 nut



Ramp Actuator Damage

Some Hercules aircraft operators have experienced ramp actuator piston rod damage due to insufficient clearance between the ramp actuator rod and a bolt and nut used to attach a fitting to the sloping longeron (P/N 342986-3L or -3R) at fuselage station 847, butt line 61.625.

The clearance problem occurs during in-flight operation of the ramp. The actuator rod position shifts due to component wear and forces acting on the ramp during in-flight operation.

Lockheed recommends that Hercules operators check the clearance between the ramp actuator rod and the fitting attachment bolt installation to be sure that the actuator is not striking an attachment bolt.

If insufficient clearance between the ramp actuator rod and a fitting attachment bolt is observed, the following corrective action is recommended:

- Reference Figure 1 – Remove the lower bolt P/N MS20005-20 (A), washer P/N MS20002C5 (B), washer P/N MS20002-5 (C), and nut P/N EB054 (D) that the actuator rod is striking.

- Reference Figure 2 – Reinstall the bolt (A) in the opposite direction (head outboard). Make certain that the washer (B) is under the bolt head and washer (C) is placed under the nut (D). Retorque the bolt to the torque value specified in the applicable maintenance manual.

This simple modification should prevent any future clearance problems between the ramp actuator rod and the fitting attachment bolt installation and help prevent the costly expense of replacing damaged actuators.

SERVICE NEWS

View **of the left ramp actuator with the fitting bolt reversed (modified fitting installation).**

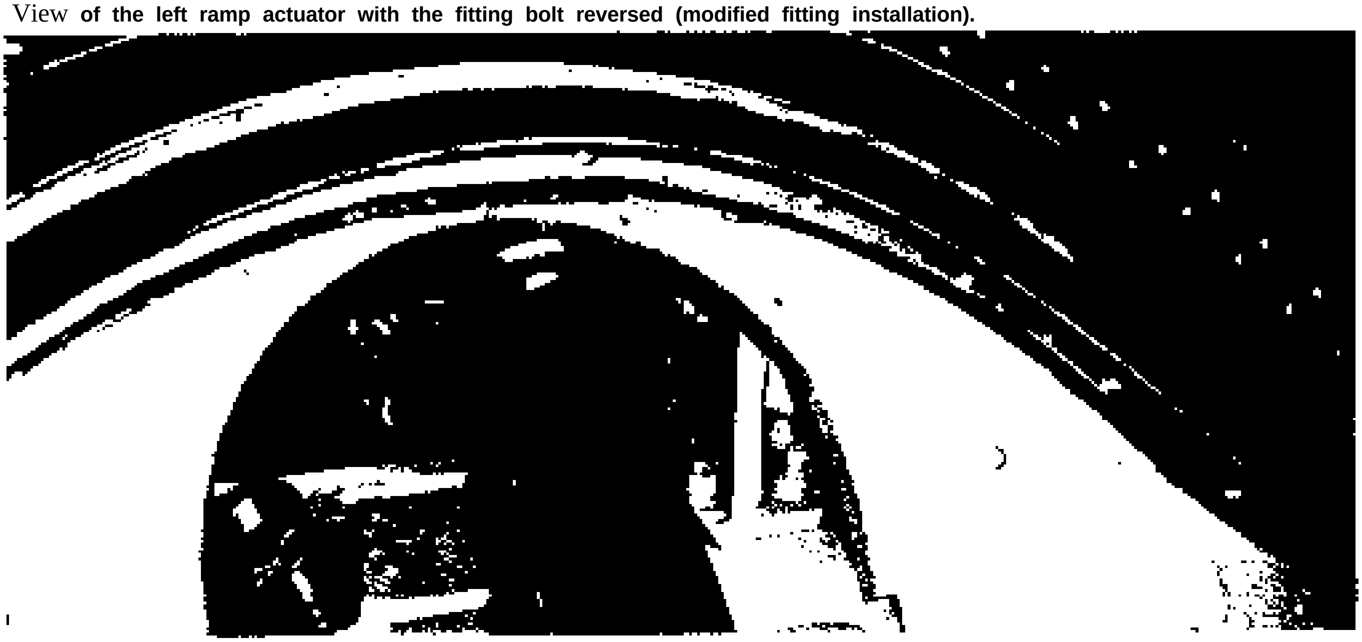


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Figure 1. Unmodified fitting installation.

