A-10A

CLOSE-SUPPORT ATTACK AIRCRAFT

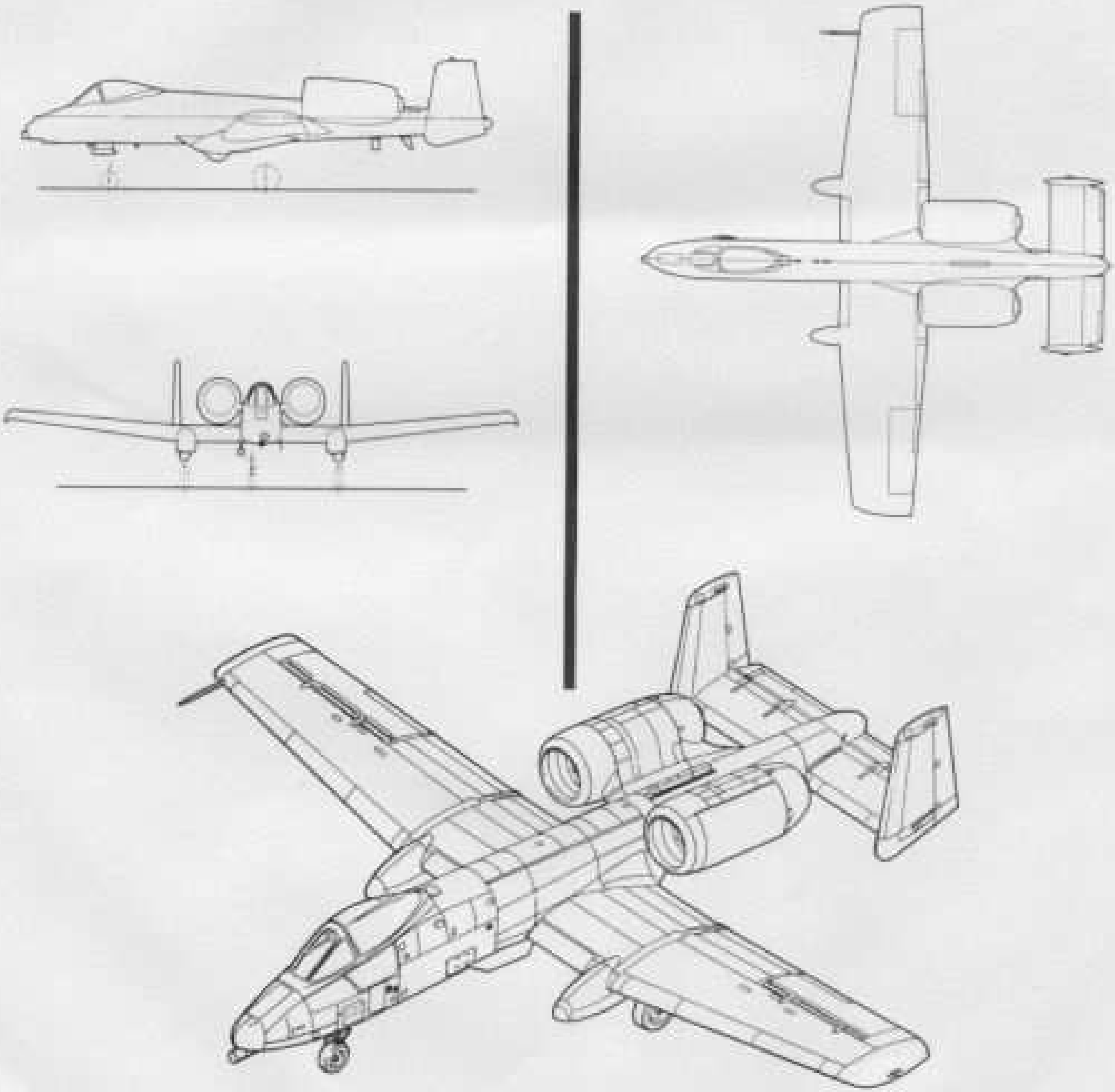
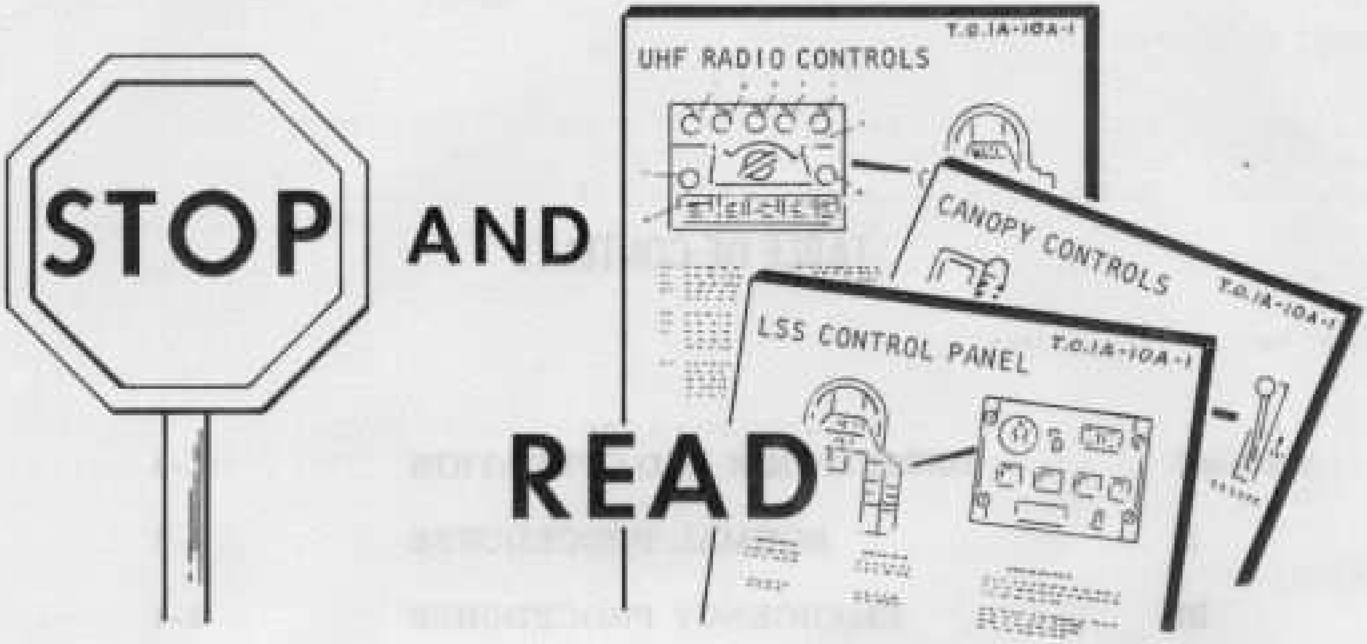


Figure 1-1



SCOPE

This manual contains the necessary information for safe and efficient operation of your aircraft. These instructions provide you with a general knowledge of the aircraft and its characteristics and specific normal and emergency operating procedures. Your experience is recognized; therefore, basic flight principles are avoided. Instructions in this manual are for a crew inexperienced in the operation of this aircraft. This manual provides the best possible operating instructions under most circumstances. Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures.

PERMISSIBLE OPERATIONS

The flight manual takes a "positive approach" and normally states only what you can do. Unusual operations or configurations are prohibited unless specifically covered herein. Clearance must be obtained before any questionable operation, which is not specifically permitted in this manual, is attempted.

HOW TO BE ASSURED OF HAVING LATEST DATA

Refer to T.O. 0-1-1-5 for a listing of all current flight manuals, safety supplements, operational supplements, and checklists. Also, check the flight manual cover page, the title block of each safety and operational supplement, and all status pages contained in the flight manual or attached to formal safety and operational supplements. Clear up all discrepancies before flight.

ARRANGEMENT

The manual is divided into seven independent sections to simplify reading it straight through or using it as a reference manual.

SAFETY SUPPLEMENTS

Information involving safety will be promptly forwarded to you in a safety supplement. Supplements covering loss of life will get to you within 48 hours by teletype, and supplements covering serious damage to equipment within 10 days by mail. The cover page of the flight manual and the title block of each safety supplement should be checked to determine the effect they may have on existing supplements.

OPERATIONAL SUPPLEMENTS

Information involving changes to operating procedures will be forwarded to you by operational supplements. The procedure for handling operational supplements is the same as for safety supplements.

CHECKLISTS

The flight manual contains itemized procedures with necessary amplifications. The checklist contains itemized procedures without the amplification. Primary line items in the flight manual and checklist are identical. If a formal safety or operational supplement affects your checklist, the affected checklist page will be attached to the supplement. Cut it out and insert it over the affected page but never discard the checklist page in case the supplement is rescinded and the page is needed.

HOW TO GET PERSONAL COPIES

Each flight crew member is entitled to personal copies of the flight manual, safety supplements, operational supplements, and checklists. The required quantities should be ordered before you need them to insure their prompt receipt. Check with your publication distribution officer - it is his job to fulfill your T.O. requests. Basically, you must order the required quantities on the appropriate Numerical Index and Requirement Table (NIRT). T.O. 00-5-1 and 00-5-2 give detailed information for properly ordering these publications. Make sure a system is established at your base to deliver these publications to the flight crew immediately upon receipt.

FLIGHT MANUAL BINDERS

Looseleaf binders and sectionalized tabs are available for use with your manual. They are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Group 75, Office Supplies, Part 1). Check with your supply personnel for assistance in procuring these items.

DEFINITION OF WORDS "SHALL," "WILL," "SHOULD" AND "MAY"

The words "shall" and "will" indicate a mandatory requirement. The word "should" indicates a nonmandatory desire or preferred method of accomplishment. The word "may" indicates an acceptable or suggested means of accomplishment.

WARNINGS, CAUTIONS AND NOTES

The following definitions apply to "Warnings", "Cautions" and "Notes" found throughout the manual.

WARNING

Operating procedures, techniques, etc., which will result in personal injury or loss of life if not carefully followed.

CAUTION

Operating procedures, techniques, etc., which will result in damage to equipment if not carefully followed.

Note

An operating procedure, technique, etc., which is considered essential to emphasize.

YOUR RESPONSIBILITY—TO LET US KNOW

Every effort is made to keep the flight manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. We cannot correct an error unless we know of its existence. In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the flight manual program are welcomed. These should be forwarded through your command channels on AF Form 847 to: Sacramento ALC/MMSRBD McClellan AFB, California, 95652 and a copy to ASD/TAXAT Wright Patterson AFB, Ohio 45433, for engineering review.

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SECTION I

DESCRIPTION AND OPERATION

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THE AIRCRAFT

The A-10A is a single-place close-air support aircraft (figures 1-1 and 1-2) manufactured by Fairchild Republic Company, Farmingdale, New York. The aircraft is a low wing, low tail configuration with two high bypass turbo fan engines installed in nacelles mounted on pylons extending from the aft fuselage. Twin vertical stabilizers are mounted on the outboard tips of the horizontal tail. The tricycle forward retracting landing gear is equipped with an antiskid system and a steerable nosewheel. The nose gear is installed to the right of the aircraft centerline to permit near centerline gunfire. The nose gear retracts fully into the fuselage while the main gears partially retract into streamlined pods in the wings. A titanium armor installation completely surrounds the cockpit below the longerons to protect the pilot and the central controls against penetration by armor piercing projectiles. The primary flight controls, consisting of elevators, ailerons and rudders are equipped with artificial feel devices to simulate aerodynamic feel for the pilot. The elevator and aileron controls split into redundant separate systems before leaving the armor protection. The controls are powered by two independent hydraulic systems. Either system has the capability of controlling the airplane. If both hydraulic systems fail the airplane can be flown using a manual reversion system. The ailerons consists of an upper and lower panel which become speed brakes when opened. The windshield front panel is bullet resistant for protection against small arms fire and bird-resistant

GENERAL ARRANGEMENT DIAGRAM

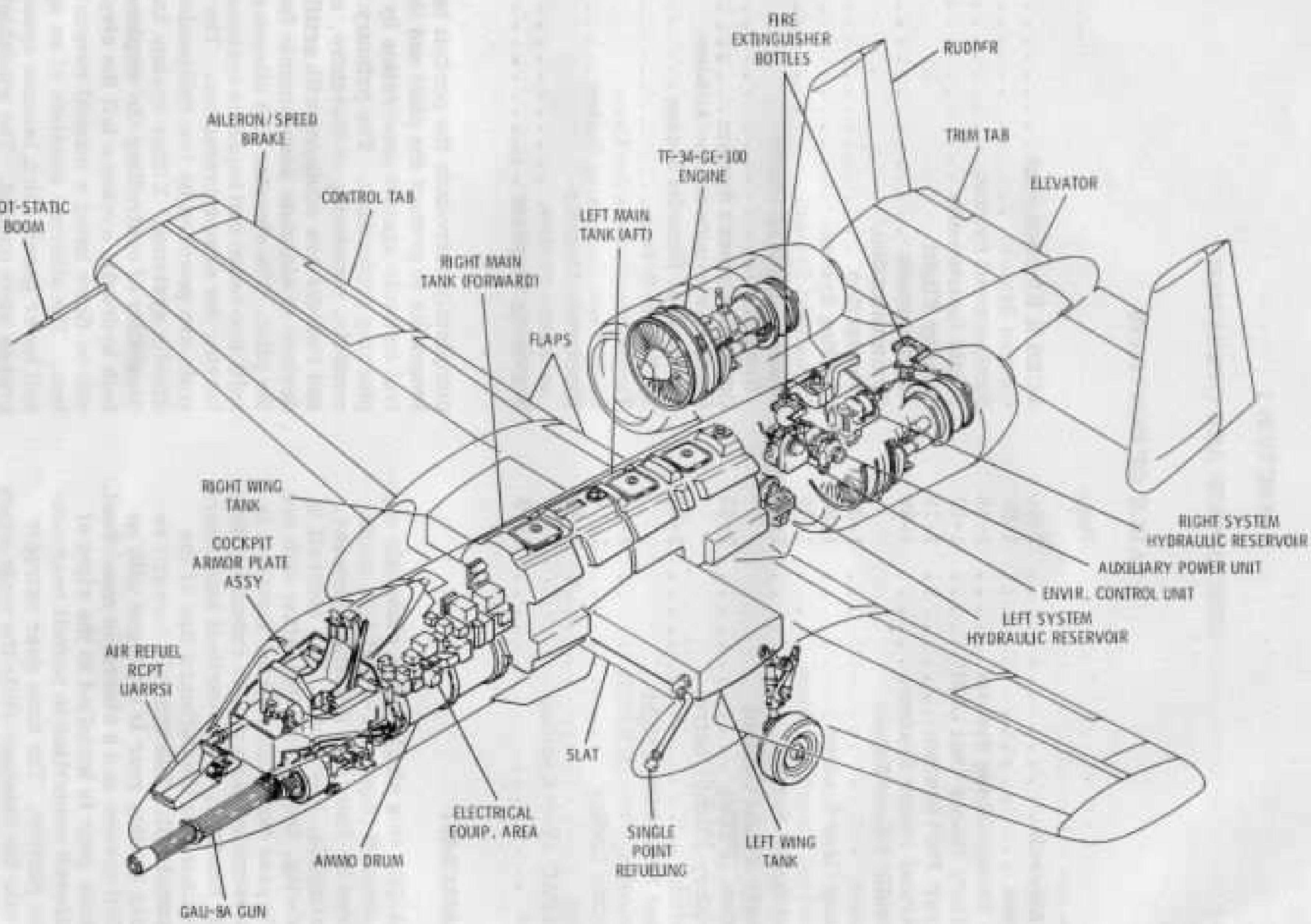


Figure 1-2

against medium sized birds at operational speeds. The windshield side panels are spall resistant for protection from spall spray caused by penetrations. The fuselage fuel cell sumps are self-sealing on the lower portion and tear resistant on the upper portion. The cells are filled with a flexible foam to prevent fuel tank explosion. Single point ground refueling and engine feed lines are self-sealing to prevent leaks due to combat damage. The escape system provides a zero/zero capability (zero velocity and zero pitch and roll attitude) either with the canopy removed or through the canopy. The armament system includes a high fire rate 30mm seven barrel gun with ammunition stored in a drum. A variety of stores are carried on eleven pylons, four on each wing and three on the fuselage.

AIRCRAFT DIMENSIONS

The overall dimensions of the aircraft under normal conditions of gross weight, tire and strut inflation are as follows:

Overall length	52 ft. 7 in.
	53 ft. 4 in. with gun and tail light
Wing span	57 ft. 6 in.
Horizontal tail span	18 ft. 10 in.
Height to top of fin	14 ft. 8 in.
Wheel base	17 ft. 9 in.
Wheel tread	17 ft. 3 in.

Refer to Section II for minimum turning radius and ground clearance dimensions.

AIRCRAFT GROSS WEIGHT

Refer to Section V for gross weight limitations and to the most recent copy of the weight and balance clearance Form 365F and to the aircraft's T.O. 1-1B-40, Chart C for the exact weight for the aircraft to be flown. The following are typical weights not to be used for computing aircraft performance.

Operating weight (includes pilot, gun, 11 empty pylons, oil and unusable fuel)	24,513 lbs.
Max. T/O weight (1350 rds ammo, 18 MK-82 bombs, full internal fuel)	50,000 lbs.

ENGINES

The aircraft is powered by two General Electric TF34-GE-100 engines (figure 1-3). Sea level, standard day, static thrust for an uninstalled engine is 9065 pounds (approximately 8900 pounds installed) at maximum thrust. The engine incorporates a single stage bypass fan rotor and a 14 stage axial flow compressor rotor. The bypass air produces over 85 percent of the engine thrust. The inlet guide vanes are variable and are automatically modulated throughout the engine operating range. An accessory gear box drives a hydraulic pump, fuel pump and fuel control, engine starter, oil pump and an electric generator. An air bleed for aircraft systems is provided. Engine acceleration time from IDLE to MAX thrust will be approximately 10 seconds at sea level. Engine thrust droop results from differential expansion of the engine turbines and casings during transients from low to high thrust operation. The duration and extent of the thrust droop is dependent upon the applied engine thrust excursion. Thrust droop is decreased if the engines have been idling for a period of time. Thrust droop is further decreased if the engines have been run up before takeoff. An example of the worst condition would be a scramble takeoff where takeoff is accomplished shortly after engine start. Maximum droop occurs approximately 10 seconds after the throttle is advanced from IDLE to MAX. After approximately 4 minutes of operation at MAX thrust, power output returns to normal.

Aircraft prior to serno 75-00280 not modified by T.O. 1A-10-508 and T.O. 1A-10-764 contain an automatic engine starting system. This system only

TF-34 ENGINE

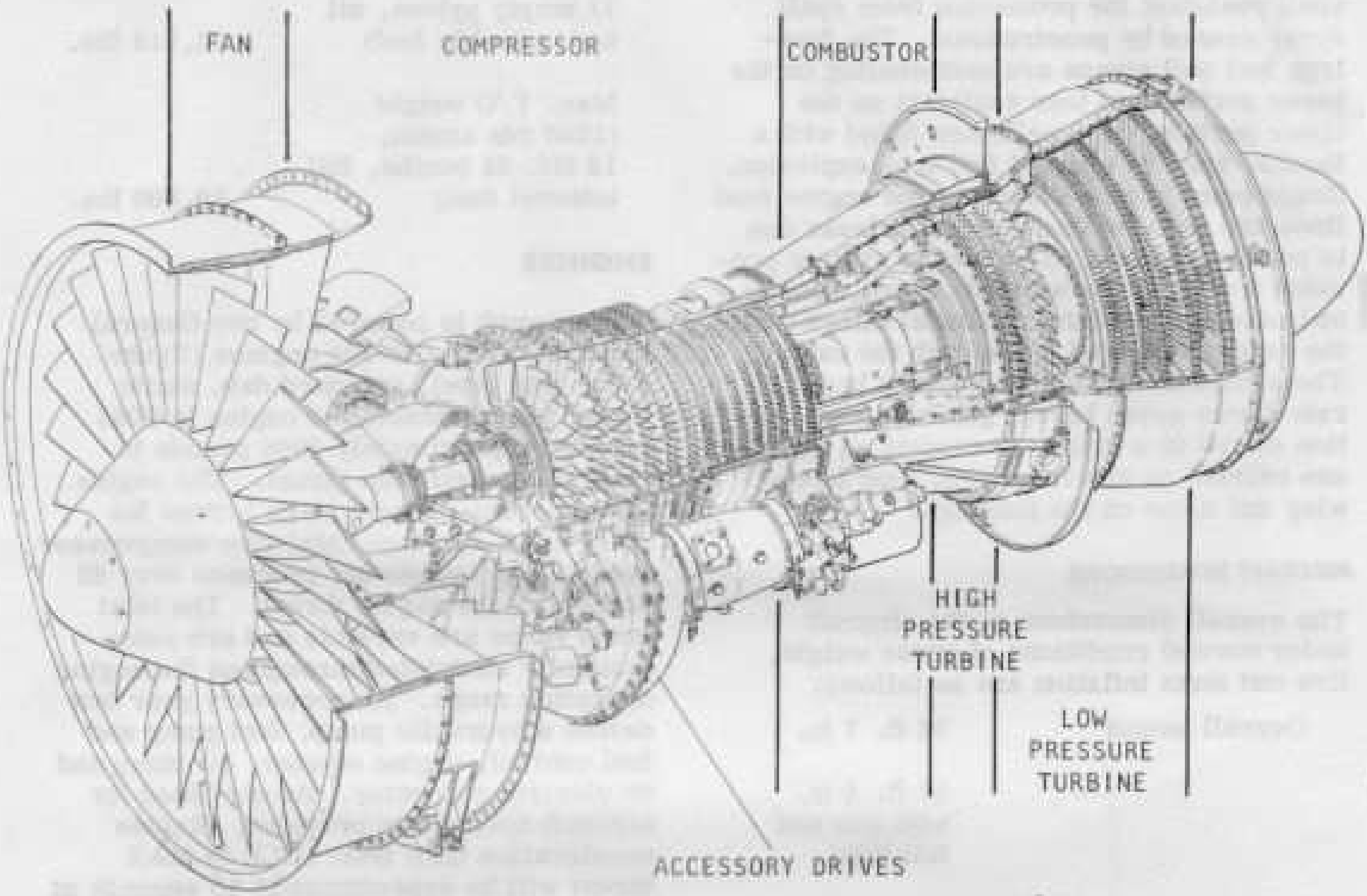


Figure 1-3

requires the pilot to advance the throttle to the IDLE position. The start cycle will automatically terminate at approximately 56 percent RPM core speed. A backup starting system is provided by depressing the ignition button on the throttle and placing the engine operate switch to the MOTOR position.

Aircraft sernos 75-00280 through 77-0226 not modified by T.O. 1A-10-764 and those modified by T.O. 1A-10-508 contain a manual engine starting system. This system requires the pilot to place the engine operate switch to the MOTOR position, depress the engine ignition button and then advance the throttle to the IDLE position. The pilot must terminate motoring by returning the engine operate switch to the NORM position approximately 10 seconds after engine reaches 56 percent RPM core speed.

Aircraft serno 77-0227 and subsequent and those modified by T.O. 1A-10-764 contain an automatic engine starting system. This system only requires the pilot to advance the throttle to the IDLE position. The start cycle will automatically terminate

Note

Because of tolerances on the fuel flow schedule of the TF34-GE-100 engine fuel control, certain engines may exhibit a characteristic of slow or limited acceleration from low power at high altitude (usually above 25,000 feet). The condition can result in either a long time period elapsing (over one minute) to accelerate an engine(s) to maximum power parameters from a low power setting or of not reaching maximum power parameters unless a descent to lower altitude is made. In no case will any overtemperature or overspeed occur nor any engine damage be sustained. This condition will not cause a power loss but can retard accelerations. If this condition occurs, write up on AFTO Form 781.

Engine Fuel Flow Indicators

A fuel flow indicator (43, figure FO-1) is provided for each engine. The indicators are placarded FUEL FLOW PPH X 100. During engine start, fuel flow will be indicated when the throttle is positioned from OFF to IDLE. The indicators are powered by the right AC system bus.

Engine Fuel Flow Switches

Two engine fuel flow switches (figure 1-4), one for each engine, are located on the engine control panel. These switches are placarded ENG FUEL FLOW L and R and each switch has two positions, placarded NORM and OVERRIDE. With the switch in the NORM position the engine fuel flow is scheduled on the basis of throttle position and ITT limiter. In the event of an ITT limiter failure, as noted by an abnormal ITT or when unstable engine operation is observed at MAX power, the temperature control system can be deactivated by placing the appropriate switch in the OVERRIDE position. When this is done, the engine will be speed controlled by the throttle position alone. However, undertemperature operation may be observed as a result of the engine achieving maximum available power at less than limiting ITT under low ambient temperature, high speed, high altitude operation and is normal. Selection of OVERRIDE in this condition will produce no change in engine operation and the switch should be returned to NORM immediately. The engine fuel flow switches are powered by the auxiliary AC essential bus.

CAUTION

Prior to selecting the OVERRIDE position, retard throttle until core speed starts to decrease. This will prevent a possible overtemperature condition since the temperature may rise significantly when the OVERRIDE position is selected.

