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J79-15/-17 TURBOJET ENGINE ACCIDENT INVESTIGATION PROCEDURES

F-4 SYSTEM PROGRAM OFFICE

AUGUST 1975

TECHNICAL REPORT ASD-TR-75-19

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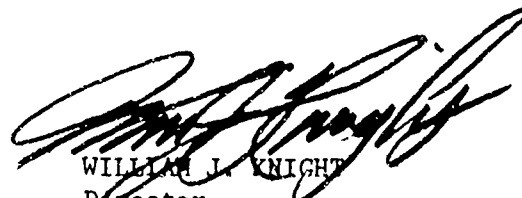
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This report was submitted by the author in August 1975.

This technical report has been reviewed and is approved.


 WILLIAM J. KNIGHT
 Director
 F-4 System Program Office
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20. ABSTRACT (Continue on reverse side if necessary, and identify by block number) The author found, when he was called upon to participate in an accident investigation, the existing documents and manuals covering accident investigations were very general and were written basically for the large aircraft engines. Since the smaller engines, such as the J85, with their high rotation speed, low polar moment of inertia, and small mass will not have the same evidence to substantiate such vital operating conditions as RPM, variable exhaust nozzle area, and operating temperatures at the time of impact as the larger engines,		

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it became apparent that the existing publications would be of marginal help in investigating an accident involving the J85 engine. The other personnel participating in the investigation with the author had their ideas about what to look for and how to analyze the evidence available. While each participant's ideas had merit, it became apparent that it would be of benefit to the Air Force to consolidate the ideas, prepare check lists on what to look for, describe the operation of the various systems of an engine and their influence on other systems, and prepare charts and graphs that will provide an investigator with sufficient information to enable him to do a systematic and accurate investigation of an accident involving the J85. It is toward these objectives that the author prepared ASNJ-TN-68-1 entitled "J85 Turbojet Engine Accident Investigation Manual." The manual for the J85 became extremely popular after its distribution. It was apparent that a similar manual for the J79 would be of great benefit to the Air Force. It was toward this need that the author prepared this report.

This document should be of great benefit to the experienced accident investigator in conducting an accident investigation involving a J79 engine. It should enable an inexperienced investigator to conduct an accurate accident investigation by following the procedures described herein.

It is recommended that this document be read and understood by all personnel who might be required to investigate an accident involving a J79 engine. It is further recommended that the investigator take this manual to the field as a handy reference when requested to investigate an accident involving a J79 engine.

PREFACE

This report was prepared by Mr. Frederick K. Ake, F-4 System Program Office, Aeronautical Systems Division (AFSC), Wright Patterson AF Base, Ohio, under Project 327C.

The report is intended to give the accident investigator means of estimating J79 engine operating conditions before and at the time of an accident, and to aid in determining if an accident was caused wholly or in part by failure or malfunction of the J79 engine.

The author wishes to acknowledge the help provided by the General Electric Company in the preparation of this report.

SUMMARY

The following documents are appropriate as guides for an accident investigation: Air Force Manuals 127-1 and 127-2, and Air Force Regulation 127-4. An accident investigator should be familiar with the contents of these documents, which cover the procedures to be followed and the duties and responsibilities of the team members.

This report details a large number of technical terms, some of which are finitely measurable, others observational. The following essentials should always be considered. (It would be advantageous to memorize them.)

ESSENTIALS OF GOOD INVESTIGATION

Promptness

The investigator should get to the scene of an accident as soon as possible, before the evidence is disturbed, and should prevent unnecessary handling or moving of the wreckage.

Thoroughness

The investigator should:

Examine all evidence in minute detail.

Take nothing for granted.

Never jump to conclusions.

Investigate all possibilities although the probable cause may be known. Never be dismayed if the cause is not readily apparent, but collect every scrap of information. These scraps can frequently, like the

pieces of a jigsaw puzzle, be put together to produce a revealing picture.

Follow every clue to the limits of usefulness.

Preserve wreckage or evidence until the investigation is satisfactorily completed.

Make a photographic record of all evidence which might be removed, effaced, lost, or destroyed.

Consider nothing which will help prevent a similar accident too much trouble to investigate.

System

The investigator should:

Conduct a planned investigation which makes the best possible use of the personnel and facilities available.

Lay out logical courses of procedure.

Follow each course in a systematic manner.

Avoid hasty conclusions which tend to curtail the investigation.

Accuracy

The investigator should remember that:

Guesses, rumors, or half truths have no place in an accident record.

Statements and theories must be verified by facts.

NOTE: This report was basically prepared for use with U.S. Air Force engines. It can also be used for U.S. Navy engines by applying the information given for the J79-15 engine for the Navy's J79-8 and the J79-17 engine for the Navy's J79-10. For the purpose of accident investigation, there are two differences

between the Air Force and the Navy's engine models. The Air Force engines have the compressor vane position indicator on the 6th stage variable vanes whereas the Navy's vane position indicator is on the 4th stage. (There is a slight difference in indications between the Air Force and the Navy models. For the differences refer to the maintenance manuals) The Air Force models use a torque booster for throttle shaft actuation whereas the Navy's models use a power level control. Therefore, for the Navy's use of this report, those portions dealing with the torque booster should be ignored. Keeping the above factors in mind, the Navy can use this report as described above.

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EXPLANATION OF TERMS

A/C	Aircraft
A8	Afterburner Nozzle Area
A/B	Afterburner
ABC	Afterburner Control
ABF	Afterburner Pump
EGT	Exhaust Gas Temperature
Fg	Thrust
FOD	Foreign Object Damage
GPM	Gallons Per Minute
IGV	Inlet Guide Vanes
VPI	Vane Position Indicator
MFC	Main Fuel Control
MFP	Main Fuel Pump
A/BFP	Afterburner Fuel Pump
NPI	Nozzle Position Indicator
O/H	Overhaul
CSD	Constant Speed Drive
P.E.	Periodic Inspection
RPM	Revolution Per Minute
TCTO	Time Compliance Technical Order
T.O.	Technical Order
T ₅	Turbine Discharge Temperature
T _{5H}	Turbine Discharge Harness Temperature
VEN	Variable Exhaust Nozzle
VG	Variable Geometry
K W/F	Fuel Flow in Thousands of Pounds Per Hour

SECTION I
INTRODUCTION

As the propulsion member of an accident board, you will probably be asked questions such as: "Did a fire develop in flight or after impact? What was the source of the combustible material and what was the ignition source? Were the engines operating at time of impact? How much thrust was being produced?" These questions may appear to be insurmountable. However, by following the procedures and techniques outlined in this report, these and other questions pertinent to the accident can be answered with a high degree of accuracy. Therefore, if you are assigned to an accident investigation board and, upon arriving on the accident scene, you find only twisted and burned parts that once were an F-4 airplane, don't despair; many clues remain in the debris that can be used to determine the operation of the engines at time of impact. You must know how to interpret your findings and determine which findings are most reliable.

It is not the intent of this report to make an expert accident investigator of you, but rather to assist you in establishing whether the propulsion system contributed to the accident so the Accident Board President can determine the most probable cause.

Expertise in any field is a result of doing the same thing over and over again in essentially the same manner. Since each aircraft accident is different from any other, and the accident that you have been assigned is your first or perhaps the only one during your career, you cannot be expected to be an expert. Slight changes in the circumstances surrounding the accident

can greatly influence the evidence to be considered. All of the ramifications influencing an accident could not possibly be covered in a document such as this. Therefore, after following the procedures outlined in this document, careful analysis and common sense must still be applied in order to establish the "most probable cause".

Attempting to establish the RPM of an engine at time of impact by observing the condition of the compressor rotor or turbine are highly inaccurate methods and should not be used. There may be as much damage resulting from low RPM and high impact damage as there is from high RPM and low impact damage. More accurate methods of determining RPM consist of analysis of components that move as a function of RPM and would be expected to retain their position at time of impact or leave impact marks. Components that have these characteristics are the main fuel control 3D cam, variable stator actuators, variable stator feed back cable, variable exhaust nozzle actuators, variable exhaust nozzle feed back cable and nozzle area control. Section V of this report explains how the above mentioned components can be used to determine RPM. This section also covers related parameters such as the throttle angle and compressor air inlet temperature.

Variable exhaust nozzle area can occasionally be measured after an accident. However, the exhaust nozzle parts have a tendency to move due to impact forces. Therefore, a better method of determining the nozzle area is through examination of the nozzle actuators internally and externally, nozzle feed

back system, nozzle area control and selected parts of the nozzle. In each case it is better to consider the impact marks on a part to determine its probable position at time of impact rather than its position after the accident. It is the intent of the section on variable exhaust nozzle area (Section VII) to explain the correct methods for the area determination.

Section III is devoted to determining the fuel flow going into the engine at time of impact. Knowing the above parameters, RPM, VEN, W_F and CIT, the thrust produced at time of impact can be determined. Interior and exterior air flow characteristics that might influence an inflight fire are covered in Section IX. Fires within the engine bay, their origin and influence are covered in Section X.

As stated previously, it is not the intent of this report to make you an expert in aircraft accident investigations but it should be of assistance in conducting a systematic investigation.

Throughout this report generalized terms such as "normally" "generally", etc. are used. These terms are necessary because no two accidents are alike. What may apply to one accident may not apply to another. It is extremely important, therefore, to establish and evaluate all factors prior to drawing any conclusions.

It is suggested that the investigator read this report in its entirety before proceeding with an investigation in order to understand the various means of determining the important parameters. In most cases, evidence which leads to a conclusion can be substantiated from several systems or components.

Particular emphasis should be placed on the preliminary investigation at the crash site since valuable evidence may be lost when the wreckage is disturbed or removed.

Appendices A, B, and C contain checklists designed to assist the investigator in making a systematic and complete investigation. Reproduced copies of the checklists should be used.

1. OBJECTIVES

The objectives of the accident investigator when examining the aircraft engines after an accident are:

a. To determine if the engines were operating normally at the time of the accident.

b. To determine engines' operating condition, i.e., engines RPM and thrust at the time of or immediately prior to the accident.

c. To determine the cause of any indicated malfunction so that actions can be taken to prevent recurrence of an accident.

2. FACTORS TO BE CONSIDERED:

There are factors that must be considered and that often provide clues to possible cause or causes for an accident. These clues often provide the investigator with information on which areas of the engine should be most thoroughly investigated.

If a pilot has indicated he has encountered trouble, his radio transmissions will be recorded on tape. These should be reviewed for possible clues as to the type of emergency the pilot encountered prior to the accident.

Weather is often a factor in an accident. The weather office should be contacted to determine the meteorological conditions at the time of the accident. Particular attention should be made to the temperature, dew point, wind velocities, freezing levels, visibility, and turbulence. If icing is believed a factor, other pilots flying in the same vicinity should be interviewed to determine if they experienced icing problems.

Maintenance records should be reviewed. If the engine has a history of a particular problem, this area of the engine should

receive particular emphasis. Recently overhauled engines or engines that have just completed a periodic inspection, have a high infant mortality rate. This should be considered. Other recently completed maintenance actions should be recorded as possible clues to the accident.

Ground witnesses to the accident may be of some help in determining how an accident happened. However, the witness' relative position to the accident, technical training, and general education must be taken into consideration when evaluating ground witness' reports.

The pilot should be interviewed, if possible. His verbal description of the events preceding the accident should be carefully evaluated. Questions should be asked of the pilot tactfully in order to assure him that it is the investigator's duty to determine the cause or causes of the accident and not to convict him of pilot error.

In the event the engine or its components were not damaged during the accident, the engine or its components should still be inspected and/or tested in accordance with applicable T.O.'s. If components are serviceable they can be substituted on other operational engines in a test cell to substantiate that they are capable of operating satisfactorily.

Aircraft accidents are often accompanied by fire. It is extremely important to determine if the fire occurred in flight and is related to the cause of the accident or if the fire occurred after impact and is, therefore, less significant. This may be determined by examination of the fire damage pattern

caused by flames, smoke, or explosion.

In-flight engine bay fires may be recognizable from the patterns of soot and metal deposits. Engine bay cooling air or the aircraft slipstream may cause these deposits to streak horizontally along the fuselage. Molten metal spray may be deposited on aft components such as the afterburner casing, variable exhaust nozzle housing, and/or related airframe parts.

(See Figure 1.)

Fires resulting from impact may be recognizable from vertical patterns of soot and metal deposits. Molten metal will tend to puddle, and fusion from contact with molten metal is not likely to occur.



Figure 1 View Showing Deposits of Fused Metal on Pigtail and Several Pinholes in Tubing.

SECTION II

INVESTIGATION PROCEDURES

1. PRELIMINARY BRIEFING

The President of the accident investigation board normally assembles the board members prior to visiting the crash site for dissemination of information available at that time. Appendix A is a check list of information to be obtained, if available, at the preliminary briefing.

It should be kept in mind that each member of the accident investigation board is responsible to the president of the board for his conduct.

No statements concerning the accident should be made to anyone except the board president and other board members who have the proper security clearance and "need to know."

Unauthorized or irresponsible statements may prove embarrassing to the board president and the Air Force in general.

Each member of the team has the responsibility to report facts and conclusions accurately without prejudice and free of outside influence.

2. PRELIMINARY INSPECTIONS (AT THE CRASH SITE)

Detailed inspection of the engines and aircraft involved in accidents is done somewhere away from the accident site. This makes it imperative that the investigator inspect the engines and the aircraft thoroughly before they are moved and note any items that may be of significance. Appendix B is a check list for the crash site investigation. The investigation should not be limited to those items listed. Every accident should be

investigated as an individual occurrence.

In addition to a visual inspection, as many photographs as practical should be taken of both the exterior and interior (cockpit and engine bay) of the aircraft. Later examination of these pictures may show significant items that were missed on the first visual examination. A recommended minimum list of photographs to be taken at the crash site is as follows:

- a. Engine inlet duct
- b. Engine inlet
- c. Engine exhaust
- d. Fire damage
- e. Improperly connected or assembled components such linkages, tubes, wires, and cables
- f. Engine instruments and switch positions

NOTE: Make certain the photographs taken will cover the items intended. For critical items an overall photograph should be taken and a close up to illustrate details with identification of the item including nomenclature and serial number if appropriate.

As much information as possible should be obtained regarding the way in which the aircraft contacted the ground, i.e., direction, attitude, impact angle, estimated speed, and sink rate, since these factors may have considerable bearing on the type of damage incurred by the engines. Evaluation of this damage, together with the way in which it occurred, is often the only way that engine thrust immediately prior to the accident can be estimated with any degree of accuracy. Ground witnesses may be of help in determining the above. Items to be considered to determine angle of impact are:

a. Contact with other objects such as trees or buildings prior to final impact

b. Type of terrain

A detailed examination of the wreckage should be made to obtain evidence on the origin of fire, the damage patterns due to fire, explosion, or impact, and the material failures or system malfunctions that could have contributed to the accident. At the same time, the investigator must search for clues which may reveal pilot error, sabotage, or weather disturbances as direct causes. The following items are typical of the evidence which the accident investigator must look for.

a. Fuels or combustibles consumed and soot formation.

b. Fuel tank damage and amount of fuel spilled.

c. Ruptured lines or loose fittings in fuel, hydraulic fluid, and lubricating systems.

d. Ruptured lines or loose fittings in oxygen supply systems.

e. Intensity and spread of fire as indicated by discoloration, fusion, or consumption (combustion) of aircraft structural materials.

f. Intensity and spread of fire in aircraft cockpit.

g. Electrical overloads or faults in wiring, relays, starters, generators, accessory motors, navigational equipment, and other electrical equipment where failure can provide a source of ignition; these faults may be revealed by a study of any localized breakdown of insulation, "weld-like" fusions and erosions of metals produced by arcings, and other signs of

shorted or overloaded circuits.

h. Failures of engine power plants, pumps and powered accessories as indicated by broken turbine blades, damaged bearings, eroded gaskets or seals, or any evidence of seizure.

i. Abnormal functioning of after-burner as evidenced by burn-through of fuselage or other severe fire damage in this area.

j. Fuel explosion occurrence as indicated by some fragmentation and wide dispersal of aircraft components.

k. Ordnance fires and explosions as indicated by intense heating, fragmentation, and damage to surroundings, e.g. ground craters.

l. Positions of flight control systems.

m. Location and physical condition of victims.

Fuel lines should be checked for fuel content and fuel and lube samples must be taken as soon as possible, even though the taking of samples may not seem necessary. If samples are not taken at the accident site it may be impossible to obtain them later due to broken lines or damaged engine parts that can allow leakage, and fuel will eventually evaporate out of fuel lines. The presence or absence of fuel in the engine system can be of major significance to an investigation.

NOTE: Up to one gallon of fuel should be extracted from between the pump and fuel control. Lubrication oil (one quart if possible) should be extracted from the engine's lube tank.

3. ENGINE TEARDOWN AND INSPECTION

After the engines are removed from the aircraft, photographs

should be taken of the engines in order to record impact damage or other items of significance. If the engines are suspected as a cause of the accident, a complete and thorough inspection should be conducted. In other cases the investigation should be conducted to the extent determined by the board.

Examination of the various engine components will normally provide considerable information on engine operating conditions immediately prior to or during an accident, but only if full consideration is given to the type of impact that was sustained by the engine during an accident. It should be clearly understood that items, such as severe rubs, that might be of major significance if an engine sustained only minor impact damage, could be of minor or no significance if an engine impacted severely

A completely different pattern of damage due to debris ingestion can result due to difference in engine deceleration and the manner in which material is ingested into an engine.

Prior to disassembly of the engine, the rigging should be completely checked. Linkage, linkage pins, cotter pins, lock wire, tubing, and hose should be examined. Indications of fire should be examined.

Engine teardown will proceed per standard T.O. procedure, except when engine damage makes normal teardown impossible. Teardown should be done in steps with close inspections made of engine conditions before and after each step and notes made of all findings. Photographs should be taken as often as practical to further document engine conditions. Appendix C

lists items to be checked during disassembly of the engines in the engine shop.

There will be many cases where hacksaws, hammers, chisels and even cutting torches must be used to disassemble damaged parts. These must be used with care to avoid destroying what may be significant evidence. The cutting of control cables and electrical harnesses can make it impossible to check out a component that could have contributed to a malfunction, and could, in extreme cases, lead to the formation of erroneous conclusions regarding an accident. Therefore, a functional check of systems should be made, if possible, prior to any cutting operation.

SECTION III

FUEL FLOW DETERMINATION

Determination of the fuel flow rate at time of impact can be accomplished by examining the fuel flow transmitter.

The vane of the damping mechanism assembly (Item 77, Figure 2) is under spring tension and rests against the tongue in the zero flow position. As flow rate increases, the vane rotates. The vane angle is a function of the fuel flow rate (Figure 3). The vane angle is measured on the damping mechanism assembly between the tongue on the zero flow side and the vane.

To determine fuel flow rate at time of impact, remove the bottom cover (Item 66, Figure 2) of the fuel flow transmitter. The vane may be stuck in the position it was in at time of impact. If so, measure the vane angle and from Figure 3 determine the fuel flow rate. If the vane is not stuck, examine the face of the damping mechanism assembly to see if the vane made an imprint on the face at time of impact. From this vane imprint, determine the vane angle at time of impact and determine the flow rate from Figure 3.

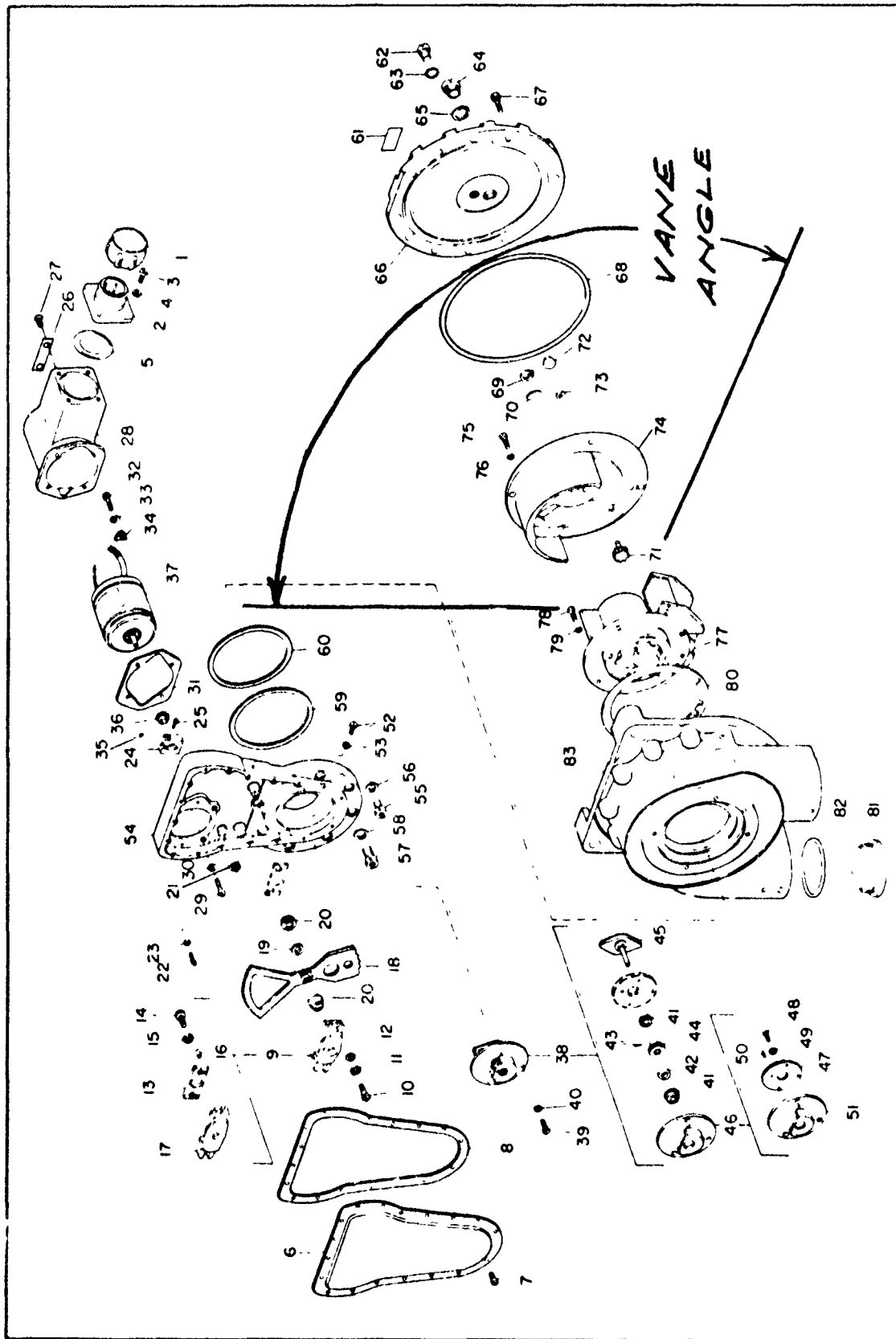


Figure 2 Exploded View - Main Assembly, Fuel Flow Transmitter, Part No. 9115-16-A1A

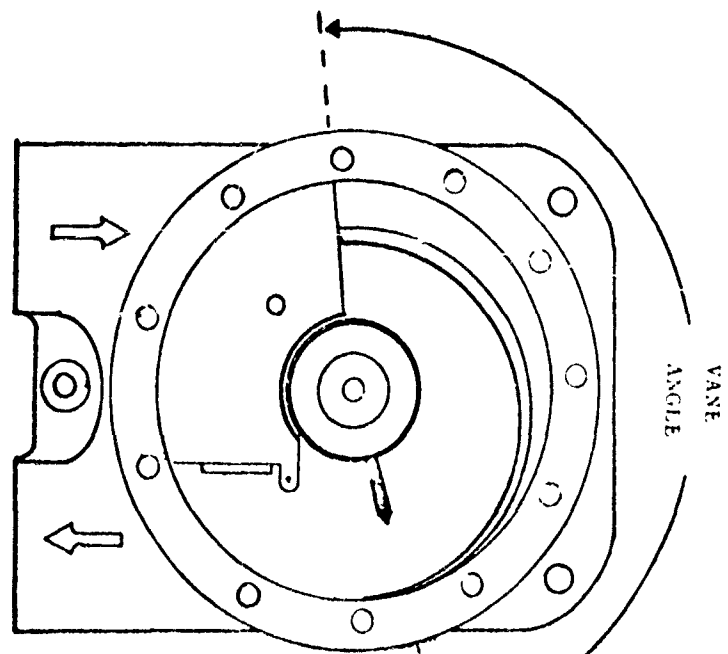
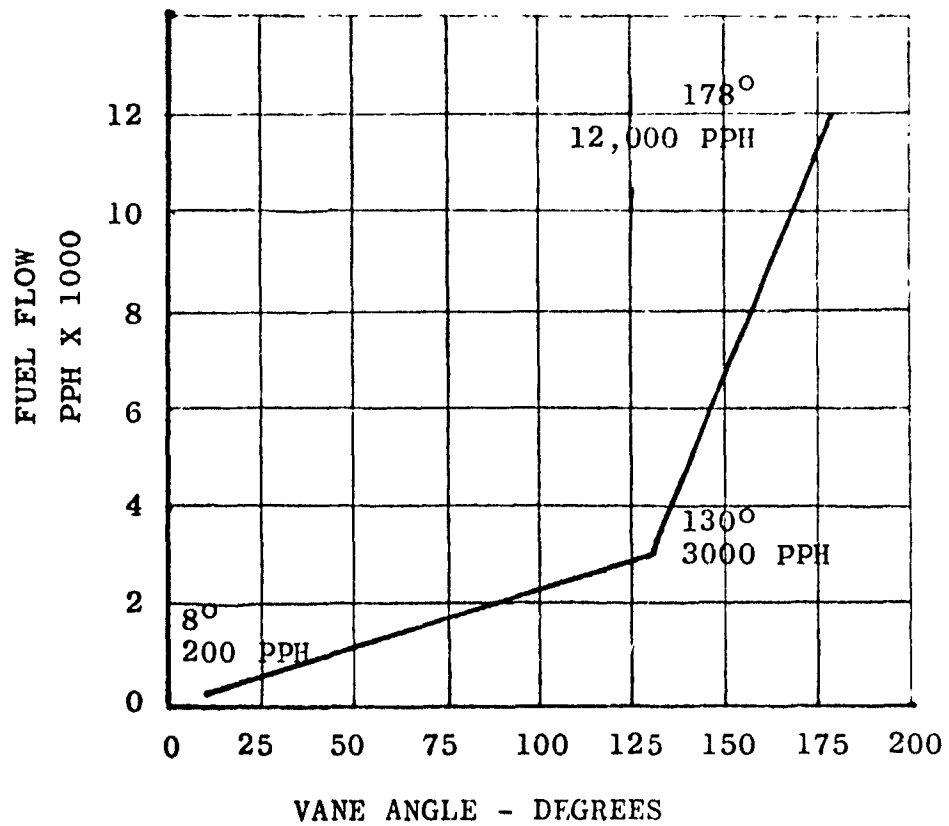


Figure 3 - Fuel Flow Transmitter Flow vs Angle (Typical For F4/J79 Installation)

SECTION IV

BEARING FAILURES

1. INTRODUCTION

A bearing may fail gradually over a long period of time, but when it finally goes there is little or no forewarning. The pilot hears what seems to be an explosion, RPM drops, EGT rises rapidly, and there is an immediate loss of thrust. The explosive sound occurs when the bearing gives way and allows the high speed compressor-turbine rotor assembly to crash against other parts of the engine.

Bearings usually fail because of lack of lubrication, improper mounting, contamination, or fatigue. This report will describe evidence that can be obtained from visual inspection of a damaged bearing and will show how each item of evidence can be related to the cause of failure.

2. EVIDENCE OF INSUFFICIENT LUBRICATION.

The most common cause of bearing failure is lack of lubrication. Within a very short time loss of the cooling agent will cause bearing failure through overtemperature. When the overheated bearing fails, there will be subsequent failure of other bearings due to severe vibration or component seizure. When a complete loss of lubrication occurs throughout a jet engine, the first bearing that fails will usually be the thrust bearing because it bears a greater load than any other bearing in the engine. When the main thrust bearing of a jet engine fails, the compressor rotor will normally shift forward. Bearings that have failed because of lack of lubrication display a burned and melted appearance

and are usually flattened and possibly fused together. The bearing races are severely gouged and pitted. Retainer rings may be broken and partially fused together with the rollers. See Figure 4.