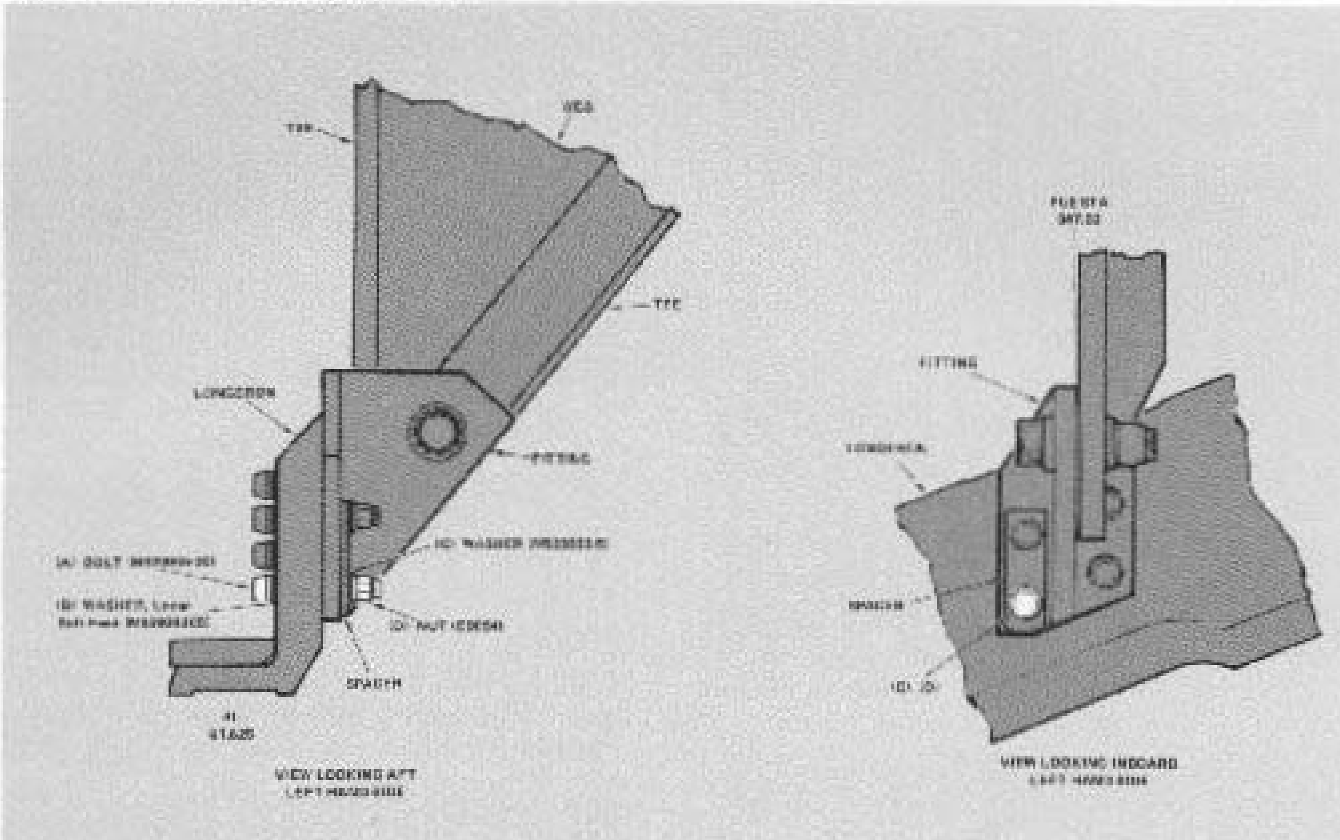
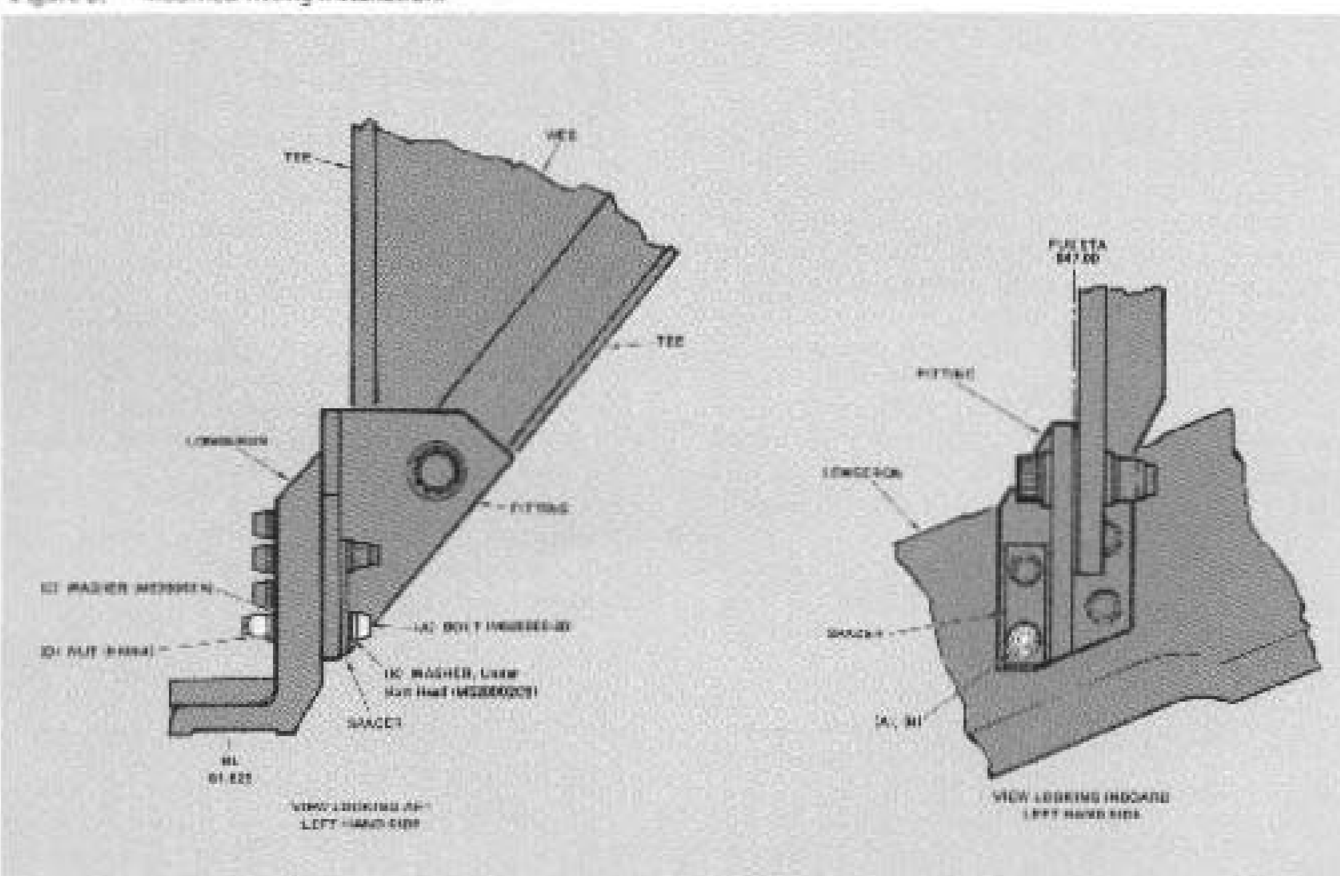
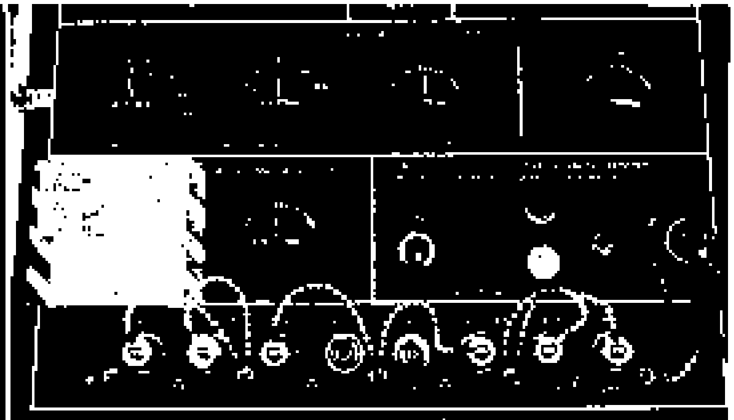