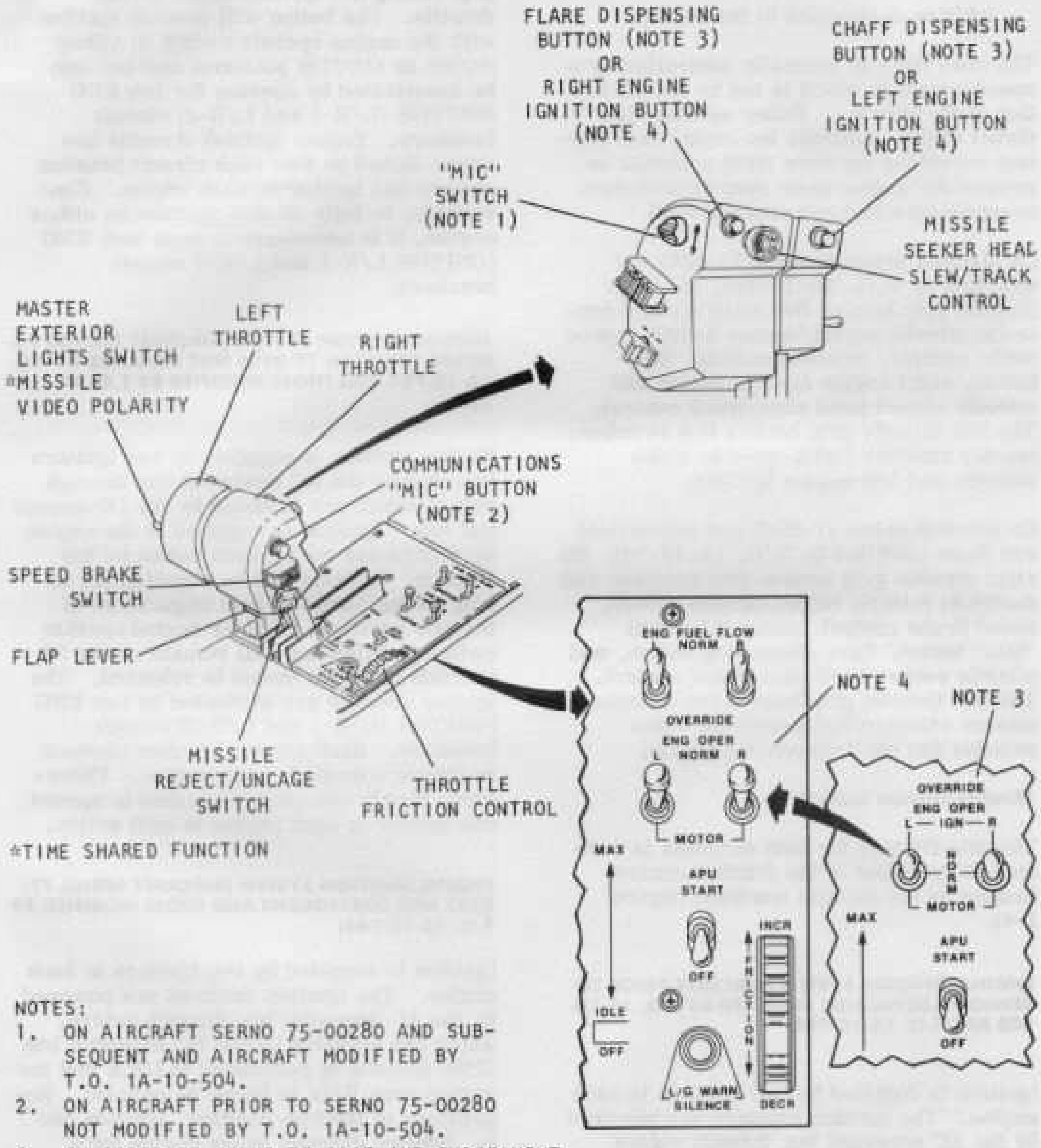
THROTTLES

A mechanical throttle control system (figure 1-4) controls the operation of each engine. Each throttle has three positive stop positions placarded OFF, IDLE and MAX. The throttles move from OFF through IDLE range to MAX position with 50 degree travel. To move from OFF to IDLE the throttle is raised and moved forward to the first stop position. To move to the OFF position the throttle is retarded to the IDLE stop, then raised and moved aft to OFF. The DC fuel pump is energized when either throttle is positioned to IDLE or above and there is no pressure from left main tank boost pump. On aircraft prior to serno 75-00280 not modified by T.O. 1A-10-508, aircraft serno 77-0227 and subsequent and those modified by T.O. 1A-10-764, when the throttle is at the IDLE stop, the following actions take place if core engine RPM is below 54 percent.

- Air turbine starter control valve opens and the ENG START CYCLE light comes on.

- Engine bleed air shutoff valve opens.

THROTTLE QUADRANT INCLUDING ENGINE CONTROL PANEL



NOTES:
1. ON AIRCRAFT SERNO 75-00280 AND SUBSEQUENT AND AIRCRAFT MODIFIED BY T.O. 1A-10-504.
2. ON AIRCRAFT PRIOR TO SERNO 75-00280 NOT MODIFIED BY T.O. 1A-10-504.
3. ON AIRCRAFT SERNO 77-0227 AND SUBSEQUENT AND AIRCRAFT MODIFIED BY T.O. 1A-10-764.
4. ON AIRCRAFT PRIOR TO SERNO 77-0227 NOT MODIFIED BY T.O. 1A-10-764.

Figure 1-4

- Environment control system shutoff valve closes.

- Ignition is supplied to the engine.

The core RPM is normally controlled by a speed governor which is set by the position of the throttle. Under certain high thrust flight conditions the engine fuel control overrides the core RPM schedule to protect the engine from overtemperature, overpressure and compressor stall.

On aircraft prior to serno 77-0227 not modified by T.O. 1A-10-764, the right throttle grip houses five switches and controls; missile reject/uncage switch, speed brake control, communications "Mic" button, right engine ignition button and missile seeker head slew/track control. The left throttle grip houses two switches; master exterior lights-missile video polarity and left engine ignition.

On aircraft serno 77-0227 and subsequent and those modified by T.O. 1A-10-764, the right throttle grip houses five switches and controls; missile reject/uncage switch, speed brake control, communications "Mic" button, flare dispensing button, and missile seeker head slew/track control. The left throttle grip houses two switches; master exterior lights-missile video polarity and chaff dispensing button.

Throttle Friction Control

Throttle friction for both throttles is controlled by means of the friction control located on the throttle quadrant (figure 1-4).

ENGINE IGNITION SYSTEM (AIRCRAFT PRIOR TO SERNO 75-00280 NOT MODIFIED BY T.O. 1A-10-508 AND T.O. 1A-10-764)

Ignition is supplied by two ignitors in each engine. The ignition exciters are powered by the AC essential bus through relays which are actuated by the DC essential bus if the throttle is positioned to IDLE and the engine core RPM is below 54 percent. Regardless of throttle position, ignition can be supplied directly to the engine for a minimum of 30 seconds by momentarily depressing the ignition button on the throttle. The button will provide ignition with the engine operate switch in either NORM or MOTOR positions and can only be deactivated by opening the two ENG IGNITOR (L/R-1 and L/R-2) circuit breakers. Engine ignition circuits are cross-linked so that each circuit breaker powers one ignitor on each engine. Conversely, to fully disable ignition on either engine, it is necessary to open both ENG IGNITOR L/R-1 and L/R-2 circuit breakers.

ENGINE IGNITION SYSTEM (AIRCRAFT SERNOS 75-00280 THROUGH 77-0226 NOT MODIFIED BY T.O. 1A-10-764 AND THOSE MODIFIED BY T.O. 1A-10-508)

Engine ignition is supplied by two ignitors powered by the AC essential bus through relays which are actuated by the DC essential bus. Ignition is supplied to the engine by depressing the ignition button on the throttle. Ignition will be supplied for as long as the button is held regardless of throttle position, RPM or engine operate switch position and will remain on for 30 seconds after the button is released. The ignitor circuits are protected by two ENG IGNITOR (L/R-1 and L/R-2) circuit breakers. Each circuit breaker protects an ignitor circuit in each engine. Therefore, if only one circuit breaker is opened one ignitor in each engine is still active.

ENGINE IGNITION SYSTEM (AIRCRAFT SERNO 77-0227 AND SUBSEQUENT AND THOSE MODIFIED BY T.O. 1A-10-764)

Ignition is supplied by two ignitors in each engine. The ignition exciters are powered by the AC essential bus through relays, which are actuated by the DC essential bus if the throttle is positioned to IDLE and the engine core RPM is below 54 percent. Regardless of throttle position, ignition can be supplied directly to the engine for a minimum of 30 seconds by momentarily placing the engine operate switch to the IGN position. Engine ignition circuits are protected by the ENG IGNITOR L/R-1 and

L/R-2 circuit breakers and are cross-linked so that each circuit breaker powers one ignitor on each engine. Conversely, to fully disable ignition on either engine, it is necessary to open both ENG IGNITOR L/R-1 and L/R-2 circuit breakers.

Engine Ignition Button (Aircraft Prior to Serno 77-0227 not Modified by T.O. 1A-10-764)

An engine ignition button (figure 1-4) is located on the forward side of each throttle. Momentarily depressing the button on either throttle will supply ignition to the corresponding engine for 30 seconds regardless of the throttle position or engine RPM. The left and right engine ignition buttons are powered by the DC essential bus through the L ENG START and R ENG START circuit breakers. The engine ignitors are powered by the AC essential bus through a relay which is actuated by DC essential bus power.

ENGINE BLEED AIR SYSTEM

Each engine has an air bleed port. The air from each engine, from the APU compressor and from a ground receptacle are ported to a common manifold (figure 1-29). A temperature-sensitive conductor is routed adjacent to the manifold to indicate a bleed air leak by sensing high bleed air temperature. The bleed air supply system furnishes air for the following systems:

- Engine starter system.
- Cockpit environment control system.
- Rain removal system.
- Canopy de-fog system.
- Canopy seal.
- Anti-G suit.
- External tank pressurization.
- Air refuel line purge system.
- Gun purge system.

Each of the above systems in described in detail under the respective sections. Bleed air is controlled by a shutoff valve adjacent to each engine. Both valves are opened or closed simultaneously by the bleed air switch.

Bleed Air Switch

The bleed air switch (figure 1-30) on the environment panel, is a two-position switch with positions placarded BLEED AIR and OFF. The BLEED AIR position opens both bleed air valves so that the bleed air from the compressor of each engine will be supplied to the systems requiring air. The OFF position closes the valves except during engine start. The switch must be raised to move it from one position to the other. Positioning the bleed air switch from OFF to BLEED AIR will provide bleed air to the systems requiring air within approximately 7 seconds. The switch is powered by the DC essential bus.

Fire Detect/Bleed Air Leak Test Button

The fire detect/bleed air leak test button (figure 1-51) is a push-to-test button placarded FIRE DETECT BLEED AIR LEAK TEST. Depressing the switch checks the integrity of bleed air sensors, fire detection sensors and associated warning lights. If the circuit is intact, the BLEED AIR LEAK caution light on the caution light panel and the MASTER CAUTION, FIRE (L ENG) PULL, FIRE (R ENG) PULL, FIRE (APU) PULL lights will come on. The test button is powered by the auxiliary DC essential bus.

Bleed Air Leak Caution Light

The bleed air lines upstream from the precooler are monitored by a leak detection system using temperature as the parameter for detecting excessive bleed air leakage. A continuous loop of heat-sensitive elements on the bleed air manifold is connected to a control unit. Upon sensing a temperature of 400°(F) or more, the system responds by activating the

BLEED AIR LEAK caution light on the caution light panel (figure 1-51).

ENGINE START SYSTEM (AIRCRAFT PRIOR TO SERNO 75-00280 NOT MODIFIED BY T.O. 1A-10-508 AND T.O. 1A-10-764)

Engine starts require low pressure air to power the air turbine starter (ATS) unit mounted on the engine. Air may be obtained from the following sources:

- Auxiliary power unit (APU).
- Crossbleed air from an operating engine.
- External pneumatic power unit (ground start).

Air from any of these sources (figure 1-5) is ducted through the bleed air shutoff valves to the air turbine starter control valves. The air turbine starter control valve opens automatically when the applicable throttle is moved to the IDLE position or when the engine is being motored by use of the engine operate switch. The electrical circuits controlling the two air turbine starter control valves are interlocked to preclude both valves being open concurrently, as insufficient air pressure is available to start both engines simultaneously. The air turbine starter control valve closes automatically during the start cycle when the engine reaches 54 percent RPM core speed. Engine ignitor firing and fuel flow scheduling is initiated when the applicable throttle is moved to IDLE regardless of engine operate switch position. Simultaneously, the aircraft environment control system is shut off to eliminate bleed air drain during the engine start cycle.

Note

If the right engine is started first, using the battery and inverter for electrical power, the main boost pumps will be inoperative. The crossfeed switch should be positioned to CROSSFEED to allow the DC boost pump located in the left main tank to supply fuel to the right engine. Brakes will be available after engine start if the emergency brake handle is pulled prior to starting. The handle should be pushed in after the left engine is started.

The left engine is normally started first. Normal brakes will be available after the left engine is started. Electrical power for starting the engines may be obtained from an external AC power unit, the aircraft battery and inverter or the APU generator.

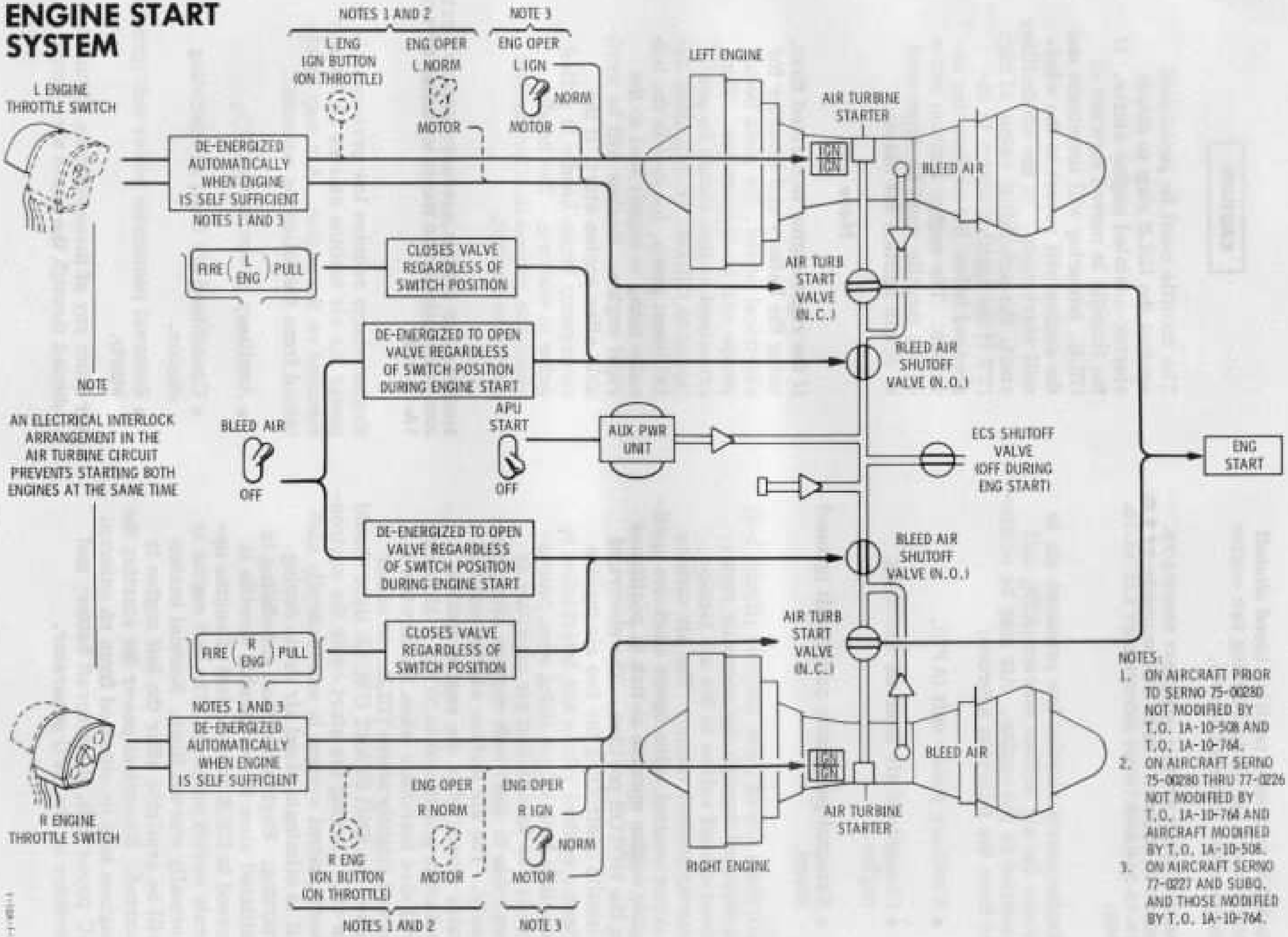
CAUTION

If both engines are inoperative and windmilling, selection of IDLE throttle position will open both bleed valves and the starter valve on one engine, allowing the faster starting engine to bleed into the open starter. This will lead to slow acceleration or prevent a successful start.

The throttle must be positioned against the IDLE stop to obtain starter-assisted engine starts. Assisted starts will be indicated by illumination of the ENG START CYCLE advisory light. If the throttle is moved forward of IDLE the bleed air to the starter will be shut off, ignition will terminate after 30 seconds, and the engine will revert to the windmill start mode. In the windmilling start, the possibility exists of high ITT temperature if the ignition button is depressed below the windmilling envelope. The engine ignition button on the throttle must be depressed to obtain ignition for windmill starts above IDLE RPM.

For windmilling starts, the bleed air switch should be positioned to OFF and the throttles advanced out of IDLE (preferably

ENGINE START SYSTEM



NOTES:
1. ON AIRCRAFT PRIOR TO SERNO 75-00280 NOT MODIFIED BY T.O. 1A-10-508 AND T.O. 1A-10-764.
2. ON AIRCRAFT SERNO 75-00280 THRU 77-0226 NOT MODIFIED BY T.O. 1A-10-764 AND AIRCRAFT MODIFIED BY T.O. 1A-10-508.
3. ON AIRCRAFT SERNO 77-0227 AND SUBQ. AND THOSE MODIFIED BY T.O. 1A-10-764.

Figure 1-5

to MAX) to insure positive bleed shutoff and starter air shutoff during two engine inoperative windmill starts.

ENGINE START SYSTEM (AIRCRAFT SERNOS 75-00280 THROUGH 77-0226 NOT MODIFIED BY T.O. 1A-10-764 AND THOSE MODIFIED BY T.O. 1A-10-508)

Engine starts require low pressure air to power the air turbine starter (ATS) unit mounted on the engine. Air may be obtained from the following sources:

- Auxiliary power unit (APU).
- Crossbleed air from an operating engine.
- External pneumatic power unit (ground start).

Air from any of these sources (figure 1-5) is directed through the normally open bleed shutoff valves to the air turbine starter control valves. The air turbine starter control valve opens when the applicable engine operate switch is positioned to the MOTOR position. The electrical circuits controlling the two air turbine starter control valves are interlocked to preclude both valves being open concurrently, as insufficient air pressure is available to start both engines simultaneously. The air turbine starter control valve closes when the engine operate switch is positioned to NORM. Motoring will also terminate when the throttle is moved slightly above IDLE. In either case, the ENG START CYCLE light should go off. During the start cycle the environment control system is automatically shut off to eliminate bleed air drain during starting. Engine fuel flow scheduling is initiated when the applicable throttle is moved to IDLE independent of engine operate switch position. The left engine is normally started first. Normal brakes will be available after the left engine is started. Electrical power for starting the engines may be obtained from an external AC power unit, the aircraft battery and inverter or the APU generator.

CAUTION

The throttle must be positioned against the IDLE stop to obtain starter-assisted engine starts. If the throttle is moved forward of IDLE, motoring will terminate and the engine will revert to the windmill start mode. In the windmilling start, the possibility exists of high ITT if the ignition button is depressed below the windmilling envelope. The engine ignition button on the throttle must be depressed to obtain ignition for starts.

Note

If the right engine is started first, using the battery and inverter for electrical power, the main boost pumps will be inoperative. The crossfeed switch should be positioned to CROSSFEED to allow the DC boost pump, located in the left main tank, to supply fuel to the right engine. Brakes will be available after engine start, if the emergency brake handle is pulled prior to starting. The handle should be pushed in after the left engine is started.

ENGINE START SYSTEM (AIRCRAFT SERNO 77-0227 AND SUBSEQUENT AND THOSE MODIFIED BY T.O. 1A-10-764)

Engine starts require low pressure air to power the air turbine starter (ATS) unit mounted on the engine. Air may be obtained from the following sources:

- Auxiliary power unit (APU).
- Crossbleed air from an operating engine.
- External pneumatic power unit (ground start).

Air from any of these sources (figure 1-5) is ducted through the bleed air shutoff valves to the air turbine starter control valves. The air turbine starter control valve opens automatically when the applicable throttle is moved to the IDLE position or when the engine is being manually started or motored by use of the engine operate switch. The electrical circuits controlling the two air turbine starter control valves are interlocked to preclude both valves being open concurrently, as insufficient air pressure is available to start both engines simultaneously. The air turbine starter control valve closes automatically during the start cycle approximately 10 seconds after the engine reaches 54 percent RPM core speed. Engine ignitor firing and fuel flow scheduling is initiated when the applicable throttle is moved to IDLE, regardless of engine operate switch position. Simultaneously, the aircraft environment control system is shut off to eliminate bleed air drain during the engine start cycle.

Note

If the right engine is started first, using the battery and inverter for electrical power, the main boost pumps will be inoperative. The crossfeed switch should be positioned to CROSSFEED to allow the DC boost pump located in the left main tank to supply fuel to the right engine. Brakes will be available after engine start if the emergency brake handle is pulled prior to starting. The handle should be pushed in after the left engine is started.

The left engine is normally started first. Normal brakes will be available after the left engine is started. Electrical power for starting the engines may be obtained from an external AC power unit, the aircraft battery and inverter or the APU generator.

CAUTION

If both engines are inoperative and windmilling, selection of IDLE throttle position will open both bleed valves and the starter valve on one engine, allowing the faster starting engine to bleed into the open starter. This will lead to slow acceleration or prevent a successful start.

- The throttle must be positioned against the IDLE stop to obtain starter-assisted engine starts. Assisted starts will be indicated by illumination of the ENG START CYCLE advisory light. If the throttle is moved forward of IDLE, the bleed air to the starter will be shut off, ignition will terminate after 30 seconds, and the engine will revert to the windmill start mode. In the windmilling start, the possibility exists of high ITT if the engine operate switch is placed to the IGN position below the windmilling envelope. The engine operate switch must be placed to the IGN position to obtain ignition for windmill starts above IDLE RPM.

- On aircraft sernos 77-0227 through 78-0633 not modified by T.O. 1A-10-898, the bleed air shutoff valve of the operating engine will not automatically open if the bleed air switch is in the OFF position. Therefore, the bleed air switch should be placed in the BLEED AIR position prior to initiating a crossbleed start.

For windmilling starts, the bleed air switch should be positioned to OFF and the throttles advanced out of IDLE (preferably to MAX) to insure positive bleed shutoff and starter air shutoff during two engine inoperative windmill starts.

Engine Operate Switches (Aircraft Prior to Serno 75-00280 Not Modified by T.O. 1A-10-508 and T.O. 1A-10-764)

Two engine operate switches (figure 1-4), one for each engine, are located on the engine control panel. These switches are placarded ENG OPER and each switch (L and R) has two positions placarded NORM and MOTOR. The switch must be raised to move out of the NORM position. The switch remains in the NORM position unless it is necessary to motor the engine. With the switch in the NORM position, the following is accomplished if the throttle is positioned to IDLE and the engine core RPM is below 54 percent.

- Opens the air turbine starter control valve.
- Closes the environment control system shutoff valve.
- Opens both engine bleed air shutoff valves.
- Supplies ignition to the engine.
- ENG START CYCLE light comes on.
- Supplies fuel to the engine.

The MOTOR position accomplishes all of the above with the exception of ignition and DC fuel pump with the throttle in the OFF position; however, ignition will be supplied to the engine when the throttle is positioned to IDLE or the ignition buttons are depressed. This position motors the engine for maintenance purposes or for air-purging the engine of excessive raw fuel or for cooling the engine. The engine operate switches are powered by the DC essential bus.

Note

To avoid nuisance bus cycling and flickering of the cockpit lights during engine motoring on aircraft prior to serno 75-00262 not modified by T.O. 1A-10-551, the AC generator switch should be positioned to OFF/RESET whenever the engine operate switches are in MOTOR position.

Engine Operate Switches (Aircraft Sernos 75-00280 through 77-0226 not Modified by T.O. 1A-10-764 and Those Modified by T.O. 1A-10-508)

Two engine operate switches (figure 1-4), one for each engine, are located on the engine control panel. These switches are placarded ENG OPER and each switch (L and R) has two positions placarded NORM and MOTOR. The switch remains in the NORM position unless it is necessary to start or motor the engine.

The MOTOR position accomplishes the following:

- Opens the air turbine starter control valve.
- Closes the environment control system shutoff valve.
- Opens both engine bleed air shutoff valves.
- ENG START CYCLE light comes on.

Engine Operate Switches (Aircraft Serno 77-0227 and Subsequent and Those Modified by T.O. 1A-10-764)

Two engine operate switches (figure 1-4), one for each engine, or located on the engine control panel. These switches are placarded ENG OPER, and each switch (L and R) has three positions placarded IGN, NORM, and MOTOR. The switches are spring-loaded from the IGN to the NORM positions. The switches must be raised to move out of the NORM position to the MOTOR position. The switches remain in the NORM position unless it becomes necessary to motor the engine. With the switch in the NORM position, the following is accomplished if the throttle is positioned to IDLE and engine core RPM is below 54 percent:

- Opens the air turbine starter control valve.

- Closes the environment control system shutoff valve.
- Opens the bleed air shutoff valve of only the engine being started. Opens both engine bleed air shutoff valves on aircraft sernos 78-0634 and subsequent and those modified by T.O. 1A-10-898.
- Supplies ignition to the engine for a minimum of 30 seconds.
- ENG START CYCLE light comes on.

The MOTOR position accomplishes all of the above with the exception of ignition and DC fuel pump with the throttle in the OFF position; however, ignition will be supplied to the engine when the throttle is positioned to IDLE. This position motors the engine for maintenance purposes or for air-purging the engine of excessive raw fuel or for cooling the engine.

Momentarily placing the engine operate switch to the IGN position will supply ignition to the corresponding engine for 30 seconds regardless of the throttle position or engine core RPM. The engine operate switches are powered by the DC essential bus.

Engine Start Cycle Light

The engine start cycle advisory light (figure 1-51) on the caution light panel is a yellow advisory light placarded ENG START CYCLE. The light will come on whenever the air turbine start solenoid valve is opened. It is powered by the DC essential bus.

Engine Core Speed Indicators

An engine core speed indicator (40, figure FO-1) is provided for each engine. The indicators display the speed of the compressor. They are placarded CORE PERCENT RPM with a scale from 0 to 100. The system is independent of the aircraft electrical system except for instrument lighting purposes.

Engine Fan Speed Indicators

A fan speed indicator (42, figure FO-1) is provided for each engine. The indicators display the fan speed. They are placarded FAN PERCENT RPM with a scale from 0 to 100. The system is powered by the auxiliary AC essential bus.

Engine Interstage Turbine Temperature Indicator (ITT)

An interstage turbine temperature indicator (ITT) (39, figure FO-1) is provided for each engine. The indicators display the temperature between the high and low pressure turbine sections. Each indicator is placarded TEMP °C X 100. A warning flag placarded OFF will appear in a window to indicate power loss. The indicating system is powered by the auxiliary AC essential bus.

Note

> On aircraft serno 76-0512 and subsequent and aircraft modified by T.O. 1A-10-619 an engine time temperature recorder has been provided for each engine. The time temperature recorder is installed in the avionics compartment.

Engine Overheat Caution Lights

An engine overheat caution light (figure 1-51) on the caution light panel is provided for each engine. The lights are yellow and placarded L ENG HOT and R ENG HOT. The lights will come on if the interstage turbine temperature exceeds 835°C. They are powered by the DC essential bus.