When scavenge pumps fail, oil level in the bearing sump area rises. Rotary action of the bearing and lack of oil flow will raise the temperature of the stagnant oil to a point where it loses its lubricating and cooling ability. The bearing will show overheat by blue discoloration with darker blues indicating higher temperatures. There will also be some metal smearing. However, the damage will not be nearly as severe as failure caused by oil starvation.

3. BEARING MISALIGNMENT

Bearings may fail because of off-squareness or misalignment during installation. For example, an electric pencil used to etch a number on the outer ring face can raise the metal on the bearing surface from .0004 to .0006 inch. The face then far exceeds the raceway to face run-out tolerance of 0.00015 inch. Burrs and foreign particles will have the same effect as the electric pencil. Any misalignment causes overloading of the bearing. Improper loading is indicated by metal transfer or a cocked ball-path on the bearing races. Misalignment can also cause broken retainer rings and can split bearing balls into two equal half balls. See Figure 5.

Remember to find out why misalignment occurred because we cannot prevent future failures unless we know the cause of misalignment.



Figure 4 - Thrust Bearing Failure From Overtemperature Possibly
From Lack of Lubrication



Figure 5 - Number Three Bearing Failure Type of Failure Associated with Misalignment

4. FATIGUE AND SPALLING

Fatigue in bearings occurs on the rolling contact surfaces. It looks like pitting and shows up as irregular sharp edged cavities. However, the cavities are of a greater depth than pitting and this, in turn, progresses into spalling. Fatigue failure of the material is evidenced by breaking out the surface layer of steel. Such a failure starts in a small area, spreads rapidly, and would eventually spread over both the races and bearing surfaces.

Bearing fatigue is caused by repeated shock, stress, or excessive loading. Bearing age is also a contributing factor. See Figure 6.

5. BRINELLING

Brinelling is indicated by depressions in metal. There are two types of brinelling, true and false. True brinelling leaves an imprint of the bearing area on the race and the dent radius corresponds with roller or ball radius. True brinelling is caused by heavy shock loading of the bearing. It can be the result of hard landings or an off center blow during mounting. The damage is a measurable dent on the races and can be determined by a feel test such as fingernail, ball point pen, or by rotation test. False brinelling leaves no measurable indents on the bearing races. It occurs when bearings do not rotate for extensive periods. Loads may be relatively light but slight changes in the surfaces of the raceways may occur as the result of minute axial or rotational movements. False brinelling can also occur in the presence of vibration without rotation. The appearance



Figure 6 - Thrust Bearing Failure from Fatigue and Spalling of the Balls

of false brinelling is the same as true brinelling but the apparent dents cannot be felt or measured.

6 IMPACT FAILURES

Impact failures will show instantaneous stress rupture. Races and bearings may shatter and normally no overtemperature indication will be displayed. Exceptions to this may be friction marks from skid at impact.

7. DAMAGE CAUSED BY FOREIGN MATTER

Ball bearings are particularly sensitive to dirt or foreign matter because of the very high unit pressure between balls and race. The damage caused by different types of foreign matter varies considerably with the nature of the foreign material. Races become worn in the ball paths and the bearings become loose and noisy. The lapping action increases as the fine steel removed from the bearing surfaces adds more lapping material. Hard and coarse foreign matter, such as metallic particles, produces small depressions. Jamming of the hard particles between the bearings and the races may cause the inner race to turn on the shaft or the outer race to turn in the housing.

8. RACE SKIDDING

Smearred skid marks on the balls, rollers, or races are an indication of improper lubrication or sudden acceleration. Skid marks will appear as a film on the otherwise highly polished surface of the race. Excessive wear in the pockets of the cage and discoloration of the balls and rollers are defects that usually accompany race skids. Microscopic examination of a race skid will usually show that the film is an actual transfer of metal.

SECTION V

DETERMINATION OF RPM, COMPRESSOR INLET AIR TEMPERATURE AND THROTTLE ANGLE

1. DETERMINATION OF RPM

In order to use the components within the main fuel control for RPM determination, remove the control cover assembly by removing the fifteen screws around the periphery of the assembly, See Figure 7. This exposes the tach rack, tach servo piston, 3D bracket, and 3D shaft assembly consisting of corrected fuel cam and variable vane scheduling cam.

The tachometer servo piston may be used for RPM determination, Figure 7. The distance (X) as seen in Figure 8 should be measured prior to disturbing the remainder of the internal parts of the control. After this measurement has been taken, the percent RPM may be determined from the chart at the top of Figure 8.

On the left end of the 3D shaft assembly, as seen in Figure 7, may be found the cam end plate and the cam end pointer. The zero degree cam angle for the end plate is 95° clockwise from the index hole in the end plate. Change in engine RPM rotates the 3D shaft assembly. Changes in compressor inlet temperature (CIT) translates the 3D shaft. If the shaft assembly is jammed as a result of the impact, the number of degrees that the end pointer indicates from the above described zero degree mark should be measured in a counter clockwise direction. After this has been accomplished, the chart in Figure 9 should be used to determine the percent RPM that the engine was operating at the time of impact.

If the shaft assembly is not jammed, the surfaces of the cams

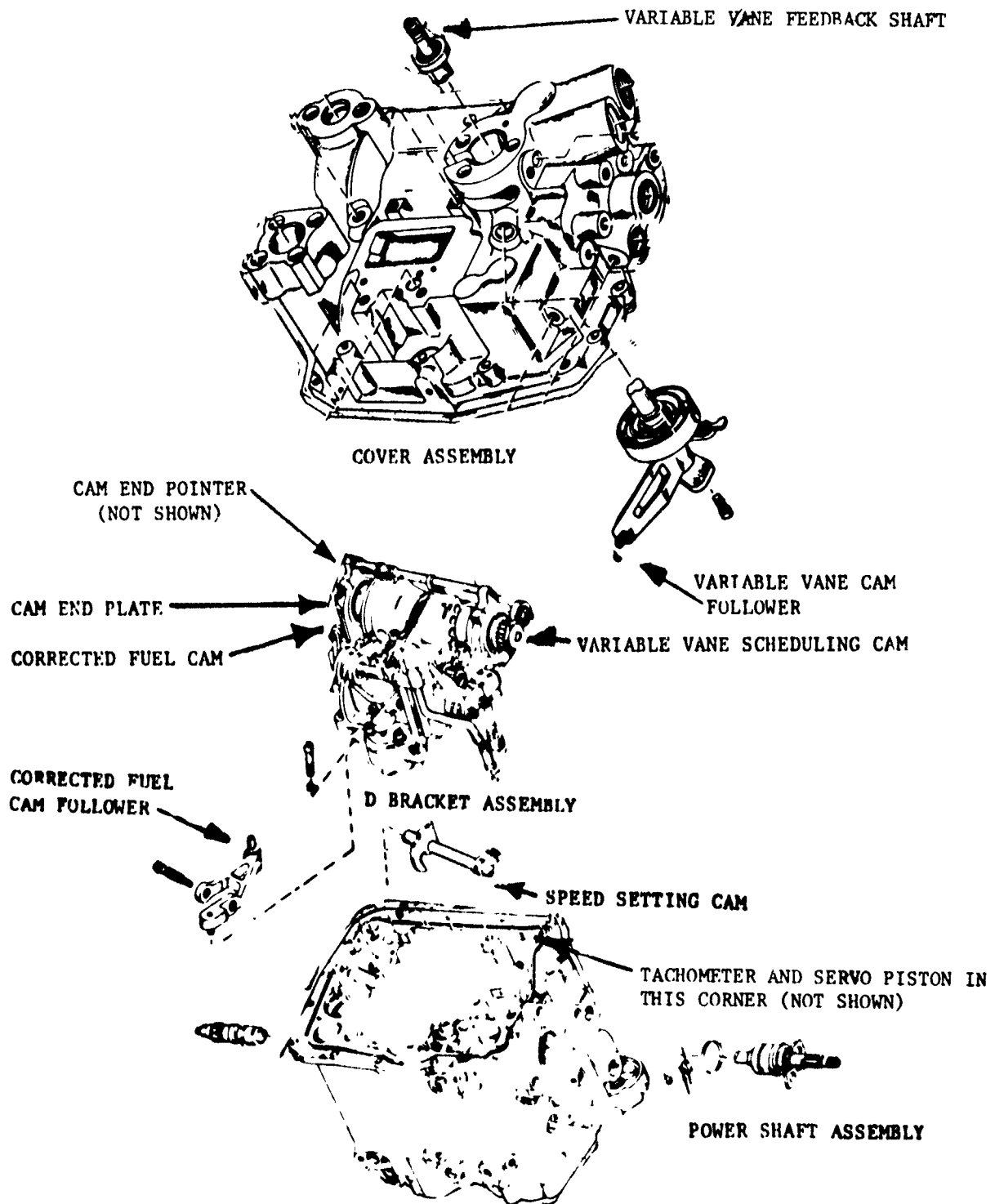


FIGURE 7 MAIN FUEL CONTROL J79-15 AND -17

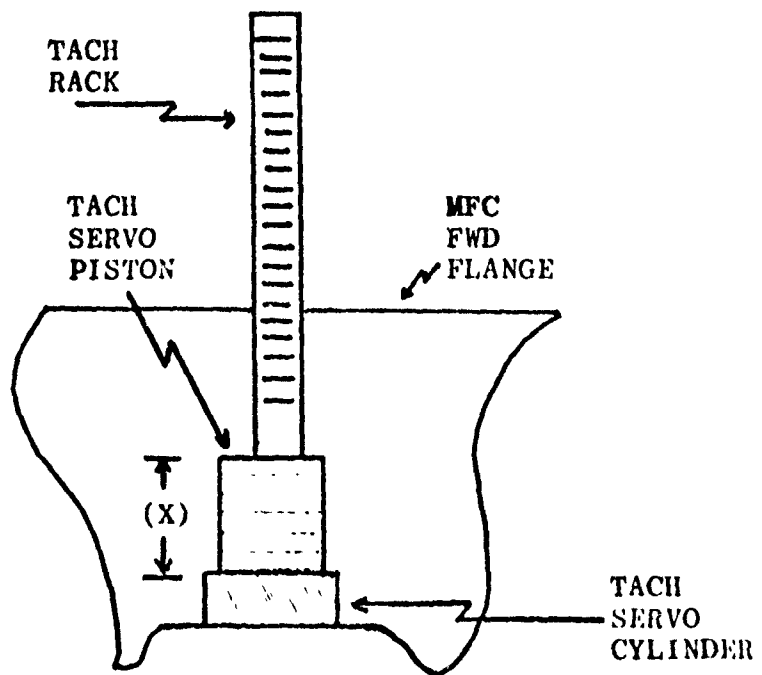
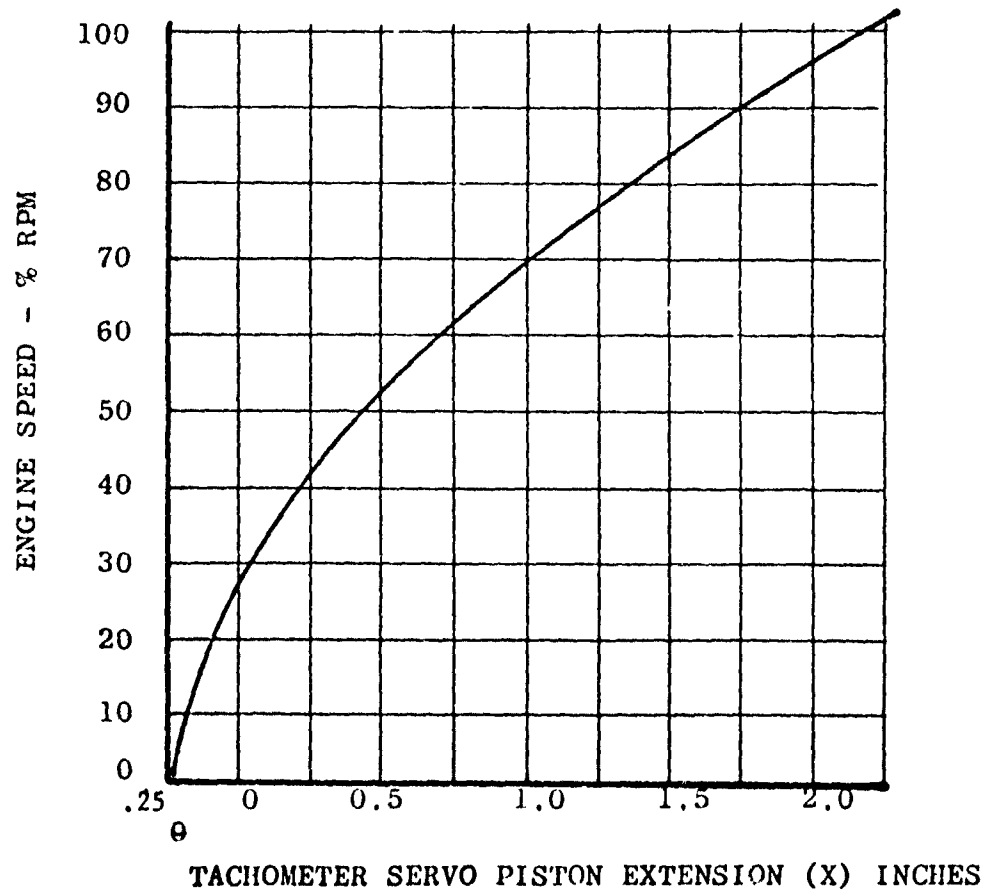


Figure 8 - J79-15/-17 Main Fuel Control Tachometer Servo Piston Extension vs Percent RPM

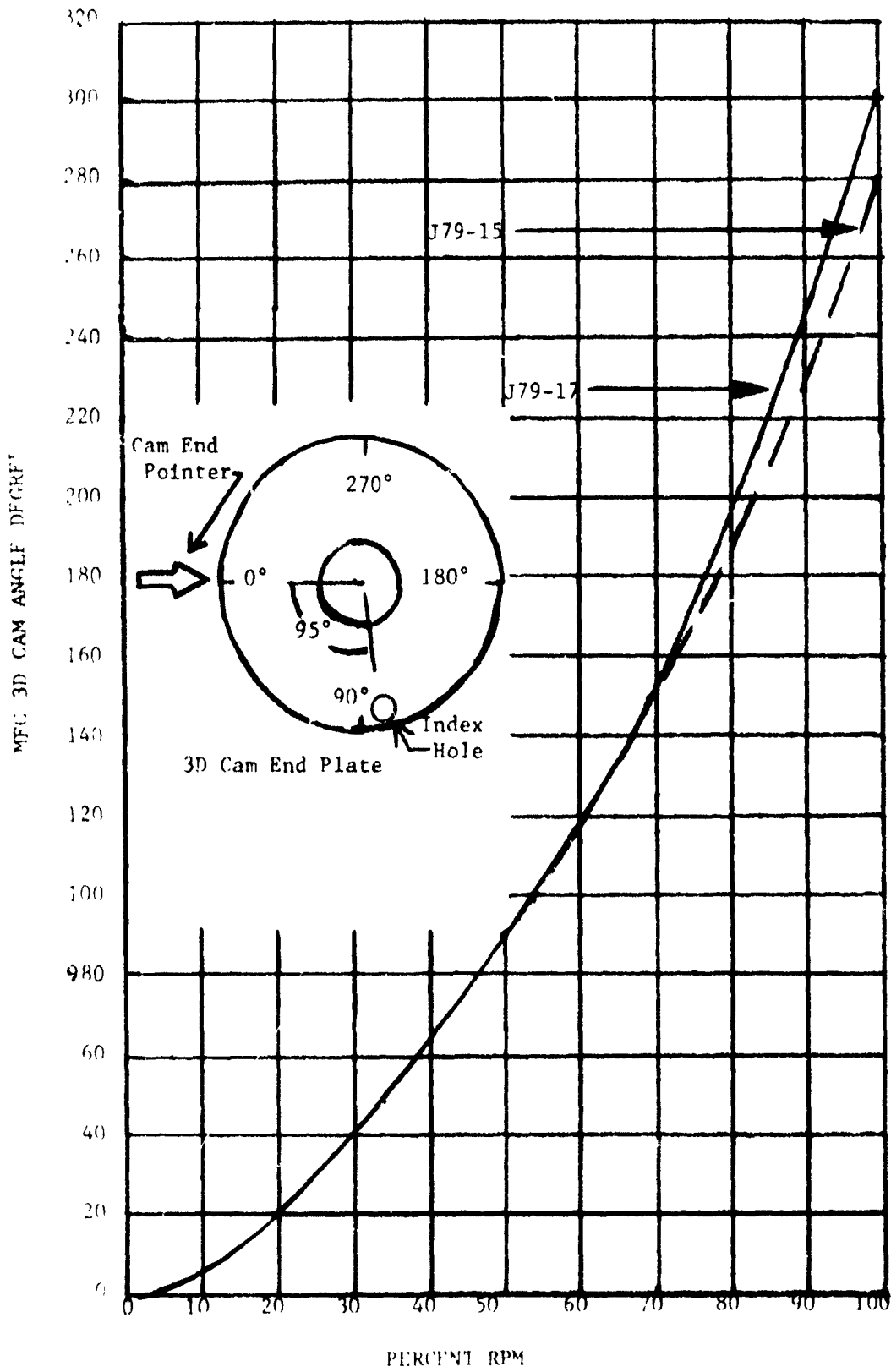


Figure 9 - 3D Cam Angle vs RPM

for corrected fuel and variable vane scheduling should be examined for impact marks from their respective cam followers. If these marks are prevalent, the shaft can be rotated to align the impact marks with the cam followers. The 3D cam angle in degrees can then be determined and the RPM established as described above.

The position of the variable vane actuators can, under some circumstances, give an indication of engine RPM. These actuators are positioned by fuel pressure from the main fuel control, between approximately 63% and 95% corrected engine RPM. The variable vanes are positioned in accordance with the variable vane schedule (Figure 12) which are functions of engine RPM and compressor inlet temperature (CIT). Below 63% the vanes are fully closed and above 95% fully opened. However, the conclusion should not be drawn that because an actuator is found fully extended, the engine RPM was below 63% or that because an actuator is found fully retracted the RPM was above 95%. For CIT determination see Paragraph 2.

The VC actuators will normally extend on engine coast down, and if an engine coasts to below 63% RPM, the variable vanes will usually be fully closed. However, heavy impact forces can force an actuator to unusual positions.

If in an accident, impact has been severe enough to seize the compressor rotor immediately, or to freeze the actuators, the position of the actuator extension will indicate engine RPM at time of impact. However, the use of actuator extension position to estimate engine RPM must be done very cautiously as the actuator

is quite likely to have been torn loose from the engine and its extension changed. Normally, the actuators' extension position should be used to substantiate estimates of engine RPM with the understanding that RPM estimates obtained in this manner may be in error.

Impact marks inside the actuators may assist in locating the position of the actuator at the time of impact. However, these must be used with caution as there may have been more than one impact, and the actuators may have moved between the first and second or subsequent impacts.

This is usually readily discernible. The initial impact marks normally are distinct circumferential marks and secondary impact marks will normally produce metal smearing appearances in the axial direction of the actuator cylinder.

As may be seen in Figure 12, there is a definite relationship between the variable vane positions and RPM for a given compressor inlet temperature. The position of the variable vane actuators may be used to determine the position of the variable vanes. For this determination measure the distance X as shown on the bottom of Figure 10. After this extension has been measured, use the chart at the top of Figure 10 to determine the degree open or closed for the variable vanes.

The inlet guide vane feedback box mounted on the variable vane feedback shaft of the main fuel control (Figures 7 and 13) may be used for variable vane position determination. Measure the over-travel X, as shown at the bottom of Figure 11. Use the chart at the top of Figure 11 to determine the degree open

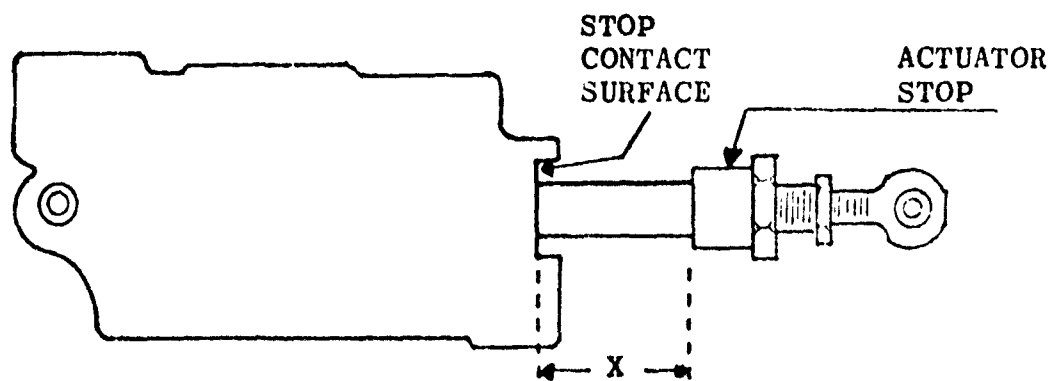
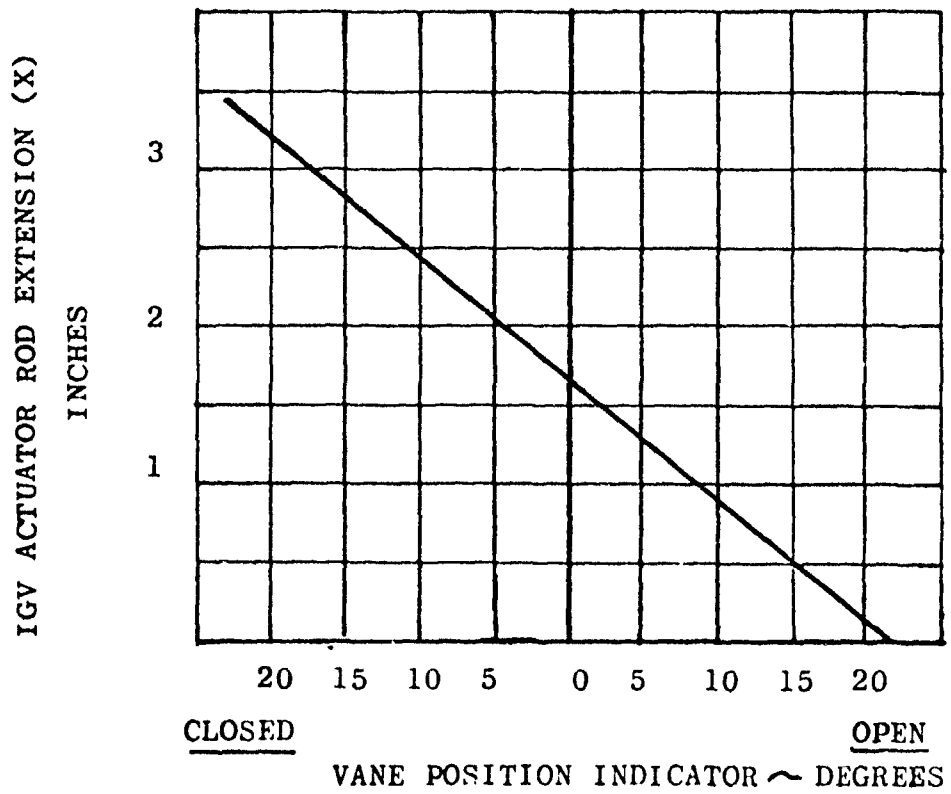


Figure 10 - J79-15/-17 IGV Actuator Rod Extension vs Vane Position Indicator

IGV FEEDBACK
OVERTRAVEL "X" INCH

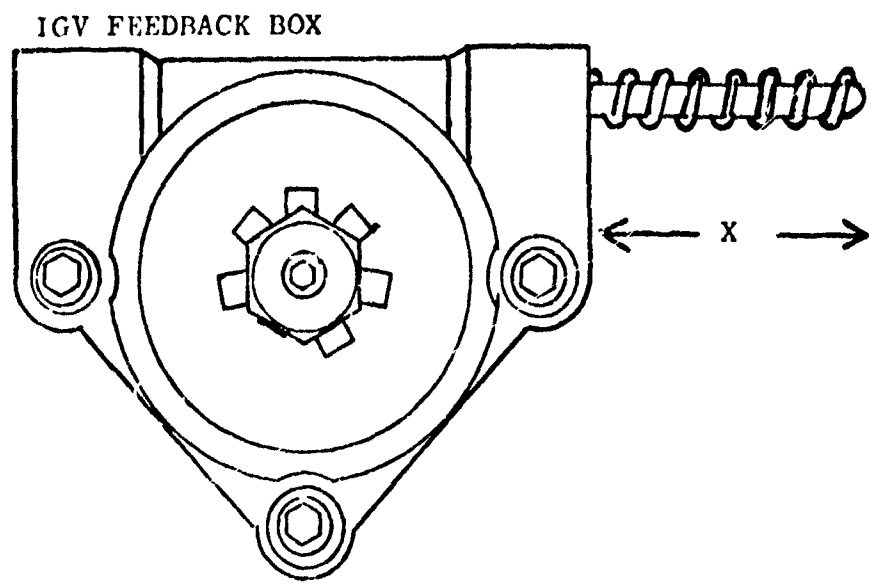
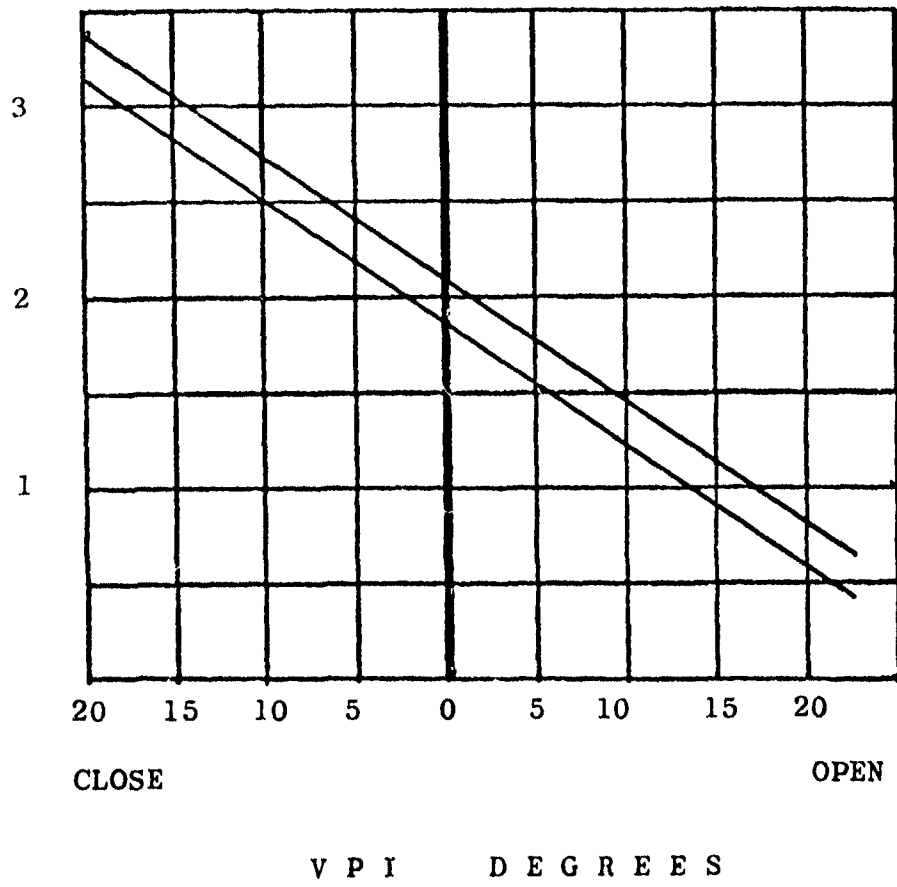