Figure 2. Modified fitting installation.





Testing the
GTF-6 Test Set

Over the years, malfunctioning test sets have been suspected of causing numerous aircraft ground accidents. A situation that came to light while a GTF-6 capacitance-type liquid quantity system test set was being used at Lockheed-Georgia Company is a case in point, and clearly demonstrates the need for proper care and inspection of test equipment.

During a fuel quantity indicating system engineering evaluation which was being carried out in conjunction with C-5 wing modification activities, a GTF-6 test set was brought to the flight station and connected to the aircraft's 115~volt AC power. Prior to hookup with the fuel quantity indicating system circuits in the tanks, and before ground wires were connected to the set, the electric potential difference between the case of the set and the aircraft structure was measured. With the AC power connected and the test set power switch on, a reading of 8 volts AC was measured; with the switch off, the potential was 6 volts AC. When the power plug was disconnected, the voltage reading dropped to zero.

An inspection of the interior of the GTF-6 unit revealed approximately l/4-inch of water in the case, and many areas of corrosion were found on and between conducting strips on the printed circuit boards.

As a result of examples like this, a test procedure has been developed to ensure that excessive current levels are not being applied through the GTF-6's aircraft tank unit connectors. For your information, here is the suggested test, which is intended to be performed in the shop on a periodic basis (once a month, initially).

With the GTF-6 test set connected to 115 volt AC power, and with the switch on, measure the AC current between the points indicated under the conditions described in items A and B below.

A. With the DISPLAY SELECT switch set to CAP, and as the CAPACITANCE FUNCTION switch is switched through all positions, measure the current between the following points on the AIRCRAFT TANK UNITS connectors:

1. COAX A shell and POWER GND
2. COAX A center conductor and POWER GND
3. UNSH B center conductor and POWER GND
4. COMP C center conductor and POWER GND
5. COAX A center conductor and COAX A shell
6. UNSH B center conductor and COAX A shell
7. COMP C center conductor and COAX A shelf
8. COAX A center conductor and UNSH B center conductor
9. COAX A center conductor and COMP C center conductor
10. UNSH B center conductor and COMP C center conductor

B. With the DISPLAY SELECT switch set to RES, and as the RESISTANCE FUNCTION switch is switched through all positions, measure the current between the same ten sets of points as in A, above.

C. Any current reading above 20 milliamps indicates a defective test unit, and it shall not be used until the fault has been corrected.

Lockheed Safety Engineering urges that full consideration be given to adopting a policy of regularly checking all GTF-6 test sets according to this procedure. It only requires a few minutes to carry out the test, but like many other valuable safety precautions, it is a step that can forge an important link in the chain of sound operational and maintenance practices that helps a truly professional organization protect both lives and property.

NEW Ice Detectors
for Hercules Aircraft

by R. S. Johnston, Staff *Engineer*

To avoid the problems that could result from the formation of ice on critical structures, Hercules aircraft are equipped with an electrical anti-icing and de-icing system for the propellers and a bleed air anti-icing system for the engine inlet, wing, and empennage areas. The engine, propeller, and radome systems are automatically activated by an ice detector which is located in the right forward side of the engine nacelle and has a probe that protrudes into the engine inlet housing (see Figure 1). Each engine is equipped with one of these units, but only the ones on engines 2 and 3 are wired into the anti-icing systems. The units on engines 1 and 4 are mounted for the purpose of interchangeability of the QEC.

Figure 1. **View of engine air inlet and installed DNE ice detector probe.**

A new design in ice detectors is bringing some improvements to the process. Cook Electric, the company which has supplied ice detectors for a number of years, has stopped making them. A new vendor, Dataproducts New England (DNE), has been selected. The DNE ice detectors are of an updated and improved design.

The new unit is interchangeable with the old one (see Figure 2). The most obvious difference between the two systems is that the Cook unit uses tiny holes in the probe to detect the formation of ice, whereas the new DNE unit uses a small platinum wire wrapped around the probe.

The Cook unit depends on changes in air pressure to initiate operation. Its probe directs air into a bellows. The detector is armed when air pressure equivalent to 40 knots pushes the bellows past the arming contact point. When the pressure drops because of icing, which stops up the tiny holes and thereby reduces the inside pressure, the bellows comes down past the arming point and then relays the message that starts the de-icing and anti-icing processes. Unfortunately, dirt, oil, mud and other contaminants may also clog the small holes. This can start the anti-icing systems when they are not needed.

The DNE ice detector works differently in that its vehicle of operation involves the physical and chemoelectrical properties of ice. Exactly how it detects the presence of ice is proprietary information retained by DNE. Like many other solid state "black box" items, when there is a problem with it, it is simply replaced. Since the new detector has no moving parts, it is expected to far outlast the Cook unit, provided care is taken when handling the probe.

Figure 2. New DNE unit, below, left, is interchangeable with the Cook unit shown below, right.