ENGINE OPERATION (AIRCRAFT PRIOR TO SERNO 75-00280 NOT MODIFIED BY T.O. 1A-10-508 AND T.O. 1A-10-764)

When compressed air is available, the engine start cycle is automatic when the throttle is moved from OFF to IDLE as long as battery and AC essential bus power is available. In this case the starter, fuel control and ignition systems operate to accomplish a start with no further action by the pilot. During air starts, sufficient cross bleed is available if the engine providing the air is above 85 percent RPM (core) and the engine to be started is below 54 percent RPM.

ENGINE OPERATION (AIRCRAFT SERNOS 75-00280 THROUGH 77-0226 NOT MODIFIED BY T.O. 1A-10-764 AND THOSE MODIFIED BY T.O. 1A-10-508)

When compressed air is available, the engine start cycle is initiated when the engine operate switch is positioned to MOTOR, the ignition button is depressed, and the throttle is positioned to IDLE. During air starts, sufficient crossbleed air is available if the engine providing the air is above 85% RPM (core).

ENGINE OPERATION (AIRCRAFT SERNO 77-0227 AND SUBSEQUENT AND THOSE MODIFIED BY T.O. 1A-10-764)

When compressed air is available, the engine start cycle is automatic, via throttle internal switches, when the throttle is moved from OFF to IDLE as long as battery and AC essential bus power is available. In this case the starter, fuel control, and ignition systems operate to accomplish a start with no further action by the pilot. An alternate method of energizing the engine start cycle is to momentarily place the engine operate switch to the IGN position, then to MOTOR, and place the throttle to the IDLE position. This action supplies ignition to the engine for a minimum of 30 seconds, opens the air turbine starter control valve, and provides fuel to the engine without the use of the throttle internal switches.

During air starts, sufficient crossbleed is available if the engine providing the air is above 85 percent RPM core speed and the engine to be started is below 54 percent RPM core speed.

FIRE EXTINGUISHING SYSTEM

The fire extinguishing system is available to both engines and to the APU compartment/area. It consists of fire extinguishing agent stored in two independently actuated pressurized bottles located in the fuselage. Either bottle may be discharged to either engine nacelle or the APU compartment/area by pulling the appropriate fire handle and actuating the discharge switch. The system is dearmed by pushing the appropriate fire handle in. The fire extinguishing system operates on battery bus power.

FIRE DETECTION SYSTEM (AIRCRAFT PRIOR TO SERNO 76-0512)

Fire detection is provided for in both engine nacelles and in the APU compartment by continuous temperature-sensitive elements. The APU fire detection is confined within the fire proof walled compartment. The fire warning light in the applicable left or right engine fire handle will come on when the temperature of the element reaches 600°F. The APU fire and overheat system is similar to the engine fire system except that the warning light in the APU fire handle will come on when the temperature in the element reaches 400°F. Both systems are powered by the auxiliary DC essential bus. The system is tested by depressing the FIRE DETECT BLEED AIR LEAK TEST button.

FIRE DETECTION SYSTEM (AIRCRAFT SERNO 76-0512 AND SUBSEQUENT)

Fire detection is provided for in both engine nacelles and in the APU area by continuous temperature-sensitive elements. The APU fire detection includes coverage for the adjacent hydraulic, fuel, electrical, flight control and environmental control subsystems equipment installed in the fuselage between the fuel tank aft bulkhead and the frame aft of the APU. The fire warning light in the applicable left or right engine fire handle will come on when the temperature of the element reaches 600°F. The APU fire and overheat system is similar to the engine fire system except that the warning light in the APU fire handle will come on when the temperature in the element reaches 400°F. Both systems are powered by the auxiliary DC essential bus. The system is tested by depressing the FIRE DETECT BLEED AIR LEAK TEST button.

Fire Detect/Bleed Air Leak Test Button

The fire detect/bleed air leak test button (figure 1-51) is a push-to-test button placarded FIRE DETECT BLEED AIR LEAK TEST. Depressing the switch checks the integrity of bleed air leak sensors, fire detection sensors and associated warning lights. If the circuit is intact, the BLEED AIR LEAK caution light on the caution light panel and the MASTER CAUTION, FIRE (L ENG) PULL, FIRE (R ENG) PULL, FIRE (APU) PULL lights will come on. The test button is powered by the auxiliary DC essential bus.

ENGINE AND APU FIRE HANDLES

Three T-shaped handles (8, 9, 10, figure FO-1), located in the glare shield on the instrument panel, provide the pilot with a fire warning for the engine nacelles or the APU. These handles display, from left to right, FIRE (L ENG) PULL, FIRE (APU) PULL and FIRE (R ENG) PULL when on. The lights are powered by the auxiliary DC essential bus through a temperature sensitive element. By pulling the appropriate fire handle the pilot initiates the following actions:

Engine Fire

- Cuts off fuel flow to the affected engine by closing the motorized main fuel shutoff valve.
- Closes the air bleed shutoff valve to the affected engine.
- Arms the fire extinguishing system to respective engine nacelle.

APU

- Cuts off fuel flow to the APU fuel control by closing the solenoid operated APU fuel shutoff valve.
- Arms fire extinguishing system to APU compartment.

CAUTION

- The engine throttle should be positioned to OFF if the engine fire handle is to be actuated. On aircraft prior to serno 75-00280 not modified by T.O. 1A-10-508, aircraft serno 77-0227 and subsequent and those modified by T.O. 1A-10-764, if the throttle is left in the IDLE position, ignition will automatically be available when the core RPM drops below 54 percent RPM and residual fuel in the fuel manifold may cause a fire.
- With more than one fire handle pulled, the fire extinguishing agent will be discharged into all areas selected. The quantity then discharged into the engine compartment may be insufficient to extinguish that fire.

Note

Moving an engine/APU fire handle while the ALE-40 chaff/flare system is armed for release, may cause an inadvertent chaff/flare release.

Fire Extinguishing Agent Discharge Switch

The fire extinguishing agent discharge switch (11, figure FO-1), placarded FIRE EXTING DISCH, is located on the right side of the glare shield above the instrument panel. The switch has three unplacarded positions, with travel as indicated by arrows above the switch. When the switch is moved in either direction the indicated extinguisher bottle is discharged and directs the agent to the engine or APU compartments selected by the fire handle. The switch will remain in the selected position to indicate which extinguisher bottle was discharged. The fire extinguisher bottles are discharged utilizing battery bus power.

AUXILIARY POWER UNIT (APU)

The auxiliary power unit (APU) (figure 1-2) supplies air for engine starting and drives a 10 KVA generator and a 10 GPM hydraulic pump. The unit is located in the aft fuselage between the two engines and is provided with safety devices which shut down the APU in the event of overspeeding or loss of oil pressure. The APU hydraulic pump is used during ground operation to pressurize the aircraft hydraulic systems. The door covering the APU hydraulic valve handle cannot be fully closed unless the handle is in the STOWED/FLT POSITION. In this position the APU hydraulic pump does not supply pressure to the aircraft hydraulic systems. The APU should be started under a no load condition. Fuel for APU starting is supplied by a DC fuel pump located in the left main tank. The DC pump and the APU control are powered by the DC essential bus.

APU SWITCH

The APU switch (figure 1-4) is a two position switch placarded START and OFF. The START position supplies DC essential bus power to operate the DC fuel pump, open the APU fuel valve, open the APU ejector solenoid valve, and energize the APU starter.

APU GENERATOR SWITCH

The APU generator switch (figure 1-9), placarded APU GEN, is a two position lever-lock switch placarded PWR and OFF/RESET. When placed in the PWR position, the APU generator powers an APU hydraulic pump cooling fan and the aircraft left and right AC system busses provided these busses are not powered by an engine generator or external power. On the ground the first source of power selected (APU or external) automatically locks out the other. However, after engine start when either left or right aircraft generator is selected, it automatically locks out the APU generator and the APU GEN light will come on. If the APU generator limits are exceeded, the system may be reset by momentarily placing the APU generator switch in the OFF/RESET position and returning it to the PWR position. The APU generator control is powered by the auxiliary DC essential bus.

CAUTION

The APU generator is the only source of power for electric fan cooling of the APU hydraulic pump. Therefore, do not operate the APU for more than five minutes with the APU generator OFF.

APU GENERATOR CAUTION LIGHT

The APU generator caution light (figure 1-51) is placarded APU GEN. The light is inoperative when the APU generator switch is in the OFF/RESET position.

With the APU generator switch in PWR position,

Light on indicates:

- Inoperable generator.
- APU operating with generator switch in PWR but aircraft busses being powered by either external power or engine generator.

CAUTION

During this mode of operation the caution light is on regardless of APU generator output. There is no indication that the APU hydraulic pump cooling fan is not receiving power. Overheating of the pump could result from extended operation with a failed APU generator either in the air or on the ground.

- APU not running and generator switch in PWR position.

Note

If the APU is operating with the APU generator switch in the PWR position and APU is shut down and restarted, the APU generator will remain inoperative until the APU generator switch is momentarily positioned to OFF/RESET, then returned to the PWR position.

Light off indicates:

- APU powering both aircraft busses.

APU TACHOMETER

The APU tachometer (44, figure FO-1) indicates the speed of the APU unit. The APU tachometer is placarded APU % RPM. The tachometer is independent of the aircraft electrical system except for instrument lighting and is enabled by the APU switch.

APU TEMPERATURE INDICATOR

The APU temperature indicator (45, figure FO-1) indicates the turbine discharge temperature. The indicator is placarded APU EGT° C X 100. The indicator is powered by the DC essential bus and is enabled by the APU switch.

APU OPERATION

APU starting requires only DC essential bus power and a fuel supply. When the APU start switch is positioned to START the DC essential bus power operates the DC fuel pump, opens the APU fuel valve (aft fuel tank mounted), and energizes the APU starter. The starter rotates the APU compressor and at approximately 10 percent RPM the APU fuel valve (APU mounted) opens and fuel and ignition are supplied to the APU. Acceleration of the APU continues until at approximately 60 percent RPM the starter disengages. At approximately 95 percent RPM ignition is terminated and the APU is self-sustaining and bleed air is available. APU speed and turbine discharge temperature are automatically controlled. The APU will automatically shut down if the RPM is excessive or the oil pressure is low. The APU will stabilize at 101 (± 2) percent RPM in approximately 60 seconds. APU starts can be made up to an altitude of 15,000 feet (most cases up to 20,000 feet) and the APU output will be sufficient to start an engine up to an altitude of 10,000 feet (most cases up to 15,000 feet). The APU will operate during negative g conditions for approximately 10 seconds.

Note

On aircraft serno 76-0535 and subsequent and those modified by T.O. 1A-10-869, the APU will automatically shutdown during ground operation if the APU EGT is excessive, APU RPM is excessive, APU oil pressure is low, or the APU fire warning system is activated. On aircraft serno 78-0634 and subsequent and those modified by T.O. 1A-10-869 and T.O. 1A-10-898, the APU overtemperature shutdown is disabled during ground engine start cycle plus four seconds. Once the weight is off the landing gear, the APU will automatically shutdown only if the RPM is excessive or the oil pressure is low.

AIRCRAFT FUEL SYSTEM

The aircraft fuel supply system (figure FO-4) consists of two integral wing tanks (left and right wing) and two tandem-mounted fuselage tanks (left main-aft and right main-forward). Up to three external (pylon) tanks may be carried; one tank on each wing and one on the fuselage centerline. The fuel supply system operates as two independent subsystems, with the left wing and left main tank feeding the left engine and the APU, and the right wing and right main tank feeding the right engine. The two subsystems can be interconnected by opening crossfeed valves (controlled by a single switch in the cockpit) to allow pressurized fuel flow to both engines and the APU from either subsystem. In addition, the two main tanks can be interconnected by opening a tank gate valve to permit equalization of the fuel in both tanks in level flight. The main tank sumps are self-sealing bladder cells. Each self-sealing sump contains approximately 900 pounds of fuel. The upper portion of the cells are of tear-resistant bladder construction. The wing tanks are integral construction within the wing structure and do not have bladder cells. Foam is incorporated in each tank to prevent fuel tank explosion. Boost pressure is provided by boost pumps located in each main and wing tank. Also a DC boost pump, located in the left main tank and powered by the DC essential bus, is used during engine and APU starts if the left main boost pump is inoperative. For negative g flight, collector tanks will supply the engine with sufficient fuel for 10 seconds operation at MAX power. In the event of a main tank boost pump failure, the affected engine will suction-feed from the failed tank for all power settings up to an altitude of 10,000 feet (most cases up to 20,000 feet). The wing tank boost pumps operate at a higher output pressure and override the main tank boost pumps to automatically empty the wing tanks first.

The main fuel feed lines to each engine, and to the APU, contain shutoff valves that are controlled by the fire handles. These shutoff valves allow for isolation of the fuel feed system outside the tanks.

Fuel in the external tanks is transferred to the main or wing tanks by pressure from the bleed air system. Fuel tank sump drains are provided for each tank. Drain valves protrude through structure and can be opened externally. Fuel cavity drains are provided in each main tank, and protrude through the aircraft skin to give an indication of fuel cell leaks.

The wing tanks have a dual level refueling shutoff valve. The valve closes when the tank is full and will not reopen unless the fuel level drops approximately 400 pounds or a time delay of approximately 10 minutes has elapsed. This characteristic will not allow the wing tanks to be topped off unless the fuel level is below approximately 1625 pounds or the fuel manifold has been unpressurized for the time delay period. This assures even fuel transfer from the external tanks. Therefore, during fuel transfer from the external tanks, the wing tank fuel quantity will drop approximately 400 pounds, then will fill to capacity. This cycling repeats until external fuel is depleted. During air refueling the wing tanks will not accept fuel unless the fuel level in the tanks has dropped approximately 400 pounds or the time delay has elapsed. The total fuel on board after refueling could be approximately 800 pounds less than total capacity. If total fuel capacity is required during air refueling, the external tanks can be turned off sufficiently prior to refueling so that the wing tank quantity drops approximately 400 pounds or the time delay has elapsed. Refueling will now fill all tanks to capacity.

A single point ground refueling receptacle, located in the leading edge of the left landing gear nacelle, permits refueling of each internal and external tank. When the receptacle is opened, refueling valves in each tank are automatically opened allowing fuel to enter each tank. When the tanks are full the refueling valves are closed by a float valve in each tank. A control panel, adjacent to the refueling receptacle, provides a means of ground checking the refueling valve shutoff. The panel also permits selective loading of any internal tank or external tank. The panel is deactivated and components restored to the normal flight active position when the door is closed. The refueling valves are powered by the auxiliary DC essential bus. Fuel tank capacities are shown in the usable fuel quantity data table, figure 1-6. Fuel grade and specification to be used are covered in the servicing diagram, figure 1-57.

The fuel system is designed for maximum survivability. Protection provisions are as follows:

- Foam inside and void filler outside of tanks for fire protection.
- Self-sealing main tank sumps. Each sump contains approximately 900 pounds "get home" fuel.
- The main tanks above the sumps are tear-resistant bladder construction.
- Single point ground refueling and engine feed lines outside the tanks are self-sealing to prevent leaks.
- The air refueling line is automatically purged of fuel after air refueling.
- The air refueling purge system can be used to check the integrity of the air refueling line prior to use.

USABLE FUEL QUANTITY DATA

		POUNDS		
	GALLONS	JP-4 (NOTE 1)	JP-5 (NOTE 2)	JP-8 (NOTE 3)
L. MAIN	511	3,325	3,475	3,425
R. MAIN	511	3,325	3,475	3,425
L. WING	311	2,025	2,125	2,075
R. WING	311	2,025	2,125	2,075
TOTAL INTERNAL	1,644	10,700	11,200	11,000
CENTERLINE	600	3,900	4,080	4,020
L. WING	600	3,900	4,080	4,020
R. WING	600	3,900	4,080	4,020
TOTAL EXTERNAL	1,800	11,700	12,240	12,060
TOTAL FUEL	3,444	22,400	23,440	23,060

NOTES:

1. FUEL WEIGHT BASED ON JP-4 AT 6.5 LBS PER GAL.
2. FUEL WEIGHT BASED ON JP-5 AT 6.8 LBS PER GAL.
3. FUEL WEIGHT BASED ON JP-8 AT 6.7 LBS PER GAL.

Figure 1-6

- The fuel is in four separate tanks arranged so that each acts independently.
- The fuel feed shutoff valves are inside the tanks to keep the engine feed lines dry after shutoff.
- Fill disable switches are provided to close off a damaged tank when air refueling.

FUEL QUANTITY INDICATOR AND SELECTOR

The fuel quantity indicator (figure 1-7) is provided to monitor the total fuel remaining or fuel remaining in a specific pair of tanks. The digital readout is a continuous display of total fuel remaining including external, in pounds. The pointer display provides an indication of fuel in specific tanks as selected by an adjacent rotary

FUEL QUANTITY INDICATOR AND SELECTOR

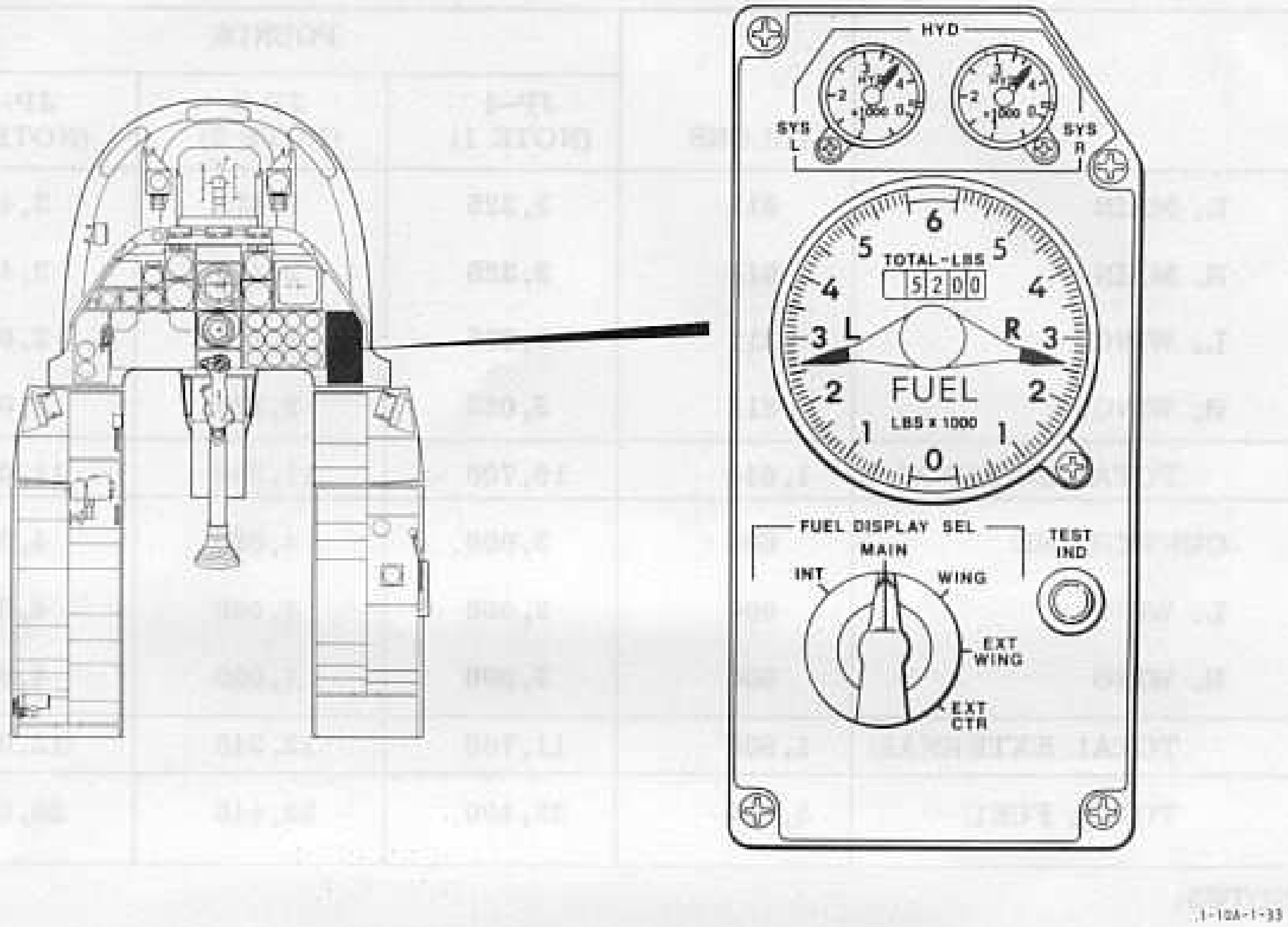


Figure 1-7

selector switch. The left and right pointers indicate for the left and right fuel systems, respectively. The fuel quantity indicator is powered by auxiliary AC essential bus. Positions of the selector are as follows:

- INT - Left and right pointers indicate total internal fuel for respective system.
- MAIN - Left and right pointers indicate fuel in the respective main tank.
- WING - Left and right pointers indicate fuel in the respective wing tank.
- EXT WING - Left and right pointers indicate fuel in the respective wing pylon tank.
- EXT CTR - Left pointer indicates fuel in the fuselage pylon tank. The right pointer will zero.
- TEST IND - When button is depressed the left and right pointers will read 3000 ± 300 pounds each and the digital readout will read 6000 ± 400 pounds. When the TEST IND switch is released the pointers and digital readout will return to the normal positions.

Note

The fuel quantity totalizer will read high if the left main tank quantity is below approximately 500 pounds and considerable quantity of the fuel remains in the other tanks. The percent error will decrease as the fuel remaining decreases. However, the individual tank readings obtained by utilizing the fuel quantity selector are not affected.

- The fuel quantity indicator system is calibrated using external AC power. The indicator totalizer reading will record the same as when calibrated if powered by the engine generators or the instrument inverter. However, on aircraft prior to serno 77-0187 not modified by T.O. 1A-10-772, if the APU generator is the source of power for the fuel quantity indicator, readings may be erroneous. Individual tank readings are not affected and they can be totaled as a check of the totalizer reading.

LEFT AND RIGHT MAIN FUEL LOW CAUTION LIGHTS

The left and right main fuel low caution lights, (figure 1-51) are placarded L-MAIN FUEL LOW and R-MAIN FUEL LOW, respectively. When the L-MAIN FUEL LOW caution light comes on, this indicates that the fuel quantity in the left main fuel tank is 650 (+150, -100) pounds. When the R-MAIN FUEL LOW caution light comes on, this indicates that the quantity in the right main fuel tank is 500 (+150, -100) pounds. This condition may be verified at the fuel quantity indicator. The lights are powered by the auxiliary DC essential bus.

LEFT AND RIGHT FUEL PRESSURE CAUTION LIGHTS

The left and right fuel pressure caution lights, (figure 1-51) placarded L-FUEL PRESS and R-FUEL PRESS, respectively, come on to indicate that fuel pressure is too low. Possible failure of a boost pump is indicated by one of these lights coming on or, if the boost pump caution lights are not on, a failure or a clog in the engine feed line. The lights are powered by the auxiliary DC essential bus.

FUEL TANK VENT SYSTEM

Each main and wing tank (figure FO-4) is vented independently to a vent collector tank located in the left main tank. Vent lines from the wing tanks also serve as return lines for any fuel collected in the vent tank. The vent tank is vented to ambient through an overboard vent scoop. The vent outlet is sized to maintain safe tank pressure if a tank overfills due to failure of a refuel shutoff valve.

Foam is installed in the vent tank to isolate the fuel system from the overboard vent scoop to provide fire and lightning protection for the fuel system.

MAIN TANK BOOST PUMP SWITCHES

Two main tank boost pump switches (figure 1-8) are placarded BOOST PUMPS with positions L-MAIN-R and OFF. The L position supplies left AC system bus power to operate the left main boost pump. The R position supplies right AC system bus power to operate the right main boost pump. The OFF position deactivates the respective boost pump.

LEFT AND RIGHT MAIN BOOST PUMP CAUTION LIGHTS

The left and right main fuel tank boost pump caution lights (figure 1-51) placarded L-MAIN PUMP and R-MAIN PUMP, respectively, come on when fuel pressure differential at the outlet of the indicated fuel boost pump is low, indicating a probable pump failure. The light will go off when the differential pressure increases to an acceptable level. The lights are powered by the auxiliary DC essential bus.

WING TANK BOOST PUMP SWITCHES

Two wing tank boost pump switches (figure 1-8) are placarded BOOST PUMPS with

FUEL SYSTEM CONTROL PANEL

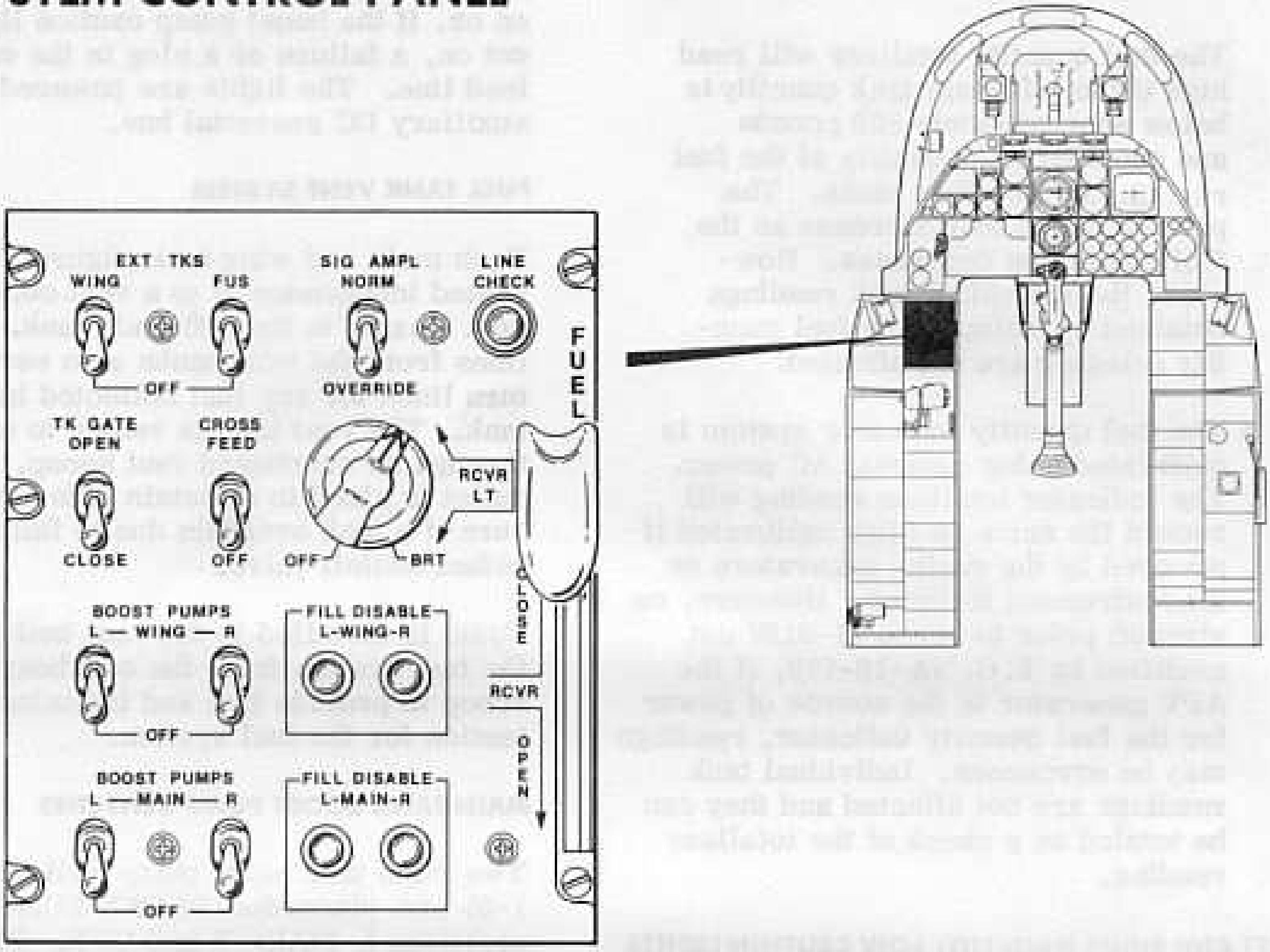


Figure 1-8

positions L-WING-R and OFF. The L position supplies left AC system bus power to operate the left wing boost pump. The R position supplies right AC system bus power to operate the right wing boost pump. The pumps will automatically stop when the tank float switch senses an empty tank. The OFF position deactivates the respective boost pump.