Figure 11 - J79-15/-17 IGV Feedback Overtravel vs VPI Degrees

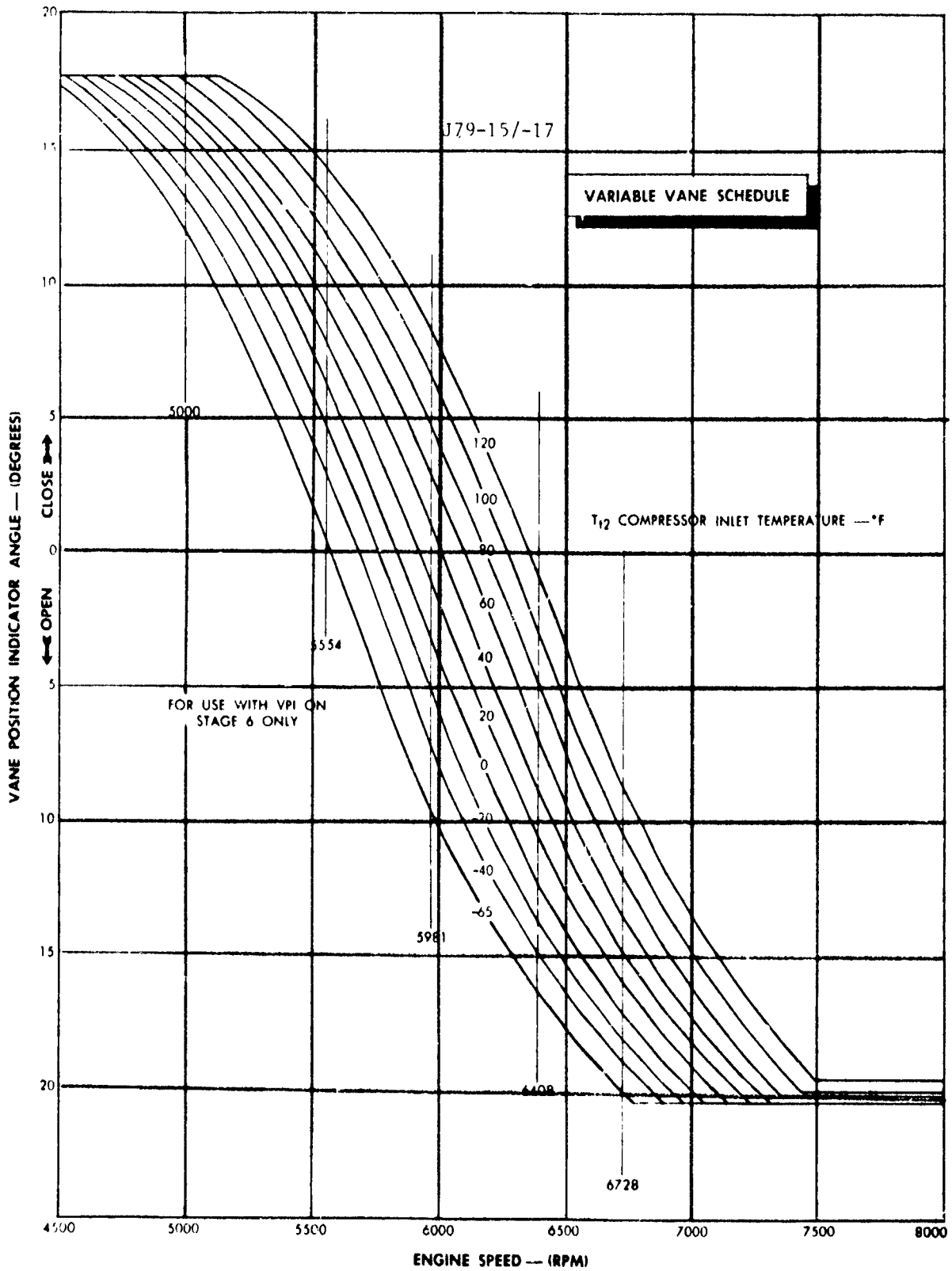


Figure 12 Variable Vane Schedule
-33-

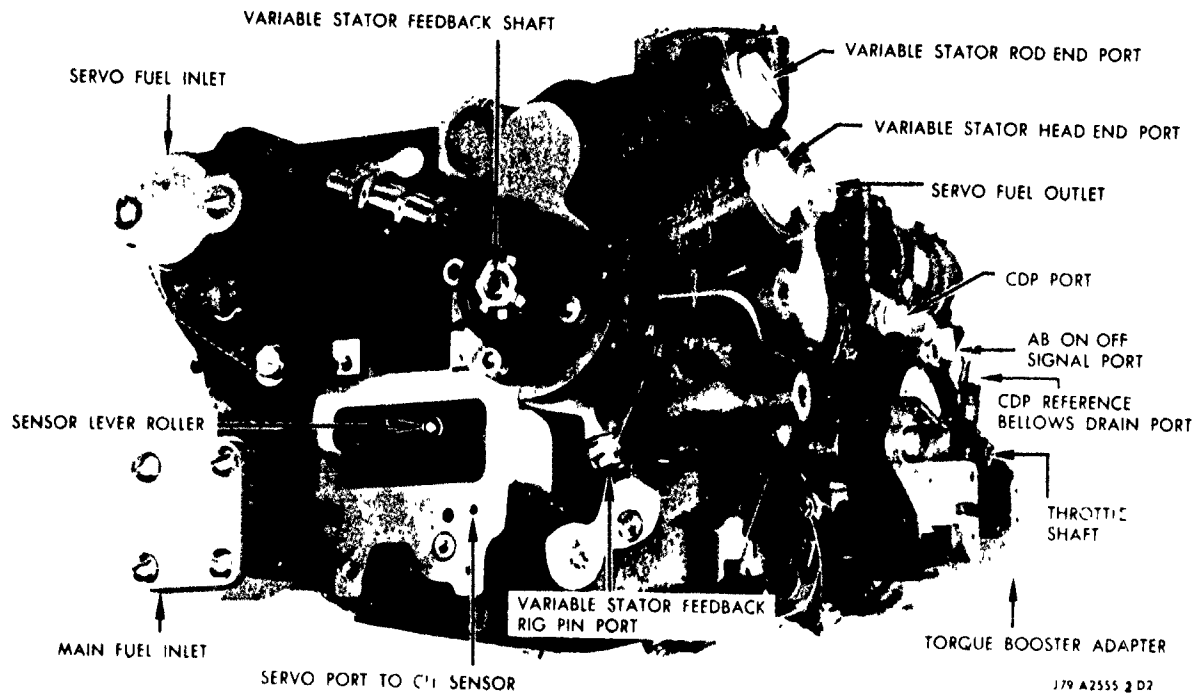


FIGURE 13 - MAIN FUEL CONTROL (FRONT VIEW)

or closed for the variable vanes.

The position of the indicator on the variable vane feedback shaft, with respect to the stop block as shown in Figure 14, may also be used for variable vane position determination. To use this method the feedback box must be removed from the feedback shaft. Care should be exercised in removing the nut on the end of the shaft to prevent turning the shaft. The degree X should be measured as shown in Figure 14. This measurement should be applied to the chart at the top of Figure 14 to determine the vane position.

Knowing the variable vane position and the CIT, the RPM can be determined from Figure 12.

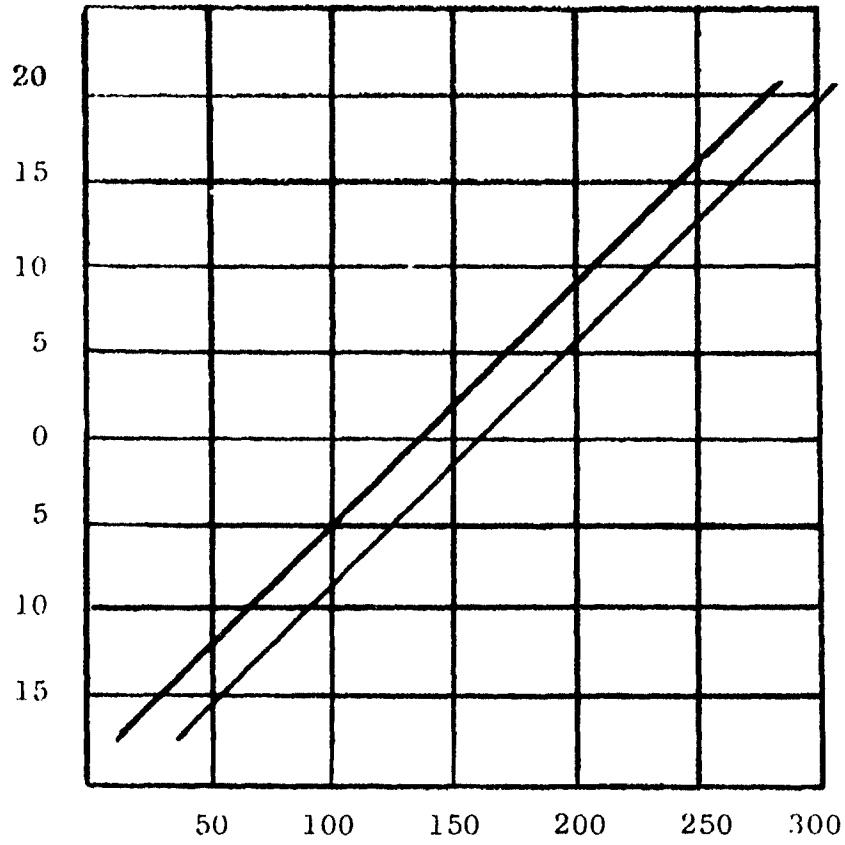
A rough estimate of RPM may be made by measuring the distance X (Figures 15 and 16) on the variable exhaust nozzle actuator as illustrated and applying this measurement to the chart at the top of the page. It must be recognized that the chart is based on the engine following a fixed mechanical schedule as a function of throttle angle. However, the nozzle will vary from the mechanical schedule whenever Exhaust Gas Temperature (EGT) exceeds the T_5 vs RPM reference schedule. For this reason only a rough estimate of RPM may be determined by this means.

2. COMPRESSOR INLET TEMPERATURE

The compressor inlet temperature (CIT) can be determined by measuring the distance between the 3D cam end plate and the end bracket (as shown in Figure 17). The 3D cam end plate and the end bracket are within the main fuel control (Figure 7). If the CIT is to be determined by another method, such as by

OPEN

VANE POSITION INDICATOR SETTING ~ DEGREES



CLOSE

MAIN FUEL CONTROL (J79-15/-17)

IGV FEEDBACK SHAFT ~ DEGREES (X)

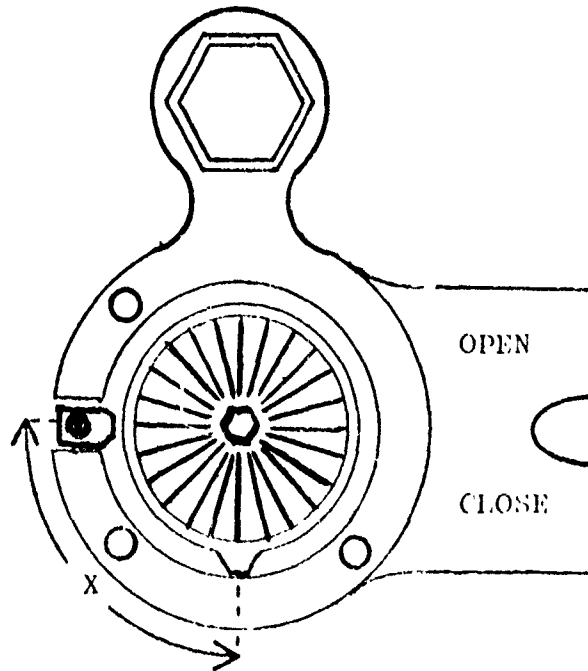


Figure 14 - Main Fuel Control IG V Feedback Position vs Vane Position Indicator Setting

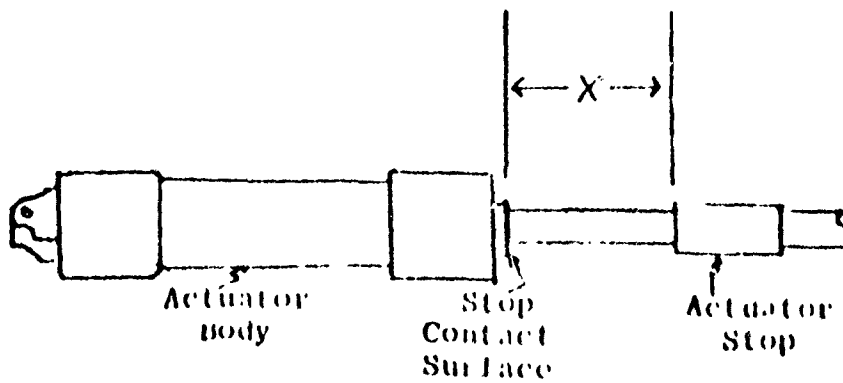
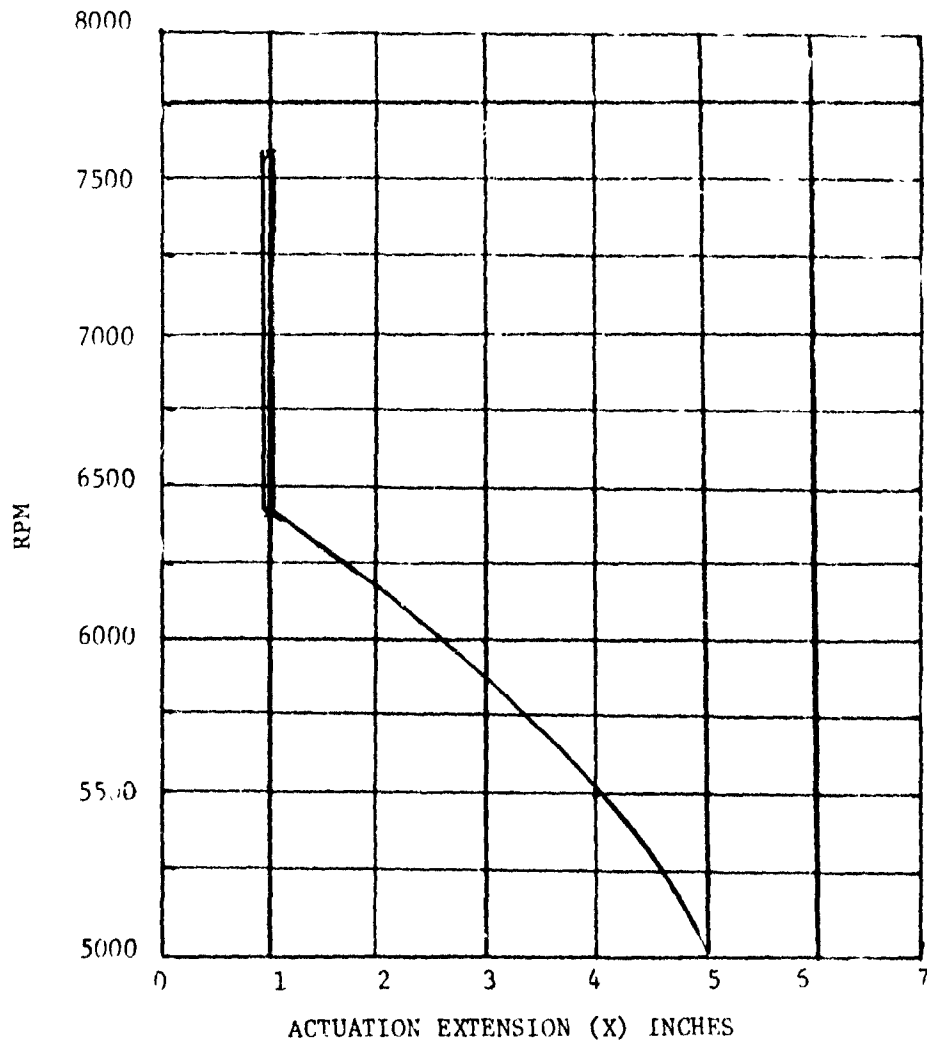


Figure 15 - J79 Exhaust Nozzle Actuator Extension vs RPM

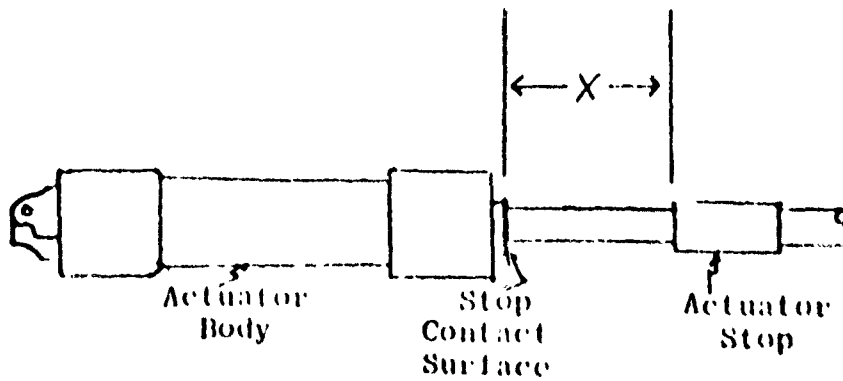
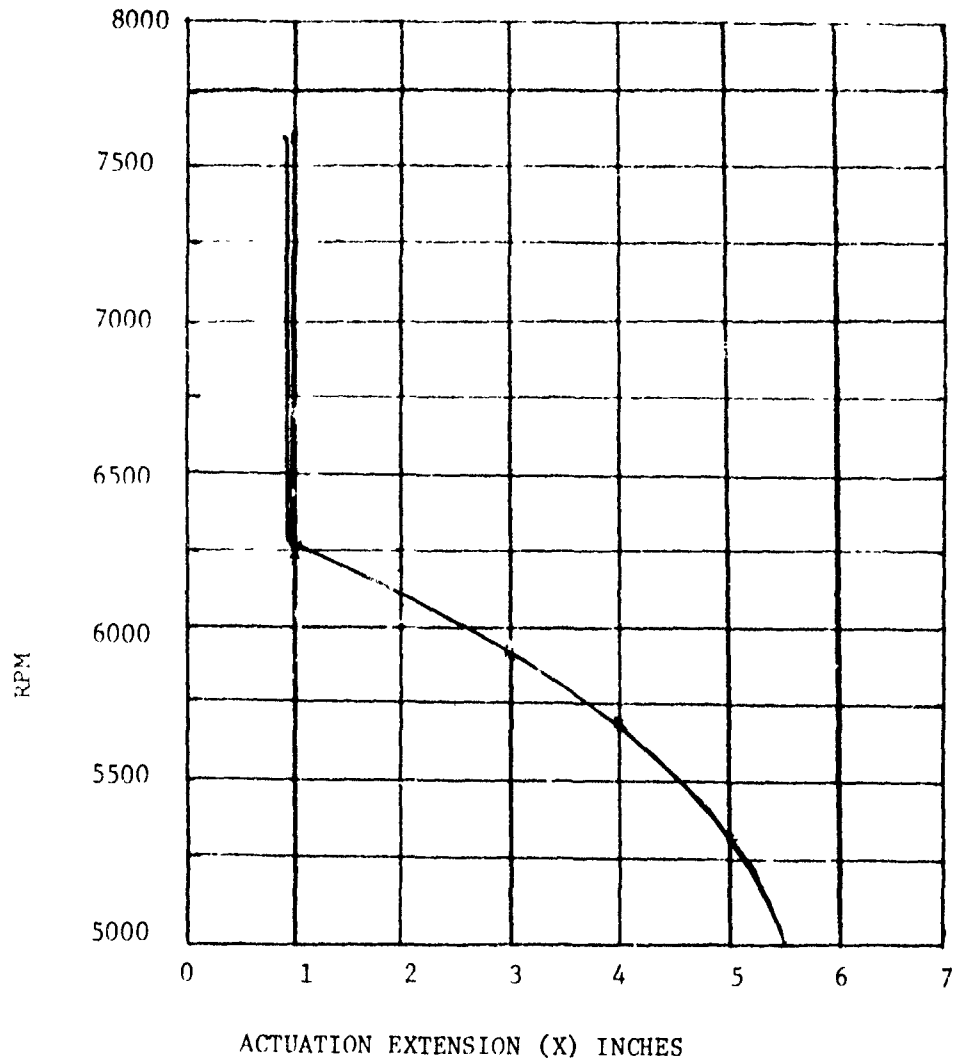


Figure 16 - J79-17 Exhaust Nozzle Actuator Extension vs RPM

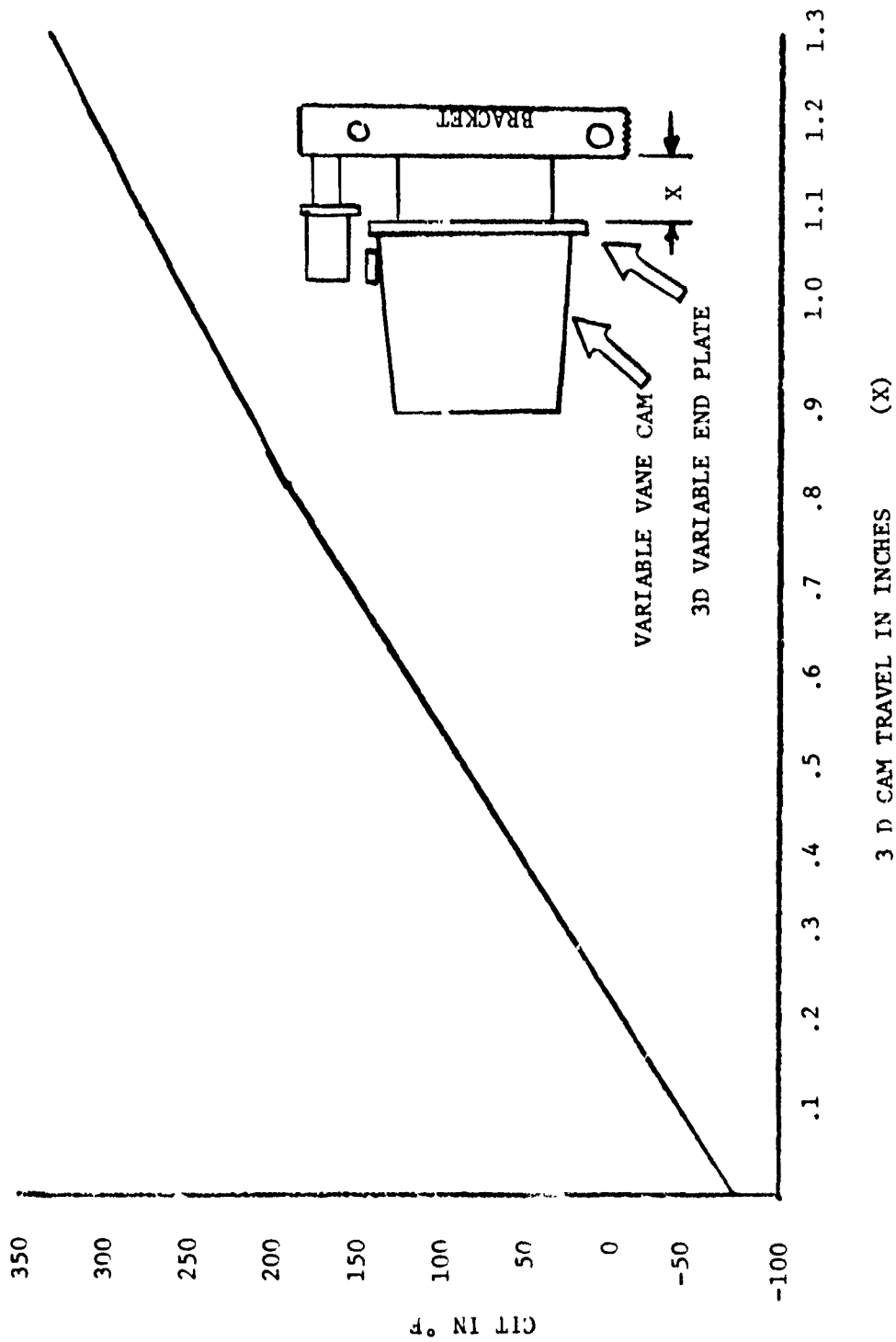


Figure 17 -- Main Fuel Control 3D Cam Travel vs CIT

knowing the ambient temperature, it must be corrected for the temperature increase due to the ram effect. Figure 18 can be used to determine inlet temperature if airspeed and ambient temperature are known.

In the case of rapid descent from colder temperatures, the temperatures as sensed by the control may be as much as 15°F cooler than actual stagnation temperature. Therefore, if high speed impact from altitude is suspected, the stagnation temperatures taken from Figure 18 should be reduced by 15°F.

Figure 19 illustrates the maximum RPM vs CIT.

3. THROTTLE ANGLE

The throttle position at time of impact can be determined by examination of the torque booster. To make this determination, the end cap (Item 73 of Figure 20) should be removed. A measurement should be taken from the end of the torque booster housing and the piston assembly, (Item 82 of Figure 20) as shown in Figure 21. After this dimension is obtained, the throttle position may be determined through the use of the chart illustrated in Figure 21.

The above procedure assumes that the piston assembly did not move as a result of the impact. Careful examination of the walls of the torque booster cylinder may reveal impact marks that indicate the piston was at a different position at time of impact. If this is the case, align the piston with the impact marks prior to making the measurement as displayed in Figure 21.