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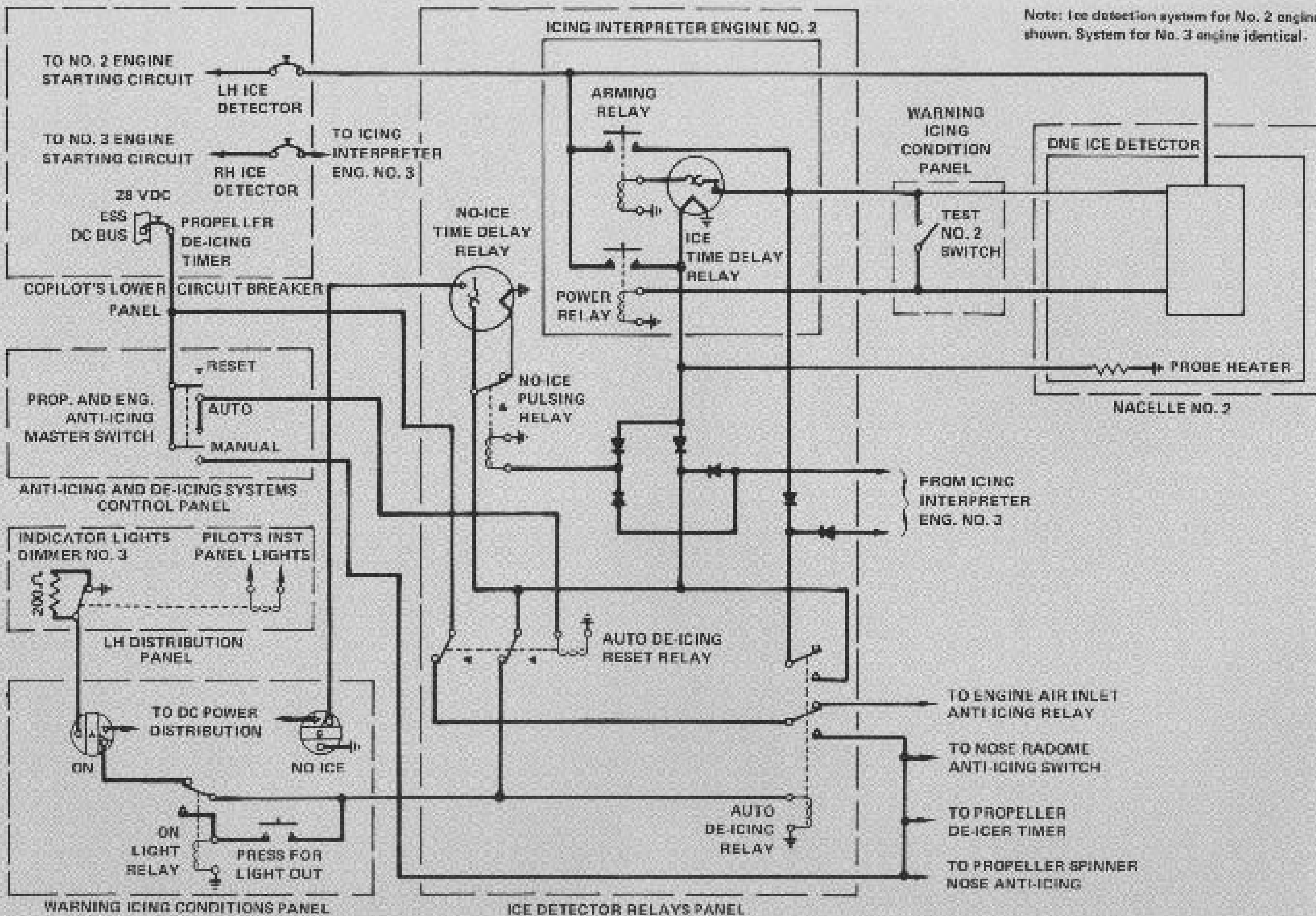


ICE DETECTION SYSTEM ELECTRICAL SCHEMATIC

CLEANING

The only maintenance required on the DNE unit is the occasional cleaning of the probe which is necessary for optimal efficiency. It is very important, however, that the cleaning be done properly.

The exposed windings of the new ice detector units are made of 0.004" diameter platinum wire. This very thin wire must be treated with care to avoid damage. Do not use soft cotton cloths to clean the probes, and in particular do not attempt to clean a probe by gripping it with a cloth rag and twisting. The loose fibers in the rag are likely to snag the thin wire and cause it to stretch and break. The only acceptable method of cleaning the probe is to spray it with Stoddard solvent and blot it dry with a paper towel.