LEFT AND RIGHT WING BOOST PUMP CAUTION LIGHTS

The left and right wing fuel tank boost pump caution lights (figure 1-51) placarded L-WING PUMP and R-WING PUMP, respectively, come on when fuel pressure differential at the outlet of the indicated fuel boost pump is low, indicating a probable pump failure. The light will go off when the differential pressure increases to an acceptable level. The lights are powered by the auxiliary DC essential bus.

EXTERNAL TANK SWITCHES

Two external tank switches (figure 1-8) placarded EXT TKS are located on the fuel system control panel. One switch is marked WING and OFF while the other is marked FUS and OFF. The WING position supplies auxiliary DC essential bus power to position the tank pressure vent valves to pressurize the external tanks, utilizing air from the bleed air supply system. The fuel is transferred to the main and wing tanks through the refueling manifold and fuel shutoff valves. When the tanks are empty or not transferring fuel, they are vented to the atmosphere. The OFF position returns the pressure vent valve in both tanks to the vent position. However, if left or right main tank low level switch is actuated due to low fuel in the tanks, external fuel, if available, will automatically be transferred to the main tanks even if the external wing tank switches are in the OFF position. When the fuselage switch is positioned to FUS, fuel is transferred from the external centerline tank in the same manner as the external wing tanks.

EXTERNAL TANK JETTISON

The external tanks may be jettisoned individually, all together or in combination with other stores. To jettison the tanks in combination with all other stores the EXT STORES JETT button is depressed. The tanks are selectively jettisoned individually or all together by following the external tank jettison procedure in Section III.

CROSSFEED SWITCH

The crossfeed switch (figure 1-8) is a two position switch placarded CROSSFEED and OFF. When positioned to CROSSFEED two auxiliary DC essential bus-powered valves in the fuel feed system open allowing any operating boost pump, to feed both engines. When positioned to OFF the valves close isolating the two fuel subsystems.

TANK GATE SWITCH

The tank gate switch (figure 1-8) is a two position switch placarded TK GATE with positions placarded OPEN and CLOSE. The OPEN position supplies auxiliary DC essential bus power to open the gate valve linking the left and right main fuel tanks and allows fuel to flow between the two tanks. Fuel in the main tanks will be below the tank gate valve and will not transfer in level flight when the fuel level is below 1300 pounds in each main tank. The sump fuel will not flow between the tanks. The CLOSE position closes the gate valve.

Note

The tank gate switch should be opened only in relatively level unaccelerated flight to preclude excessively large cg shift due to fuel transfer.

LEFT AND RIGHT TANKS UNEQUAL CAUTION LIGHT

The left and right fuel tanks unequal caution light, (figure 1-51) placarded L-R TKS UNEQUAL come on when an imbalance of 750 ± 250 pounds in fuel quantity is sensed between the two main fuselage tanks. This condition may be verified by checking the fuel quantity indicator for a specific reading of the tanks. The light is powered by the auxiliary DC essential bus.

CAUTION

With less than 300 rounds of ammunition remaining, or for configurations without ammunition but with ballast for the most aft cg limit of 29.6 percent MAC, and an L-R TKS UNEQUAL caution light coming on:

- Verify unequal fuel quantity utilizing fuel quantity gage.
- If more fuel remains in the aft (left main) tank than in the forward (right main) tank, aircraft should not exceed the calibrated airspeeds indicated in Section V, pending flight test verification of center of gravity operating limitations.

FUEL SYSTEM OPERATION

During normal operation, fuel system operation is automatic except for selecting external tank fuel flow. The main and wing tank boost pump switches are positioned to L and R so that the pumps will become operative when AC power is available after APU or engine start. The tank gate switch is positioned to CLOSE to preclude a fuel imbalance. The crossfeed switch is positioned to OFF so that the left and right main and wing tanks will supply fuel independently to the respective engine. When the battery switch is placed in the PWR position the DC boost pump is energized when the APU switch is positioned to START or if either throttle is forward of OFF position and the left main boost pump is inoperative. The DC boost pump supplies fuel to the APU and the left engine. If the right engine must be started first and AC system bus power is not available, the crossfeed switch must be positioned to CROSSFEED so that DC boost pump pressure is supplied to the right engine. When the left and right AC system busses are energized, the left and right main and wing tank boost pumps will operate. After takeoff the external wing tank switch is positioned to WING. External tank fuel will be transferred to the internal tanks as fuel is used until the external tanks are empty at which time the pressure vent valve in the tanks will return to the vent position. The wing boost pumps will then supply the respective engine with fuel until the wing tanks are empty, at which time the wing tank boost pumps will automatically shut off. The main boost pumps will then supply the respective engine with the remainder of the fuel in the airplane. In the event of a wing tank boost pump failure, wing tank fuel will gravity feed to its associated main tank. However, due to the relative head between the main tank and wing tank, gravity feed of a full wing tank will not occur until the main tank fuel level is below approximately 600 pounds. Dual check valve units in each wing tank gravity feed line prevent reverse fuel flow from the main tanks back into the wing tanks.

Note

During fuel transfer from the external tanks, either fuel level in the wing tanks will drop approximately 400 pounds, or the time delay will have elapsed before external fuel transfers. Fuel will then transfer until the wing tanks are full. This cycling will repeat until external fuel is depleted.

When carrying external tanks, fuel sequencing will be as follows:

- External wing tanks.
- External fuselage tank.
- Internal fuel.

AIR REFUELING SYSTEM

The aircraft can be refueled inflight from a tanker aircraft equipped with a flying boom. The aircraft is equipped with a universal air refueling receptacle/slipway installation (UARRSI) (figure 1-2) located in a compartment forward of the cockpit. By positioning a lever on the fuel system control panel, a flush (slipway) door folds down into the fuselage powered by the right hydraulic system to expose the air refueling receptacle and to provide a slipway to guide the tanker boom during refueling. When the tanker boom is inserted in the receptacle, the nozzle latch rollers are actuated to the locked position, and the refueling transfer commences. Fuel transfer through the receptacle is distributed to the main and wing tanks, and to external tanks if carried. However, through use of the fill disable switches, located on the fuel system control panel, the pilot can prevent fuel from entering any specific internal tank suspected of having sustained damage. As each tank is filled, float-operated fuel shutoff valves within each tank will close, preventing overfill. When refueling is completed, the disconnect of the boom nozzle will normally be accomplished by a signal from the tanker or by the receiver pilot depressing the air refuel disconnect/reset button on the control stick grip. An automatic disconnect will occur when both receiver and tanker systems are completely operational and one of the following occurs:

- Excessive fuel pressure occurs in the receiver fuel manifold, actuating the high pressure disconnect switch.

- The tanker boom limit switches (see T.O. 1-1C-1-26) are actuated due to excessive displacement of the boom.

When the slipway door is closed, the air refueling manifold forward of the main tank is automatically purged of fuel. The READY light may come on for up to three minutes during the purge cycle if one engine is operating above 85 percent core RPM or the APU is operating. The purged fuel is directed to the right main tank. Refer to T.O. 1-1C-1 for basic flight crew air refueling procedures and T.O. 1-1C-1-26 for A-10A flight crew air refueling procedures.

Note

During air refueling the wing tanks will not accept fuel unless the fuel level in the tanks has dropped approximately 400 pounds or the time delay has elapsed. The total fuel on board after refueling could be approximately 800 pounds less than total capacity. If total fuel capacity is required during air refueling, the external tanks can be turned off sufficiently prior to refueling so that the wing tank quantity drops approximately 400 pounds. Refueling will now fill all tanks to capacity.

If the right hydraulic system fails, the spring loaded slipway door will open when the air refuel control is positioned to OPEN. The LATCHED and DISCONNECT lights will not come on because of the hydraulic failure.

Note

On aircraft prior to serno 76-0512, aerodynamic effect (aircraft speed and angle of attack) and system friction will result in a condition where the door opens sufficiently to expose the receptacle lights and permit emergency "stiff boom" refueling with or without the READY light coming on. Air refueling may be accomplished at this time. Applying boom nozzle pressure on the slipway door should result in the slipway door down lock engaging and the READY light coming on. The time for the door to open sufficiently to expose the receptacle is improved by reducing speed and will occur within approximately three minutes at 150 KIAS.

Air Refuel Control

The air refuel control (figure 1-8) is placarded RCVR with two positions OPEN and CLOSE. When positioned to OPEN, power from the left DC system bus is supplied to the signal amplifier and to a solenoid in the hydraulic control valve. At the same time a slide valve is mechanically positioned to supply right hydraulic system pressure to open the slipway door and cause the READY light to come on when the door is locked open. When the boom nozzle is inserted in the receptacle an electrical signal is directed to the latch solenoid valve to close the latches securing the nozzle in the receptacle and to cause the LATCHED light to come on and the READY light to go off. After the nozzle is removed from the receptacle the DISCONNECT light will come on. The CLOSE position directs hydraulic pressure to close the slipway door and cause the DISCONNECT light to go off. In the event of loss of hydraulic pressure, the OPEN position releases a lock allowing the spring loaded slipway door to open.

Fill Disable Switches

Four fill disable switches (figure 1-8) are push-pull button type switches. Two switches are placarded L-MAIN-R and two are placarded L-WING-R. If a main or wing tank is damaged, pulling up the respective switch shuts off the shutoff valve in the tank preventing that tank from being refueled. The switches are powered by the left DC system bus.

Signal Amplifier Switch

The signal amplifier switch (figure 1-8) provides an emergency method of hook-up. The switch is placarded SIG AMPL and has two positions placarded NORM and OVERRIDE. During the normal refueling cycle, the switch remains in the NORM position and air refueling system power and actuating signals function automatically through the amplifier and the refueling sequence is automatic after contact is made. If a failure occurs, fuel may not be transferred or the tanker boom may not stay latched. When this occurs, the override switch should be placed in the OVERRIDE position. In OVERRIDE no signals are passed to the tanker, and the tanker cannot actuate the disconnect cycle. Disconnect is accomplished by depressing the air refuel disconnect/reset button on the control stick. The signal amplifier switch is powered by the left DC system bus.

Air Refuel Disconnect/Reset Button

An air refuel disconnect/reset/button (time shared with Maverick Track) is provided on the control stick grip (figure 1-15). Placing the air refuel control in the OPEN position activates this button to function for air refuel operation. With the boom nozzle inserted in the receptacle and the LATCHED light on, a disconnect may be accomplished by depressing the air refuel disconnect/reset button. If the DISCONNECT light is on and a hookup is desired, depressing the air refuel disconnect/reset button will recycle the system to the ready mode.

Air Refuel Line Check Button

The air refuel line check button (figure 1-8) is a pushbutton switch placarded LINE CHECK. Momentarily depressing this button checks the air refuel manifold integrity through a time delay relay. If the manifold is damaged, inflowing fuel will be discharged overboard with the possibility of fire and explosion. When the button is depressed before operating the air refuel door, the internal tank shutoff valves are closed and the air purge valve opens allowing air to pressurize the air refueling manifold. One engine must be operating above 85 percent core RPM or the APU operating to supply sufficient air pressure to assure the READY light coming on if the manifold is intact. The READY light should come on when the air pressure builds up in the manifold (approximately 1 to 2 minutes). The light will go out approximately three minutes after the line check button is depressed. However, the light will remain on as long as the wing tanks are above approximately 1625 pounds. If the READY light does not come on within three minutes after the line check button is depressed, the refuel manifold is damaged. In this case, air refueling should not be attempted unless absolutely necessary.

Air Refuel Status Lights

The air refueling status indication is provided by three lights (6, figure FO-1) placarded READY, LATCHED, and DISCONNECT, located to the right of the main windshield directly below the standby compass. When the slipway door is fully open and locked, circuits are energized by left DC system bus power to cause the READY light to come on. Once the tanker boom nozzle and the refueling receptacle are connected, the READY light goes out and the LATCHED light comes on. When the refueling operation is completed, or when the boom nozzle and refueling receptacle are disconnected for any reason, the LATCHED light will go out and the DISCONNECT light will come on. The DISCONNECT light will remain on until the air refuel control is moved to the CLOSE position, or the air refuel disconnect/reset button on the control stick is depressed.

Receptacle Lighting

The air refueling receptacle provides a slipway floodlight installation on either side of the receptacle for illumination of the receptacle and the tanker boom during refueling. A rheostat type receptacle lighting switch (figure 1-8) is placarded RCVR LT with a full range of settings from OFF to BRT. The lights are powered by the left DC system bus.

Receptacle Lighting Switch

A rheostat type receptacle lighting switch (figure 1-8) is placarded RCVR LT with a full range of settings from OFF to BRT. The switch is powered by the left DC system bus.

Air Refueling Intercommunications

When some tankers and the A-10A are connected, secure interphone is available between the aircraft when the signal amplifier switch is in the NORM position and the LATCHED light is on. The intercom system is powered by the DC essential bus.

Note

> The HM and INT monitor switch on the intercom control panel should be pulled out to enable communications with the tanker.

ELECTRICAL POWER SYSTEM

The electrical power system (figure FO-5) provides 115/200V, 400 Hz, three-phase AC and 28 VDC required for operation of the various aircraft systems. This power is produced by two independent systems referred to as left system and right system. For descriptive purposes, each system is further divided into an AC generator and drive system, and a DC power system.

The AC power is produced by two independent engine driven integrated drive generators (IDG's). In addition to powering the main AC buses, the output of each AC generator and drive system is applied to the DC power system by means of converters. With the exception of the engine-mounted IDG's, the components comprising the left and right electrical systems are located in electrical load centers on either side of the aircraft. In the event of a failure of either AC generators and drive systems, the load of the failed system will automatically transfer to the operating system. If a complete failure occurs, essential AC and DC power is provided by an emergency power system consisting of a battery and an instrument inverter. An additional source of AC power is provided by the APU generator which in turn powers the DC converters.

For ground operation, external AC power can be connected to the main AC busses via the external power receptacle.

APU GENERATOR SWITCH

The APU generator switch, (figure 1-9) placarded APU GEN, is a two-position switch with positions placarded PWR and OFF/RESET. When placed in the PWR position, the APU generator powers the hydraulic pump cooling fan and the aircraft left and right AC system busses; if these busses are not powered by an engine generator or external power. On the ground the first source of power selected (APU or external) automatically locks out the other.

The switch must be lifted to move it out of the PWR position and into the OFF/RESET position. If the APU generator limits are exceeded, the system may be reset by momentarily placing the APU generator switch in the OFF/RESET position and returning it to the PWR position. The OFF/RESET position removes the APU generator from the line. The APU generator control is powered by the APU generator.

APU GENERATOR CAUTION LIGHT

The APU generator caution light (figure 1-51) is placarded APU GEN. The light is inoperative when the APU generator switch is in the OFF/RESET position.

With the APU generator switch in PWR position:

Light on indicates:

- Inoperable generator
- APU operating with generator switch in PWR but aircraft busses being powered by either external power or engine generator

ELECTRICAL POWER CONTROL PANEL

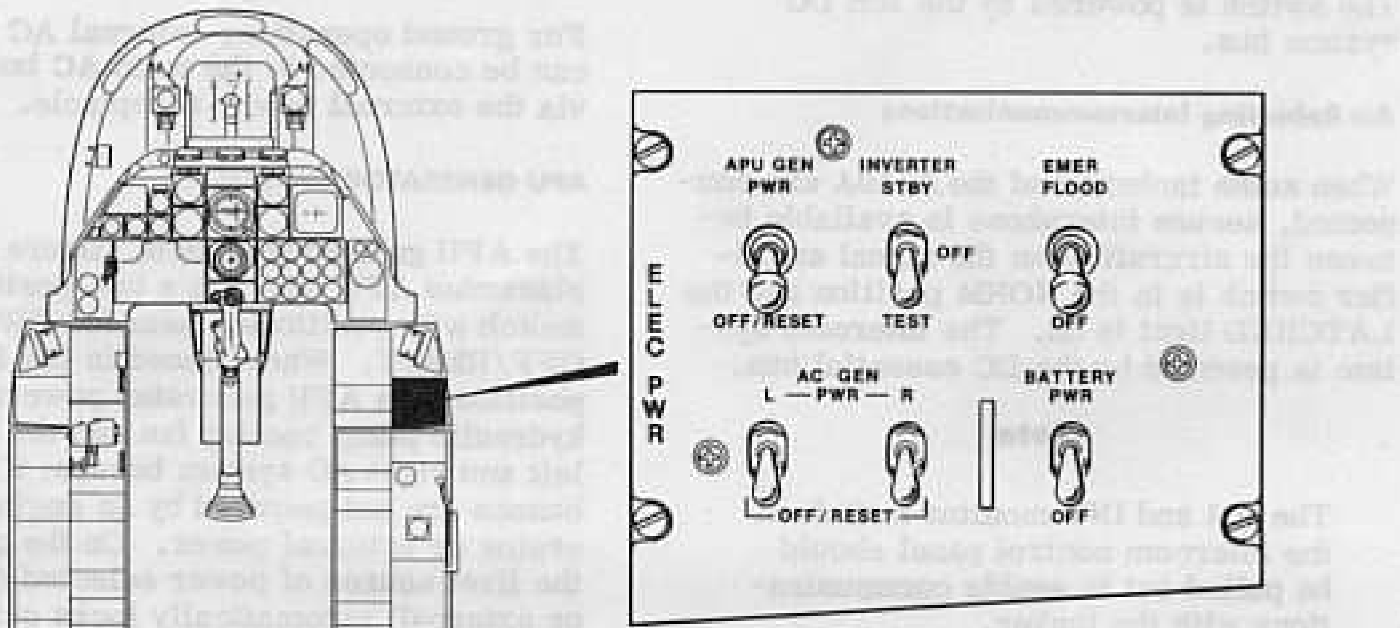


Figure 1-9

- APU not running and generator switch in PWR position.

Note

If the APU is operating with the APU generator switch in the PWR position and APU is shut down and restarted, the APU generator will remain inoperative until the APU generator switch is momentarily positioned to OFF/RESET then returned to the PWR position.

Light off indicates:

- APU powering both aircraft busses.

ESSENTIAL BUSSES

The essential and auxiliary essential busses power that equipment essential to maintain safe flight. For the AC system, these busses are normally powered from the left generator with automatic transfer to the right generator in the event of a failure with backup from the instrument inverter. For the DC system, power is provided by the converters with backup from the battery.

Left and Right System Busses

The left and right system busses power the mission support equipment and those sub-systems not deemed essential to maintain safe, controlled flight. These busses are normally powered by their respective generator for the AC system and by the converters for the DC system.

Instrument Bus

The 26 VAC instrument transformer bus supplies power to some instrumentation. This bus is powered through the instrument transformer circuit breaker from the auxiliary AC essential bus.

Battery Bus

The battery bus powers equipment requiring operation when the cockpit battery switch is OFF. This includes: canopy operation, emergency cockpit flood lights, external stores jettison, fire extinguishers and boarding ladder. An external battery switch is installed between the battery and the battery bus.

Armament Busses

The AC and DC armament busses provide the power to the armament and fire control system's equipment. Power is provided by the respective right system bus. Power to the armament system is also supplied by the left system AC and DC busses.

AC/DC Circuit Breaker Panel

Push-button type circuit breakers (figure 1-10) are provided to protect the electrical circuits in the aircraft. Those circuit breakers essential for aircraft survivability are located in the cockpit on the pedestal below the instrument panel. The remaining circuit breakers are located in two electrical load centers in the aircraft.

AC SYSTEM

The two AC generator and drive systems each consist of a 30/40 KVA engine-driven, constant speed, oil cooled integrated drive generator (IDG). The IDG's operate as two separate (isolated) power systems with provisions for the automatic transfer of bus load of a failed system to the operator system. Under normal operating conditions, the left engine IDG supplies power to the left AC system bus and the AC essential bus. The right engine supplies power to the right AC system bus.

Cockpit control of the AC generators is provided by two switches on the electric power control panel. Two lights on the caution light panel indicate generator malfunction.

AC Generator Switches

The two AC generator switches (figure 1-9) placarded AC GEN L and R are located on the electric power control panel. Each switch has two positions, placarded PWR and OFF/RESET. When placed in the PWR position, the associated AC generator is placed on the line, providing the corresponding generator control unit senses that the generator output is within specified limits.

If these limits are exceeded during operations, as indicated by a L- or R-GEN light coming on on the caution light panel, the affected generator will go off the line. The system may be reset by momentarily placing the applicable AC GEN switch to the OFF/RESET position. If the fault remains, the system will not reset. Placing the AC GEN switch in the OFF/RESET position removes the generator from the line.

AC Generator Caution Lights

The generator-out caution lights, (figure 1-51) on the caution light panel placarded L GEN and R GEN, are provided to indicate the operational status of the two AC generators. When the control unit in either left or right generator senses that the generator output is not within limits, it automatically trips the associated generator off the line.

If it was only a transient fault the affected generator system may be restored by momentarily positioning the associated generator switch to OFF/RESET then returning it to the PWR position.

CIRCUIT BREAKER PANEL

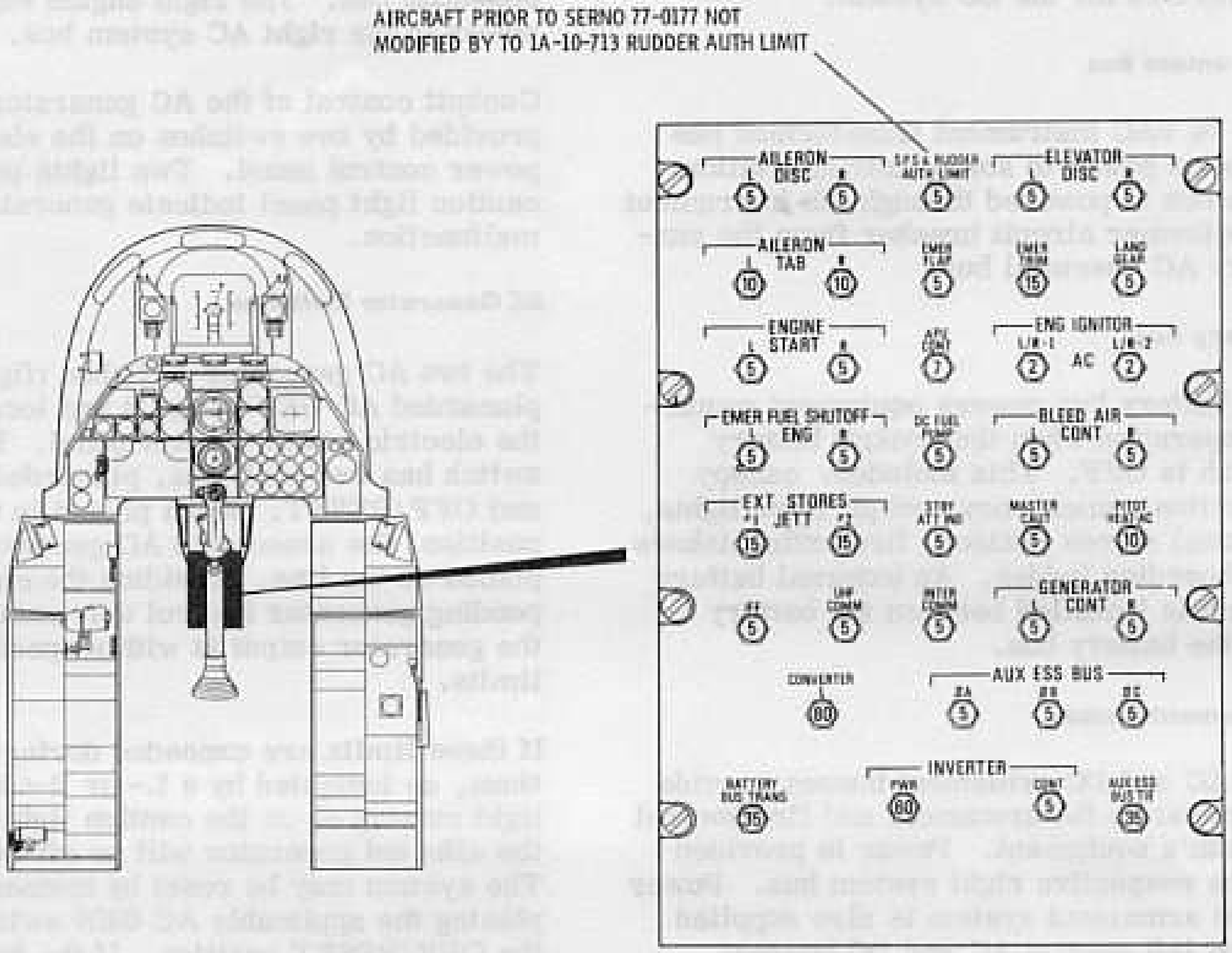


Figure 1-10

AC External Power

A standard receptacle on the forward underside of the fuselage is provided for ground connection of external power. External power is supplied to the aircraft provided power is not being supplied by the APU or aircraft generators. The first source of power selected (APU or external) automatically locks out the other. The DC essential bus can be energized from the battery or, when external power is connected, from the right system converter.

Instrument Inverter

A 750 VA instrument inverter supplies power to the AC essential, auxiliary AC essential and instrument busses when the left and right AC busses are not energized. Power for the instrument inverter is from the DC essential bus. The instrument inverter is controlled by a toggle switch located on the electrical power control panel. An instrument inverter caution light is provided to come on when power is not being supplied to the AC essential bus.