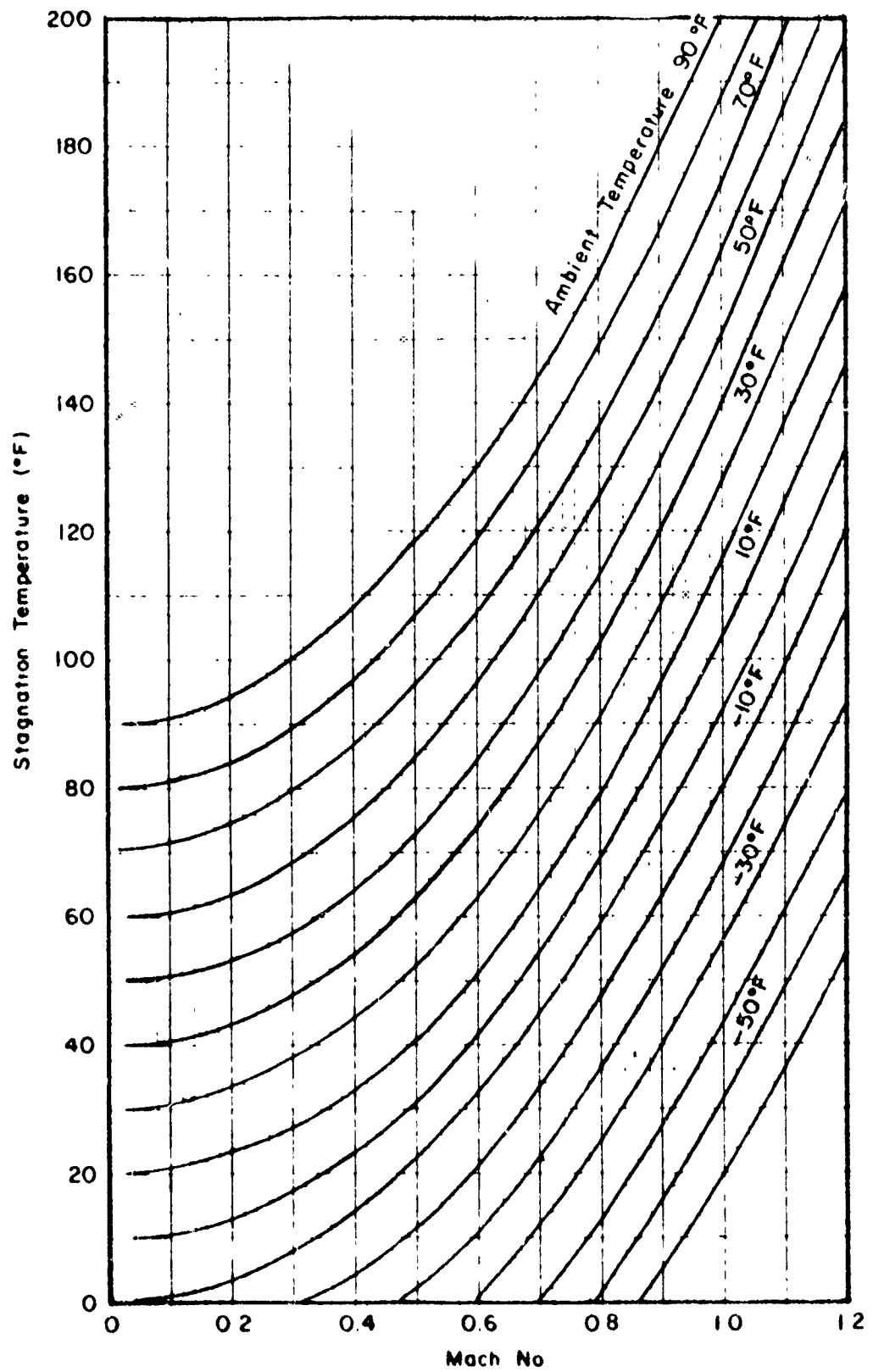


Figure 1P Mach No vs. Stagnation Temperature

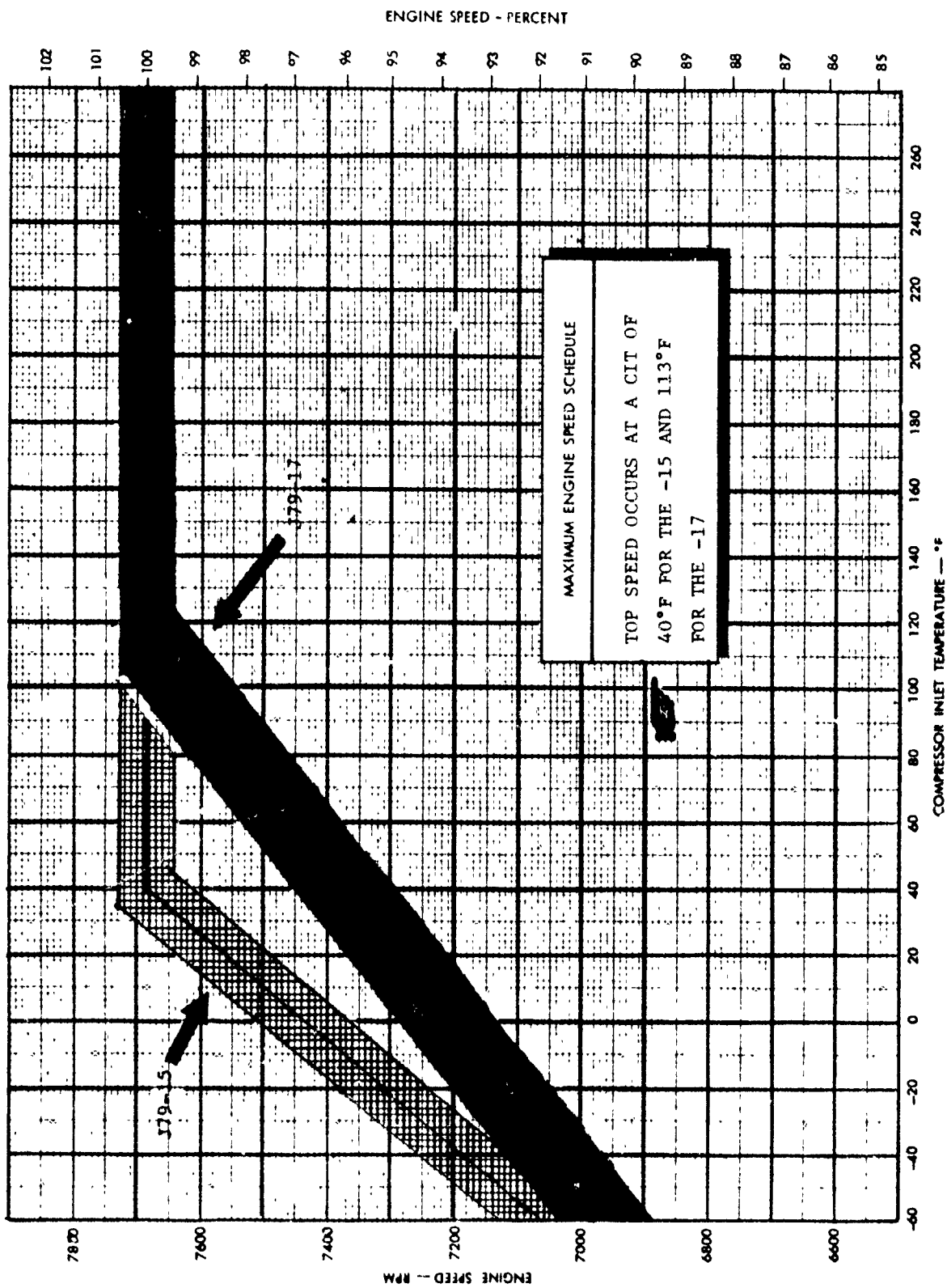
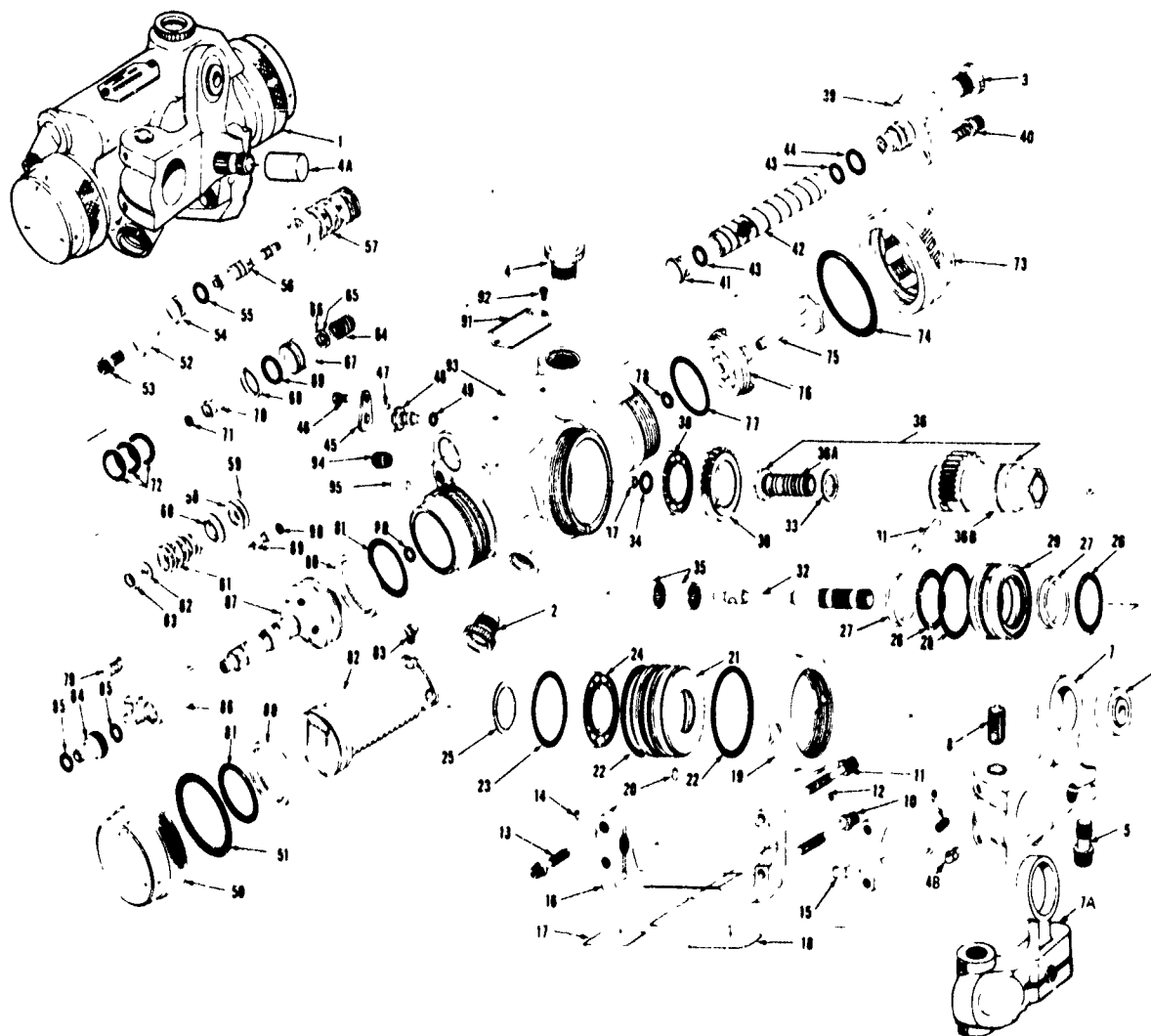


FIGURE 19 - ENGINE SPEED VS CIT



4-114

FIGURE 20 - TORQUE BOOSTER ASSEMBLY

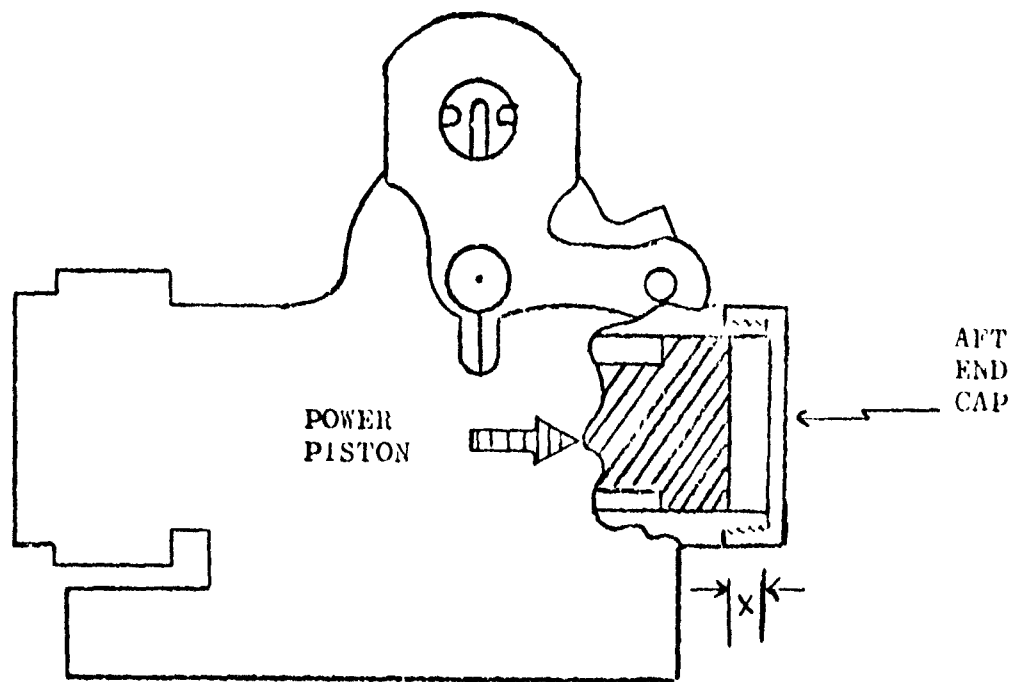
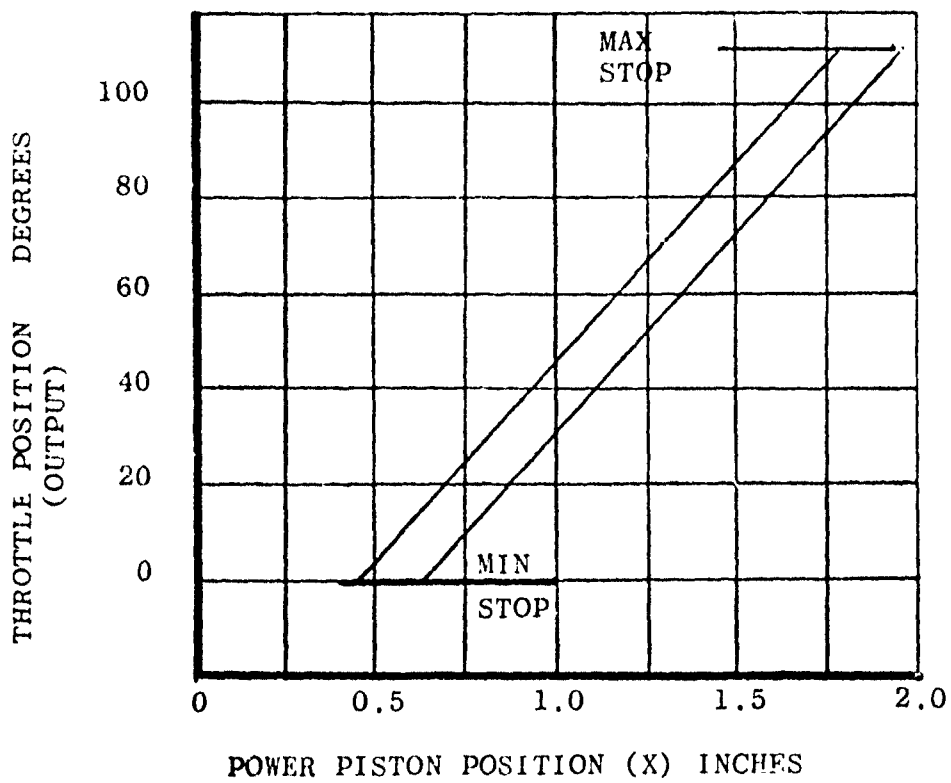


Figure 21 - Throttle Torque Booster - Power Piston Position vs Throttle Output Position

A rough estimate of throttle angle may be made by measuring the distance X (Figures 22 and 23) of the variable exhaust nozzle actuator as illustrated and applying this measurement to the chart at the top of the page. This is a rough estimate.

After the throttle angle has been established, the RPM of the engine may be determined through the use of Figure 24.

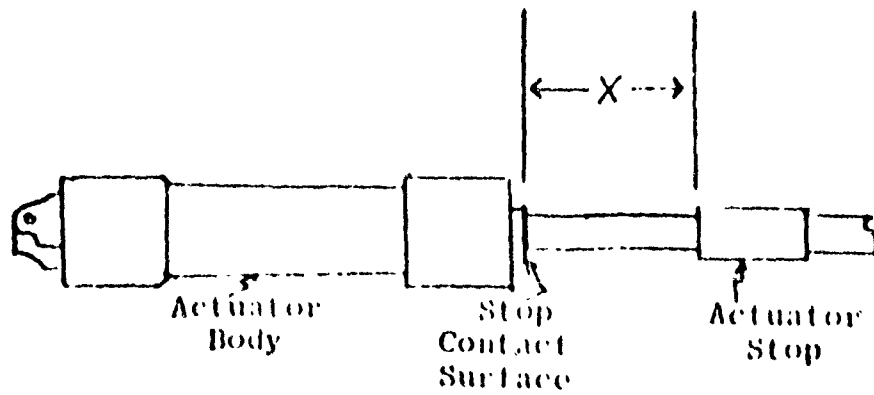
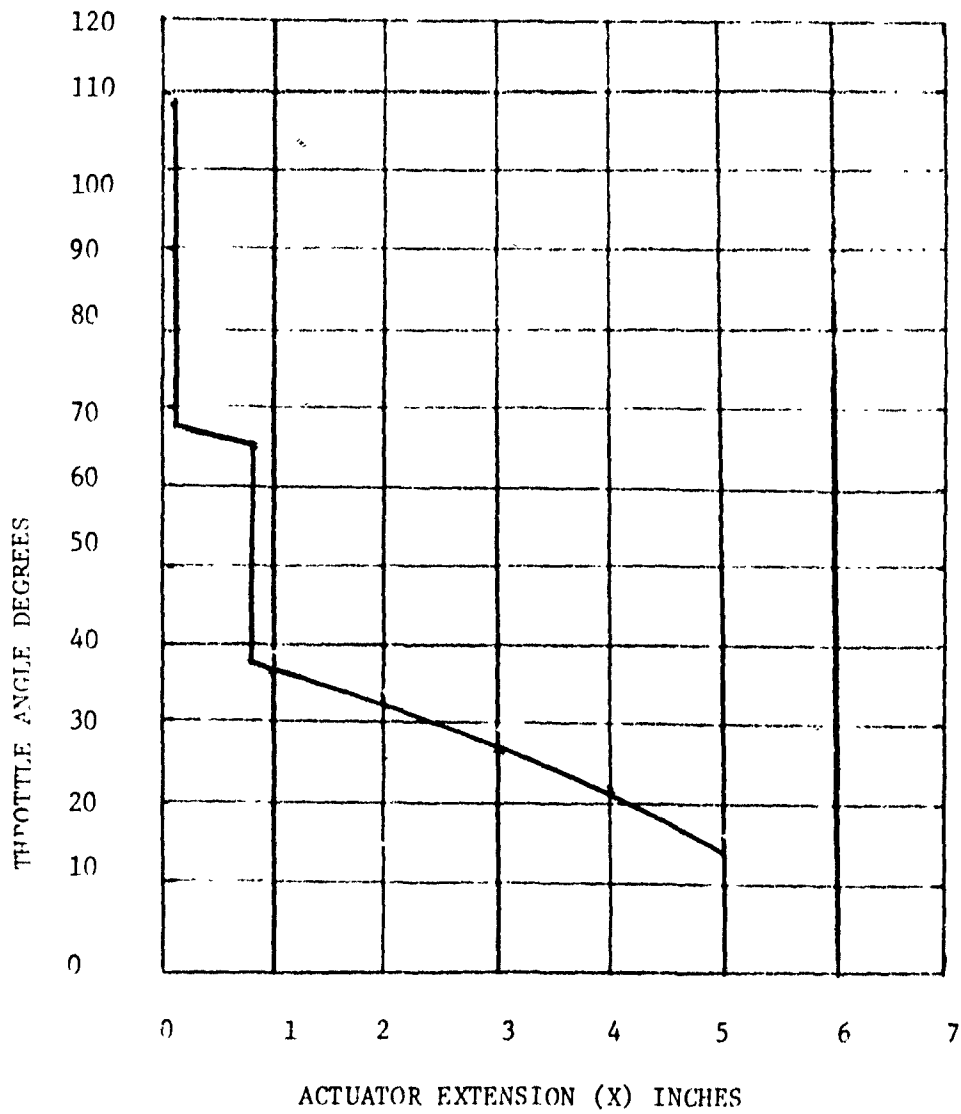


FIGURE 22 - J79-15 EXHAUST NOZZLE ACTUATOR EXTENSION VS THROTTLE ANGLE

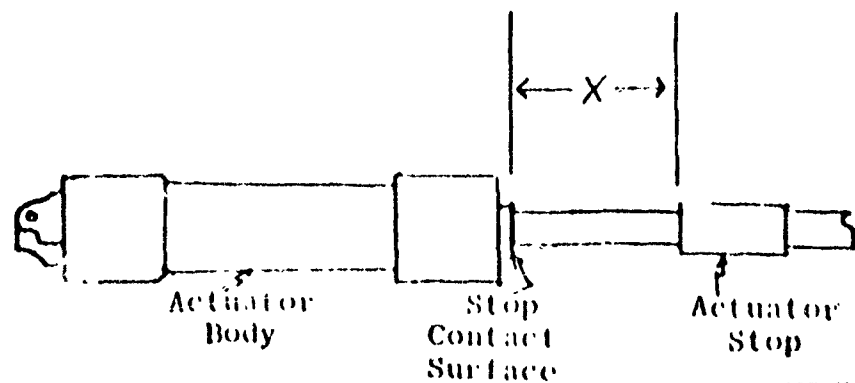
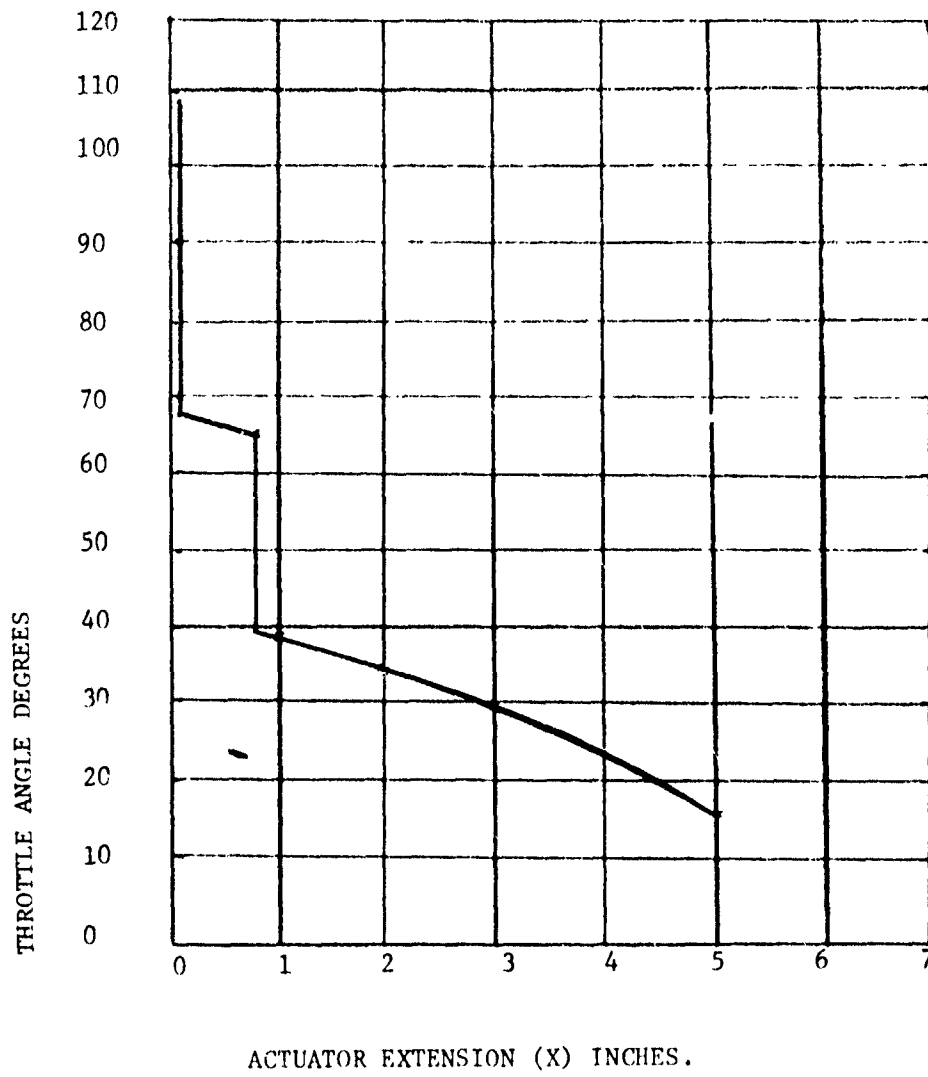


Figure 23 - .179-17 EXHAUST NOZZLE ACTUATOR EXTENSION VS THROTTLE ANGLE

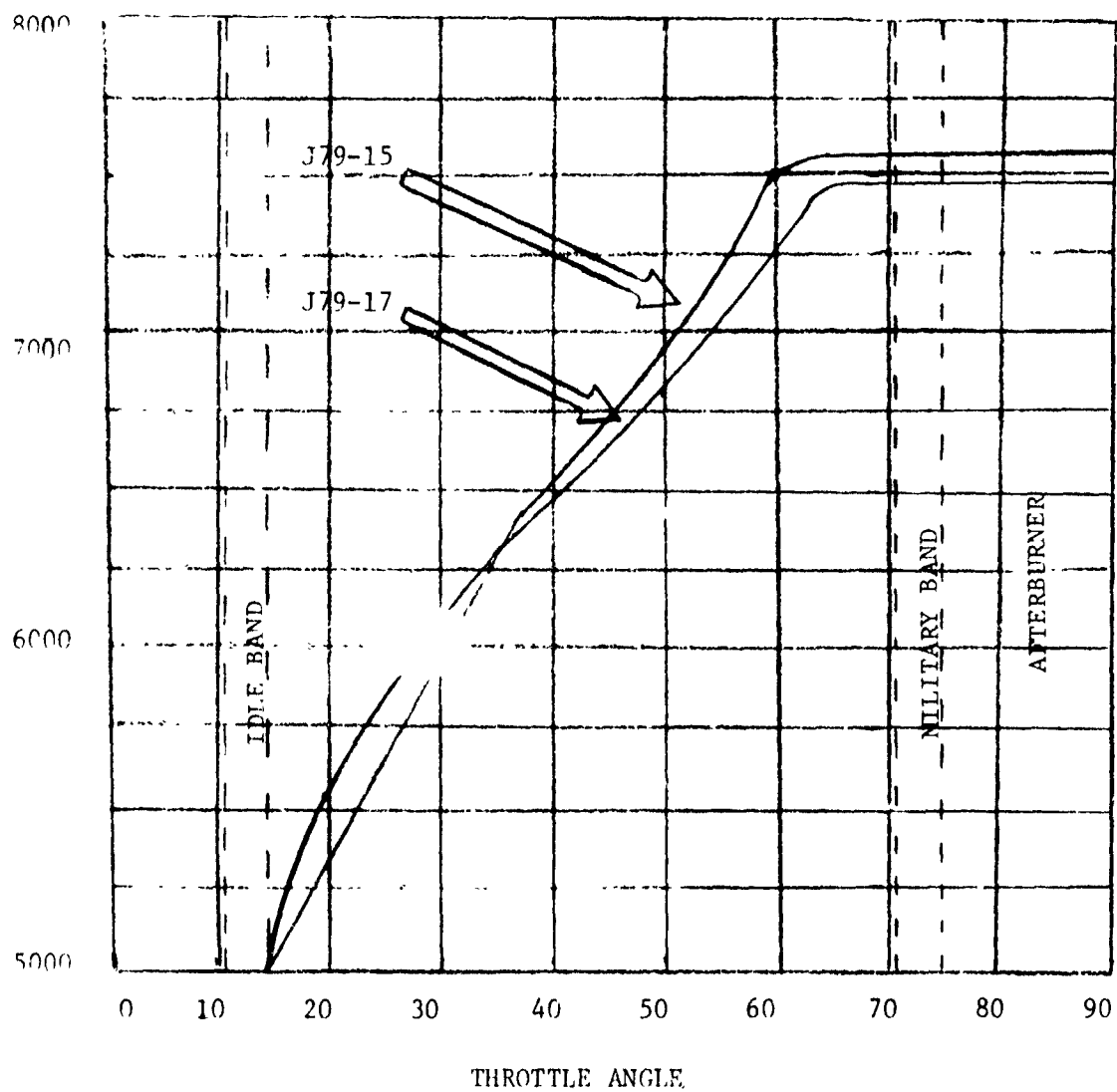


FIGURE 24 - THROTTLE ANGLE VS RPM

SECTION VI

TEMPERATURE DETERMINATION

The determination of approximate engine RPM at instant of impact is a major part of the job of estimating engine power; however, the propulsion investigator should also be alert to possible indications of the engine operating temperature.

If aluminum or foreign materials are ingested into the engine and the engine is operating at idle RPM or above, there may be enough heat to melt the material and cause it to fuse with the turbine and exhaust components. This metal fusion can be recognized by the deposits of aluminum color and may be expected to occur on the combustion liners, turbine nozzles, turbine blades, exhaust cone, and exhaust nozzle.

The location, pattern, and degree of fusion on the hot parts is indicative of the engine operating temperature at the time of ingestion. If the engine was operating at normal temperature when the ingestion occurs, the fusion will be thorough, even, and smooth appearing. The metal cannot be flaked off with a knife or eraser. If the deposits are rough and globular and can be scraped off, then the fusion took place during rapid deceleration which could have resulted from interruption of fuel flow. If there were a flameout or the engine was shut down before the ingestion no metal fusion could be expected to occur. In a matter of seconds, the engine hot parts will cool below the melting point of aluminum.

Sometimes the metal fusion will be limited to a specific

area such as a portion of the turbine nozzle or just a certain portion of the combustion chamber. This indicates that the ingestion followed impact. An even annular distribution would be indicative of in-flight ingestion. Another indication of sequence of events would be metal fusion on dents in the turbine or exhaust section. If a dent is caused by impact and fusion occurs only on its forward side, then it is obvious that the ingestion followed initial impact and the engine was operating at time of impact.

The J79 compressor uses only steel alloy. Should metal fusion be in evidence, then the material came from a source other than the compressor. The aircraft air inlet duct and associated component, should be examined as a probable source. A laboratory analysis of the fused metal should be made to help in identifying the unit from which the material originated.

Engine temperature can sometimes be estimated by the condition of ingested debris. If the crash occurs in a wooded area at a low angle of impact, considerable wood may be ingested. A charred or burnt condition of the wood pulp would indicate that the engine had been operating at time of impact.

Impact forces sometimes cause engine instrument pointers to strike the instrument face leaving a very light deposit of fluorescent material. Each instrument face should be carefully examined with a black light. There may be telltale marks which will reveal the exact engine RPM, exhaust gas temperature, oil pressure, or fuel pressure that existed at the instant of impact.