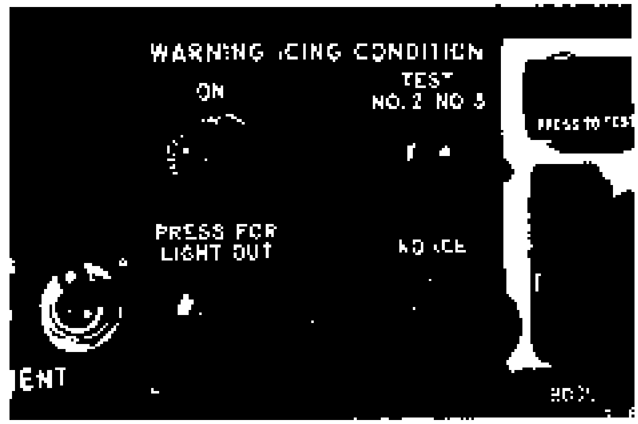
Note that it is also possible to damage the probes inadvertently in the process of carrying out other cleaning activities. The windings can be degraded by high pressure water jets during nacelle cleaning, or when walnut shelling is used to clean the engine compressor. If either operation is necessary, the ice detector should be removed first.

If a break has in fact occurred in a detector probe winding, the defect will make itself known to an alert observer because the system will malfunction in a characteristic way. With the circuit breakers closed and power available to the system, the amber ON light on the icing condition warning panel will illuminate when the condition lever for the affected engine (No. 2 or No. 3) is moved to RUN – an abnormal indication. The light will remain illuminated approximately 20 seconds. It cannot be made to extinguish when the PROP & ENG ANTI-ICING MASTER switch on the anti-icing control panel is reset, except when the switch is actually held in the RESET position. After about 20 seconds, the amber light will extinguish by itself; it then cannot be made to illuminate when an attempt is made to check the defective system by using the TEST switch on icing condition warning panel.

TESTING

Electrical shorts and other conditions that affect solid state units can cause the new detector to become ineffective; therefore, testing the unit should be part of the preflight inspection. On the Cook unit, it is necessary during ground tests to use an air hose to give enough pressure to arm the detector so it can be tested. With the DNE unit, this is not necessary. The following procedure is used to test the DNE detector:

1. Move the TEST switch on the icing condition warning panel to the No. 2 position to test the detector in the No. 2 engine inlet housing. Hold it down 4 to 5 seconds. The amber ON light should immediately illuminate when you turn on the switch. If the ON light has already illuminated



Warning icing condition panel.

before you placed the TEST switch in the No. 2 position, you have a failure in the system.

2. After releasing the TEST switch, push the PROP and ENG ANTI-ICING MASTER switch on the anti-icing control panel to the RESET position; then quickly release it. The amber light should wink off then come back on. Try this several times. It should not go out and stay out until 10 seconds after you have released the TEST switch.

3. While you are testing the unit with the panel indicators, someone should be positioned near the No. 2 engine inlet housing and put his hand close to the probe to determine if it is getting hot. If the probe is working properly, it should get hot (about 150'F) while the amber light is on.

CAUTION

If the test switch is actuated too many times or held too long, the heater element can overheat and melt the solder in the detector. The test switch should be held to TEST a maximum of 5 seconds. The test cycle lasts a maximum of 12 seconds. This sequence may be repeated once, but then wait 5 minutes for the probe to cool before testing the unit again.

Use the same steps to test the ice detector in the No. 3 engine.

DNE is also making a replacement for the interpreter unit in the ice detection system. Both the DNE ice detector and the DNE interpreter are compatible and interchangeable with the units now on the aircraft. The DNE units are the only authorized replacements for these items once the supply of Cook units is exhausted.

New DNE ice detector units are shipped with a plastic cap over the probe to protect the winding. Be sure to remove the cap carefully before installing one of these units on your aircraft.

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CUSTOMER SERVICE
LOCKHEED GEORGIA COMPANY
A DIVISION OF LOCKHEED CORPORATION
MARIETTA, GEORGIA 30063

The first Lockheed Aircraft Service Company (LAS) – modified C-130 Hercules Airborne Emergency Hospital is shown taking off on its delivery flight to the Royal Saudi Air Force. This modified Hercules aircraft is designed to handle medical emergencies that result from natural disasters, and can be used for citizens living in remote, outlying areas, where medical facilities are rare.