Instrument Inverter Switch

The instrument inverter switch (figure 1-9) is placarded INVERTER and has three positions: STBY, OFF and TEST. The STBY position must be selected to supply AC power to the AC essential and auxiliary essential busses. The STBY position also completes a circuit which causes the DC essential bus to automatically operate the inverter if the AC power system fails to provide power to the AC essential bus. All electrical power to the AC essential bus is interrupted when the inverter switch is placed in the OFF position. In the OFF position, the INST INV caution light will come on. The momentary TEST position is utilized to verify that the instrument inverter will operate properly in the event of a complete AC power failure. This position disconnects normal AC power to the essential AC bus and allows the instrument inverter to apply AC power to the bus while the switch is held in TEST. Proper inverter operation will be indicated by the INST INV caution light off or momentarily on. If the INST INV caution light remains on while the inverter switch is in the TEST position, the inverter will not power the AC essential bus if the normal AC power system fails.

Note

- After power is applied to the aircraft (external generator, or APU), the inverter switch must be in the STBY position to obtain power for the AC essential, auxiliary AC essential, and instrument busses.
- Moving the instrument inverter switch while the ALE-40 chaff/flare system is armed for release, may cause an inadvertent chaff/flare release.

Instrument Inverter Caution Light

The instrument inverter caution light, (figure 1-51) placarded INST INV, comes on to indicate failure of the instrument inverter or the instrument inverter switch is in OFF position. The light circuit may be checked by placing the inverter switch in the TEST position. This disconnects any AC power to the AC essential bus and momentarily operates the inverter to supply AC power to the bus. If the inverter system is normal, the INST INV caution light will show either a momentary light, or no light at all. If the light remains on while the inverter switch is in the TEST position, failure of the inverter or change over circuitry is indicated.

AC SYSTEM OPERATION

The first IDG to come on the line will supply power to its associated bus. The bus load of the nonoperating generator will be supplied by the external power source during ground operation. When the second IDG comes on the line the external power contactor will be deenergized. During shutdown, if external power is connected, the respective bus will be supplied power from the external source when the IDG drops off the line. If external power is not connected, the bus will be supplied power from the operating generator. The bus transfer relay will transfer to the right AC system bus whenever the left AC system bus is deenergized.

DC SYSTEM

Both DC power systems are comprised of a 100-ampere, 28 VDC converter which supplies power to the associated DC bus and to the DC essential bus. The converters are connected in parallel to provide uninterrupted power to the using equipment in the event of a failure in either system.

The DC essential bus is automatically powered from the battery if converter power is lost provided the battery switch is in PWR position. Either converter is capable of supplying the entire DC load of the aircraft.

Aircraft Battery

The aircraft battery is a 24 volt, 34 ampere-hour nickel cadmium type. The battery supplies power to the battery bus if the external battery switch is in the ON position. The battery also powers the DC essential and the auxiliary DC essential busses for self sufficient APU and engine starting and for inflight emergency power. The battery is recharged from the DC essential bus whenever the converters are operating.

External Battery Switch

The battery disconnect switch behind access door F63 provides the means to disconnect the battery bus and the battery relay from the battery, regardless of the battery switch (on the right console) position and without disconnecting the battery connection.

Battery Switch

The battery switch (figure 1-9) placarded BATTERY, is a two-position switch located on the electric power control panel. The switch positions are placarded PWR and OFF. The PWR position connects the battery to the DC essential bus. The DC essential bus, powers the battery bus when either converter is operating. If the battery switch is in the OFF position, the battery is disconnected from the DC essential bus, and powers only the battery bus.

Note

The battery bus is energized whenever the external battery switch is in the ON position. If the external battery switch is OFF the battery bus will not be energized even with the cockpit battery switch in the PWR position. When external power is connected and either left or right engine generator or the APU generator is operating, the battery bus is energized.

Converter Caution Light

The left and right system converter caution lights (figure 1-51) placarded L CONV and R CONV, are provided to indicate the operational status of the converters. Either caution light coming on indicates failure in the associated converter.

Note

If either generator fails, the affected converter caution light should remain off, indicating automatic transfer to the operating system.

DC SYSTEM OPERATION

When external AC power is supplied to the aircraft, power is directly supplied to the right system converter. Three-phase AC power is applied to the left system converter by the aircraft AC busses which in turn powers the system in parallel with the right system. The converters supply 28 VDC power to the DC essential bus, auxiliary DC essential bus, left and right DC busses and to the DC armament bus.

When the generators become operative, the right converter is powered by the right AC system bus. A failure in either converter is indicated in the cockpit by the L-CONV or R-CONV light on the caution light panel coming on.

HYDRAULIC POWER SUPPLY SYSTEM

The hydraulic power supply system (figure FO-6) consists of two fully independent hydraulic power systems, designated left hydraulic system and right hydraulic system. Both systems are pressurized by identical engine driven pumps. A small accumulator in the reservoir circuit stabilizes the pressure. Fluid temperature control is maintained through the use of non-regulated air/oil heat exchangers that remove heat from the engine driven pump case return fluid. In addition to the two system hydraulic pumps, a 10-gallon per minute hydraulic pump, driven by the APU can be selected by the ground crew to provide hydraulic power for test purposes to either hydraulic system, but not both simultaneously. The selector valve is accessible through the APU access door on the bottom of the aft fuselage. The door cannot be fully closed unless the selector valve handle is in the STOWED/FLT POSITION. In this position the APU hydraulic pump does not supply pressure to the aircraft hydraulic systems.

The left hydraulic system powers the following subsystems:

Flight control	-Left rudder, left elevator, left and right aileron, flaps
Landing gear	-Landing gear extend and retract, wheel brakes, nosewheel steering
Armament	-One half of gun drive

The right hydraulic system powers the following subsystems:

Flight control	-Right rudder, right elevator, left and right aileron, speed brakes, leading edge slats
Auxiliary landing gear	-Auxiliary landing gear extend, emergency wheel braking and associated accumulators
Armament	-One half of gun drive and is necessary to start the gun
Air Refueling	-Slipway door and receptacle lock

Each of the subsystems listed above is described within their respective system.

HYDRAULIC SYSTEMS PRESSURE GAGES

Two hydraulic pressure gages (46, figure FO-1) permit the pilot to continuously monitor both hydraulic systems operating pressures. These gages are placarded HYD SYS L and HYD SYS R. The scale is placarded, PSI X 1000. The gages are powered by the 26V instrument transformer which is powered by the auxiliary AC essential bus.

HYDRAULIC PRESSURE CAUTION LIGHTS

Two hydraulic pressure caution lights (figure 1-51) on the caution light panel are placarded L HYD PRESS and R HYD PRESS. The lights will come on if the pressure in the respective system drops below 900 ± 100 psi. The light will go off when the pressure returns to a level above 1000 psi. The lights are powered by the auxiliary DC essential bus.

HYDRAULIC RESERVOIR LOW LEVEL LIGHTS

Two hydraulic reservoir low level lights (figure 1-51) on the caution light panel are placarded L HYD RES and R HYD RES. The lights are powered by the auxiliary DC essential bus. The lights will come on whenever one reservoir fluid level falls below a preset level.

LANDING GEAR SYSTEM

The landing gear system (figure 1-11) is a tricycle configuration with the main gear retracting into pods suspended below the wing and the nose gear retracting into the fuselage. The nose gear is offset to the right of the aircraft centerline to accommodate the centerline location of the 30mm gun. All three landing gear struts retract forward to provide a free-fall auxiliary extension. The nose gear incorporates a nosewheel steering and shimmy damper unit. Landing gear extension and retraction is controlled by the landing gear handle and powered by the left hydraulic system. In the gear-retracted position, the system is depressurized and isolated. The main gear wheel brakes are differentially operated through pressure control valves that are pressurized from the gear down circuit and mechanically actuated by linkage from the rudder pedals. Automatic main gear wheel braking (anti-spin) occurs on gear retraction.

LANDING GEAR SYSTEM SCHEMATIC

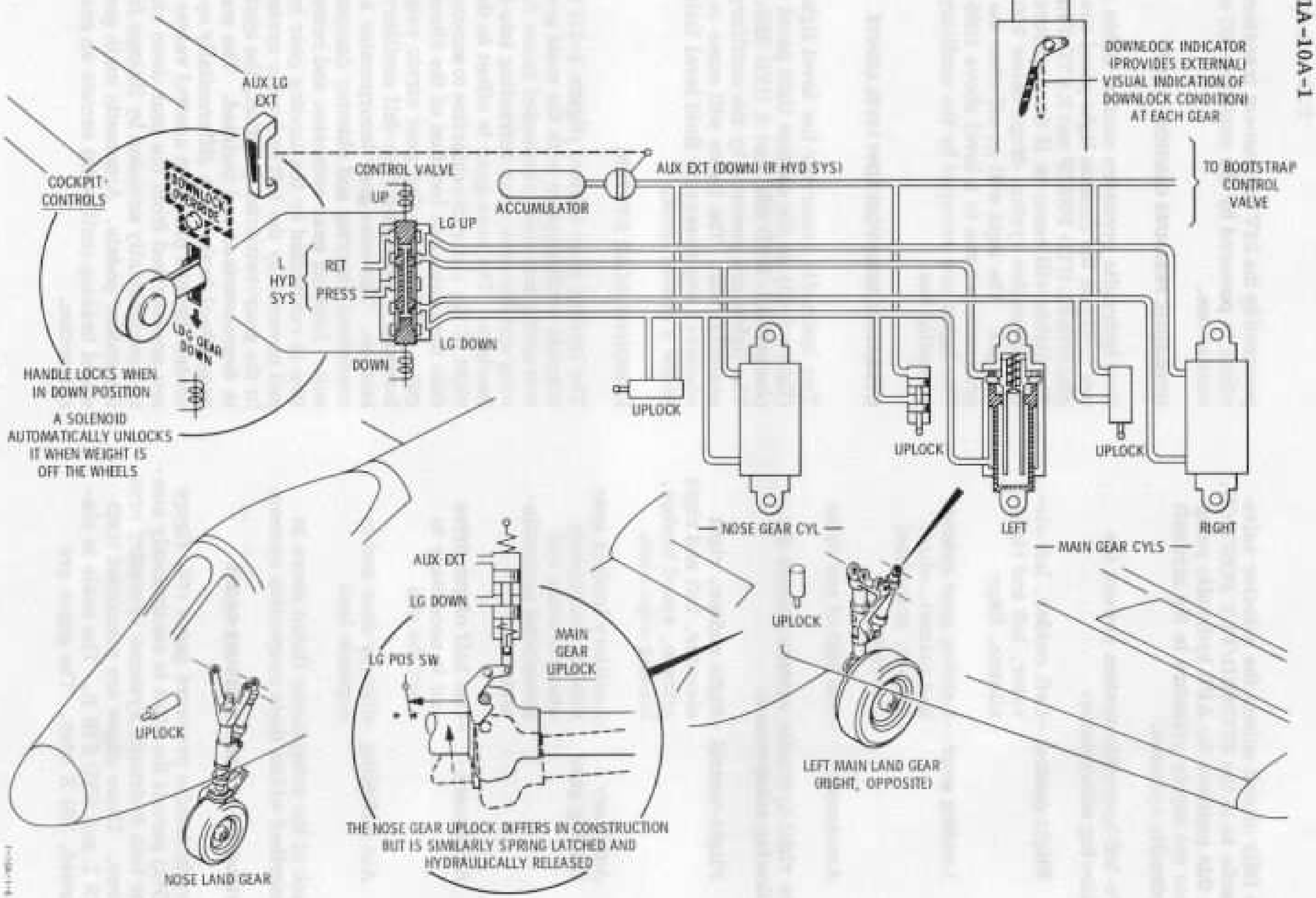


Figure 1-11

Anti-skid control is contained in the normal wheel brake control subsystem. An electrohydraulic valve in each normal brake pressure line regulates the pilot's applied pressure upon command from the wheel skid detection devices. Nosewheel steering is accomplished with an electrohydraulic steering assembly pressurized from the normal gear "down" line. In the event of electrical or hydraulic supply loss, the steering assembly automatically transfers to the shimmy damping mode.

An auxiliary means for extension of the landing gear has been provided in the event left hydraulic system pressure is not present or the landing gear handle or valve is jammed or failed. This is accomplished by providing hydraulic power to actuate the landing gear uplocks, permitting free fall of the gear to the down and locked position. The auxiliary landing gear extension system pressure supply is stored in an accumulator located in the nose wheelwell. This accumulator is pressurized by, but isolated from, the right hydraulic system. To actuate the auxiliary landing gear extension system, the pilot must pull the landing gear auxiliary extension handle to its stop.

When the handle is pulled to its stop, a valve is mechanically actuated directing right hydraulic system pressure to the landing gear uplock actuators releasing the uplocks. If right hydraulic system pressure is not present, the landing gear emergency accumulator automatically serves as the pressure source. Upon release of the uplocks, all three gears will extend by gravity, aided by aerodynamic forces, to the down and locked position, provided left hydraulic system pressure is not present. Should left hydraulic system pressure be present, landing gear extension by the auxiliary system can be accomplished by first opening the LAND GEAR circuit breaker to deactivate the landing gear control valve circuit and then pulling the auxiliary landing gear extension handle.

CAUTION

Minimize use of flight controls and flaps whenever the auxiliary landing gear extension handle is in the out position and left hydraulic system pressure is present to avoid left hydraulic system pump cavitation.

The pulling of the auxiliary landing gear extension handle, in addition to releasing the uplocks, directs the same hydraulic pressure to the left hydraulic system reservoir bootstrap pressure control valve, depressurizing the left hydraulic system reservoir and thereby minimizing the back pressure against which the gear must fall.

Should the auxiliary landing gear handle be pulled, with the LAND GEAR circuit breaker closed and with left hydraulic system pressure present, the landing gear control valve will be electrically powered to the up position as soon as the uplocks are released, and the landing gear will be held in the retracted position by hydraulic pressure, precluding extension by the auxiliary system. Auxiliary landing gear extension can be accomplished when in the manual reversion flight control mode without opening the LAND GEAR circuit breaker, as both the left and right hydraulic pressure systems are shut off in this mode of operation.

The main components of the landing gear system are the main landing gear, nose landing gear, wheel brake system, emergency brake system, anti-skid devices, and a nosewheel steering system. In addition, the landing gear system includes a landing gear position and warning system, and a downlock override control. Switches sense gear and uplock position to provide cockpit indication of gear security and to automatically depressurize the landing gear hydraulic subsystem after retraction.

Isolation of the subsystem reduces overall vulnerability of the hydraulic system.

MAIN LANDING GEAR

The shock strut is a cantilever configuration providing a long/soft stroke for rough field taxi capability. The shock strut retracting cylinder is also the drag brace. The retracting cylinder includes a spring powered mechanical downlock (figure 1-11) which automatically engages both for powered and free-fall gear extensions. Internal switches provide cockpit indication of downlock engagements.

For gear retraction, application of hydraulic pressure unlocks the downlock and then extends the retracting cylinder piston to push (rotate) the gear forward and up. As the gear approaches the upstop, a roller is engaged by a spring loaded uplock hook (see figure 1-11). Also, gear up pressure automatically applies brake pressure to stop wheel rotation before the wheels retract into the gear pods.

For gear extension, hydraulic pressure is applied to hydraulic cylinders to disengage the uplock hooks and simultaneously to the retracting cylinder to pull down the gear. Extend pressure is maintained with the gear handle in the DOWN position.

When retracted, the gear is housed in a pod on the under surface of the wing with approximately one third of the wheel exposed. A spring loaded snubber contacts the tire to prevent air drag rotation of the wheels after gear retraction. One door and two fairings, mechanically supported/actuated by shock strut attachments, seal off the remainder of the gear.

NOSE LANDING GEAR

The nose landing gear operates similar to the main gear and gear actuation. As the shock strut piston extends when weight comes off the tire, an internal shock strut cam centers the nosewheel. Two doors, mechanically actuated by nose strut, seal off the fuselage compartment after gear retraction.

LANDING GEAR HANDLE

The landing gear handle (33, figure FO-1) is wheel-shaped and placarded LDG GEAR DOWN. The handle can only be moved from the DOWN to the UP position when DC essential power is available and the aircraft weight is off the wheels, or when the landing gear DOWNLOCK OVERRIDE button is depressed while moving the landing gear handle to the UP position. The hydraulic retract circuit is automatically depressurized after all gear are up and locked.

The handle must be pulled aft before moving it to the DOWN position.

The time for the gear to extend or retract is approximately 6 seconds.

DOWNLOCK SOLENOID OVERRIDE BUTTON

The downlock solenoid override button (33, figure FO-1), is located on the landing gear control panel and placarded DOWNLOCK OVERRIDE. Depressing the button allows the landing gear handle to be moved to the UP position and retract the gear even if aircraft weight is on the main gear. The button is powered by the DC essential bus.

WARNING

Attempting to retract the gear when the aircraft is on the ground will result in nose gear retraction only.

CAUTION

If the downlock override is used with a broken scissors or uninflated strut, damage to the gear or aircraft could result.

AUXILIARY LANDING GEAR EXTENSION HANDLE

An auxiliary landing gear extension handle (48, figure FO-1), placarded AUX LG EXT, is provided to permit extension of the landing gear in the event left hydraulic system pressure is not present or the landing gear handle or valve is jammed or failed. A button at the top of the auxiliary landing gear extension handle must be depressed before the handle can be pulled out.

Extension of the landing gear by the auxiliary system when left hydraulic system pressure is not present should be accomplished by first placing the landing gear handle in the DOWN position and then pulling out the auxiliary landing gear extension handle. The auxiliary landing gear extension handle should be returned to its stowed position as soon as a landing gear down and locked indication is present.

CAUTION

> Minimize use of flight controls and flaps whenever the auxiliary landing gear extension handle is in the out position and left hydraulic system pressure is present to avoid left hydraulic system pump cavitation.

Extension of the landing gear by the auxiliary system when left hydraulic system pressure is present should be accomplished by first opening the LAND GEAR circuit breaker, then placing the landing gear handle in the DOWN position, and finally pulling out the auxiliary landing gear extension handle. The auxiliary landing gear extension handle should be returned to its stowed position as soon as a landing gear down and locked indication is present to preclude left hydraulic system pump cavitation in the event a heavy demand is imposed upon the system.

CAUTION

> Allow at least 15 seconds to elapse between returning the auxiliary landing gear extension handle to the stowed position and the resetting of the LAND GEAR circuit breaker to provide sufficient time for hydraulic system stabilization after reservoir repressurization to avoid hydraulic pump cavitation.

Landing gear retraction after extension by the auxiliary system with left hydraulic system pressure present should be accomplished by first checking that the auxiliary landing gear extension handle is in its stowed position, then closing the LAND GEAR circuit breaker, and finally raising the landing gear handle to the UP position.

CAUTION

> Allow at least 15 seconds to elapse between returning the auxiliary landing gear extension handle to the stowed position and placing the flight control mode switch in NORM to avoid left hydraulic system pump cavitation.

The landing gear can be retracted after extension by the auxiliary system when in the manual reversion flight control mode, provided left hydraulic system pressure is available. The retraction should be accomplished by first checking that the auxiliary landing gear extension handle is in its stowed position and that the LAND GEAR circuit breaker is closed, then placing the flight control mode switch in NORM, and finally raising the landing gear handle to the UP position.

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM

The landing gear position indicating and warning system consists of three separate green landing gear display lights (32, figure FO-1), red warning lamps within the wheel-shaped knob of the landing gear handle (33, figure FO-1), and an audible warning signal (beeper).

The three landing gear display lights are placarded L SAFE, N SAFE and R SAFE. Each display contains two bulbs and comes on green to indicate the respective gear is down and locked.

When the landing gear is up and locked all display lights are off. When the gear handle is placed to the DOWN position, the warning light and beeper come on and remain on until all three gear are in their locked positions. When the handle is moved to the UP position each safe-down display light will go off and the warning light and beeper will come on and remain on until all gear are in their up and locked positions.

The beeper will sound and the warning light will come on if the following conditions occur concurrently:

- Gear handle in the UP position.
- Below 10,000 ±500 feet MSL (decreasing altitude).
- Below 160 ±5 KIAS.
- A throttle positioned at approximately half of maximum throttle.

The landing gear position indicating and warning system is powered by the auxiliary DC essential bus.

SIGNAL LIGHTS LAMP TEST BUTTON

The signal lights lamp test button (figure 1-51) is a push-button placarded SIGNAL LIGHTS LAMP TEST. Depressing the button causes the landing gear display lights and the landing gear warning light to come on and tests the audible warning signal. The lights coming on test the lamps only and not the complete circuit. The button is powered by the auxiliary DC essential bus.

LANDING GEAR HORN SILENCE BUTTON

The landing gear horn silence button (figure 1-4) on the throttle quadrant, is push-button placarded L/G WRN SILENCE. Depressing the button will silence the beeper. If the beeper sounds due to an unsafe gear and is silenced, it will not sound again until the gear is recycled. If the beeper sounds due to aircraft configuration (gear not down and locked, decreasing altitude below approximately 10,000 ±500 feet, airspeed below 160 KIAS and throttle retarded) and is silenced, it will sound again if the throttle is advanced and again retarded. The button is powered by auxiliary DC essential bus.

NOSEWHEEL STEERING SYSTEM

The nosewheel steering system provides sufficient steering control to effect a 180 degree turn on a 50-foot wide runway. When steering is deenergized ±150° free swivel of the nosewheel is available. The system is pressurized by the left hydraulic system. Damping is provided to prevent nosewheel shimmy in the steering and free swivel modes.

Nosewheel steering can be energized only when the landing gear handle is DOWN and weight is sensed on either main gear. The electrical circuit is a fail-safe design; that is, failure of the circuitry or loss of electrical power will revert the system back to the swivel mode to prevent a hardover. A compensator on the steer/damp unit provides sufficient hydraulic fluid and pressure to retain the shimmy damping function in event of loss of the hydraulic power source.

Note

Nosewheel steering must be engaged, at least momentarily, prior to each flight to insure full charge within the damping mode compensator.

NOSEWHEEL STEERING BUTTON

The nosewheel steering button (figure 1-15) is located on the control stick grip. On aircraft prior to serno 79-0167, the button also functions as the HARS fast-erect button. On aircraft serno 79-0167 and subsequent, the button also functions as an airborne INS mark/update button. Steer engagement is indicated by a green STEERING ENGAGED light coming on, located at the top of the instrument panel.

Auxiliary DC essential bus power arms the engage switch when weight is on either main gear. Subsequent press and release of the button engages the steering mode. When in steering mode, a press and release of the button disengages steering mode. A sustained press of the button, regardless of sequence, engages steering mode.

Any interruption of electrical power disengages the steering mode until the engage switch is rearmed and the button is again pressed. Thus, on landings, steering is automatically disengaged until the button is pressed after main gear ground contact.

In flight, the nosewheel steering button serves only as the HARS fast-erect button. Authority transfer is automatic as weight comes off both main gear.

NOSEWHEEL STEERING ENGAGED ADVISORY LIGHT

The nosewheel steering advisory light (14, figure FO-1) placarded STEERING ENGAGED, will come on to indicate the nosewheel steering system is activated. The light is powered by auxiliary DC essential bus.

WHEEL BRAKE SYSTEM

The normal wheel brake system is fully powered from the left hydraulic system landing gear-down circuit.

During landing gear retraction hydraulic retract pressure is directed to separate ports on the brake valves and produces sufficient brake pressure to stop the wheels prior to wheel engagement of the snubbers within the pods. This brake pressure is automatically released when the landing gear retract circuit is unpressurized after gear safe indication.

EMERGENCY BRAKE SYSTEM

In event of a failure in the left hydraulic system, emergency braking power is provided by an accumulator serviced by but isolated from the right hydraulic system. In event of loss of both hydraulic systems, sufficient accumulator fluid pressure is available for at least three full brake applications.

The system is activated by pulling the emergency brake handle, which is mechanically connected to a selector valve and then actuating the brake pedals. Pulling the handle also actuates a switch which disables the anti-skid control system.

With the left hydraulic system failed and the right hydraulic system operative the emergency brake system has the same capabilities as the normal system without anti-skid. The emergency braking system is fully independent of the normal system down to but not including the wheel brake cylinder. In event normal pressure (left hydraulic system) becomes available while emergency braking is selected, the emergency system retains control of the brakes.

EMERGENCY BRAKE HANDLE

The emergency brake handle (1, figure FO-2) is a manual control placarded EMERG BRAKE. The emergency brake system is engaged by pulling the emergency brake handle aft which mechanically positions an emergency brake selector valve which directs pressure from the accumulator to the emergency half of the power brake valves. If the right hydraulic system is intact, unlimited braking will normally be available. However, on aircraft prior to serno 75-00295 not modified by T.O. 1A-10-599, an isolation fuse is installed in the pressure line to the emergency brake and landing gear accumulators. This fuse is installed to provide protection for the right hydraulic system in the event of a leak or combat damage in the nose wheelwell where both systems are in close proximity. Closing of the isolation fuse will limit the emergency braking to a minimum of three full brake applications (accumulator volume). Since there is no cockpit indication of fuse closing, it is recommended that the pilots always employ the emergency brakes as though only the accumulator volume is available. When the emergency brake handle is pulled, the anti-skid control system is deactivated.

ANTI-SKID CONTROL SYSTEM

The anti-skid control system enables efficient maximum braking for all runway conditions. During light and moderate braking, the anti-skid control system usually does not operate. During heavy braking, however, the anti-skid control automatically limits brake pressure to the skid threshold regardless of how much pedal force is exerted.

The anti-skid control does not function below 10 knots groundspeed.

Cockpit controls and displays consist of an engage switch, an emergency disengage switch and a caution light. The anti-skid control system reduces pressure to both brakes regardless of which wheel experiences the incipient skid.

Locked wheel/touchdown protection circuitry is integrated in the control unit which provides safeguards, in addition to the anti-skid control circuitry, as follows:

- Prevents application of brake pressure until both wheels have spun up after touchdown. Either or both squat switches arm the protection circuit to maintain a pressure release until wheels spin up to 25 knots.

- Provides an overriding pressure release signal in the unusual conditions, such as hydroplaning, when the normal anti-skid control circuit allows a wheel to decelerate to less than one half aircraft speed. The pressure release signal is maintained until wheel speed recovers to near aircraft equivalent speed.

- In event a squat switch fails to activate after touchdown the spin-up wheel signal overrides the switch failure and normal skid-control performance is available until aircraft speed reduces to approximately 15 knots.

CAUTION

At speeds below 15 knots with the anti-skid system engaged and a squat switch failure, both brakes will release and the pilot must disengage anti-skid control to regain braking. For this failure mode the caution light does not come on.

ANTI-SKID SWITCH

The anti-skid switch (30, figure FO-1) is placarded ANTI-SKID and OFF. The switch must be manually moved to the ANTI-SKID setting where it is electrically held. The switch can be manually moved to OFF and is electrically released to OFF whenever:

- Emergency disconnect lever is actuated
- Emergency brake handle is pulled
- The auxiliary DC essential bus is de-energized.