SECTION VII

DETERMINATION OF VEN AREA

If the afterburner section of the engine is intact and undamaged, nozzle area can be estimated by taking measurements across the inside of the outer leaves. Quite often, however, the variable exhaust nozzle (VEN) is severely damaged, and the estimation of the VEN area must be determined by other means.

The position of the actuator rod with respect to actuator body may be used to determine nozzle area. The actuator extension X as shown on the bottom of Figure 25 should be used with the chart at the top of the figure to determine the nozzle area.

For the J79-15 engine the overtravel distance X as shown on the bottom of Figure 26 should be applied to the chart at the top of the figure for nozzle area determination.

The position of the feed back rig wheel may be used for exhaust nozzle diameter determination, see Figure 27. Remove the cover plate as shown in Figure 27 and insert a feedback rig pin. Place a mark on the feedback shaft and a corresponding mark on the adjacent housing. Rotate the feedback shaft until the feedback rig pin engages the slot in the feed back rig wheel. Measure the angle between the two marks described above. The chart at the top of the page may then be used to determine the exhaust nozzle diameter.

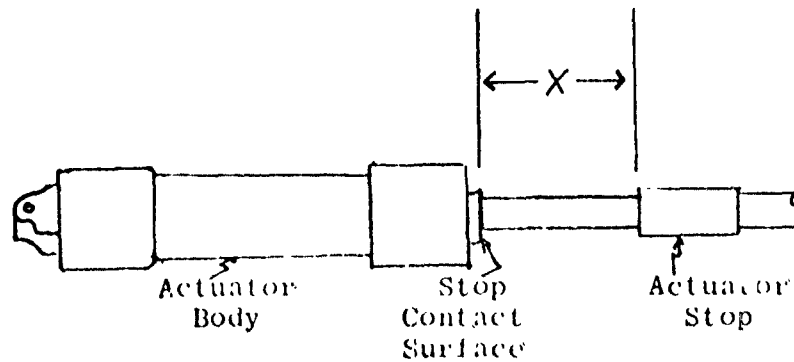
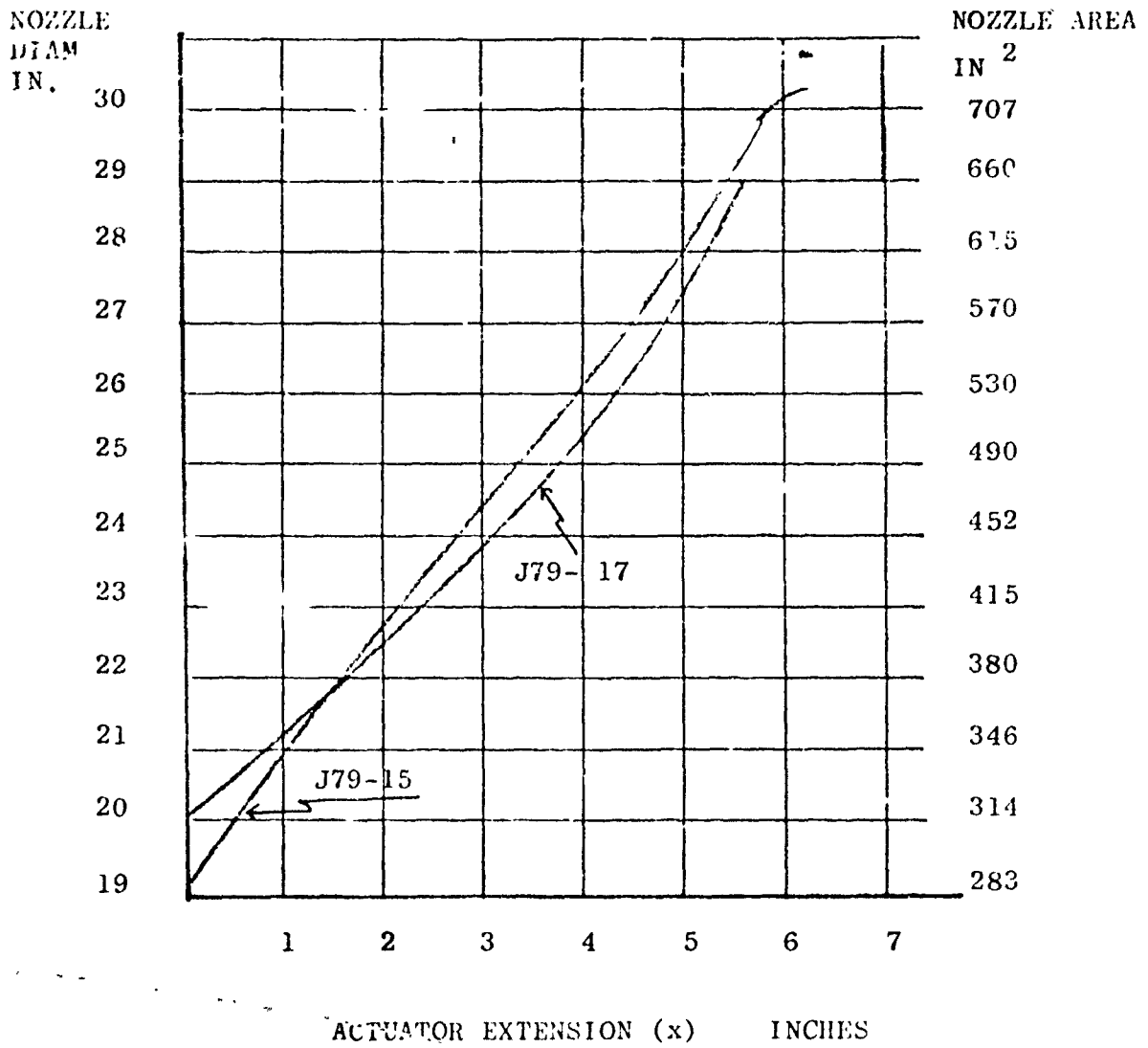


Figure 25 - Actuator Extension vs Nozzle Position

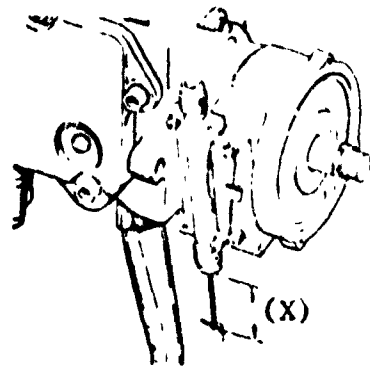
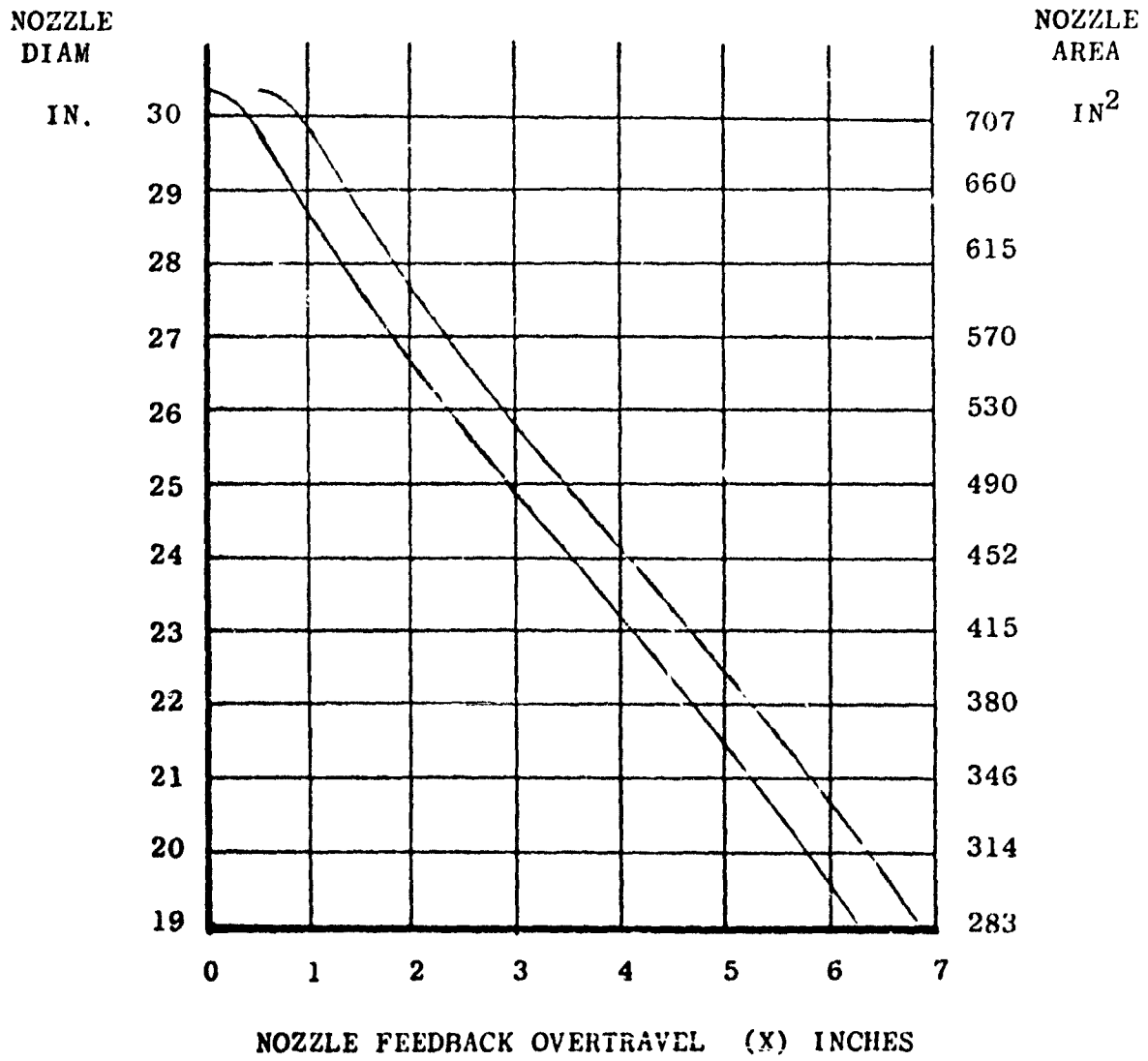


Figure 26 - J79-15 Exhaust Nozzle Feed Back Overtravel vs Nozzle Position

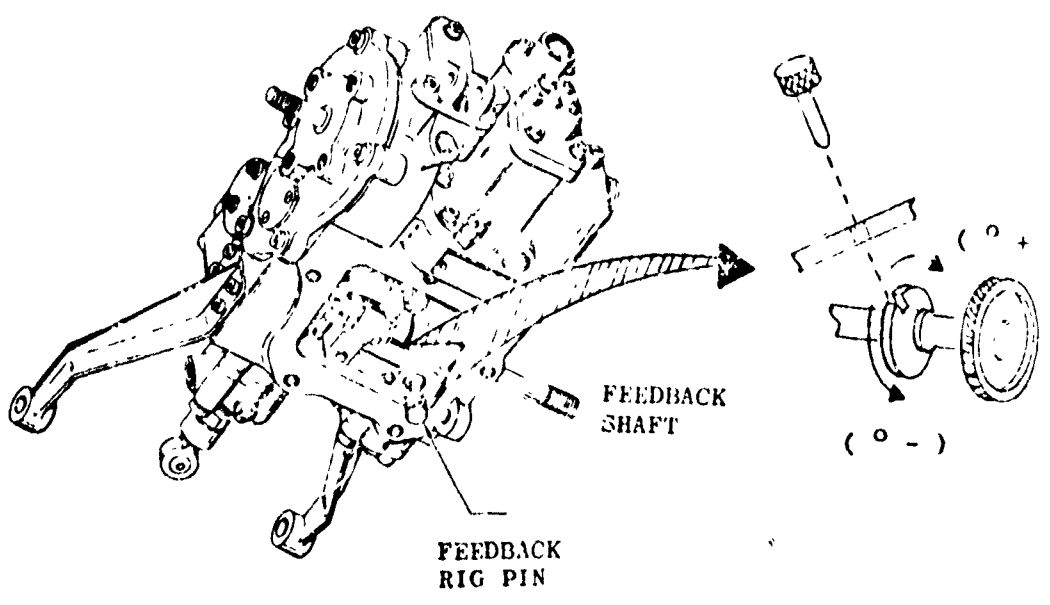
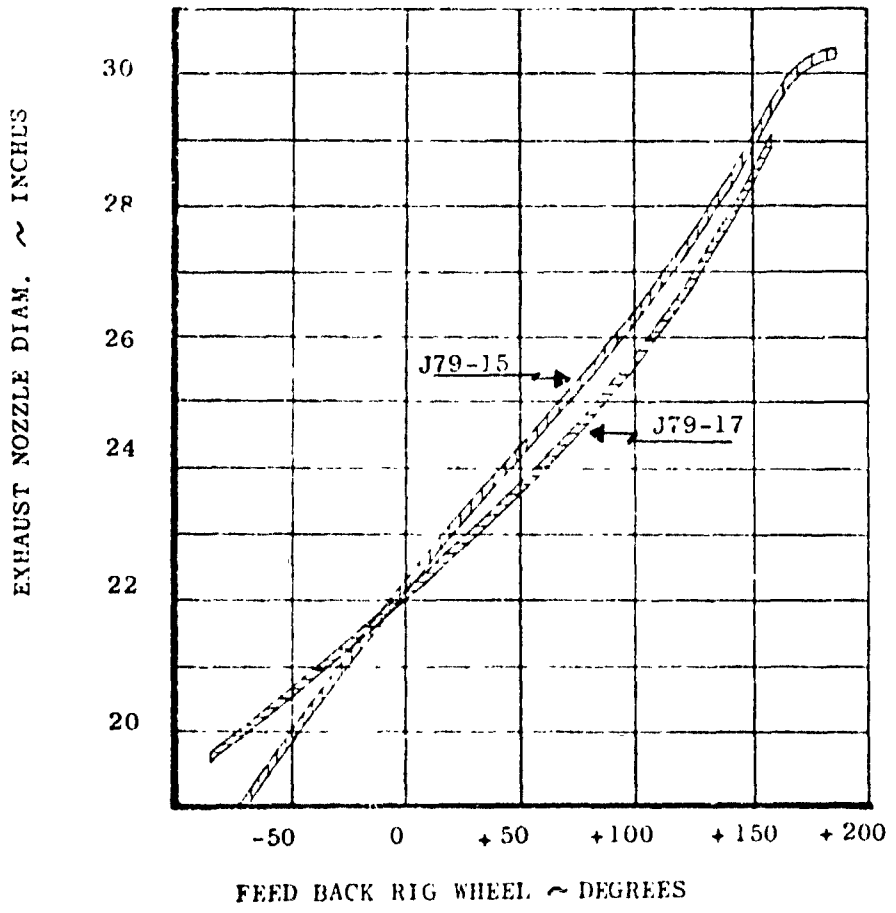


Figure 27 - J79-15/-17 Exhaust Nozzle - Nozzle Area Control Feed Back Position vs Nozzle Diameter

SECTION VIII
AFTERBURNER OPERATION

To determine whether the afterburner was in operation at the time of impact, the normal variables associated with the basic engine operation cannot be used. The RPM and afterburner nozzle area may reach their maximums at less than military operation. Therefore, these parameters could be the same at less than military operation as they are during afterburner operation. The fuel flow meter cannot be used to determine whether the engine was operating in afterburner because it records only the fuel passing through the main fuel control. For these reasons other parameters must be analyzed to determine whether the afterburner was in operation at time of impact.

It is imperative that an accident investigator understand the operation of the afterburner in order to determine whether it was in operation at the time of impact.

Afterburner operation is initiated and terminated by an on-off signal from the main fuel control. When the main fuel control throttle is moved into the afterburner range (76.5 ± 1.5 degrees throttle range), a lever on the power shaft moves the afterburner pilot valve downward. Then, if engine RPM exceeds 90.3 percent, the on-off signal is ported to the on-off speed adjustment piston and to the inlet valve of the afterburner pump.

Inlet fuel pressure, acting on the on-off speed adjustment piston, forces the on-off speed pilot valve to a position which opens the port to the afterburner pilot valve further. This permit engine RPM to decrease to 83.2 percent, once the signal exists, before afterburning is terminated.

The on-off signal from the MFC enters the A/B pump at the on-off signal port, see Figure 28. This signal pressure is ported to in the inducer side of the piston. The piston moves the valve diaphragm from its seat allowing fuel to enter the inducer of the pump, see Figure 29.

The method for determining the position of the main fuel control throttle was described in Section V. The nominal afterburner range for the throttle is 80% for minimum A/B to 113° for maximum A/B. The various ways of determining RPM was described in Section V. Through these methods, it can be determined if the two conditions, throttle angle and RPM, have been satisfied for A/B operation.

Examination of the impact marks of the piston within the cylinder or the position of the valve diaphragm with respect to its seat, see Figure 29, should enable the investigator to determine whether the valve was open for A/B operation at time of impact.

After it has been determined that the afterburner was in operation, the next question to be resolved is the extent of modulation. The afterburner fuel pressurizing valve can be used for this determination, see Figure 30. The valves are opened by fuel pressure from the afterburner fuel control. Sequence of operation opens the primary core, primary annulus, secondary annulus and secondary core in that order. They may jam in the position they were in at time of impact. Air pressure or fluid flow introduced at the core and annulus inlets will indi-

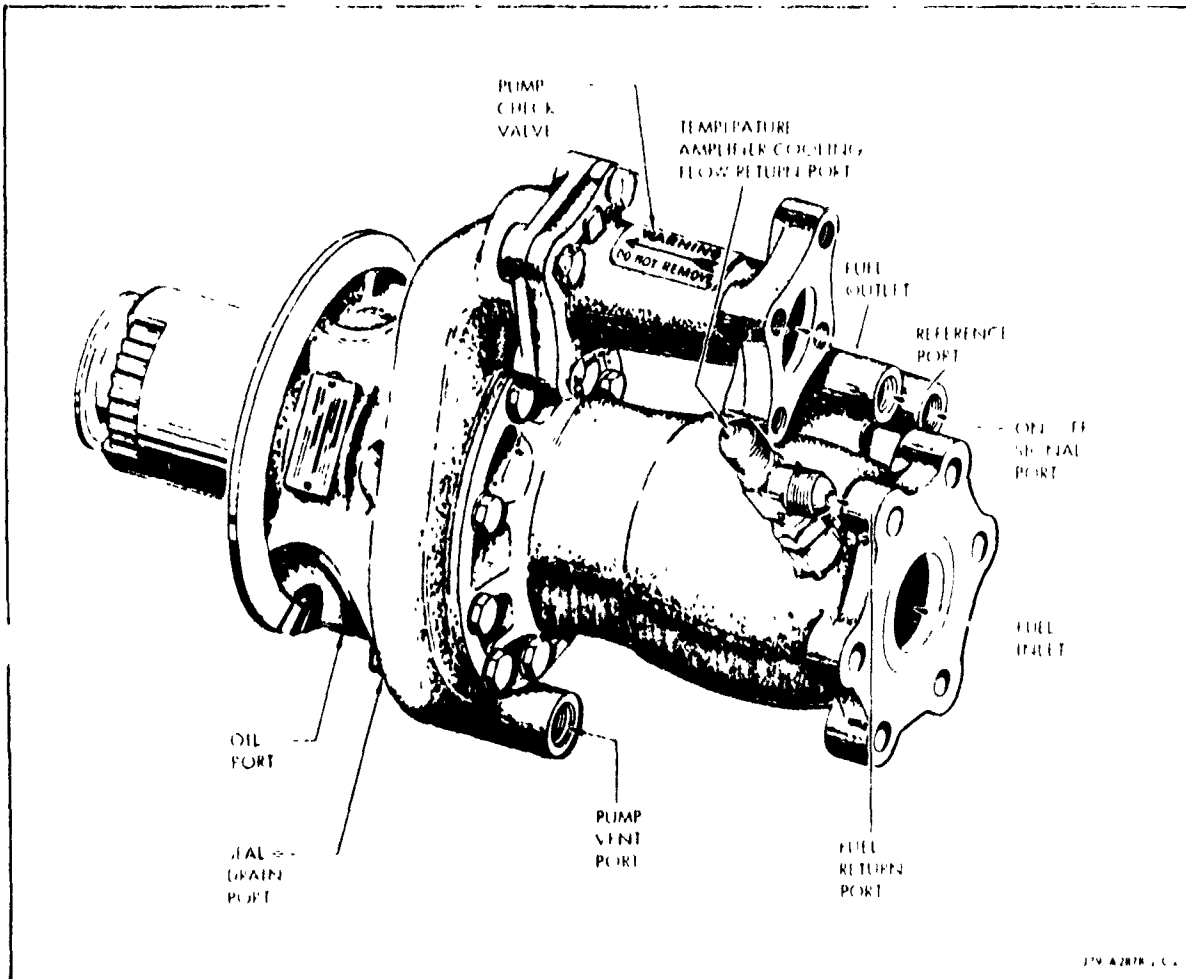
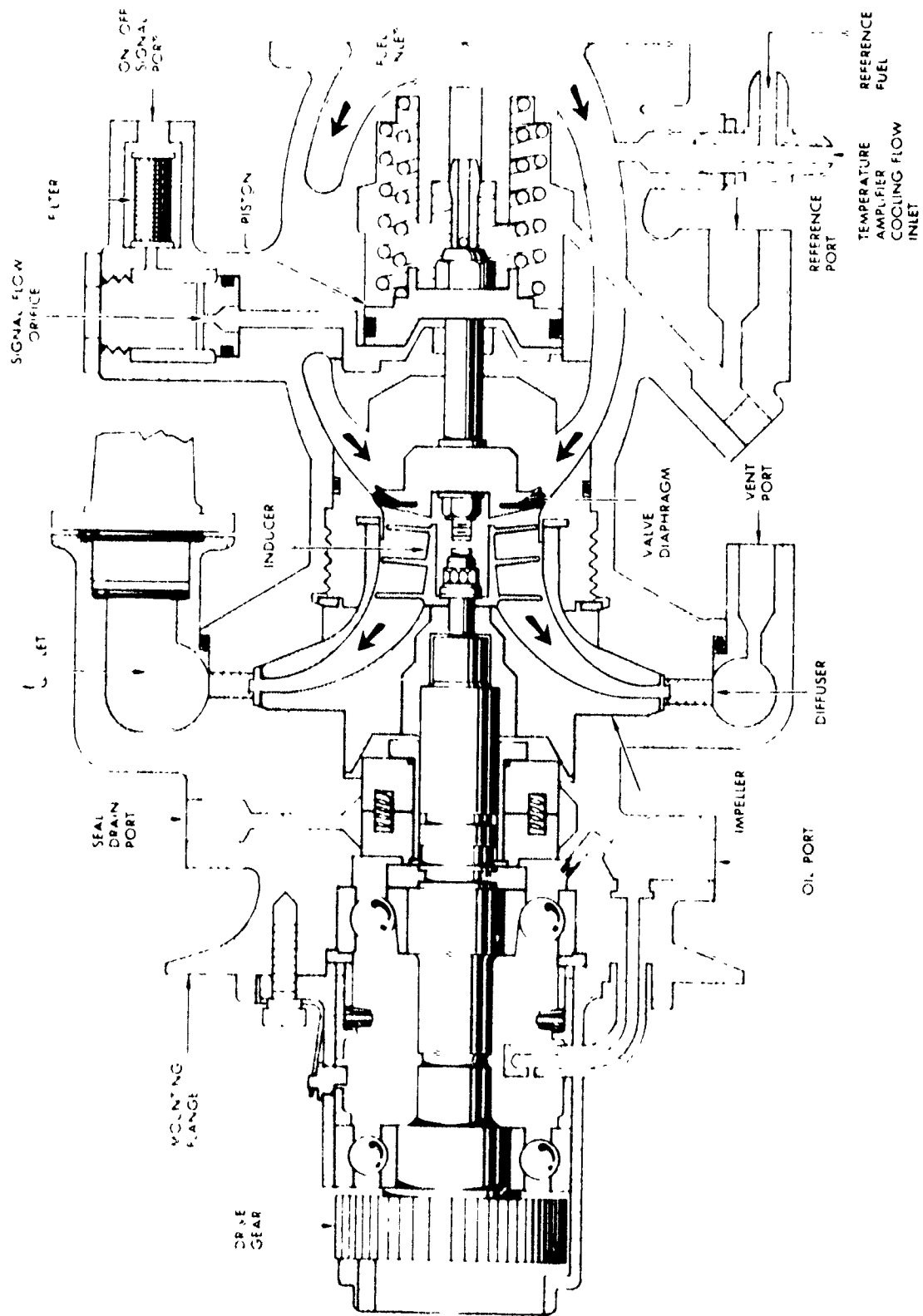


FIGURE 28 - EXTERNAL VIEW OF THE AFTERBURNER FUEL PUMP



379 A 67 4 4 2

FIGURE 29 - CUTAWAY VIEW OF THE AFTERBURNER FUEL PUMP

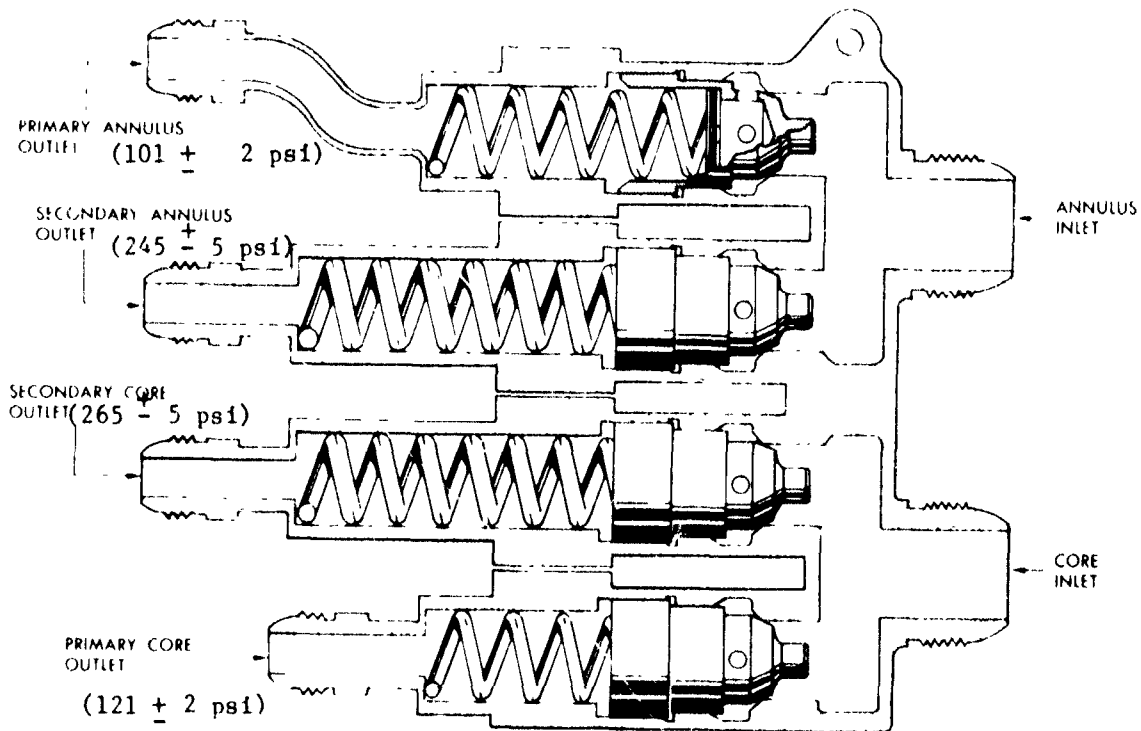
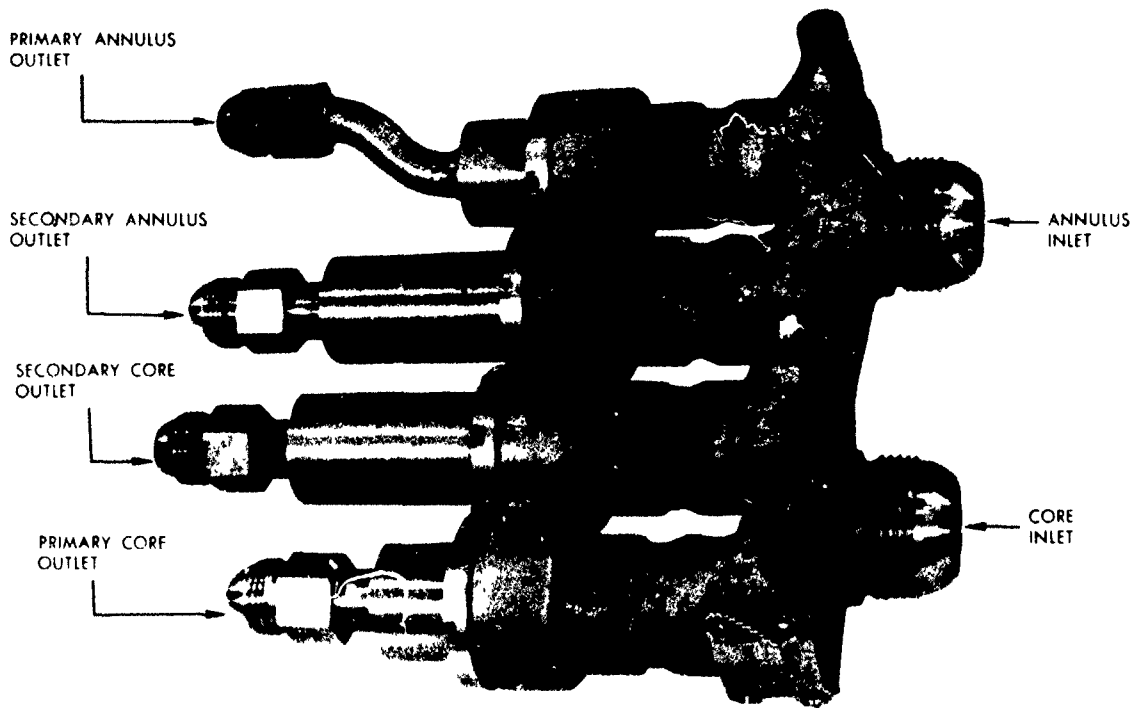


Figure 30 - Afterburner Fuel Pressurizing Valve Schematic

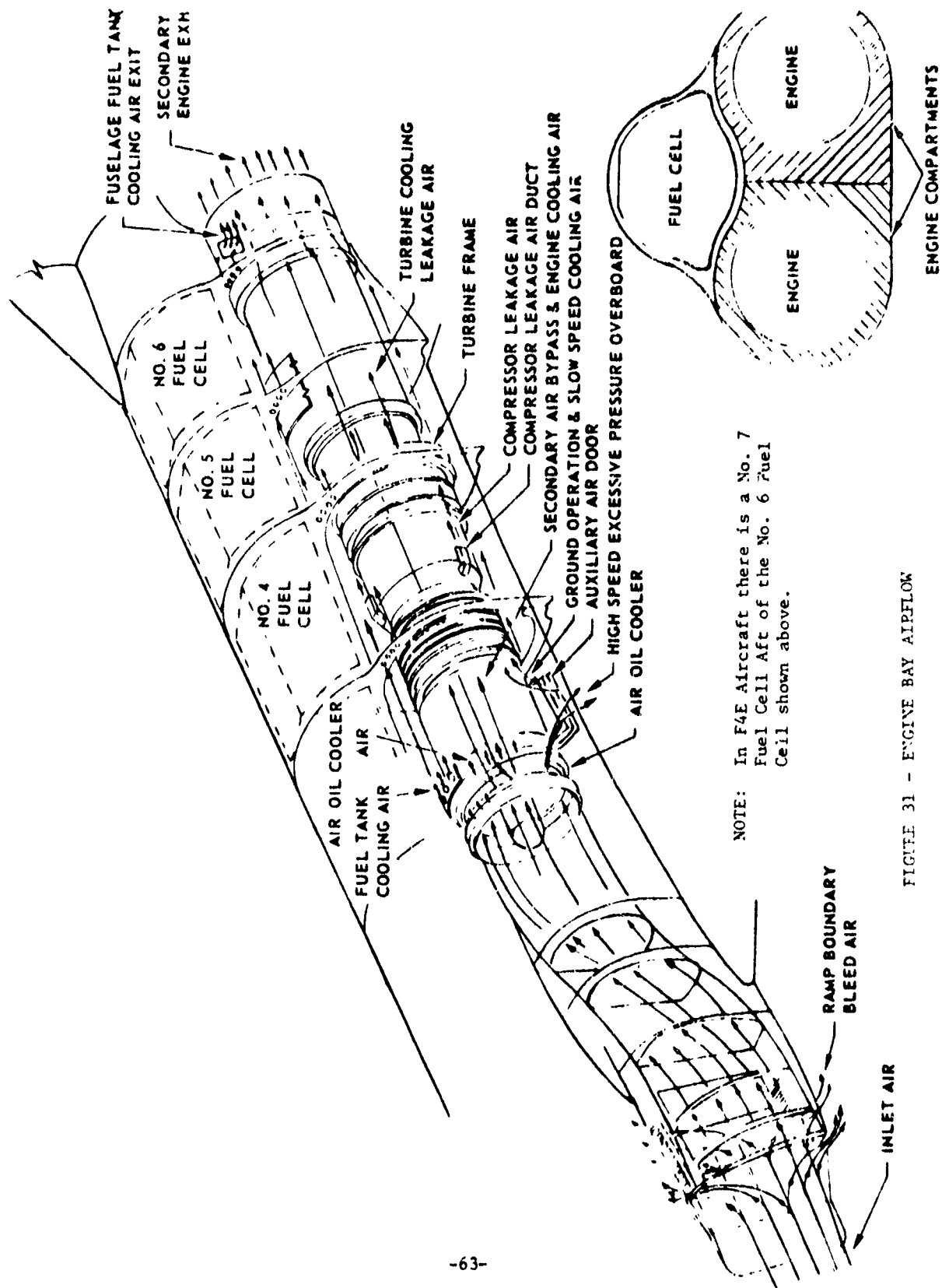
cate if the valves are jammed. It may be necessary to inspect the cylinders of the valves for impact marks to determine their position. Minimum A/B operation would be the primary core valve only being opened and maximum A/B operation would be indicated by all four valves opened.

SECTION IX

INTERIOR AND EXTERIOR AIR FLOW

In order for an accident investigator to understand and properly analyze the events preceding and at the time of ignition of combustibles it is imperative that he has knowledge of the air flow both exterior to the aircraft and within the engine bay. See Figures 31, 32 and 33. He must also understand the normal functions of the auxiliary air doors and pressurization of the 600 gallon centerline fuel tank.

Except at high Mach conditions, where the pressure within the engine compartment must be relieved, the auxiliary air doors open when the landing gear is down and conversely are closed when the landing gear is retracted. With respect to inflight fires, with the aircraft operating at the high Mach numbers that would necessitate relieving the pressure within the engine bay by opening the auxiliary air doors, the air flow velocity through the engine bay is too high for a combustible fluids to ignite. However, a combustible fluid might be ignited by the afterburner in the vicinity of the afterburner secondary flaps at relatively high Mach numbers. These flaps can serve as flameholders. The flame will not flash forward until the air flow velocity in the engine bay decreases. Whenever the engine's operating mode is changed from afterburner to non-afterburner operation, the efficiency of the ejector formed by the secondary flaps of the A/B nozzle is decreased during the transition period. This in turn will result in a decrease in velocity through the engine bay and a flash forward can take place.



NOTE: In F4E Aircraft there is a No. 7 Fuel Cell Aft of the No. 6 Fuel Cell shown above.

FIGURE 31 - ENGINE BAY AIRFLOW

F-4 ENGINE COOLING AIR FLOW GROUND AND LOW SPEED OPERATION - GEAR DOWN

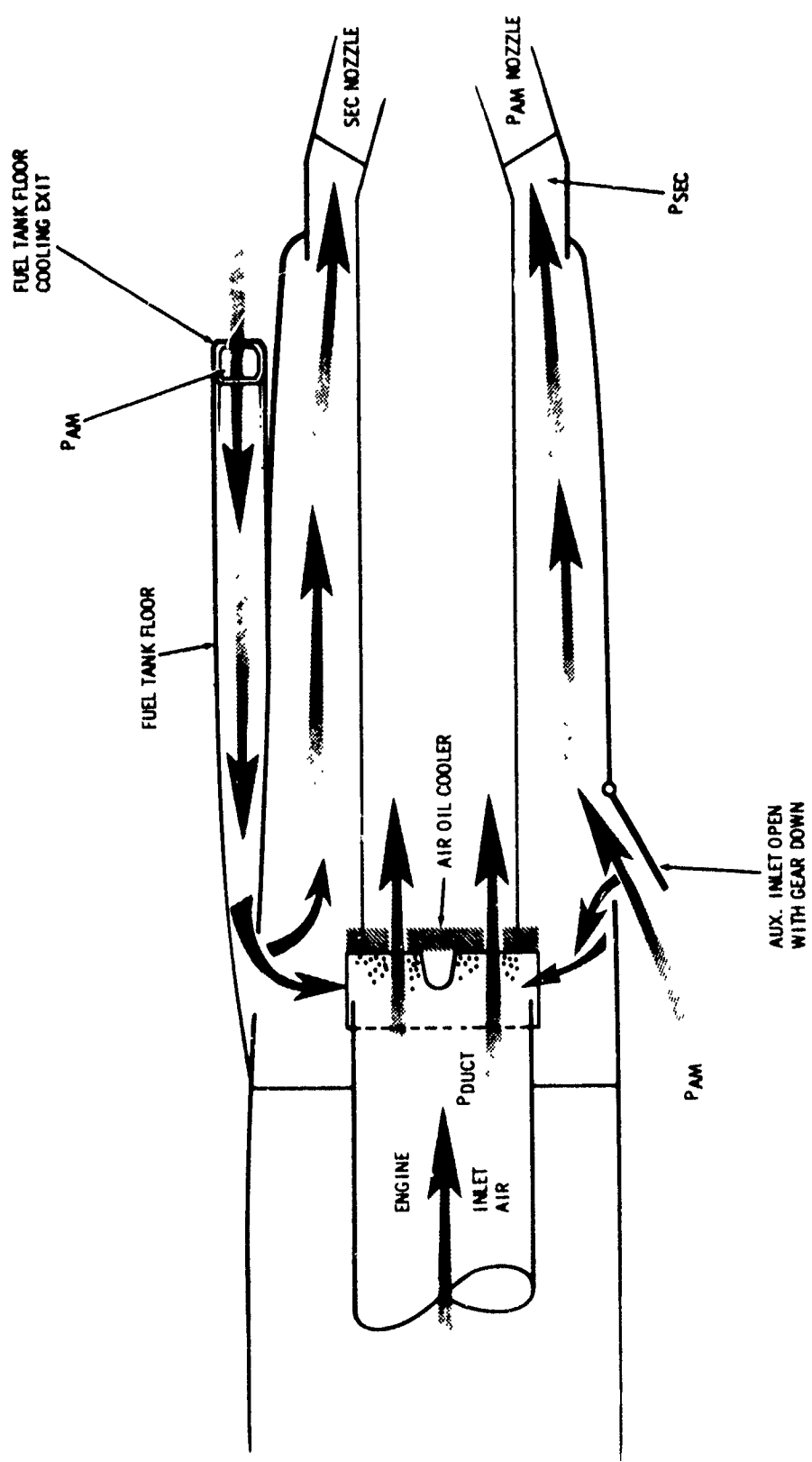


FIGURE 32

F-4 ENGINE COOLING AIR FLOW

IN-FLIGHT OPERATION - GEAR UP

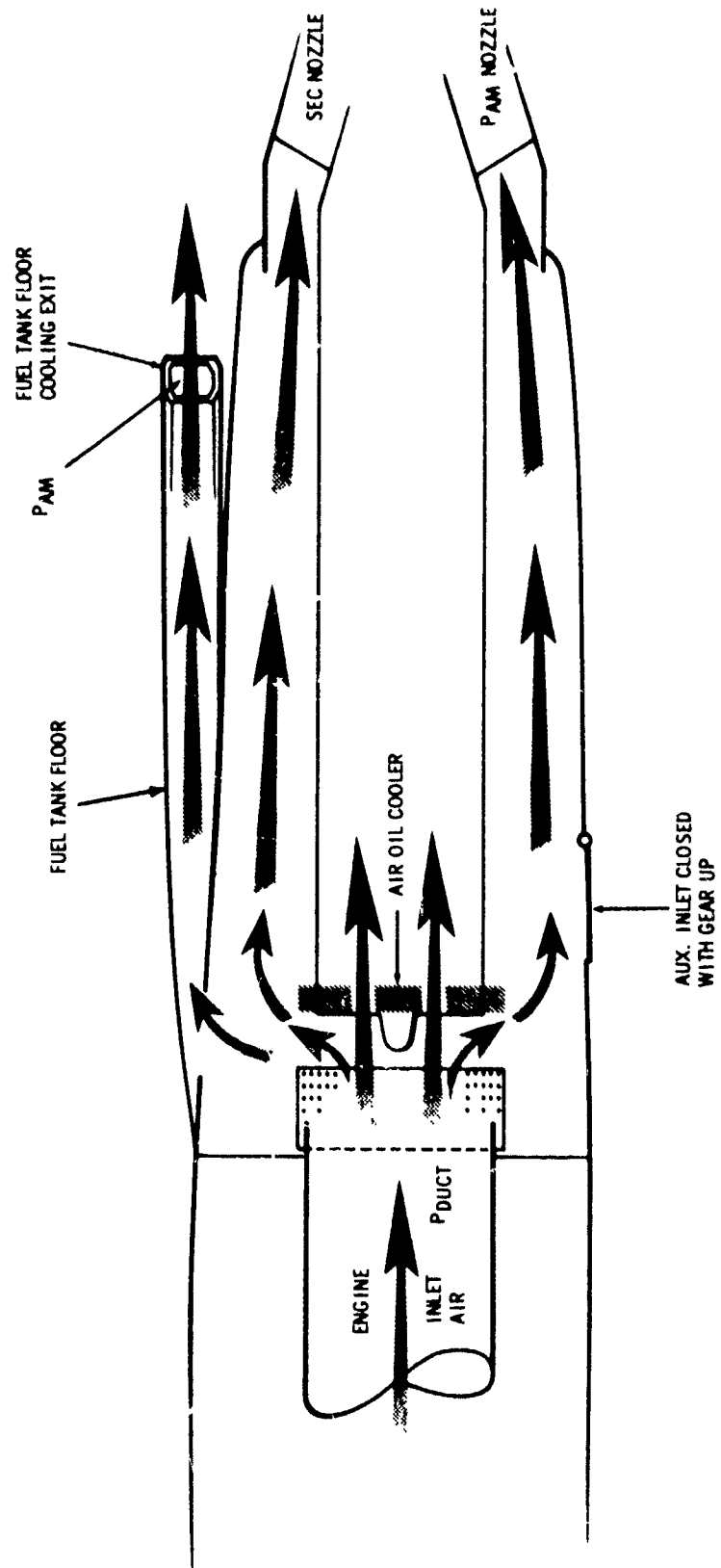


FIGURE 33

During ground operations and takeoff the auxiliary air doors are open. The normal air flow path is through the auxiliary air doors, into the engine compartment, forward into the engine inlet or aft out through the A/B nozzle. The maximum landing gear down speed is 250 knots (approximately M.38). As may be seen from Figure 34, the engine compartment is operating at pressures less than atmospheric, (The air flow is through the auxiliary air doors into the engine bay) during the climb in military or Max A/B with the landing gear down and auxiliary air doors open. For this reason any combustible fluid that contacts the aircraft skin or flows around the various doors during this phase of flight will have a tendency to be sucked into the engine bay.

As may be seen from Figure 35, retracting the landing gear which in turn closes the auxiliary air doors (even at minimum climb speed approximately M.25), the airflow at the front of the engine compartment changes from airflow from the engine compartment into the inlet to airflow from the inlet into the engine compartment. Therefore, there is a short period of time during the retraction of the landing gear that the airflow is not flowing in either direction. It is at this time that a combustible coming into contact with the hot parts of the engine would be most susceptible to igniting. With the auxiliary doors closed, the engine bay operates with positive pressure above approximately M.22 under all operational modes, see Figure 35. For air flow rates through the engine compartment, see Figures 36 and 37.

There is a squat switch in the landing gear mechanism that

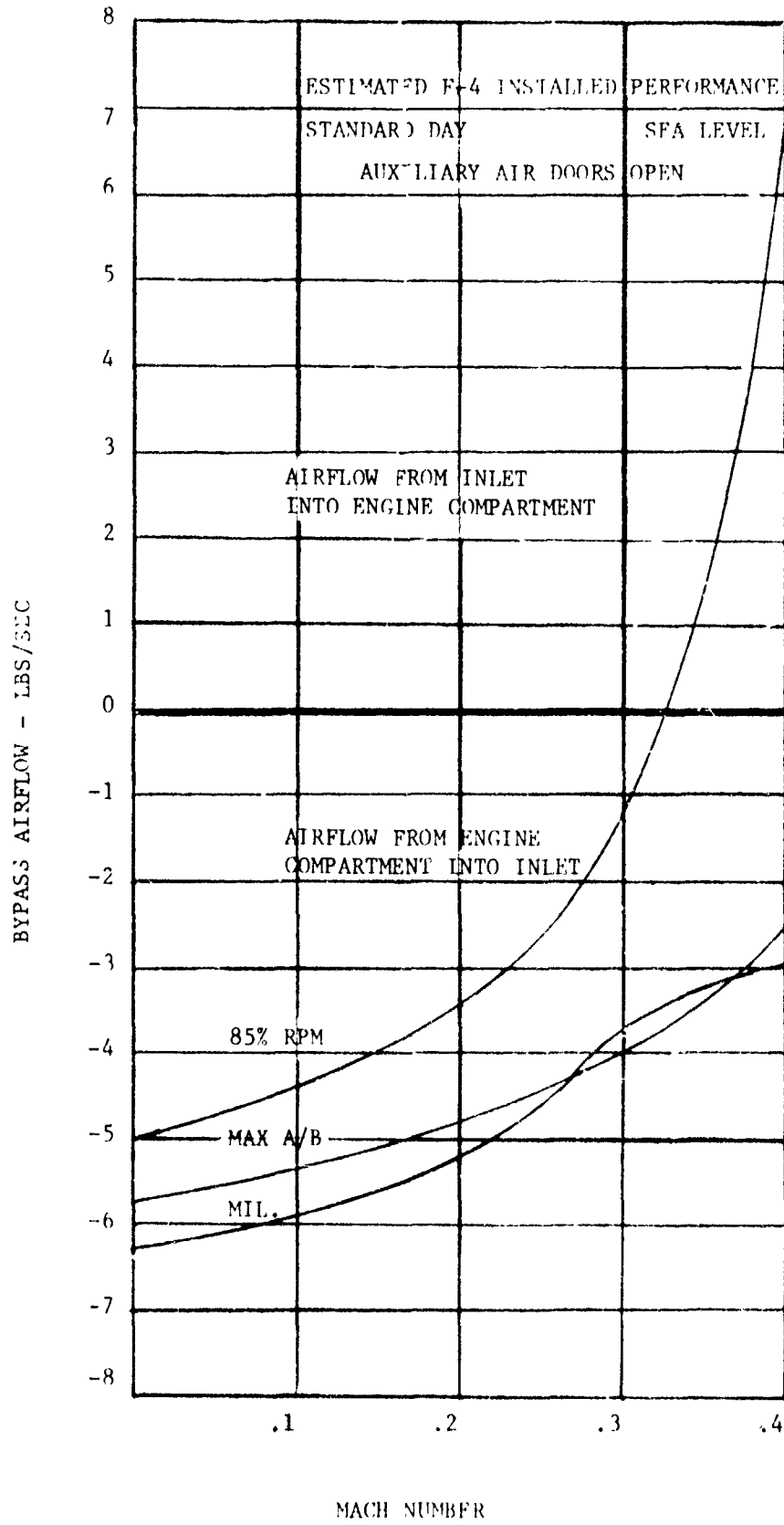


Figure 34 - Bypass Airflow vs Mach Number

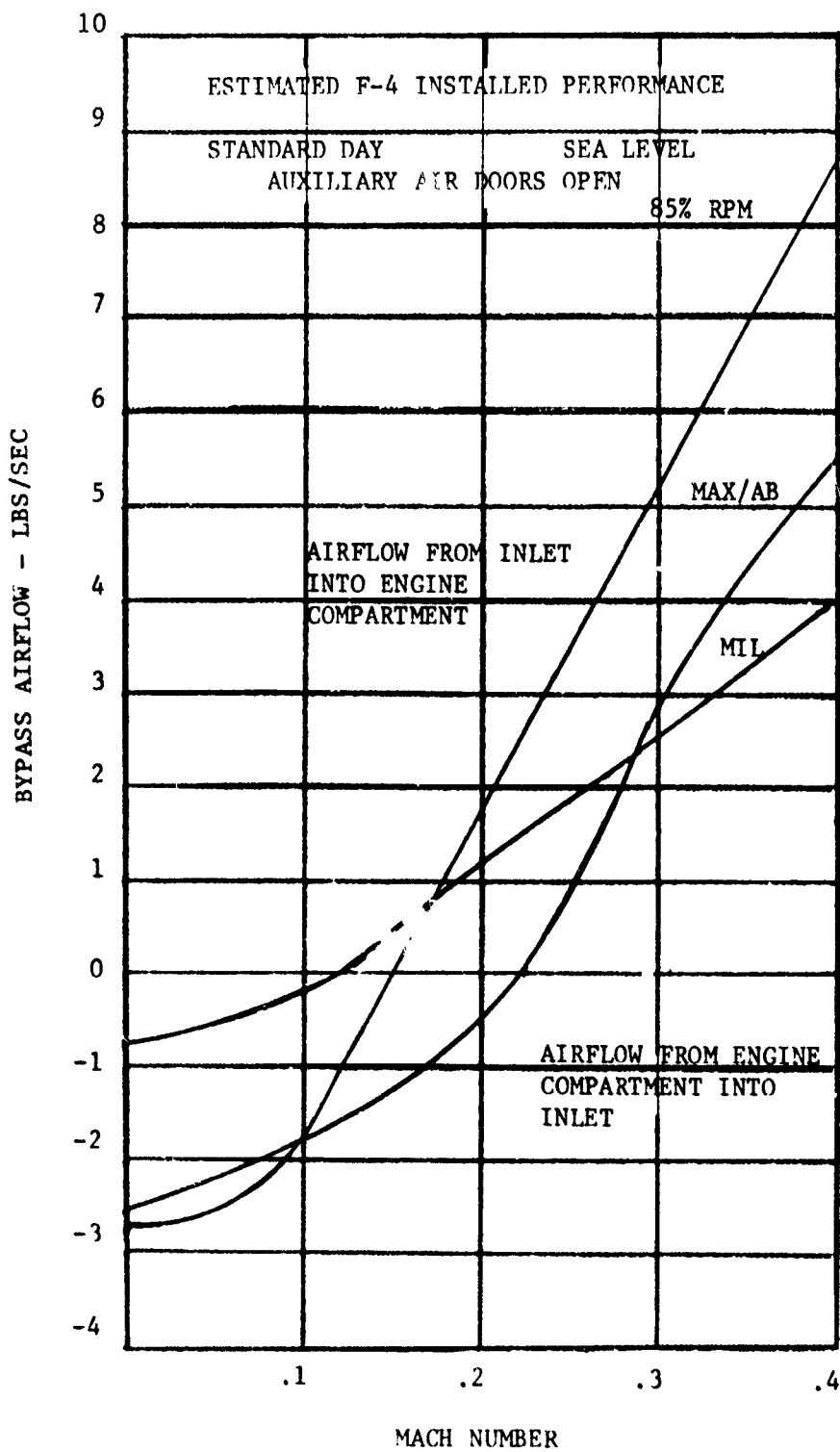


Figure 35 - Bypass Airflow vs Mach Number

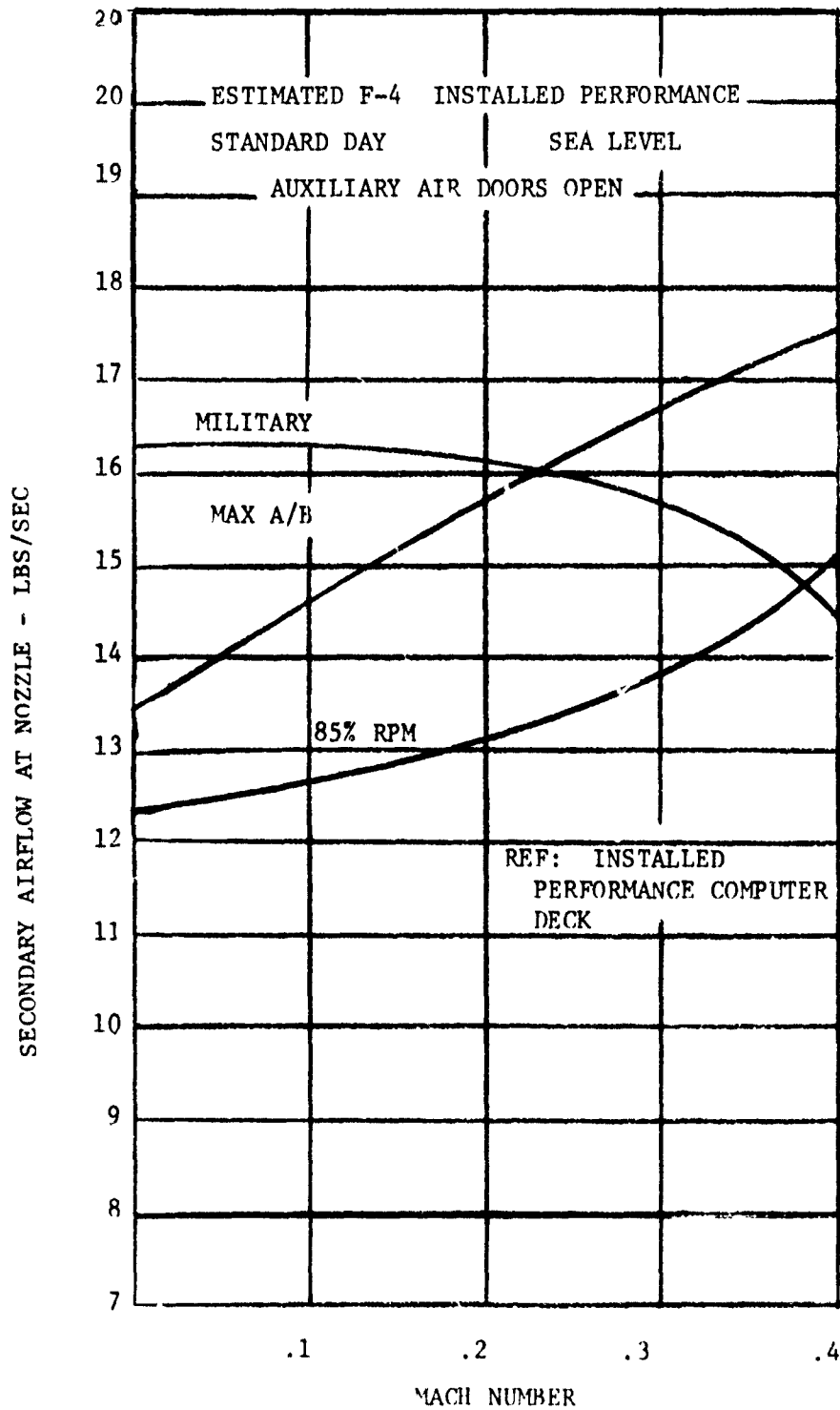


Figure 36 - Secondary Airflow at Nozzle vs Mach Number

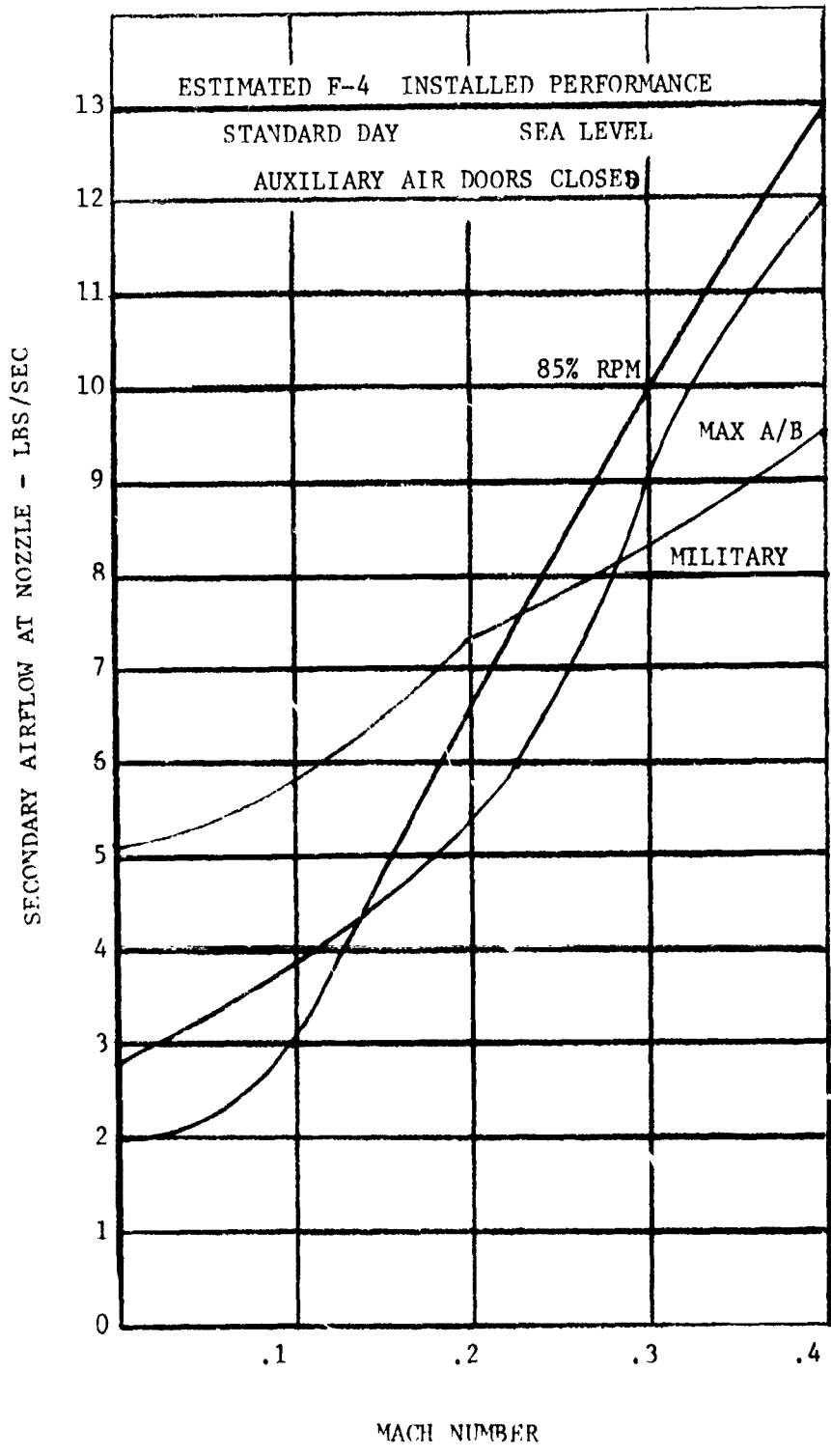


Figure 37 - Secondary Airflow at Nozzle vs Mach Number

allows air pressure to pressurize the 600 gallon centerline fuel tank after the aircraft weight is no longer on the aircraft's landing gear. If there is a leak in the attach tube of the 600 gallon tank fuel, it would begin leaking when the tank becomes pressurized.