When the landing gear is raised the anti-skid control elements are deenergized however the switch remains engaged. The OFF position deactivates the system and causes the ANTI-SKID caution light to come on if the landing gear handle is in the DOWN position.

Note

If the anti-skid system is turned on prior to engine start it may drop off the line when the APU is started. This is due to the normal high voltage drop as the result of the battery drain for an APU start.

ANTI-SKID CAUTION LIGHT

The anti-skid caution light (figure 1-51) is placarded ANTI-SKID. The light serves two functions:

- Indicates the anti-skid system is not engaged when the landing gear handle is DOWN.
- Indicates anti-skid system has automatically deactivated in response to a self-detected failure.

Subsequent to failure indication the anti-skid switch remains in the ANTI-SKID position. The light is powered by the auxiliary DC essential bus.

CAUTION

The anti-skid system should not be reactivated for normal use after a failure indication. Certain failure modes, either intermittent in nature or associated with specific braking conditions, may not be immediately reindicated after the anti-skid switch is recycled.

EMERGENCY DISCONNECT LEVER

The emergency disconnect lever (figure 1-51) is located on the forward side of the control stick just below the grip. Momentary actuation of the lever immediately deactivates both the anti-skid and SAS subsystems on the ground or during flight and the anti-skid and SAS switches return to the OFF position.

CAUTION

The emergency disconnect lever should be immediately actuated when anti-skid failure is suspected. To avoid skidding at the instant anti-skid is deactivated, relieve brake pedal force momentarily as the emergency disconnect lever is depressed, then reapply brake pedal force. All possible failures are not indicated by the caution light.

PRIMARY FLIGHT CONTROL SYSTEM (PFCS)

Dual redundant surfaces are provided for control of the aircraft in the pitch, roll, and yaw axes. Pilot commands are transmitted from the stick via dual redundant parallel run cables to the elevators and ailerons. Pedal commands are transmitted by a single mechanical path. Loss of one hydraulic power source does not affect pitch and roll response but does cause moderate increase in pedal force gradient for yaw. Jams in any pitch or roll control surface or command path may be isolated to free the control stick for control of the unjammed surface.

Dual redundant control circuits are provided for trim controls in all three axes.

Dual redundant stability augmentation systems (SAS) provide rate damping in the pitch and yaw axes. The SAS includes crossfeed from the speed brakes to the elevators and an aileron/rudder interconnect (ARI). Fail-safe provisions include automatic SAS shutdown in response to a failure of any part of the dual systems.

CONTROL STICK

The conventional control stick (figure 1-15) is connected, by push-pull rods and cable assemblies, to hydraulic control valves at the roll and pitch control actuators. Movement of the stick positions these control valves so that pressure from the appropriate hydraulic systems is directed to the actuators to move the control surfaces. A follow-up system automatically closes the control valves when the desired movement of the control surfaces has been obtained. Stick authority at the surfaces actuators exceeds SAS authority.

In the event of loss of both hydraulic power supplies, valve motion is automatically locked out and the stick is directly connected to the elevators and to the aileron tabs when the flight control mode switch is positioned to MAN REVERSION.

PITCH CONTROL SYSTEM

The pitch control (figure 1-12) is affected with two independent elevators connected together with a cross over, powered by independent actuators and hydraulic power sources, and controlled by two widely separated cable and linkage transmission paths which combine and connect to the control stick.

The powered controls are irreversible. Stick feel is provided by redundant spring cam devices located close to the surface actuators, and a bob weight located in the armored cockpit compartment. Redundant control circuitry is provided for trim control.

Loss of one hydraulic power source or one mechanical control path has no discernible effect on stick/surface response. Electrically operated mechanical disconnects in the cockpit and frangible torque shaft connections at the input and output connections of the surface actuators provide means to identify and isolate a jammed elevator surface or control path downstream of the elevator disconnectors.

The input arms of both elevator actuators are interconnected by a transverse torque shaft which includes a shear rivet. The cross over connection enables both actuators to be commanded in unison when either disconnect is actuated and in event of a separation type failure of one control path. In event of a jam of pitch controls on either side, relatively low pilot effort is sufficient to shear the rivet. This action in combination with pilot operation of the disconnector frees the stick for control of the unjammed system.

The artificial feel devices consist of spring loaded rollers which resist rotation of cammed cranks to provide a force/deflection gradient at the stick. Operation of the feel device changes the zero load position of the stick (and elevator position) to effect pitch trim. Two independent, redundant electrical control circuits lead from the cockpit to the trim actuator, the normal pitch/roll circuit, and the emergency override pitch/roll circuit.

Geared/trimmable tabs are mounted on the outboard trailing edges of both elevators. The tabs are only required for the MRFCS and not required in the powered mode. However, the geared function (-1:1) is also active in powered flight control modes. The geared tab motion reduces elevator aerodynamic hinge moments at all times to levels satisfactory for instantaneous transfer of pitch control mode from powered to manual. Refer to PITCH MRFCS section for a description of pitch manual reversion operation.

Two identical and independent pitch SAS are installed which provide rate damping to enhance tracking and speed brake/elevator crossfeed signals to minimize pitch transients during speed brake deployments.

Elevator Emergency Disengage Switch

A three-position switch (figure 1-16) placarded ELEVATOR EMER DISENGAGE, is mounted on the emergency flight control panel. The switch is normally centered. In the event of a jam of the elevator controls downstream of either elevator disconnector, a L/R ELEV light, adjacent to the switch will come on as the pilot exerts abnormal stick force countering the jam. The stick is disconnected from the jammed side by moving the switch toward the light. The stick becomes immediately free to control the operable surface. However, stick force gradient is momentarily moderately higher than normal until the controllable elevator is displaced approximately 3 degrees relative to the jammed surface at which point the cross over connection shaft at the two elevator actuator input levers is sheared. Thereafter, the pilot experiences normal stick force per g, but the stick has to be moved or trimmed twice as much for a given maneuver. When a control path is not fully engaged at the disconnect an ELEV DISENG light on the caution light panel comes on.

WARNING

Flight control jams that cannot be identified by a jam indicator light cannot be overcome by disengaging a flight control path.

When the switch is subsequently moved to neutral setting or to disengage the opposite side, a latch becomes spring loaded for reengagement and reengages as soon as the stick is moved in alignment with surface position. The disengage circuits are powered by the DC essential bus.

Elevator Disengaged Caution Light

The elevator disengaged caution light (figure 1-51) on the caution light panel is placarded ELEV DISENG. The light coming on indicates that either or both elevator control paths are not fully engaged at the disconnect units. The light is powered by the auxiliary DC essential bus.

Elevator Jam Indicator Lights

The elevator jam indicator lights (figure 1-16) on the emergency flight control panel are placarded L ELEV and R ELEV and bracket the disengage switch. The lights are actuated by the flight control jam circuit from the auxiliary DC essential bus.

PITCH CONTROL SYSTEM SCHEMATIC

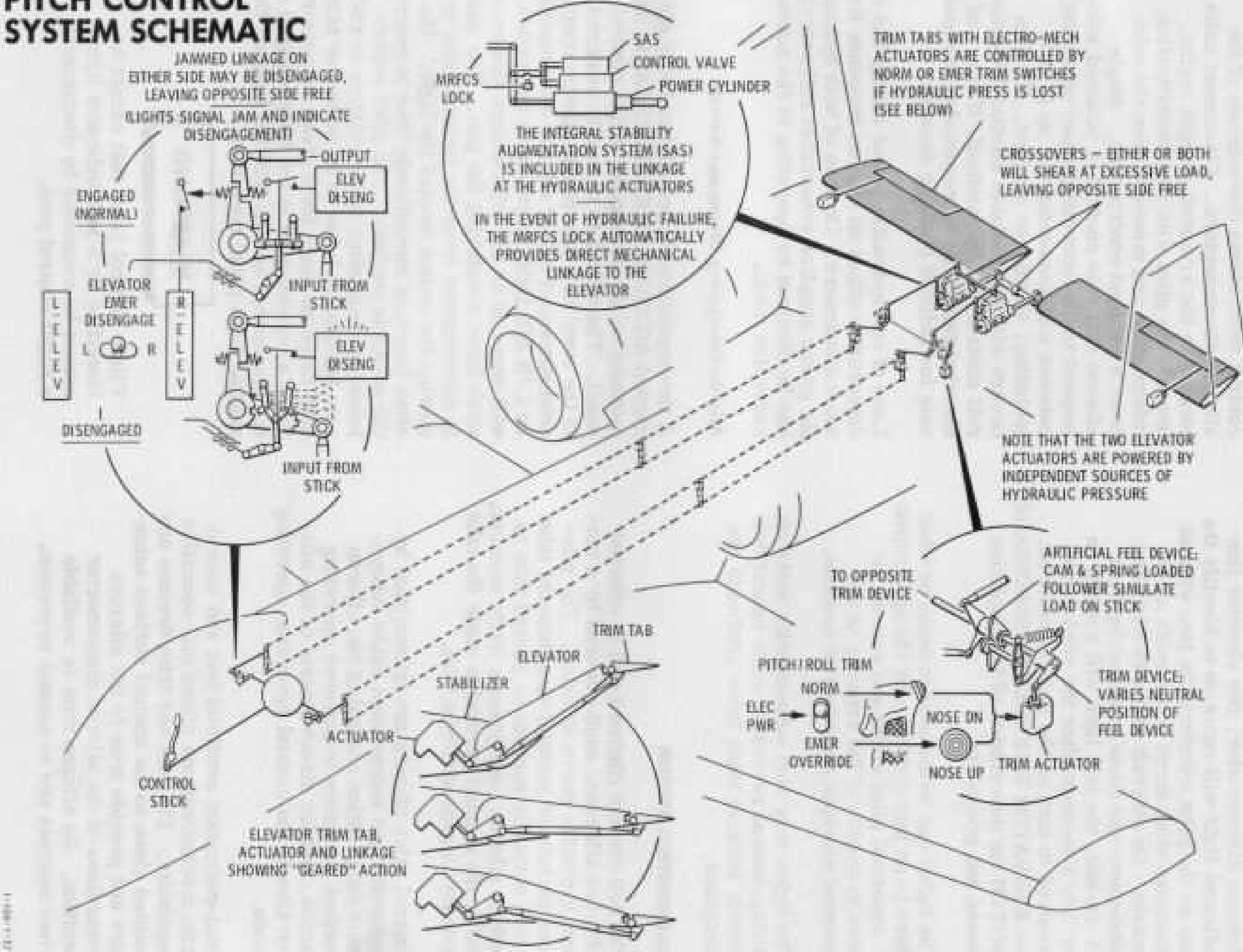


Figure 1-12

In event of an actual jam downstream of an elevator disconnector, one elevator jam indicator light will come on to identify the jam as the pilot counters the jam with an abnormal stick force of as much as 65 pounds. On aircraft serno 79-0167 and subsequent and those modified by T.O. 1A-10-883, the jam light will remain on for 3 to 5 seconds after the stick force required to turn on the light is reduced. See ELEVATOR EMERGENCY DISENGAGE SWITCH for description of jam isolation means and procedure.

The lights can be induced to come on when there is no jam condition by the application of abnormal pilot effort and rate at the stick in excess of the capacity of the powered elevator actuators to respond.

The lights may also momentarily come on during manual reversion mode operations merely because stick force gradients are increased.

ROLL CONTROL SYSTEM

The roll control (figure 1-13) is effected with two ailerons, each powered by a tandem hydraulic servo actuator. The ailerons include 50 percent span inboard tabs which normally operate as geared tabs to reduce aileron aerodynamic hinge moment, and in the manual reversion mode the tabs operate as geared/servo tabs.

Lateral stick signals are transmitted by separated and independent cable subsystems for position control of the aileron surface actuators in powered operating modes, and for direct control of the aileron tabs in the manual reversion operating mode.

Dual-redundant artificial feel and lateral trim mechanizations afford fail-operative capability. The feel and trim devices are located close to the control surface actuators and provide trim to the ailerons regardless of the aileron disconnector position. No aileron trim is available when controls are in manual reversion.

Jam protection is afforded for both roll control subsystems downstream of the aileron disconnectors. Disconnect units, located in the armored cockpit region, provide the pilot with jam identification information and means to free the stick from a jammed control side. Single aileron roll rate control is retained subsequent to disconnecting a jam occurring between a disconnector and up to and including, an aileron surface. If a jam occurs with appreciable aileron deflection, roll control will be minimal, if any, and elevator input will be required to compensate for induced pitching moment.

Loss of one hydraulic power supply has no discernible effect on aileron response for most maneuvers. On loss of both hydraulic power supplies reasonable roll control can be obtained by operating in the manual reversion mode.

Aileron Emergency Disengage Switch

A three-position switch (figure 1-16) placarded AILERON EMER DISENGAGE, is mounted on the emergency flight control panel. The switch is normally centered. In the event of a jam of the aileron controls downstream of either aileron disconnector an L/R AIL light, adjacent to the switch, will come on as the pilot exerts abnormal stick force countering the jam. The stick is disconnected from the jammed side by moving the switch toward the light. The stick becomes immediately free to control the operable surface. The pilot experiences normal stick force relative to roll rate but the stick has to be moved or trimmed twice as much for a given maneuver. When a control path is not fully engaged at the disconnect an AIL DISENG light on the caution light panel comes on.

WARNING

Flight control jams that cannot be identified by a jam indicator light cannot be overcome by disengaging a flight control path.

ROLL CONTROL SYSTEM SCHEMATIC

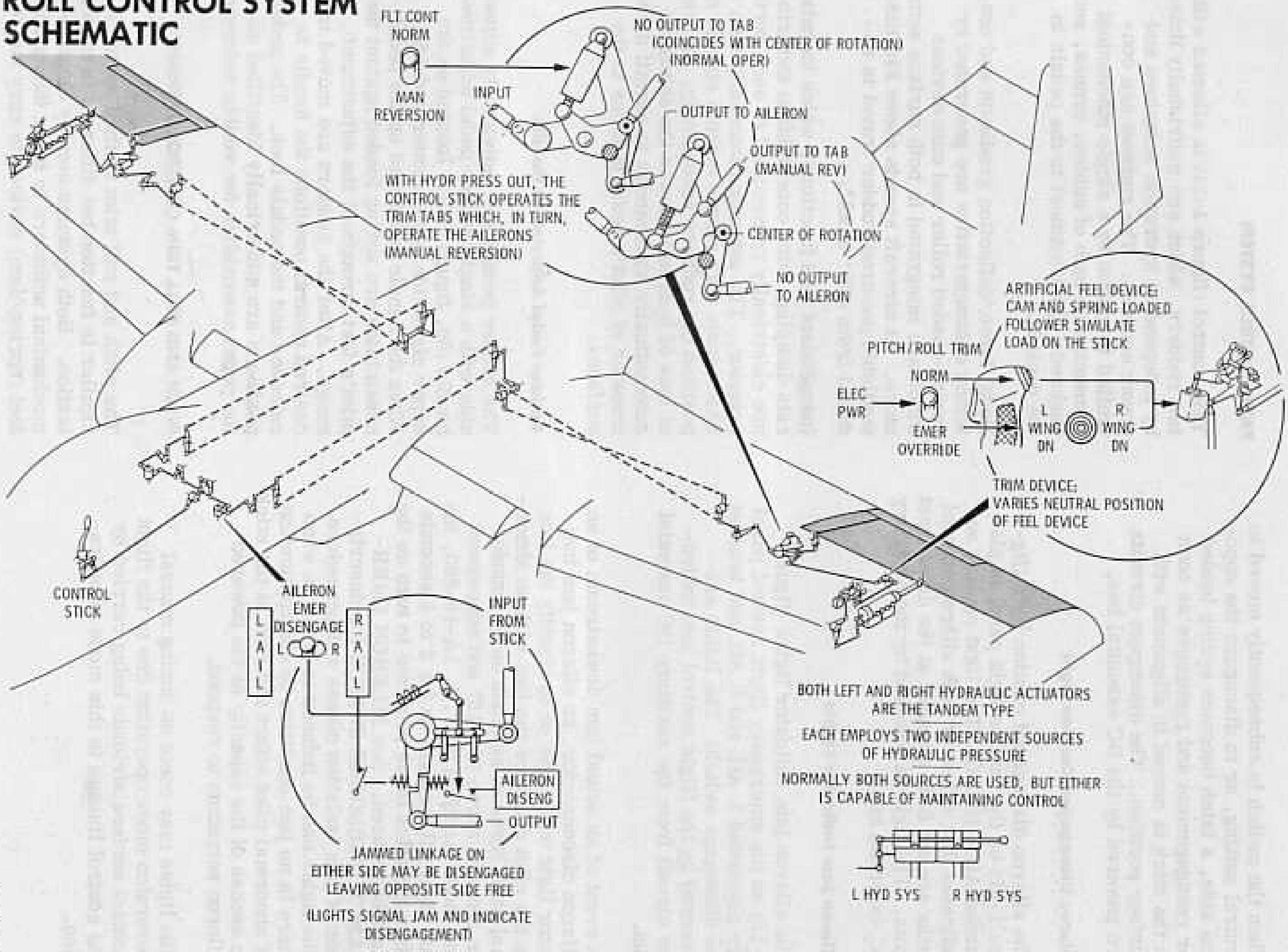


Figure 1-13

When the switch is subsequently moved to neutral setting, or to disengage the opposite side, a latch becomes spring loaded for reengagement and reengages as soon as the stick is moved in alignment with surface position. The disengage circuits are powered by the DC essential bus.

Aileron Disengaged Caution Light

The aileron disengaged caution light (figure 1-51) on the caution light panel is placarded AIL DISENG. The light coming on indicates that either or both aileron control paths are not fully engaged at the disconnect units. The light is powered by the auxiliary DC essential bus.

Aileron Jam Indicator Lights

The aileron jam indicator lights (figure 1-16) on the emergency flight control panel are placarded L AIL and R AIL and bracket the disengage switch. The lights are powered by the flight control jam indicator circuit from the auxiliary DC essential bus.

In event of an actual jam downstream of an aileron disconnector, an aileron jam indicator light will come on to identify the jam as the pilot counters the jam with an abnormal stick force of as much as 50 pounds. On aircraft serno 79-0167 and subsequent and those modified by T.O. 1A-10-883, the jam light will remain on for 3 to 5 seconds after the stick force required to turn on the light is reduced. See AILERON EMERGENCY DISENGAGE SWITCH for description of jam isolation means and procedure. The lights can be induced to come on when there is no jam condition by the application of abnormal pilot effort and rate at the stick in excess of the capacity of the powered aileron actuators to respond.

The lights may come on during manual reversion mode operation due to the flight control surface airloads being carried by the control linkage in this mode of operation.

YAW CONTROL SYSTEM

Yaw control (figure 1-14) is effected with two rudders, which are individually driven by independent hydraulic actuators and power sources. The rudders are controlled in unison by a single mechanical transmission path of cables, cranks, and pushrods which connect to the pedals in the cockpit.

Pedal force/deflection gradients and centering characteristics are generated by spring-loaded roller and cam devices which are integrated in both surface actuators. At aircraft speeds above 240 KIAS available powered rudder travel is reduced from ±25° to ±8°.

Independent SAS functions, which include rate damping, turn coordination and trim, are electrically transmitted to each surface actuator. The actuators internally sum the SAS signals with pilot's manual signals and position the rudders accordingly. In event of loss of hydraulic power the actuators automatically shift modes to permit direct transfer of pilot's pedal motions to the surfaces.

Rudder Pedal Adjustment Handle

The rudder pedals are individually adjustable with a single rudder pedal adjustment handle (50, figure FO-1) located on the upper right side of the center pedestal. When the handle is rotated up the pedal assemblies are spring loaded against the pilot's feet throughout the adjustment range. After the rudders are moved to the desired neutral positions the handle is released and the pedals lock. The pedal positions are numerically identified on the pedal assemblies for visible reference.

PITCH AND ROLL TRIM CONTROL SYSTEMS

The pitch and roll trim control systems are similar in function but different in mechanization. Both systems employ electromechanical actuators to index the artificial feel (spring/cam) devices to vary the zero

YAW CONTROL SYSTEM SCHEMATIC

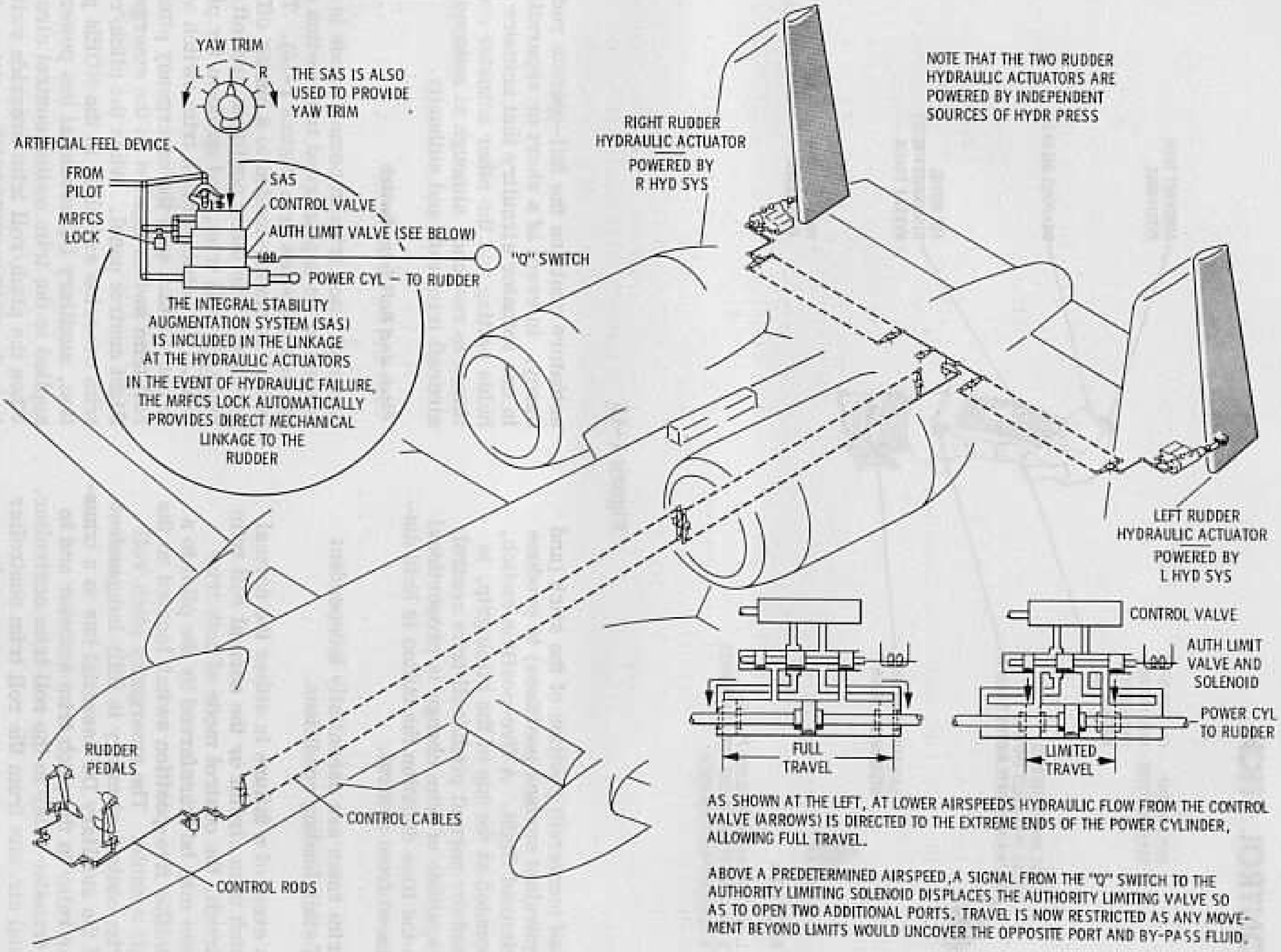
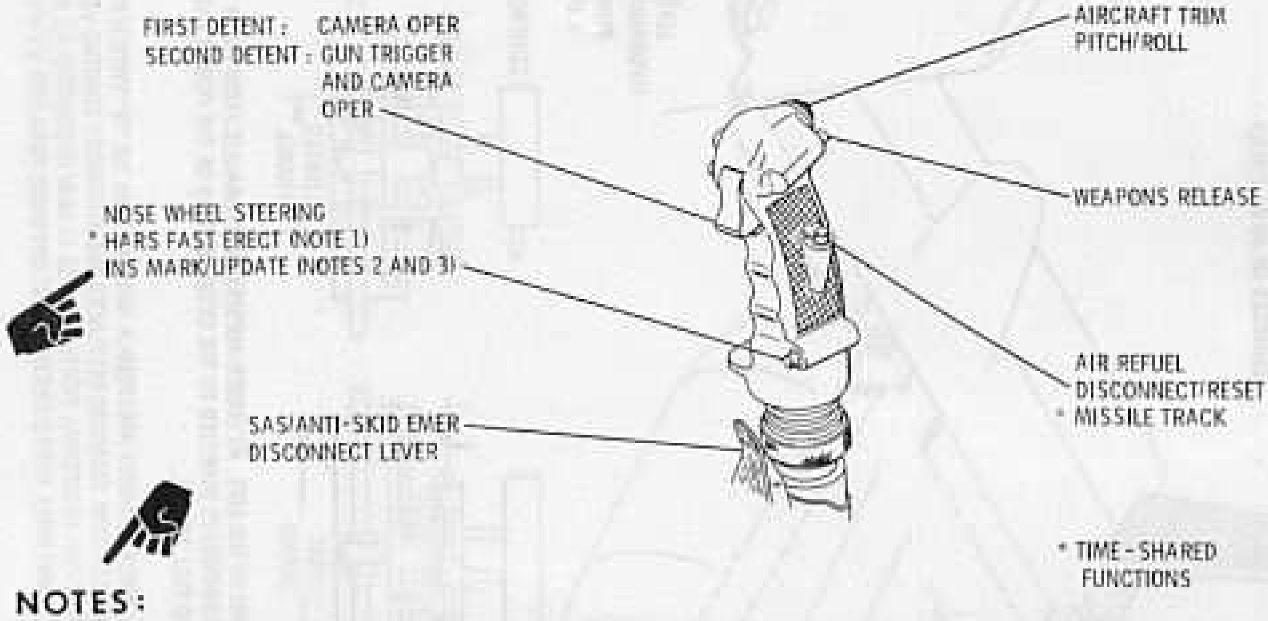


Figure 1-14

CONTROL STICK



NOTES:

1. AIRCRAFT PRIOR TO SERNO 79-0167.
2. AIRCRAFT SERNO 79-0167 AND SUBSEQUENT.
3. AIRBORNE FUNCTION ONLY.

Figure 1-15

load (centered) position of the stick (and equivalent surface positions) to produce trimmed flight. A five-position switch, mounted at the top of the stick grip, is used for normal pitch/roll trim control. Magnitude of trim change is proportional to the time duration the button is held displaced from neutral.

Trim rates are essentially independent of stick loading conditions.

In event of a failure in either the normal pitch trim circuit or the normal roll trim circuit, the control mode of both trim axes may be transferred by the pilot to a similar five-position switch located on the left console. The emergency pitch/roll trim control circuitry is fully independent of the auxiliary DC essential bus to a transfer relay in the pitch trim actuator and to a transfer relay in the roll trim controller. Dual circuits from the roll trim controller to two independent stepper motor trim actuators continue the fail-operate redundancy. In event of a short or separation in one actuator circuitry that actuator holds setting and the other actuator continues to respond, although at reduced aircraft trim rate and authority.

Pitch and Roll Trim Switch

Aircraft pitch trim in normal mode is controlled by an unplacarded trim switch on the control stick grip (figure 1-15). The switch is spring-loaded to a center off position. The other positions are left/right for roll trim, and up/down for pitch trim. The control stick trim switch works in conjunction with the emergency pitch/roll trim switch located on the emergency flight control panel. When the pitch/roll trim override switch is in the NORM position, auxiliary DC essential bus power is applied to the trim switch control circuits. When the pitch/roll trim override switch is in the EMER OVERRIDE position, the

EMERGENCY FLIGHT CONTROL PANEL

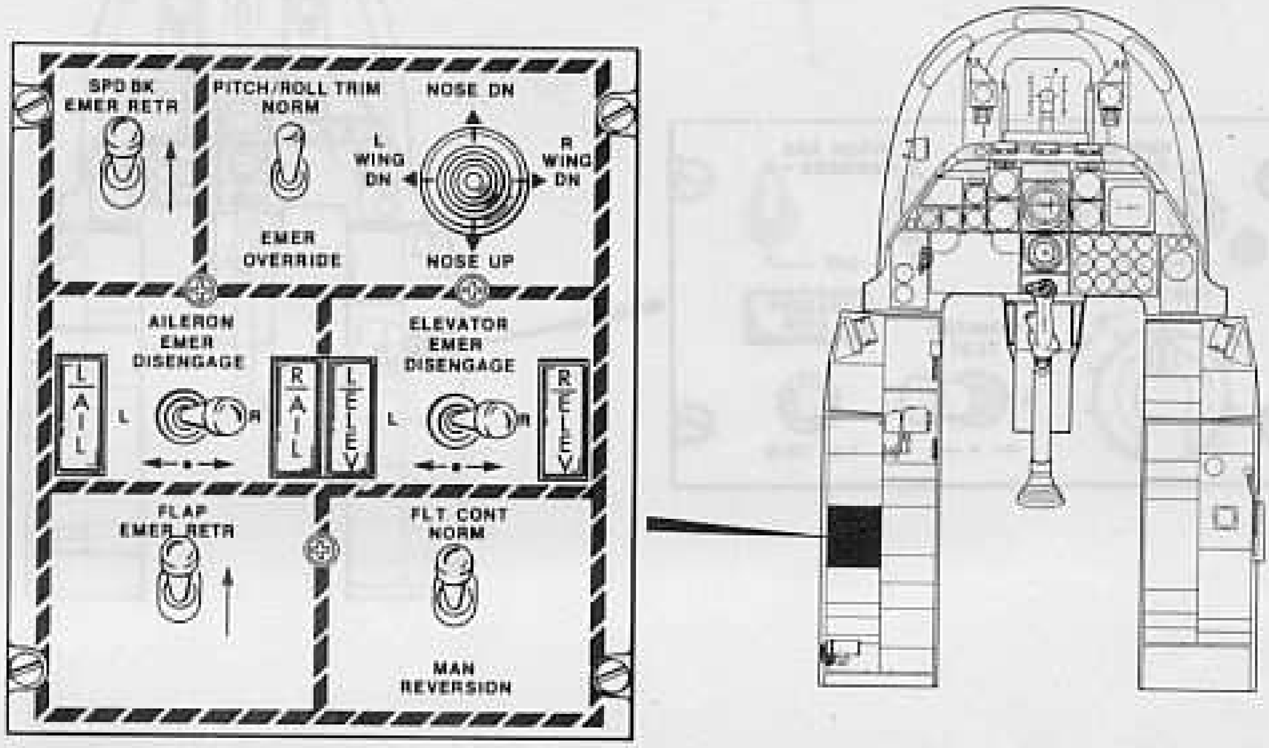


Figure 1-16

control stick trim switch is deenergized and trim control is shifted to an identical trim control switch on the emergency flight control panel.

Pitch and Roll Trim Override Switch

The pitch and roll trim override switch (figure 1-16), placarded PITCH/ROLL TRIM, is a two-positioned toggle switch located on the emergency flight control panel. When set to the NORM position, auxiliary DC essential bus power is applied to energize the five-position pitch and roll trim switch on the control stick grip, permitting normal trim control. When the pitch/roll trim override switch is set to the EMER OVERRIDE position, DC essential bus power is applied to the five-position emergency pitch and roll trim switch located on the emergency flight control panel. The pitch/roll trim override switch must be in the NORM position for the TAKEOFF TRIM pushbutton switch on the SAS control panel to be effective.

Emergency Pitch and Roll Trim Switch

The emergency pitch and roll trim switch (figure 1-16), unplacarded, located on the emergency flight control panel, is identical to the pitch/roll trim switch on the control stick grip. The switch has a spring-loaded center OFF position, and four placarded positions: NOSE DN, NOSE UP, L WING DN, and R WING DN. The L WING DN and R WING DN positions provide emergency roll trim control. The longitudinally-operated NOSE DN and NOSE UP provide emergency pitch trim control. The

STABILITY AUGMENTATION SYSTEM PANEL (SAS)

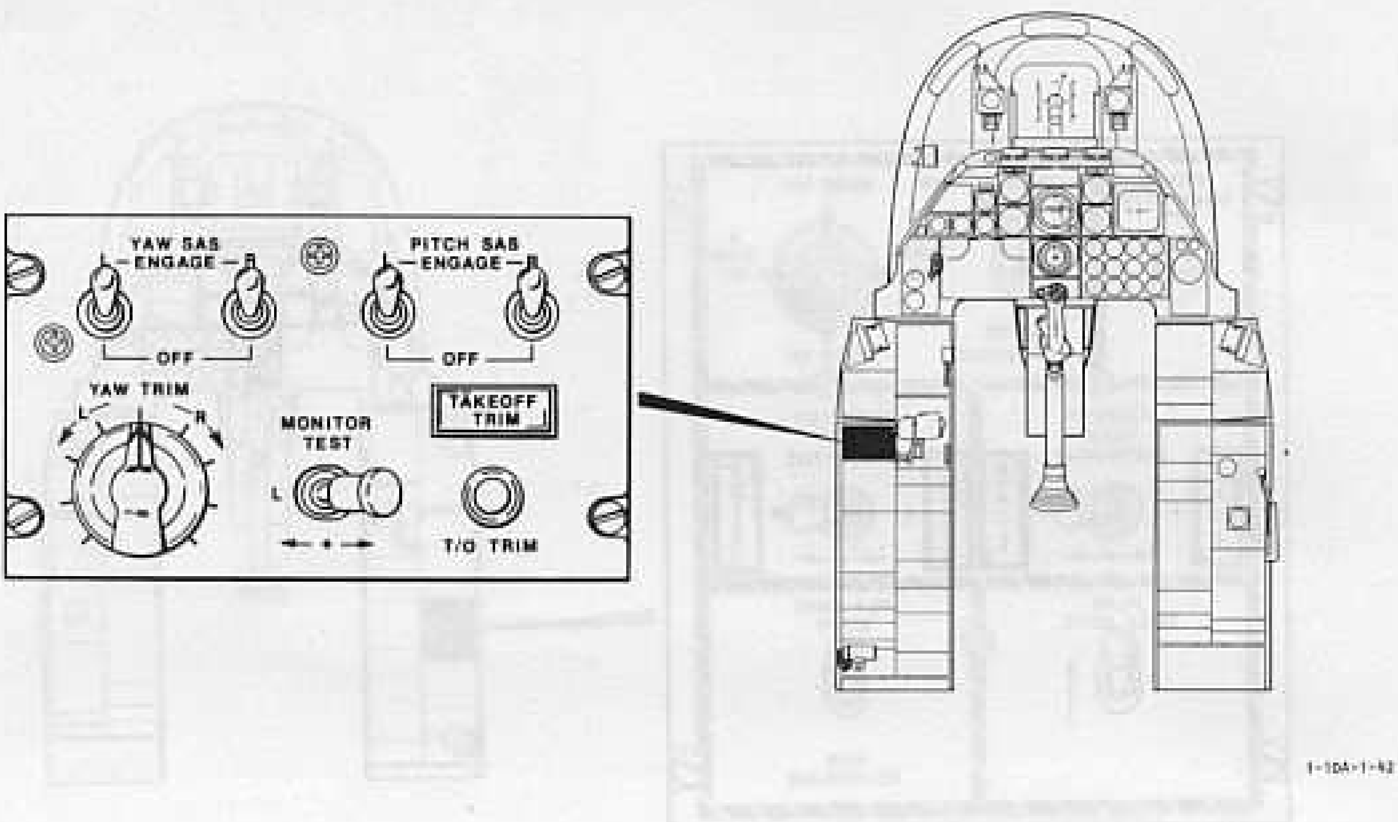


Figure 1-17

emergency pitch and roll trim switch is activated by DC essential bus power when the adjacent pitch/roll trim override switch is in the EMER OVERRIDE position. Normal trim control will revert to the trim switch on the control stick grip when the pitch/roll trim override switch is set to the NORM position.

YAW TRIM CONTROL KNOB

Yaw trim control is effected with a knob (figure 1-17) located on the SAS control panel on the left console placarded YAW TRIM. The single knob controls two independent bias circuits which are individually summed in the two yaw SAS resulting in equal trim displacements of both rudders. Zero pedal load positions do not change as trim is commanded. The knob position indicates the degree of trim commanded. Rudder deflections are limited to ±10° at speeds below 240 KIAS and ±8° above 240 KIAS. A mechanical detent is provided in the zero trim position. The yaw trim system is powered by the right DC and AC busses.

Note

- The YAW TRIM control must be in the zero position to cause the TAKEOFF TRIM light to come on after depression of the T/O TRIM button.
- The yaw SAS must be engaged to obtain operation of the yaw trim function.

In event of loss of one SAS or one hydraulic power supply, 50 percent yaw trim authority is retained with one SAS engaged.

TAKEOFF TRIM CONTROL SYSTEM

When the T/O TRIM button (figure 1-17) is depressed, the roll and pitch actuators and the two-pitch trim tab actuators are driven to a neutral deenergized setting. When the

T/O TRIM button is depressed, the yaw trim knob is in neutral setting, and the five actuators are at neutral setting, a TAKEOFF TRIM indicator light comes on. The T/O TRIM circuitry does not operate when pitch/roll trim is switched to emergency mode. This characteristic provides assurance that the normal pitch/roll trim control at the stick grip is activated prior to takeoff. The takeoff trim circuit is powered by the auxiliary DC essential bus.

CAUTION

> The trim to neutral indication circuitry is the only positive check the pilot has that the pitch trim tab actuators are properly positioned for a potential inflight transfer to the manual reversion flight control mode.

Takeoff Trim Switch

The takeoff trim switch (figure 1-17), placarded T/O TRIM, is a push-button switch located on the SAS control panel. When the T/O TRIM push-button switch is depressed, the elevators, ailerons, and the elevator trim tabs are trimmed simultaneously to the proper takeoff neutral-trim position utilizing auxiliary DC essential bus power.

The push-button must be depressed until the TAKEOFF TRIM indicator light comes on, indicating that the surfaces have reached the desired position. When the T/O TRIM push-button is released, the TAKEOFF TRIM indicator light will go off. Prior to activating the takeoff trim system, the pitch/roll trim override switch on the emergency flight control panel must be set to the NORM position, and neutral yaw trim selected at the YAW TRIM rotary selector switch on the SAS control panel.

Takeoff Trim Indicator Light

The takeoff trim indicator light (figure 1-17), placarded TAKEOFF TRIM, is located on the SAS control panel. The TAKEOFF TRIM indicator light indicates that all trim surfaces have achieved proper trim for takeoff. The yaw trim control switch must be in neutral and the pitch/roll trim override switch in NORM position before TAKEOFF TRIM ready indicator will come on. The TAKEOFF TRIM ready light is energized by the auxiliary DC essential bus.

STABILITY AUGMENTATION SYSTEM (SAS)

The SAS serves the enhance flying qualities for target tracking, reduce pilot workload, and provide yaw trim capability.

Two SAS channels are provided in both the pitch and yaw axes. Each channel controls an electrohydraulic servo valve incorporated within the respective control surface actuator which in turn actuates the SAS piston of the actuator. Performance of the individual channel is continuously monitored by a computer which compares displacement of the SAS pistons within the surface actuators in the same axis. In the event of an excessive difference, the computer deactivates both channels in the affected axis, and triggers an associated light on the caution annunciator panel. The operable SAS channel can be reenergized by the switch on the SAS panel to permit single channel operation.

The pitch and yaw SAS failure monitor circuits can be tested by using the monitor test switch on the SAS panel.

An emergency disconnect lever, located immediately below the stick grip, disengages all SAS operation when momentarily operated in flight. The pilot's stick and pedal authorities are in excess of the total SAS authorities at the actuators. Hydraulic power is required at the surface actuator for realization of any flying quality enhancement provided by a SAS channel. The aircraft can be safely flown throughout its flight envelope with SAS disengaged. The SAS is powered by the right AC and DC system bus.

Monitor Circuit Test Switch

A test switch on the SAS control panel (figure 1-17), placarded MONITOR TEST, is used to operationally check both pitch and yaw SAS failure monitor circuits. The switch is three-positioned, spring loaded to the midposition where it is lever-locked. When the switch is held to the L position a circuit is introduced in the pitch and yaw left SAS channels to simulate a failure and the monitor circuits disengage all SAS. To complete the test SAS is reengaged and the lever is held at the R setting which exercises the right SAS channels to simulate a failure and effect a disengagement of all SAS.

Emergency Disconnect Lever

The emergency disconnect lever (figure 1-15) is located on the forward side of the control stick below the control grip. The lever functions as an anti-skid and SAS system disconnect lever. Although unplacarded, the lever is identifiable by yellow and black striping. In flight or on the ground the emergency disconnect lever is provided for immediate disengagement of both pitch and yaw axes of the SAS system and the anti-skid system. The anti-skid switch and all SAS switches will return to the OFF position.

Pitch SAS

The pitch SAS performs three basic functions: pitch rate damping, pitch trim compensation for speed brake deployment, and pitch trim compensation for gun firing.

Total SAS authority is mechanically limited within the surface actuators to 2° elevator T.E. up and 5° elevator T.E. down.

The monitor circuit senses differential between pitch SAS piston positions of the left and right elevator actuators and shuts off pitch SAS when the differential exceeds the equivalent of 0.5° difference in displacement of the left and right elevators. Control stick authority is more than sufficient to override a SAS-induced elevator displacement.

Each pitch SAS channel can be reengaged, subsequent to a commanded or uncommanded disengagement. Flight with a single pitch SAS channel engaged is not recommended, due to the repetitive loading of the elevator interconnect shear rivets during single channel pitch SAS operation and the possibility of an undesirable pitch transient should a SAS hardover failure occur in the active channel. A single pitch SAS channel will drive both elevators when only one hydraulic power source is available. The pitch transient from a SAS hardover failure during single channel operation will be greatest during flight when the control stick is rigidly restrained by the pilot and only one hydraulic power source is available. During such flight, a SAS hardover failure which drives the elevators down, to the limit of SAS authority, could result in "G" forces, in a relatively short period of time, which are in excess of the structural limitations of the airframe. Two types of events will result in an uncommanded SAS disengagement.

- Excessive differential between pitch SAS piston positions as a result of failure in a pitch SAS channel:

  This type event is identified by noting that both hydraulic pressure gage readings are normal and that the control stick is not jammed. Reengagement of pitch SAS, if desired, should be accomplished with caution, one channel at a time, in straight and level flight at a safe altitude.

- Excessive differential between pitch SAS piston positions as a result of loss of one hydraulic power source:

  This type event is identified by noting one of the HYD PRESS lights coming on and low or no pressure indication on the hydraulic pressure gage for that source. Yaw SAS may or may not become disengaged simultaneously.

WARNING

The pitch SAS fail-safe monitoring feature does not function during single channel SAS operation. If pitch SAS operation cannot be maintained with both channels engaged, pitch SAS should be turned OFF.

Pitch SAS Engage Switches

Two pitch SAS engage switches (figure 1-17) are located on the SAS control panel. These are two-position solenoid-held switches placarded PITCH SAS ENGAGE and OFF with one switch placarded L, and the other placarded R.

For normal engagement both switches are actuated simultaneously and momentarily held. The switches are both electrically released to the OFF setting if the monitor circuit signals a failure or the pilot actuates the SAS emergency disengage. The switches can also be manually moved to OFF. When either or both are in the OFF setting the PITCH SAS light will come on.

The two switches are powered by the right DC system bus.

Pitch SAS Caution Light

The PITCH SAS caution light (figure 1-51) on the caution light panel, will come on to indicate that one or both of the pitch SAS channels is disconnected. The light is powered by the DC essential bus.

Yaw SAS

On aircraft prior to serno 78-0582, the yaw SAS performs three basic functions: yaw rate damping with ±7° rudder authority, yaw trim with ±10° rudder authority, and aileron/rudder interconnect (ARI) with lateral accelerometer inputs for turn coordination with ±10° rudder authority. The total SAS authority is mechanically limited to ±10° rudder below 240 KIAS. Above 240 KIAS SAS authority is limited to ±8° rudder, which is the full powered actuator stroke in this speed range. The turn coordination command is generated by the shaping and summation of signals from the lateral stick position sensors, and yaw rate gyros. The gains of lateral stick position are decreased by 50 percent at speeds above 255 KIAS.

On aircraft serno 78-0582 and subsequent, the yaw SAS performs three basic functions: yaw rate damping with ±7° rudder authority, side slip control with ±7° rudder authority, and yaw trim with ±10° rudder authority. The total SAS authority is mechanically limited to ±10° rudder below 240 KIAS. Above 240 KIAS SAS authority is limited to ±8° rudder, which is the full powered actuator stroke in this speed range. The side slip control is generated by the shaping and summation of signals from the roll attitude synchro of the INS or HARS (aircraft serno 79-0167 and subsequent), or HARS (aircraft prior to serno 79-0167), roll rate sensors, angle-of-attack transmitter, and yaw rate sensors. The gain of the roll attitude signal is changed at 180 KIAS and again at 255 KIAS.

The monitor circuit senses differential between yaw SAS pistons of the left and right rudder actuators and shuts off yaw SAS when the differential exceeds the equivalent of a 3.2° difference in displacement of the left and right rudders. Rudder pedal authority is more than sufficient to override a SAS induced rudder displacement.

Each yaw SAS channel can be reengaged, subsequent to a commanded or uncommanded

SPEED BRAKE SYSTEM SCHEMATIC

(Aircraft Prior To Serno 75-00280 Not Modified By T. O. 1A-10-509)

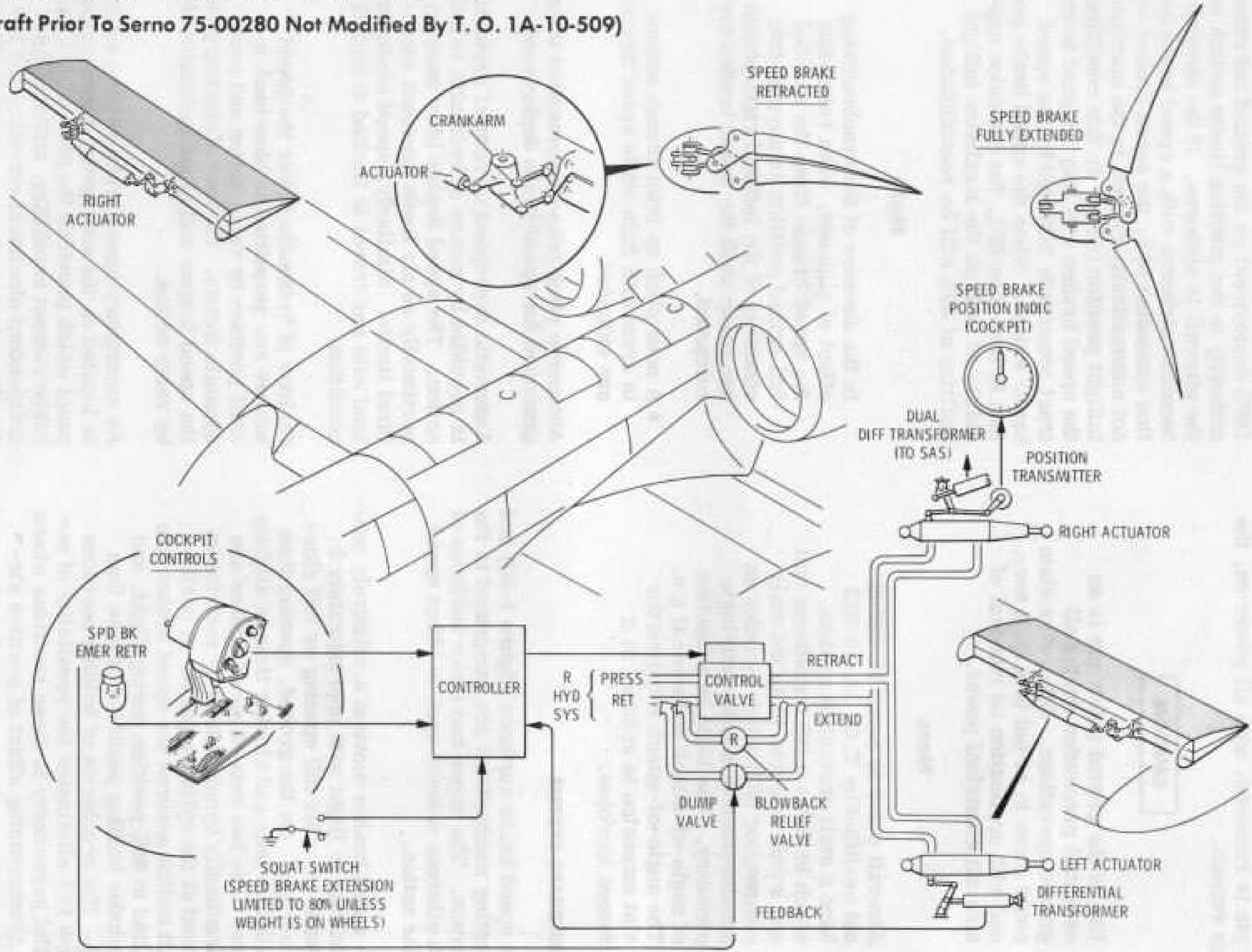


Figure 1-20. (Sheet 1 of 2)

SPEED BRAKE SYSTEM SCHEMATIC
(Aircraft Serno 75-00280 and Subsequent and Aircraft Modified By T. O. 1A-10-509)

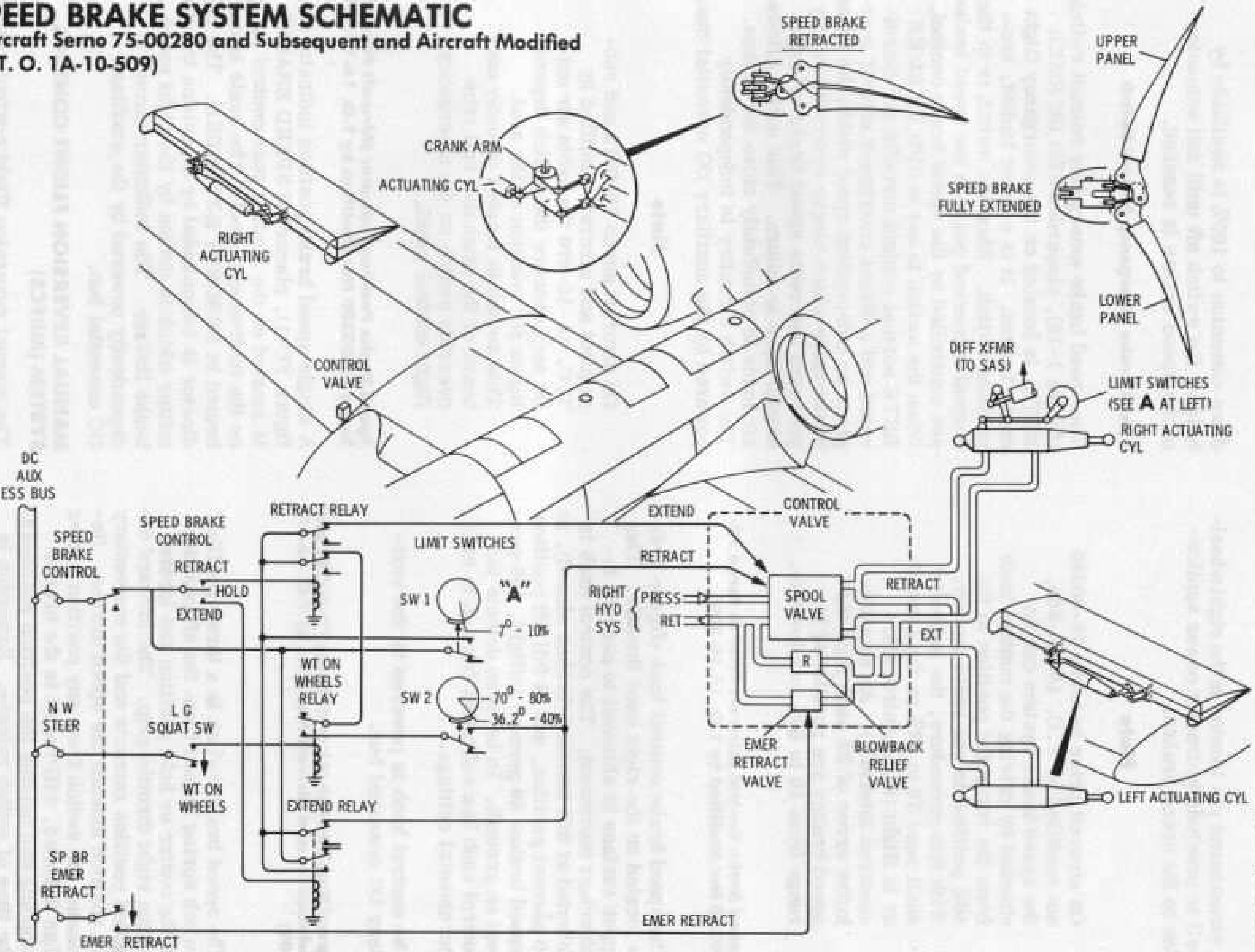


Figure 1-20. (Sheet 2 of 2)

a streamered pin located in the right wheel-well to preclude hydraulic power application to the speed brakes.