SECTION X

FIRE

1. GENERAL

The area with the greatest fire damage should be examined carefully because this area is frequently the source of the fuel or oxidizer. Since the fire may originate in flight, it is important to know that the fire intensity will be more severe in the areas exposed to an air stream; also the fire pattern will tend to follow the slip stream. The damage from in flight fires, as well as those involving the rupture of a high pressure hydraulic fluid line or an oxygen line, will tend to be similar to that produced by a torch. To assist the investigator in evaluating the fire damage, the properties of aircraft materials is included in Table 1. Figure 38 illustrates aluminum splatter resulting from an in flight fire.

In the fire or explosion analysis, it is necessary to account for the source of the combustible, the probable source of ignition, the history of the fire, and the observed fire damage. Any assumptions that are made must be reasonably consistent with the evidence on system malfunctions, material failures, and the sequence of events.

2. SOURCE OF COMBUSTIBLES.

Aircraft jet fuels frequently account for the major amount of fire damage in an aircraft fire but they may not necessarily be involved in the initiation stage. In a ground or in flight fire, the leakage of jet fuels as well as other volatile flammable fluids should be suspected as a combustible source, depending

TABLE I - PROPERTIES OF AIRCRAFT MATERIALS

Metal	At. Wt.	Specific Gravity (Water=1)	Melting Point °F	Heat of Combustion Btu/lb	Ignition Temperature, °F In O ₂ Slab or Sheet	In Air Dust Cloud	Adiabatic Flame Temperature in Oxygen at Atmospheric Pressure, °F
Aluminum	26.97	2.70	1220	13,400	---	1200	6450
Brass	---	8.4 - 8.8	1600	---	---	---	---
Cadmium	112.41	---	610	1,000	1400	1070	---
Chromium	52.01	6.93	3430	4,750	---	1080	5480
Copper	63.57	8.92	1980	1,070	---	1290	---
Iron	55.85	7.86	2795	2,000	---	1620	---
Lead	207.21	11.34	620	460	1710	1310	2780
Magnesium	24.32	1.74	1200	11,600	1600	1150	5570
Molybdenum	95.95	10.20	4750	3,430	1160	1330	---
Nickel	58.69	8.90	2810	1,780	1380	NI	---
Silver	107.88	10.50	1760	65	---	---	---
Stainless Steel	---	7.6 - 7.9	2600	2,000	1710	NI	---
Tin	118.70	7.31	450	2,160	1590	1170	4400
Titanium	47.90	4.50	3100	6,820	1600	630	5480
Tungsten	183.92	19.30	6100	1,970	---	1350	---
Zinc	65.38	7.14	785	2,300	1650	1260	3500



FIGURE 38 - Circled is a Splatter of Aluminum on a Part
of Aircraft Structure

upon the evidence on ruptured lines, loose fittings, etc. Aircraft jet fuels, particularly JP-4 is usually a prime suspect in an explosion or sudden widespread fire because of its high volatility and great ease of forming flammable vapor-air mixtures; the lower volatility fuels, such as JP-5, can be equally hazardous at slightly elevated temperatures or reduced pressures, or when atomized to form flammable mists. The hydraulic fluids and lubricating oils or greases have rather low volatilities but can be the prime suspects in the case of an engine compartment fire, particularly if no jet fuel leaks are detected. The pattern of soot formation may be useful in determining the combustible source; however, chemical analyses of the soot or other deposits are normally required when the evidence is inconclusive. In assessing the possible role of any combustible fluid, the following items should be considered:

- (1) Quantity and distribution of combustible.
- (2) Vapor pressure and flash point of combustible.
- (3) Concentration limits of flammability.
- (4) Temperature requirements for ignition.
- (5) Flame temperatures and propagation rates.
- (6) Effects of ambient temperature, pressure, ventilation rate, and other flight environmental variables.

3. SOURCES OF IGNITION.

As is known from experience, the chance of ignition after a fuel leak occurs in aircraft engine compartments or adjacent areas is relatively great. The possible sources of ignition in these areas include the combustion chamber surfaces, overheated

engine accessories, and sparks or arcs from electrical circuits and equipment; other sources are electrostatic sparks, flames, hot gases, lightning, aerodynamic heating, and frictional heat or sparks. The entrainment of afterburner gases into a fuel tank vent is an example of the hot gas ignition hazard. Generally, most ignitions are caused by hot surfaces or electrical energy sources, although in a crash situation multiple sources can be encountered. For sustained ignition to occur, flammable vapor-air mixtures must be present or the combustible liquid or solid must be heated to produce at least a flammable layer of gas at the surface. Thus, the physical state of the combustible is important in determining whether a particular heat or energy source could produce ignition. Furthermore, the investigator must be aware of the fact that ignition temperature requirements can be much higher in a flowing system than in a static system.

For this reason a fuel leak could be present in the engine bay but ignition does not occur until the air flow through the engine bay is disturbed such as initiation or termination of afterburner operation or opening or closing of the auxiliary air doors.

4. DEVELOPMENT OF FIRE.

The origin of the fire is deduced from a combination of the evidence developed in determining the sources of ignition and combustible and the material failures or system malfunctions. The spread of the fire is determined from a study of the distribution of combustibles, the flammability properties of the combustibles, the intensity and distribution of the fire damage, and the known airflow or ventilating conditions throughout the

aircraft. Witness accounts should be used to help corroborate the fire sequence indicated by the physical evidence.

The presence of soot can be used to indicate whether a jet fuel or organic combustible fire occurred in a given location. Extensive deposits of soot or char usually indicate that the combustion occurred under non-optimum conditions, e.g. insufficient air, and that the average fire temperature was probably of the order of only 1000°F, generally referred to as low magnitude fires. However, some soot deposits are expected in the aft compartments of the aircraft fuselage from normal operations, such as in taxiing, when exhaust gases are entrained by intake of coolant air. Evidence of soot formation and heat damage to aircraft structural materials should be used in determining the progress of the fire.

Generally, the fire spread rate will be greatest where flammable vapor-air moistures can form readily and where the fire is formed by wind or flowing air, as in the engine bay of an aircraft. Thus, in flight fires exposed to an airstream will spread rapidly from the point of origin to the aft part of the aircraft, depending upon the available quantity of combustible, generally referred to as high magnitude fires. In comparison, the pattern of a ground fire will be more irregular, with more vertical and lateral flame spread. The amount of fuel leakage or spillage will greatly determine how widespread the fire will be. Other factors to be considered are combustible volatility, amount of atomization, mass burning rates, and the flame speeds of fuel vapor-air mixtures.

5. DAMAGE PATTERNS.

The intensity of an aircraft fire can be determined by comparing

the temperature limitations of the aircraft materials that were consumed and those that were highly resistant to heat. Most aircraft materials, including metals and fire resistant materials, cannot withstand the temperatures reached in a hydrocarbon fuel type fire; titanium and stainless steels are among the exceptions and tend to show damage only in inflight fires or in torchlike fires. The fire temperature, exposure time, and the airflow or available oxygen must be considered in evaluating any fire damage. Where a massive fuel spillage has occurred in a fire, the entire aircraft and part of the adjacent surroundings will show widespread fire damage. In the event that incendiaries, magnesium, or other high energy combustibles are involved, the fire damage will be severe and localized in the areas where these materials are present. Damage from electrical arcing is even more localized and can be identified from the erosion, splatter, and fusion of metals that is characteristic of arc welding. In the absence of current, e.g. after a crash fire, fire damage to a wire bundle will normally not show bead-like fusions (See Figure 39) or other such intense heating unless strands of fine wire or unusually high fire temperatures are present. This should not be confused with damage from chafing as illustrated in Figure 40.

Damage from explosions is usually indicated by the rupture of an aircraft compartment and the dispersal of fragments. Fuel tanks and other aircraft compartments cannot sustain most combustible vapour-air explosions, although they may sustain partial propagations under certain conditions. Explosion pressures can be



FIGURE 39 - Wire Bundle Showing Bead Like Fusion Associated
With Electrical Arcing



FIGURE 40 - Braided Hose That Was Worn Through by Chafing.

Note the Braided Strands are Worn Sharp on the
Ends.

more severe in an "empty" fuel tank than in a filled one, depending upon the fuel volatility and the flight altitude pressure. In assessing the explosion damage, it is necessary to consider both the structural limitations of the confinement material and the degree of venting that existed; e.g., an explosion not sustained in a fuel tank can conceivably be sustained in a vented compartment, such as the engine bay. Also, the investigator must consider the damage contribution from any physical explosions or implosions, e.g., sudden decompression of compartments or rupture of fuel tanks from overpressurization, and from the detonation of ordnance items or high energy fuels that may be aboard.

An analysis of the fragmentation and air blast effects can provide an estimate of the chemical or pressure energy of the explosion; crater evidence can also be useful in this connection. Examination of metal fractures will indicate whether tension, compression, or torsion failures occurred, providing the fractured part has not been greatly damaged by fire, metallographic analyses are usually required to determine the exact nature of any fatigue failures. Combustible vapor-air explosions (deflagrations) will be evidenced by tension failures, whereas gas detonations will ordinarily produce less stretching and cleaner breaks since a detonation wave propagates faster than the material can react or stretch to its stress limit. A map or diagram showing the size and spatial distribution of fragments should be prepared to facilitate the correlation of damage with the potential explosion energy.

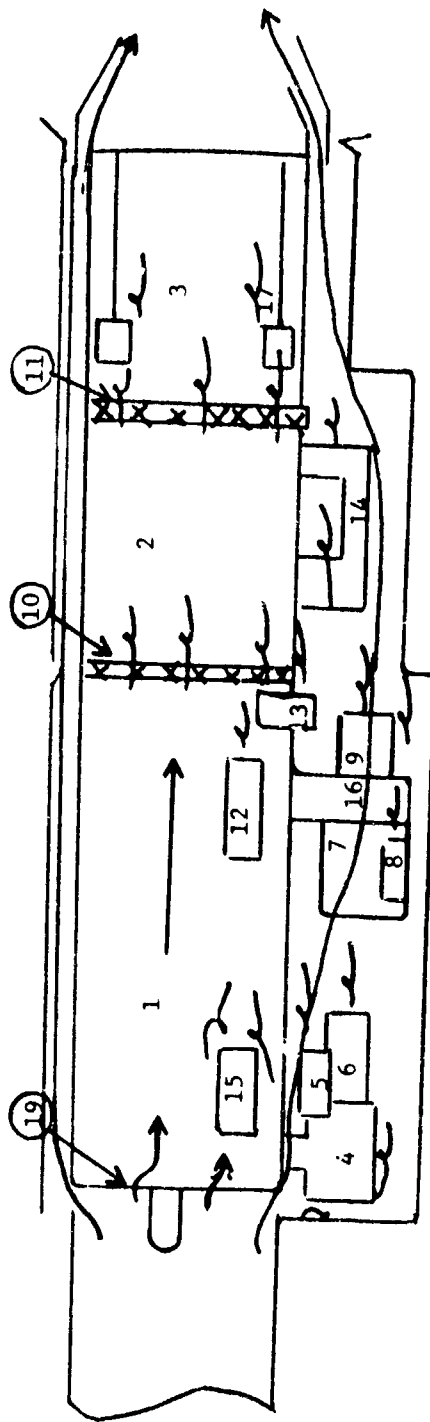
Finally, an analysis must be made of the accident casualties and the fire fighting and rescue operations. Again, a diagram is recommended in order to relate the fire development with the position of each occupant. The immediate effects of fire or explosion on the flight crew and passengers can be deduced from the analysis of the fire development and any record of radio transmissions with the crew; subsequent effects can be determined from medical records and a consideration of toxicity limits, asphyxiation limits, and physiological thresholds of heat and dynamic pressures. An evaluation of the fire fighting phase is important in developing the fire evidence as well as in determining the adequacy of the extinguishing agents and procedures for fighting aircraft fires.

6. ENGINE BAY AREAS MOST SUSCEPTIBLE TO FIRES.

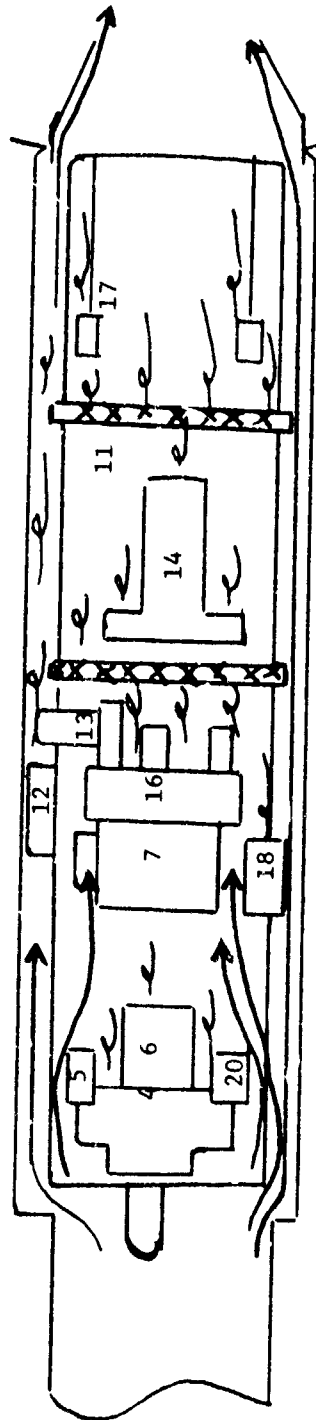
Engine mounted hardware forms an obstruction to engine bay secondary airflow and as such create disturbances to the flow pattern. Eddy's resulting from this flow disturbance are ideal flame holders and will retain and propagate flame in these quiet areas. Experience has further established definite flow tunnels and quiescent areas to secondary air flow. It is in these areas that evidence would be visible should any engine fire exist.

The tunnels are to the left and right of the gear boxes and mounted accessories. The major quiescent areas are just aft of the gear boxes and at the A/B fuel spray-bar and manifold assembly. See Figure 41.

Considering secondary air flow, fuel leakage could be present



ENGINE LEFT VIEW



ENGINE BOTTOM VIEW

- | | | | |
|------------------------|------------------------|------------------------------|-----------------------------------|
| 1. Compressor | 6. Starter | 11. A/B Fuel Manifold | 16. Rear Gearbox |
| 2. Combustor & Turbine | 7. Main Fuel Control | 12. A/B Control & Oil Cooler | 17. Nozzle Actuators |
| 3. Afterburner | 8. Throttle Boost | 13. Nozzle Area Control | 18. Ignition Boxes and Flow Meter |
| 4. Transfer Gearbox | 9. Accessories | 14. Bleed Air Manifold | 19. Anti-Icing |
| 5. A/B Pump | 10. Main Fuel Manifold | 15. Temp Amplifier | 20. Main Fuel Pump |

Figure 41. Air Flow and Eddy Currents Within The Engine Bay

and ignite along the engine bay skin. Since the source would be away from the engine and separated from the engine by secondary air flow, the fire could continue to burn without affecting engine hardware. Such a fire however, would have to be of low magnitude. A fire of this type may not cause the fire warning light to glow to full intensity and may be extinguished on A/B termination.

If it can be determined that the flame was in the area of the tail hook hinge fairing and along the exterior surface of the engine exhaust nozzle secondary flaps, it can be established that the observed fire was in the boundary layer away from the flap surface itself. It would further establish that fuel was fed from some forward exterior source and the flow pattern and eddy's created by the fuselage aft edge and fairing assembly, served as flame holders.

If a flame is observed 360 degrees around the engine exhaust nozzles, like a "mini A/B flame", it would suggest that fuel was being ingested into the exhaust stream from the top and bottom of both engines. A failure above the fuel cells probably would result in a massive fuel loss before the fuel could ever reach the engine exhaust nozzle area. To have minimal amount of fuel loss and still sustain a fire in both engines, the failure would have to be below any one of the fuel cells between number 3 and number 6. Underneath the fuel cell compartments there are fuselage fuel tank cooling cavity which draws cooling air from the forward section of the engine compartment and exhausts the air overboard to the free air stream through the louvers near the engine exhaust nozzle. The construction of the cooling cavity is such that below the number 6 fuel cell, the two cooling cavities are manifolded together. At the bottom of number

7 fuel cell (FS 493) there is a 1 1/2 inch diameter hole exists which will vent cooling air into the tail (aft of FS 515) of the aircraft.

7. FUEL LEAKAGE OUTSIDE OF AIRCRAFT.

The 600 gallon center line fuel tank is the only source of fuel external to the aircraft that can cause an inflight fire. There are two possible ways that the centerline tanks can cause fuel to enter the engine bay and subsequently an inflight fire. One is a leak around the tank attach tube and the other the tanks fill port.

Whether the problem was caused by leakage at the attach tube or the filler port both would result in ingestion of fuel into the engine at the lower portion of the compressor and towards the aircraft keel. Symptoms of fuel ingestion into the engines and method of determining entry location are covered in paragraph 8 next page. Generally both engines will have essentially the same degree of internal burning of hot section parts. Burning of hot section parts can take place within a matter of seconds.

Leakage at the attach tube differs somewhat from leakage at the filler port. In the case of an improperly installed filler port, the fuel may splash out during the take off roll. The fuel would normally remain within the slipstream and not come into contact with the aircraft skin until the aircraft is rotated. At the time of rotation, the fuel would be carried aft along the lower surface of the aircraft and would be drawn through the seams and openings since the engine bay is operating at a negative pressure with respect to

ambient pressure during this mode of operation. (See Section IX for discussion of engine bay pressures and airflows.) The intensity of fuel leaking from the fill port will increase after the landing gear squat switch is actuated following removal of aircraft weight from the landing gear. This switch allows the tank to be pressurized.

Some of the fuel may adhere to the aircraft skin and be ignited by the afterburner flame. The fire that would result on the exterior skin of the aircraft would spread left and right to doors 83L and 83R (See Appendix E) from the tanks attach point (See Figure 42).

If an engine is shut down after a fire is detected, that engine bay would become positive in pressure with respect to the engine bay with the operating engine. This could cause flame to propagate from the side with the fire through the keel to the other side.

In the case of a leak at the attach tube, leakage would not be expected until the weight of the aircraft is off the landing gear and the squat pressurization switch is activated. The remaining discussion above applies.

8. FUEL INGESTION INTO THE ENGINE.

Fuel ingestion into the engine results in uncontrolled burning of the fuel in the combustion chamber. This results in one or more severely burned combustion cans, combustor casing, transition duct, turbine nozzle and turbine blades. (See Figure 43). This damage generally takes place within a few seconds. The entry location of the fuel into the compressor is often useful to the accident investigator. Fuel going through the compressor will rotate approximately 145° clockwise when viewing the engine from the rear. Therefore,



Figure 42 - F-4 Keel After an Inflight Fire Resulting from a Fuel Leak in the
600 Gallon Center Line Tank Attachment

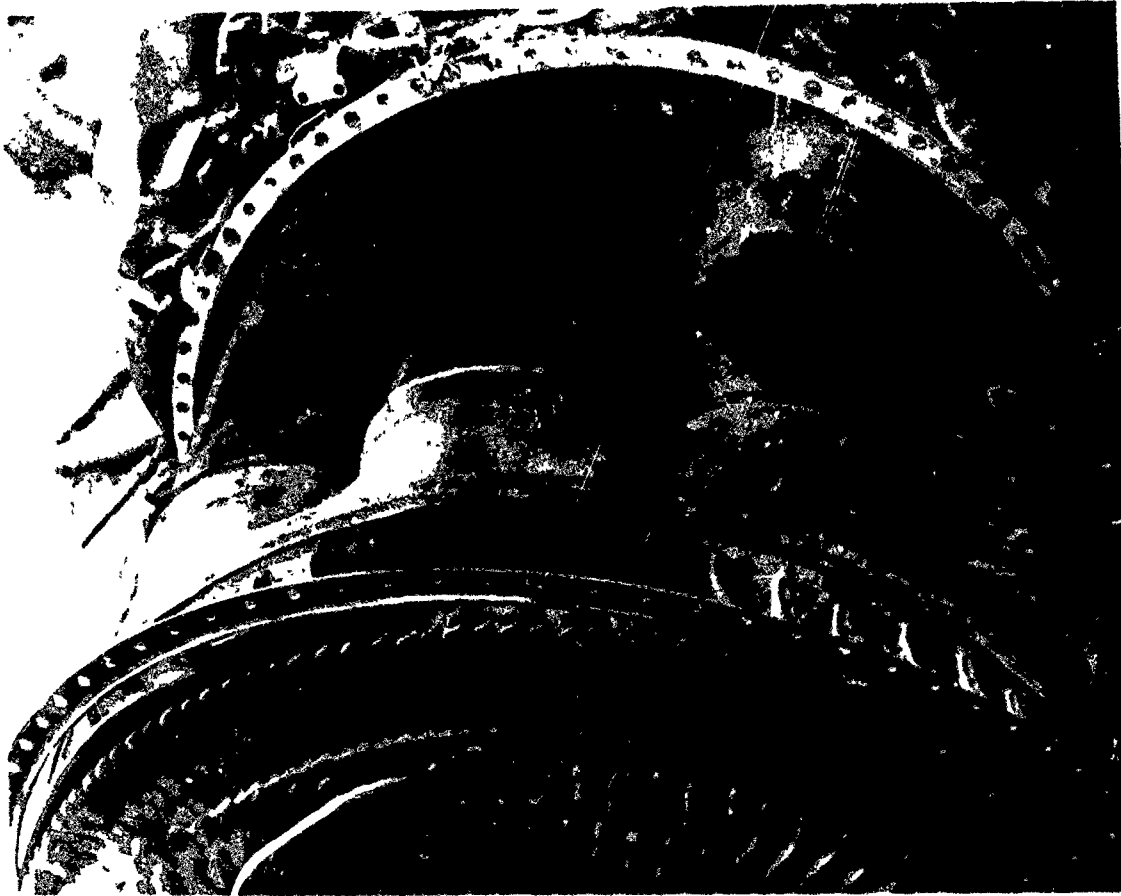


Figure 43 - Combustor Burning From Fuel Ingestion.

Note the one combustor side completely burnt away and the adjacent combustors have less burning. Also notice the turbine nozzle downstream of the most severely burned combustor is missing. Turbine blades have evidence of F.O.D. and over heating.

the compressor entry location is 145° counter clockwise from the center of the burned area as seen aft of the engine looking forward.

9. SOURCES OF FUEL.

Possible sources and paths of fuels which could result in raw fuel injection are:

Rupture of upper fuel cells or transfer lines. Fuel escaping from these components could follow a path forward through the turtle-back cavities, downward around fuel cell no. 1 and existing through openings around door no. 22, See Appendix E.

At this point the fuel can travel aft on exterior skin until it is drawn through skin crevices (e.g. missile launcher attachment points auxiliary air door hinge openings, and ordnance (cartridge access panels) and into the engine bellmouth inlet.

Fuel may leak from the fuel cell floor into the cooling cavity and then forward through the cavity inlet screens into the engine bellmouth. Normal cavity airflow would not allow this to occur; however, turbulent inlet conditions resulting from unusual aircraft flight attitudes could account for such a flow. These unusual flight attitudes might well be encountered during loss of flight control following aircrew ejection. Further, airflow reversal can occur during conditions of low airspeed and high engine speed.

10. FUEL LEAKAGE IN EVENT OF FAILURE IN THE SADDLE BACK SECTION OF THE FUEL CAVITY/CELL.

The aircraft construction is such that the cooling vent chamber between fuel cell cavity floor and the upper engine bay skin (saddle-

back) is common to both the left and right engine bay. Louvers located in the aft fuselage at the approximate twelve o'clock position of each engine exhaust nozzle section vent air from this chamber. The chamber is also vented to the aircraft tail section by a 1.5 inch diameter passage in the louver center section of the rigid former at fuselage station 493. Experience has established that the sealing integrity of the seals of the aircraft and engine bay skin may be reduced through aircraft time and usage.

In the event of a failure in the saddleback section of the fuel cavity cell, leakage would enter directly into the cooling vent chamber. From there it could seep through seams and joints into the engine bay. Normal venting would carry leakage to the tailhook actuator and hinge area and to the exhaust louvers. Leakage into the engine bay, tailhook area and out the exhaust louvers would be ignited by conditions occurring at time of afterburner initiation. Leakage from a fuselage seam or joint however, may not possess the conditions necessary for combustion and would therefore, appear as a vapor. Leakage required for these conditions may not result in significant change of the fuel quantity indicating system.

11. COMBUSTOR BURN THROUGH.

Combustor burn through can be the result of combustor cowl snout failing causing air blockage, fuel nozzle malfunction, or streak burning. Generally these burn throughs are progressive failures that require several hours of operation before serious problems are encountered. These failures are generally confined to one combustor. The combustion casing will develop a hot spot outside of the hole that

develops in the combustor cowl. The hot spot will develop into a bubble followed by a burn through of the combustion casing that could progress into the fuel cell number four above the combustion section. Burn through of the combustion casing is generally about four inches down stream of the forward combustion casing flange. See Figure 44. However, a combustor cowl snout failure can result in an air flow reversal with a subsequent burn through at the forward combustion casing flange.