Note

> On aircraft prior to serno 75-00280 not modified by T.O. 1A-10-509, the speed brake system can be checked by driving the control knob from the retracted position to the 40% position in one rapid motion. With this procedure, the indicator shall read 38 to 42% on the ground or in flight (with control stick centered laterally). An accumulative error of 2% each time the speed brakes are positioned in range from 10 to 80% is allowable.

Speed Brake Control (Aircraft Prior to Serno 75-00280 Not Modified by T.O. 1A-10-509)

The speed brake control knob (figure 1-4) is located on the right hand throttle. The upper surface is shielded to prevent inadvertent movement. The control knob is detented at full forward (brakes closed), at 40 percent position, and at full aft position (speed brakes 80 percent inflight, 100 percent on ground). In between detents the control knob has sufficient friction to hold incremental settings.

The control knob is powered by the auxiliary DC essential bus.

Speed Brake Switch (Aircraft Serno 75-00280 and Subsequent and Aircraft Modified by T.O. 1A-10-509)

The speed brake switch is a three-position switch spring loaded from the aft position to the center or hold position and located on the right throttle grip. The forward detented position retracts and the momentary aft position extends the speed brakes. Releasing the switch from any position, other than detented, will result in the brakes stopping and holding the position reached at the time of switch release. Extension is limited to 80% during flight. After touchdown extension to 100% is available by holding the switch aft until full extension of the speed brakes is reached.

Speed Brake Emergency Retract Switch

The speed brake emergency retract switch (figure 1-16), placarded SPD BK EMER RETR, is located on the emergency flight control panel. It is a lever locked, two-position switch. When the switch is in the normal unmarked position the speed brakes are controlled by the speed brake control. When the switch is set to SPD BK EMER RETR normal control circuits are deactivated and a direct emergency circuit energizes an independent spool within the speed brake valve which blocks hydraulic supply pressure and vents speed brake cylinder extend lines to return. This action allows air loads to completely close the brakes. The switch circuitry is independently powered by the auxiliary DC essential bus.

Note

> On aircraft serno 76-0512 and subsequent and aircraft modified by T.O. 1A-10-670 the cabin air outlet may obscure the switch depending on the position of the seat. This switch is located directly outboard of the pitch and roll trim override switch on the emergency flight control panel.

Speed Brake Position Indicator (Aircraft Prior to Serno 75-00280 Not Modified by T.O. 1A-10-509)

A single speed brake position indicator (35, figure FO-1), placarded SPEED BRAKES, is located on the landing gear control panel on the instrument panel. The scale is calibrated in PERCENT EXTENDED. The indicator is commanded by a position transmitter which is driven by the right speed brake linkage. The indicator circuit is independently powered by the auxiliary DC essential bus.

MANUAL REVERSION FLIGHT CONTROL SYSTEM (MRFCS)

The manual reversion flight control system is an emergency flight control system without the use or availability of any hydraulic power. The mode is adequate for safe recovery from maneuvers, for executing moderate maneuvers, and for safe and comfortable return to base and landing.

In the MRFCS mode longitudinal stick and pedal commands are directly transmitted to the elevator and rudder surfaces through the power actuators which are in a hydraulic bypass mode. The lateral stick commands are either similarly transmitted through surface actuators to the ailerons or, after pilot actuation of the flight control mode switch to MAN REVERSION, the stick commands are freed from the ailerons and directly transmitted to aileron servo tab control linkage. Manual reversion trim capability is provided only in the pitch axis.

Emergency transitions to MRFCS are automatic and instantly effected in the pitch and yaw control axes. Transitions in the roll axis must be pilot initiated and are intentionally slowed, to minimize pitch trim transients, by the hydraulic damping action that limits the rate the ailerons collectively float up to assume trail (zero hinge moment positions). The float-up transition time is approximately 4 seconds dependent on aileron loads. A second transition time occurs after the flight control mode switch is set to MAN REVERSION. Stick transfer from ailerons to tabs is progressive and requires approximately four seconds to complete. Mode switch action also precludes unexpected returns to powered flight control by energizing hydraulic systems shutoff valves.

WARNING

With flaps full down, maintaining level flight following transition to manual reversion may require aft stick forces which may exceed the physical capability of the pilot. If transition to MRFCS occurs with flaps full down, it is imperative that the flap emergency retract switch be activated immediately.

For training and checkout purposes, dual hydraulic failures can be simulated and the mode change accomplished by setting the flight control mode switch to MAN REVERSION. Aileron/tab shifting is accomplished concurrently with aileron pressure bleed off. With mode switch return to NORM, hydraulic power is restored immediately to all surface actuators and immediate normal control of elevators and rudders is available. The ailerons instantly drive to neutral trim position, but lateral stick control of roll is not fully regained until completion of tab/aileron shifter operation (approximately four seconds).

PITCH MRFCS

The same mechanical and electromechanical elements are used for manual as for hydraulic power pitch control. The difference is in "mode of operation". Three logic functions act together to automatically effect the necessary mode changes at the instant usable hydraulic power has failed:

- Hydraulic - as hydraulic pressure reduces to 600 - 400 psi, elevator actuator internal piston bypass opens. This minimizes hydraulic resistance to the transmission of pilot effort to drive the surfaces. Concurrently, hydraulic restraint is removed from a plunger which is spring-loaded to close the valve stops.

- Mechanical - the valve stops, although hydraulically released, are restrained open by external mechanical logic until both elevator actuators are in bypass mode. This is accomplished by flexible linkage which interconnects the two elevator stop linkages. The purpose of this logic is to eliminate free play between the stick and elevator surfaces during manual reversion operation, and to retain the free play in the single, as well as the dual, powered flight control modes for unrestricted stick authority over the powered actuator.

- Electrical - both the normal and emergency pitch trim command circuits are switched in for control of the two trim tab actuators in addition to the artificial feel trim actuator. The transfer of trim tab actuator command to pilot's trim commands is automatically accomplished using hydraulic pressure switch signals and logic that responds only when both power sources have failed to or below the 800 - 1000 psi level.

Mode change logic is reversible. Power control of the elevators is instantly restored as pressure at one (or both) of the actuators is increased to or above the 700 - 900 psi level. Additionally, both elevator trim tab actuators are automatically transferred from cockpit trim control to "trim to neutral" control when either, or both, pressure switches sense increase of pressure to or above the 1000 - 1200 psi range.

YAW MRFCS

The same mechanical elements are used for manual as for hydraulic power yaw control. The mode changes are all automatically effected internally within the surface actuators. As hydraulic pressure at an actuator reduces to 600 - 400 psi piston bypass opens and closes down valve stops to the equivalent of ±4 degrees of rudder free play relative to pedal commands. There is partial reversion operation of the rudders when only one hydraulic system has failed. The free play improves pedal feel during single hydraulic power control operation and is not objectionable in full manual reversion operation.

Mode change logic is reversible. Power control of an actuator is instantly restored as actuator supply pressure is increased to 700 - 900 psi level.

ROLL MRFCS

The 50 percent span geared/servo tabs, mounted on the inboard upper-aileron panels, are provided only for manual reversion purposes. The tabs are geared relative to aileron motion (-.7 to 1). The geared tab action, in combination with sealed aileron nose balance, reduces aileron actuating hinge moment. The tabs are indexed 35° (serno 75-00266 and subsequent, 30° on earlier aircraft) T.E. up relative to the ailerons at powered neutral. Aircraft serno 76-0512 and subsequent and those modified by T.O. 1A-10-718 have the tabs indexed to 30 degrees. This factor serves to reduce the aileron up-float rotation on loss of hydraulic power.

Shifter mechanisms are installed in the wing outer panels forward of the inboard edges of the ailerons. The shifters are each positioned by an electromechanical actuator which responds to pilot-initiated cockpit command signals to redirect the output of lateral stick motion. Limit switches deenergize and brakes lock the shifter actuators at both settings. At the normal setting, the shifters serve as idler bellcranks to transmit stick commands to the aileron actuators and also serve as fixed support points for the geared aileron tab linkage. As the manual reversion setting is reached the shifters have completely disengaged the stick from aileron actuator connections (regardless of possible combinations of stick and aileron travels) and then function as an idler bellcrank connecting stick to the aileron tab linkage. At this setting maximum stick travel provides ±15° of servo tab travel which in turn aerodynamically back drives the ailerons. Feel at the stick is proportional to the differential hinge moment at the tabs.

Aileron Float-Up Transition

Due to the aileron airfoil camber, the characteristic aerodynamic trail position (unpowered zero hinge moment position) of the ailerons is higher than the powered neutral position. The aileron float-up induces an aircraft nose-up pitch trim change. To soften pitch trim onset, while transitioning to manual reversion mode, the aileron float-up rate is limited by prescribed damping at the surface actuators.

Fluid is trapped in the right hydraulic system chambers of both aileron actuators. The trapped fluid is allowed to slowly bleed past the pistons until the ailerons have floated up to positions where hinge moments are reduced to the equivalent of 5 percent of total actuator normal capacity. At this point bypass porting automatically opens simultaneously in both actuators. Snap-action, bypass valve logic is similar to that within the elevator and rudder actuators except the settings are lower, 300 - 250 psi, for failing pressure. The elapsed time for the ailerons to float-up to the bypass actuation point after bleed off of the two hydraulic power supply pressures, is approximately 4 seconds. Aircraft at high speed will require the longer time. Bleed off of supply pressures can take up to 10 seconds or more.

The valve input linkage stops are permanently fixed at settings equivalent to ±4° aileron travel. The collective aileron float-up action engages the stick mechanical transmission system at the stops and then stretches the forward cable systems within allowable limits. If the aileron/tab shifters have not been actuated at this point of time, the stick acts directly on the surfaces through actuator linkage with no free play and effectively no roll control.

Aileron/Tab Shifting Transition

The flight control mode transfer switch, when set to MAN REVERSION position performs two direct functions; it energizes two hydraulic solenoid valves to block all hydraulic power to all services, and it energizes the two aileron/tab shifter mechanisms to shift to the manual reversion setting. The shifting cycle takes approximately 4 seconds to complete in either direction with time duration unaffected by loading conditions.

The hydraulic shutoff valve action is immediate, as are the resulting mode shifts in the pitch and yaw axes. Shifting action is progressive and roll control is normally available in the time it takes for the ailerons to float up.

As each aileron/tab shifter moves from the normal setting, shifter actuation limit switch action initiates the following actions:

- Deactivates the cockpit roll trim control circuits, both normal and emergency.
- Energizes the roll trim actuators to drive to neutral where they stop and hold position.
- Corresponding L & R AIL TAB caution light comes on.

The purpose for automatically driving roll trim to neutral during MRFCS operation is to assure that when hydraulic power is restored, the ailerons, which respond instantly to reapplication of hydraulic power, drive down to neutral settings.

Aileron Tab Caution Lights

The aileron tab lights (figure 1-51) on the caution light panel are placarded L AIL TAB and R AIL TAB. The lights come on if the corresponding aileron/tab shifter mechanism is not at the full normal position. The lights are powered by the DC essential bus aileron tab circuits.

Aileron/Tab Shifter Malfunctions

Failure of an aileron/tab shifter to initiate shift to tab drive after the Flight Control Mode switch is placed in the MAN REVERSION position is manifested by:

- Respective AIL TAB caution light off.
- Both hydraulic pressure gages indicating approximately zero pressure within 14 seconds after switch actuation.
- Both ailerons up-floated - normally within 14 seconds after switch actuation.
- Lateral displacement of control stick toward side of nonfunctioning shifter.

- Very high lateral stick force gradients - approaching locked stick feel.
- Roll-off, usually toward side of nonfunctioning shifter.
- Both aileron jam lights may be on depending upon lateral stick forces applied.
- Very limited response of one aileron servo tab with application of heavy lateral stick forces.

In the event these manifestations are experienced after positioning the Flight Control Mode Switch to MAN REVERSION and hydraulic power is available, the switch should be returned to NORM for remainder of the flight. Should hydraulic power not be available, a reasonable amount of roll control can be achieved by disengaging the aileron control system for the side with the nonfunctioning aileron/tab shifter as indicated by:

- AIL TAB caution light off.
- Side to which control stick is displaced.
- Side with nonresponsive aileron servo tab as determined by observation.

Failure of an aileron/tab shifter to initiate return to aileron drive after the Flight Control Mode Switch is placed in the NORM position is manifested by:

- Both hydraulic pressure gages indicating normal pressure almost immediately.
- Both ailerons returned to takeoff trim position.
- Lateral displacement of control stick toward side of nonfunctioning shifter - amount depending upon air speed.
- Roll-off toward side of nonfunctioning shifter if stick is unrestrained.
- Lateral stick force required to keep wings level.
- Respective AIL TAB caution light remains on when opposite side AIL TAB caution light goes off.
- Aileron tab on side with nonfunctioning shifter responds to lateral control stick displacement with aileron remaining in neutral position.
- Aileron jam light for side with nonfunctioning shifter may come on, depending upon aileron tab air loads.
- Aileron trim inoperative.

In the event these manifestations are experienced after positioning the Flight Control Mode Switch to NORM, near normal flight control can be achieved by disengaging the aileron control system for the side with nonfunctioning aileron/tab shifter as indicated by:

- AIL TAB caution light on.
- Side to which control stick tends to displace.
- Side with responsive aileron servo tab as determined by observation.

With one side disengaged, roll capability will be reduced approximately 50 percent. Lateral control stick force gradient (or feel) will also be 50 percent of normal. If desired, roll trim can be restored by opening the AIL TAB circuit breaker for side with nonfunctioning shifter. The AIL TAB caution light will go off when this circuit breaker is opened and both ailerons will respond to roll trim.

Failure of an aileron/tab shifter to complete shift to tab drive or aileron drive is manifested by degradation of roll control capability for the control mode selected. The degree of degradation is based upon the amount of shift accomplished prior to failure. Hydraulic pressure and aileron neutral position will be normal for the mode selected. Both aileron tab caution lights will generally be on. The exception will be when a shifter failure occurs while shifting back to the NORM flight control mode. In this situation, the aileron tab caution light for the shifter which sustained a failure will remain on whereas the caution light associated with the shifter which completed shift to aileron drive will be off. Other manifestations will be similar to those outlined above for failure of an aileron/tab shifter to initiate shift to the flight control mode selected. The aileron tab circuit breaker should be checked whenever a shifter operational failure is suspected. If open, an attempt at reset should be made.

Flight Control Mode Switch

The flight control mode switch (figure 1-16) is located on the emergency flight control panel. It is placarded FLT CONT, with positions NORM and MAN REVERSION. The switch is lever-locked in both positions.

In the MAN REVERSION position both hydraulic systems are shut off to all services. The switch simultaneously energizes the aileron/tab shifters to drive to the tab drive setting. All other mode transfer logic is automatic. The switch controls two independent circuits, both powered through the L & R AILERON TAB circuit breakers by the DC essential bus.

Note

> The switch is left in the NORM position during the ground storage of the aircraft so that the aileron tabs and the stick are restrained, for protection from ground gust effects.

OPERATION—MRFCS

Shifting to MRFCS Mode

During service operations, well over 99 percent of the conversions to MRFCS mode will be intentionally initiated. Reasons to transfer to MRFCS mode, while hydraulic power is still available, include the following:

- Training in the MRFCS mode.
- Checkout of the MRFCS mode.
- Precautionary transfer to MRFCS mode: for example, one hydraulic system failed and it is suspected failure of second system is imminent.

When conditions permit, the following precautions are recommended prior to planned conversions:

- Straight and level flight at moderate speed.
- Trim for 1 g (level flight).

While actuating the mode transfer switch, hold stick laterally in neutral region until it is sensed that aileron actuators have shifted to the bypass mode. This period will be approximately 4 seconds after hydraulic pressure supply bleed off which can take up to 10 seconds. The total time without lateral control after actuating the mode transfer switch can therefore be as high as 14 seconds. If control stick is moved laterally prior to bypass action, the ailerons may float up abruptly and asymmetrically.

In the rare instances of abrupt dual hydraulic failures, initial control of the roll axis (after aileron float-up) is similar to control of the pitch and yaw axes. After regaining controlled steady flight the pilot is free to actuate the flight control mode switch to shift lateral stick control to tabs and to prevent an unexpected return to powered flight mode.

CAUTION

Failure of one or both hydraulic systems to drop below 250 psi after activating the Flight Control Mode Switch to MAN REVERSION may result in locked ailerons after shift to aileron tab drive commences. Under these circumstances, control stick feel will be near normal for the manual reversion mode; however, roll capability will be slight and in opposite direction to stick displacement. Therefore, should one or both hydraulic pressure gages fail to indicate approximately zero pressure within 14 seconds after switch is activated to MAN REVERSION and roll is in opposite direction to stick displacement, return the switch to NORM providing it is known that adequate pressure is available from at least one system.

Shifting Back to PFCS Mode

Hydraulic power, if available, is immediately applied to all surface actuators when the flight control mode switch is returned to NORM setting. All logic functions are fully reversible and powered control of the elevators and rudders is immediate. Longitudinal trim force may be present in the stick which can be immediately reduced. The ailerons drive down to neutral trim position and lateral stick control is not effective in neutral region until the approximate three to four second aileron/tab shifting operation is complete. Roll trim control, both normal and emergency, is returned to the cockpit at the completion of the shifting operation. Yaw trim control is regained after YAW SAS is reengaged.

BOARDING LADDER

The boarding ladder (figure 1-21) is a telescoping ladder which stows in a compartment in the left forward fuselage, below the cockpit. The ladder steps extend fore and aft from the center column. The ladder compartment door is hinged on the forward edge and opens to rest flat against the fuselage. A permanent type magnet is attached to the door and the fuselage to hold the door in the open position. A battery bus-powered solenoid activated latch mechanism is located at the aft edge of the door. Both the compartment door and the ladder are spring-loaded to open and extend.

From the cockpit, the ladder may be extended by pressing a push-button switch (4, figure FO-3) located under a hinged cover guard placarded EXTEND BOARDING LADDER. From the exterior of the aircraft the ladder may be extended by pressing a switch located to the rear of the door. For stowage, the ladder must be pushed up manually from the ground, until it is completely telescoped.

WARNING

To avoid personal injury, stand aft of the door and press the push-button switch. Be sure the immediate area in front of the door and below the ladder are clear of personnel as the door is spring-loaded to open.

- The ladder extends by gravity.

CAUTION

Do not hold boarding ladder switch depressed for more than four seconds as the latch relay may be damaged.

CANOPY

The canopy is constructed of molded stretched acrylic plastic with no supporting structural members.

Normal operation is controlled by switches (figure 1-22). The canopy is opened and closed by means of an electromechanical actuator which operates on electrical power supplied directly from the battery bus. The

BOARDING LADDER

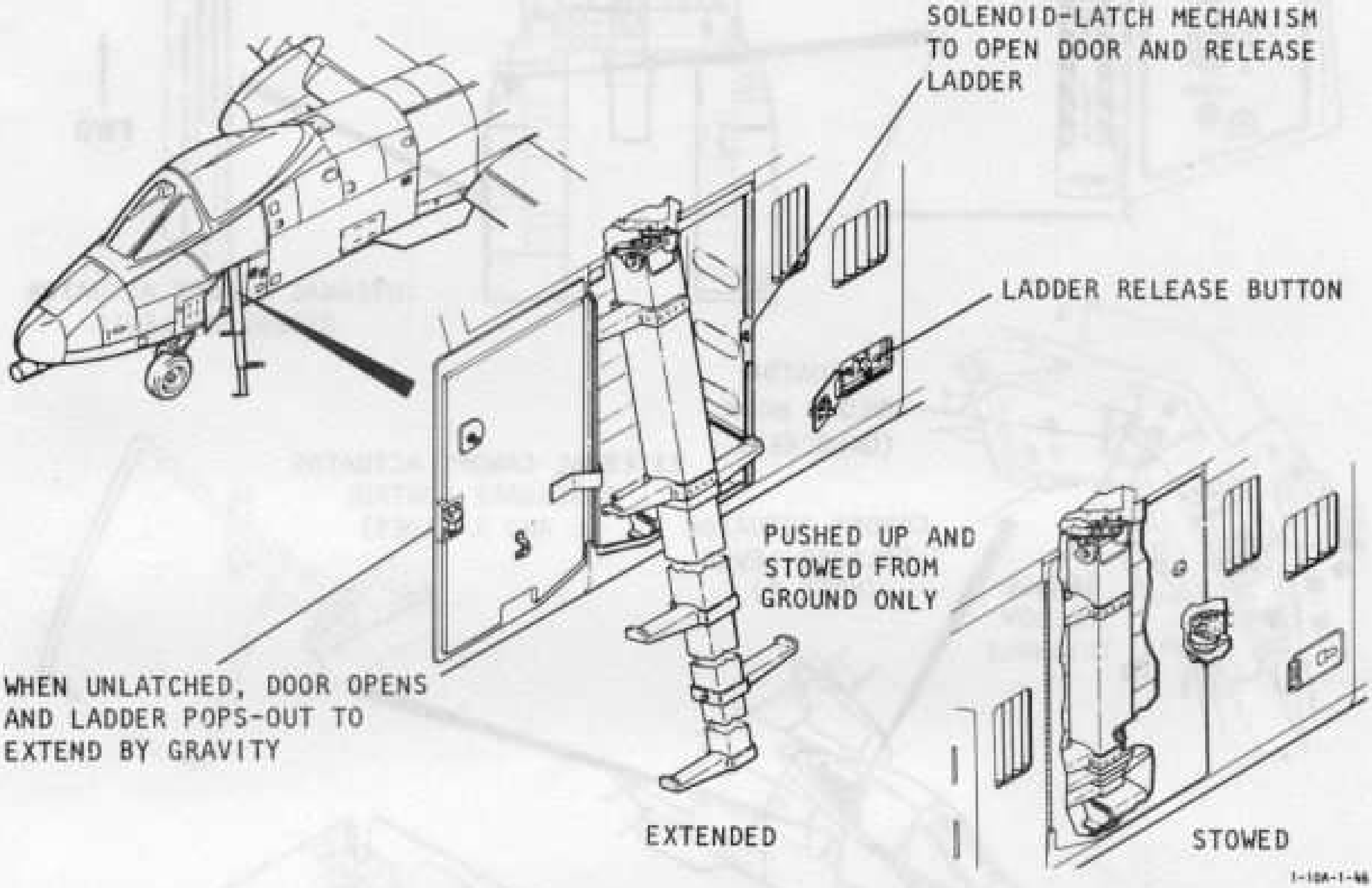


Figure 1-21

process of raising or lowering the canopy is governed by two control switches, enabling operation of the canopy from inside or outside the aircraft.

In the event of failure of the electromechanical canopy actuator or loss of battery bus power, provisions for mechanical disengagement of the canopy/actuator attachment are incorporated. The disengagement process is accomplished by three mechanical control devices enabling the pilot or ground crew to open the canopy manually from inside or outside (left or right side) the aircraft.

The canopy may be jettisoned, either in flight or on the ground, independent of the seat ejection function, by pulling a control, placarded CANOPY JETT, located on the right console. The canopy may be jettisoned from the outside by a control on either side of the aircraft which is independent of the seat ejection function. The canopy jettison sequence is initiated by opening either rescue door, and pulling the handle approximately six feet.

CANOPY CONTROLS

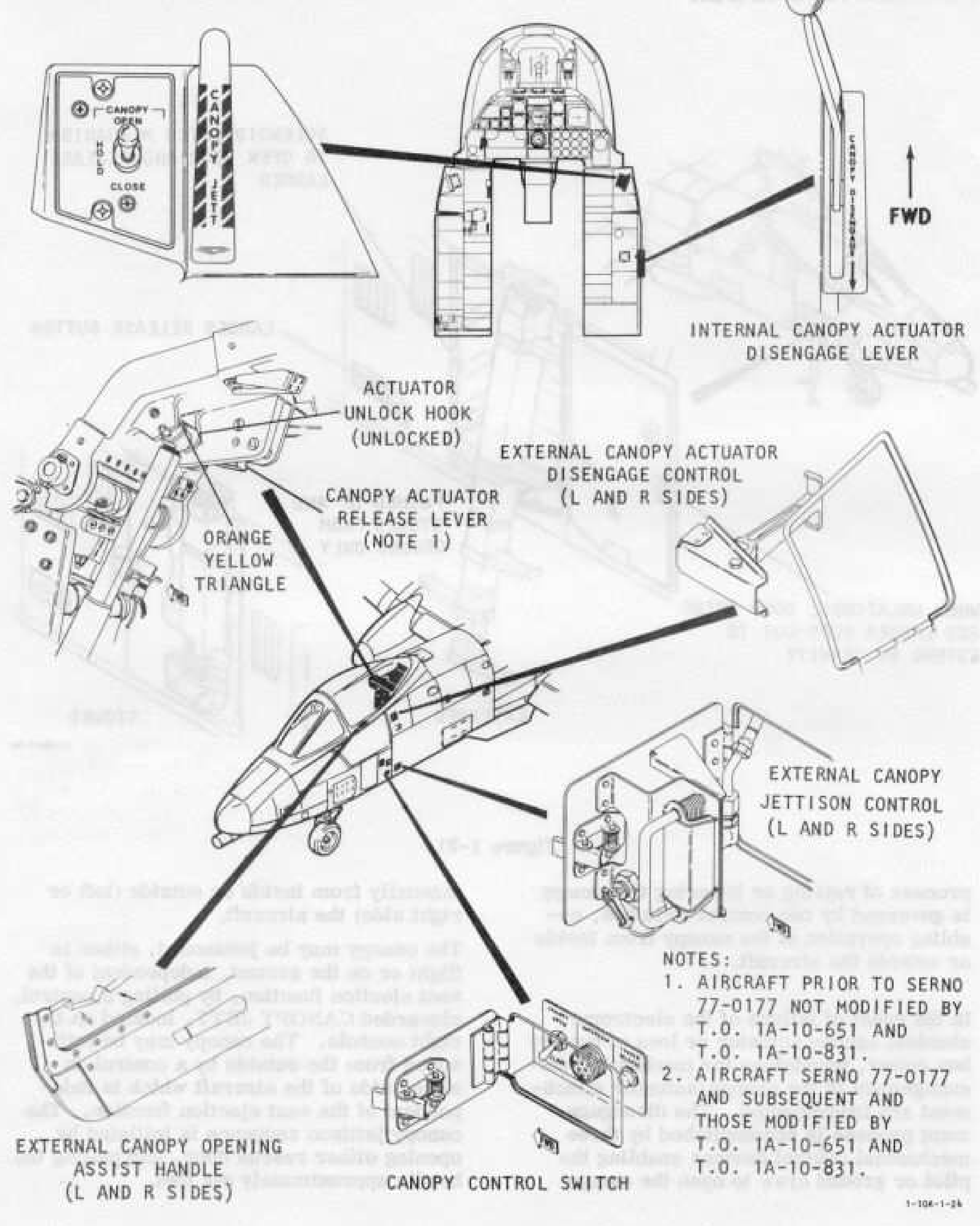


Figure 1-22. (Sheet 1 of 2)