Although not directly related to the propulsion system, the airflow around an aerodynamic surface, such as a wing follows the contour of the surface on the top and bottom until the angle of attack is increased. At slower airspeeds the airflow becomes detached from the upper surface of the wing progressing from the trailing edge forward as the angle of attack is increased. The airflow underneath the wing can bend around the trailing edge of the wing and move forward toward the area where the airflow detached from the upper surface. Therefore, if the aircraft is operating at a near stall condition and a fire is in the vicinity of the trailing edge of the wing, there will probably be evidence of metal splatter with the flow of the metal forward. This evidence can be used in estimating the airspeed of the aircraft during the inflight fire.

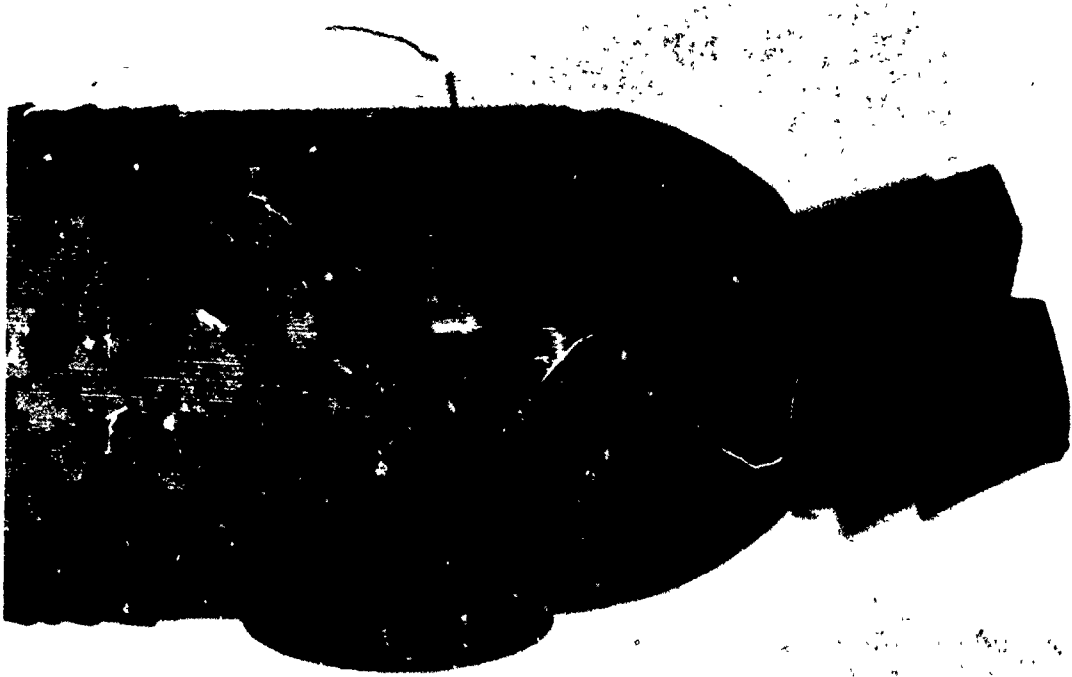


Figure 46 - Typical Combustor Burn Through

SECTION XI

INTERPRETATION OF INSPECTION FINDINGS

1. FRONT FRAME AND COMPRESSOR.

The condition of the front frame is not necessarily indicative of engine operation at time of impact. During the course of an accident, the engine front frame often sustains a large amount of impact damage which results in the front frame being pushed back into the compressor rotor. This results in a large amount of rubbing and tearing of the variable guide vanes and compressor blades. In extreme conditions, the inlet guide vanes can be torn completely free of the front frame, and will be found lodged in the compressor rotor. Care should be exercised in trying to estimate RPM from compressor rotor condition. Often a compressor rotor operating at high RPM and subjected to low impact damage will look similar to a compressor rotor operating at lower RPM but subjected to high impact damage.

The degree of rubbing and the extent of damage in this area is of interest: (1) to establish whether or not the rotor was turning at time of impact, (2) to make sure that the front frame damage occurred at time of impact, (3) to determine that the inlet guide vanes, compressor blades, or vanes did not fail first and cause engine failure. This assessment can usually be done, by evaluating the pattern of damage in the compressor rotor, together with the quantity of debris ingested by the rotor.

This material can, and usually does, cause the stage 1 blades to bend forward and impact on the variable guide vanes. This contact will appear as a number of sharp knife-like marks on the surface of the variable vane near the trailing edge, and the leading edges of the

stage 1 blades will be torn and curled back as a result of the impact. If the engine is stopped, or windmilling, at time of debris ingestion, the blades may not indicate contact with the variable vanes.

The amount of damage resulting from FOD is a function of air-speed, size of the foreign object and engine RPM. However, there are typical damage patterns from FOD. Often an imprint or other evidence can be seen on a compressor blade that may be helpful in determining the kind of foreign object that entered the engine. See Figures 45 through 48.

If the front frame was jammed into the rotor, the inlet guide vanes will usually be forced to the closed position. No significance should be attached to the position of the inlet guide vanes in this case.

The condition of the engine compressor rotor and stator will vary from almost complete destruction to virtually undamaged condition, depending on the type of impact, and the quantity and size of debris that has been ingested.

Normally, in the course of rapid deceleration after impact, a considerable amount of debris is ingested by the engine. Its effect on the compressor blades and vanes can often be used as a rough measure of engine RPM at time of impact. Methods for determining engine RPM may be found in Section V.

NOTE: The appearance of a ragged blade condition is a fairly good indication of moderate to high engine RPMs. The absence of this condition is NOT necessarily evidence of lower RPMs. It is quite possible for the engine to decelerate due to other impact damage before much debris is ingested. Considerable judgement must be used, there-



Figure 45 View of Front of Compressor After Ingesting A Foreign Object at 100% RPM Which Resulted From First Stage Blade Failure (185 Engine)

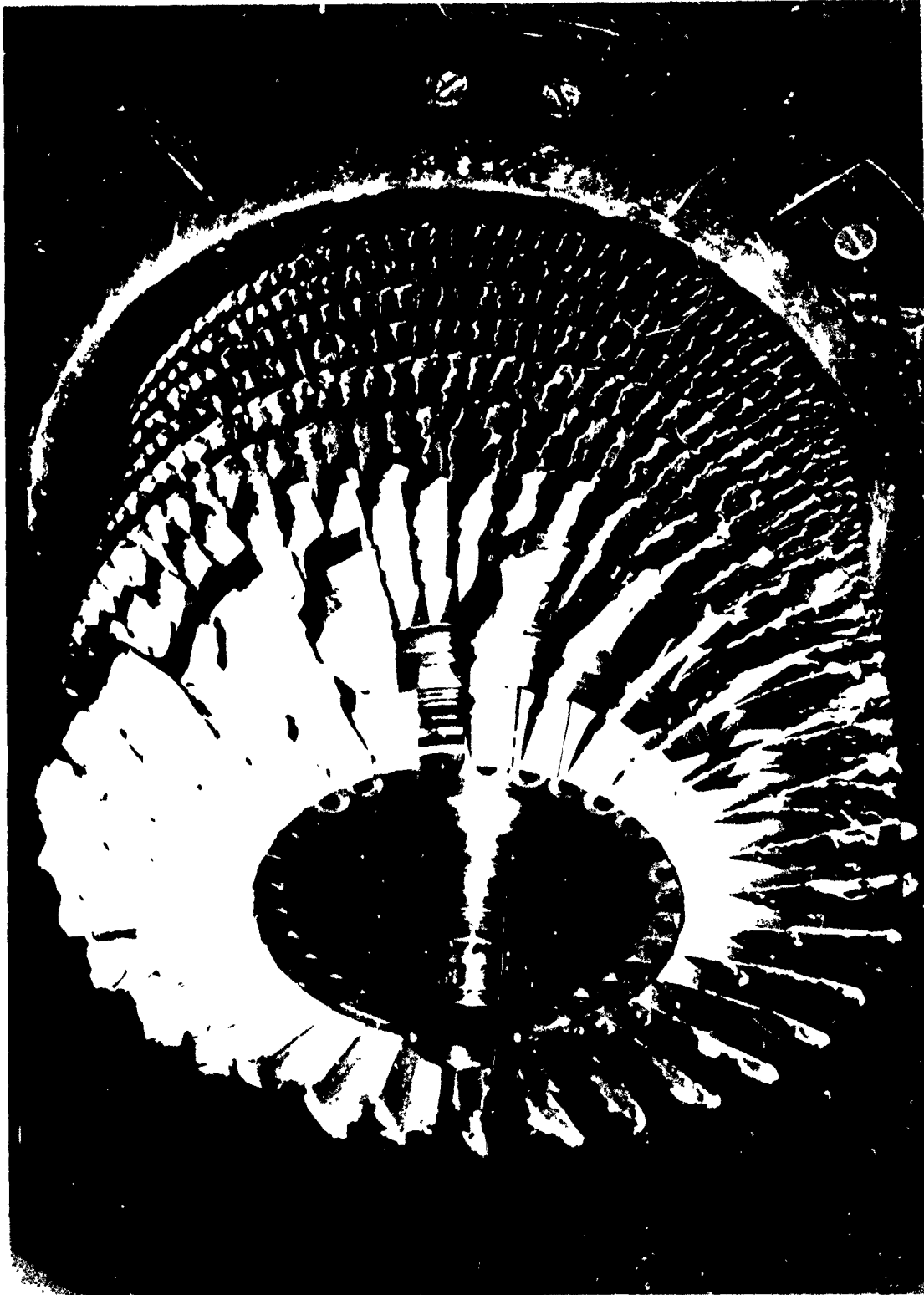


Figure 46 - View of Compressor Rotor After Ingesting a Foreign Object at 100% RPM Which Resulted from First Stage Blade Failure (I85 Engine)



Figure 47 - Second Stage Blade Failure at 1007 RPM (185 Engine)

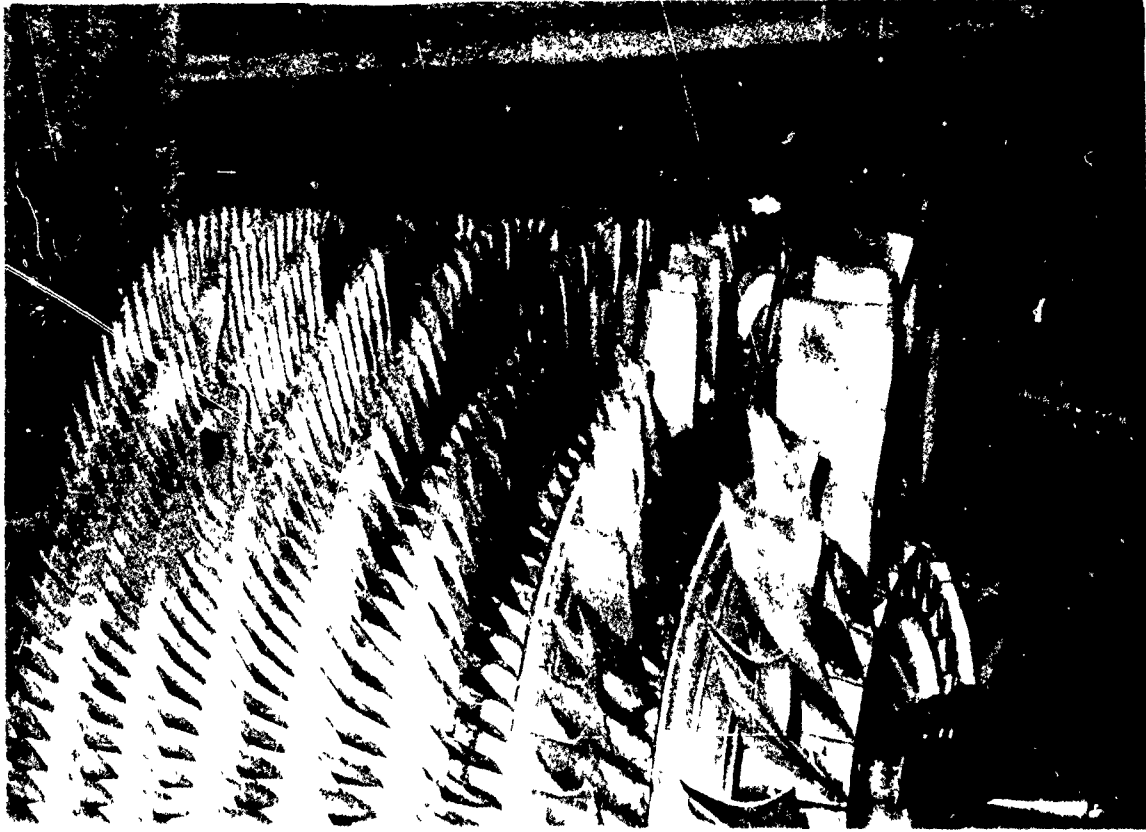


Figure 48 - Foreign Object Damage at Low RPM (J85 Engine)

fore, in evaluating rotor condition, and other factors such as aircraft incidence angle, velocity at impact, etc., must be considered in determining when the engine ingested foreign objects.

If the engine is rotating at high RPM at time of impact, foreign objects in quantity will cause the rotor blades to assume a very ragged looking appearance. A large number of nicks, gouges, and dents will appear on all surface and edges of the blades as well as on the stator vanes. The rotor blades may be forced forward into the compressor stator vanes, resulting in a curling of the stator vane's trailing edges. Compressor rotor blades may be curled, bent, or broken off in a direction opposite to rotation. Severe scoring and peening of the compressor blades, and a rounding of their tip corners, indicate high RPM.

Under extreme impact conditions, with high RPMs, the rotor may be "corn-cobbed", and all stator vanes stripped from their bases, as well. Under these conditions, the rotor and stator may actually disintegrate. Debris ingestion alone will not always result in the complete stripping of blades from a rotor, even with the rotor operating at 100% RPM, although some breaking of blades at their root may occur.

At lower engine RPMs, debris ingestion results in a different type of damage. The stator vane curling and blade leading edge curling still occurs, but the blades and vanes do not have the ragged edge appearance as they have at higher RPMs and their condition will be uniform.

In addition to blade and vane appearance, there are other types of damage that can often be used to estimate engine RPM. Blade bending

and tearing is normally associated with low RPM. At low RPM blade damage will show little evidence of peening. blade tip corners will maintain their squareness. Some low RPM damage is usually present when debris is ingested, and can be readily detected as blades or vanes will be bent into contact with one another, and yet 360 degrees contact around the rotor or stator may not occur. Breaking or shearing of blades and vanes at their roots is usually associated with ingestion of large quantities of debris or when impact results in severe distortion of the compressor casing, resulting in breaking of blades and vanes which then causes the same type damage as foreign object ingestion.

Indications of compressor rotor speed are usually available if the compressor case has been punctured during impact. Corncobbing of the compressor blades coupled with extensive damage aft of the point of puncture, is indicative of high RPM at time of impact. However, if tearing and bending of the blades occurs in the vicinity of, and aft of the point of puncture, low RPM at time of impact may be suspected.

If fatigue of blades and vanes is suspected, the broken blades and vanes should always be visually inspected for evidence of fatigue. If this visual inspection reveals that fatigue could have caused the fractures, the blades and vanes should be sent to a materials laboratory for detailed analysis. A high RPM failure of either a blade or vane will result in major damage to the compressor stages aft of the stage with the broken piece. Damage due to this type failure will usually cause a stall and flameout.

In flight FOD is generally associated with ice, munitions, or bird

ingestion. If any major FOD occurs to the engine it develops a tendency to stall. Severe FOD will destroy the pumping capability of the compressor, thus making it impossible for the engine to operate. Consequently, evidence of high RPM engine operation at the time of an accident usually eliminates the possibility of severe FOD or parts failure in the front of an engine.

If marks are found on a J79 engine that indicates FOD ingestion occurred at high RPM, a careful examination of the rotor and compressor should be made to determine when these marks were made. If impact damage or other evidence indicates a low to moderate RPM (48% - 80%) at time of impact, and high RPM is indicated at time of FOD ingestion, then ice or FOD ingestion at altitude should be suspected. This could have resulted in sufficient damage to the compressor blades to have caused engine stalls, and perhaps have caused the accident.

Ingestion of a bird in flight will usually cause damage to the compressor similar to that caused by ice ingestion. However, the bird's body parts are usually found throughout the compressor section.

NOTE: There are materials within the aircraft that resemble bird feathers when ingested in the compressor upon impact. A laboratory analysis must be made for final determination.

Ingestions of munitions in flight usually results in an imprint of the object on the compressor blade that it first comes into contact. If the object strikes the forward side of the blade, the blade would be forced aft and would probably rub on the stator vanes aft of that compressor stage. If this happens, the object is normally slowed down before it is ingested into the next stage. The damage aft in the compressor will be the result of the blades deflected forward rubbing

the stator vanes. From that point aft, the object can normally be traced from one stage to the next by the imprints or damage pattern on the blades and vanes.

The No. 1 bearing area should be inspected for presence of lube oil and the condition of the bearing should be recorded. Lack of lubrication in this area may be indicative of lack of oil to the other bearing areas, which may have caused an engine failure.

Fatigued spots or spalls on the bearing, although indicative of a bearing that is in the process of failure, should not receive undue emphasis unless the bearing has clearly failed and caused obvious additional damage to the compressor rotor or the front frame. See Section IV discussion on bearing failures.

2. ENGINE ACCESSORIES

Examination of the engine accessories will usually provide valuable information to assist in evaluating an accident. Careful inspection of the various accessories is essential in order to completely evaluate conditions found during inspection of the main engine structure.

The condition of the various drive shafts and splines should be carefully examined. If any of the shafts or spline have failed, the method of failure should be determined.

Torsional shearing of the shafts can be caused by the accessory seizing on impact, or could have been caused by seizing of the accessory during flight due to parts failure, or contamination.

Nontorsional shear is invariably caused at impact.

Fatigue failures of any shaft may cause an in-flight malfunction of the engine, and the effect of a fatigue failure should be evaluated to determine if it was the cause of an engine failure.

NOTE: Examples of torsional fatigue, rotating bending fatigue and torsional fatigue with 45° step type failures may be seen in Figures 49, 50, and 51 respectively.

Failures of the different drive shafts have different effects on the engine operation.

Failure of A/B pump shaft will prevent the engine from going into A/B operation. The A/B will flameout if shaft fails during A/B operation.

Failure of main fuel pump drive shaft will cause immediate flameout. VG actuators will not move.

Failure of lube pump shaft will cause engine bearings to start to fail in a short period of time (30 sec - 1 min).

Failure of the radial drive or horizontal shafts will cause immediate engine flameout.

3. MAINFRAME AND NO. 2 BEARING AREA.

External inspection of the mainframe will rarely provide significant information regarding engine operating conditions. However, the struts and inner support structure should be examined for cracks that would indicate that the engine had been running with excessive vibration.

Before disassembling the compressor rear frame from the compressor case, the No. 2 bearing should be carefully checked for proper seating and locking. Improper seating or locking can result in the rotor shifting forward with a resultant engine failure.



Figure 49 - Gear Shaft Failure Caused by Torsional Fatigue with 45° Steps



Figure 50 - Shaft Failure Resulting from Torsional Fatigue



Figure 51 - Shaft Failure Resulting from Rotating Bending Fatigue. Center of the Shaft Failed in Shear

The No. 2 bearing sump area, and the No. 2 bearing should be examined for signs of lubrication or lack of lubrication. The bearing itself should be examined carefully. If the bearing is intact, then the possibility of a lubrication system failure is unlikely, even if the bearing sump shows very little oil. A lubrication system failure will result in very rapid failure of this bearing due to high loads to which it is subjected. The No. 2 bearing can be disregarded as the cause of engine failure if the bearing is whole, even though there may be fatigue spalls on the races and/or balls. See Figures 52 and 53 which illustrate fatigue of the race and balls. See Section IV for a detailed discussion of bearing failures.

If the bearing races have badly deformed surfaces, over-temperature is evident, and the ball cage is broken and split, then the bearing undoubtedly failed prior to impact. See Figure 53. When a No. 2 bearing does fail, the compressor rotor probably will shift forward and engage the compressor stator vanes, resulting in severe compressor damage.

The radial drive or horizontal drive should be inspected for signs of any failure including spline failures and improper assembly of parts that could result in the cessation of power output to the accessory gearboxes. Any accessory drive power failure will cause an immediate flameout of the engine since the fuel pumps and controls are driven by the rear gearbox.

4. COMBUSTION SECTION.

Inspection of the combustion section of the engine will often show a quantity of material (dirt and vegetation from the impact area)

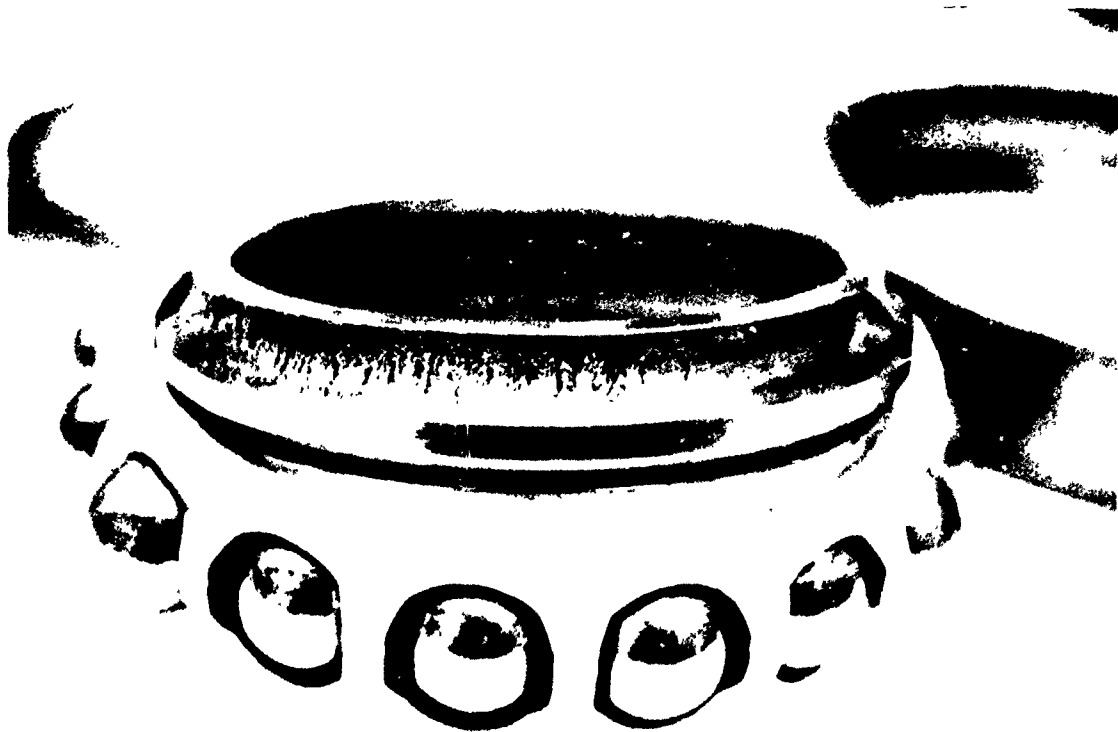


Figure 52 - Incipient No. 2 Bearing Fatigue Failure

This bearing has started to fail in fatigue. If this bearing had continued in service, it would only be a matter of a few hours before it would fail completely, and major damage to the engine would result from the rotor shifting forward.

No engine malfunction had been caused by this bearing. All that might be noticed when a bearing reaches this condition would be an increase in vibration level, and some steel particles in the lubrication filter.

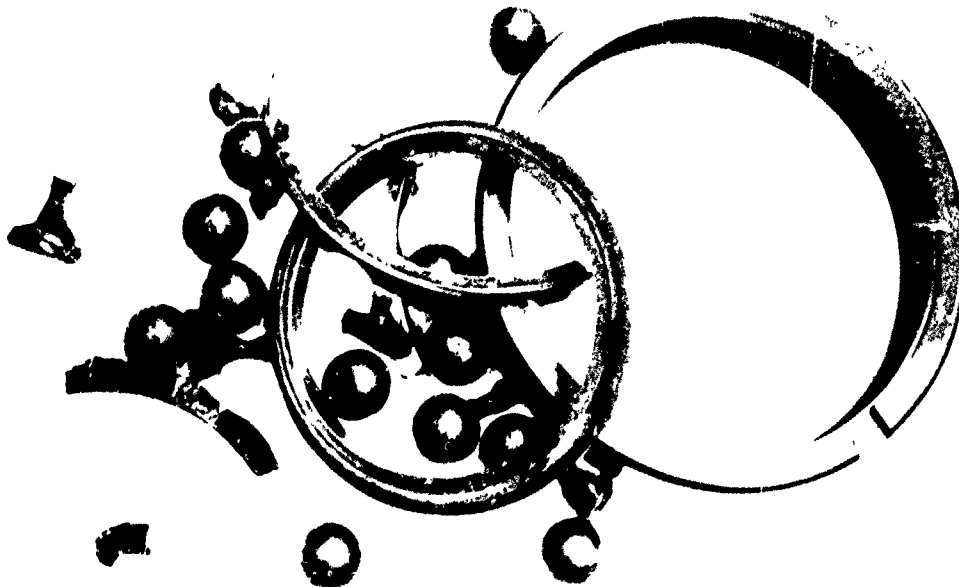


Figure 53 - No. 2 Bearing Fatigue Failure

This bearing caused an engine failure.

Fatigue of the balls and races had progressed to the point where the cage broke and the bearing would no longer support the rotor load.

Note the indentations in the balls where they were deformed from the extremely high temperatures generated during the failure.

Bearing failure of this type will allow the rotor to shift forward, engaging the compressor stator vanes, and result in complete failure of the compressor and seizure of the rotor.

packed around the aft end of the liner's outer shell. A smaller quantity of material will often be found around the inside shell. Material found in this section will have fairly fine texture and the appearance of saw dust. Large objects ingested by the engine will either be chopped up by the compressor, or will jam in the compressor. Vegetation that is ingested by the engine will be finely chopped by the compressor.

If vegetation is packed around the combustion liner, the material next to the liner itself will probably be charred or slightly burned by the heat of the liner. If charring is evident, the engine was operating at time of impact. A lack of charring should not be taken as evidence that the engine was not operating at time of impact as the ingestion of any quantity of material may cause an immediate stall or engine flameout. The only heat available to char material in the combustion chamber has to come from the combustion casing and liner shell. This residual heat may not be enough to cause extensive charring.

The combustion liner itself should be checked for any burnouts in the shells as evidence of possible fuel nozzle malfunction, combustor, snout failure or improper igniter plug installation. If large pieces of the liner are missing, it is quite possible that these could have caused turbine FOD and possibly engine stall. Any disturbance in the normal cooling air flow or fuel flow in the liners could cause "burning" of the liners, turbine nozzles, or turbine wheels. As long as burned pieces remained intact, their condition would probably not contribute to an accident. See Section X for discussion of fuel ingestion and combustor burn through respectively.

5. TURBINE AND TURBINE NOZZLES.

Both turbine nozzles should be checked for signs of burnouts that could indicate a possible combustion or fuel flow malfunction. Hot streaks from these causes will cause metal burning. See Figure 54.

High speed rotational damage will be indicated in the turbine section by the bending of turbine blades opposite to the direction of rotation by low speed rubbing of the turbine disc, blades, shroud ring, or exhaust cone. Low speed rubs are identified by lack of concentricity and little or no metal discoloration. See Figures 55 and '6.

The presence of dirt and vegetation in the compressor rotor will invariably be accompanied by dirt and vegetation in the turbine nozzle passages if the engine was rotating at time of impact.

The absence of contamination in the turbine nozzle may indicate low air flow and low engine RPM, assuming dirt and contamination were found in the compressor. Turbine nozzle partitions and turbine blades should be inspected carefully for foreign object damage and adhering metal particles. If an engine ingests small quantities of metal during operation or upon impact this metal may melt and fuse to the turbine and nozzles. Aluminum particles normally come from the aircraft. Normally parts break up and get into the turbine would completely destroy the compressor first. In this case fusion probably would not take place.

Silver, chromium or nickel particles would probably come from the No. 1 bearing area, and would be indicative of a major failure of the No. 1 bearing or associated hardware.

Turbine blades that are burned off or failed because of stress

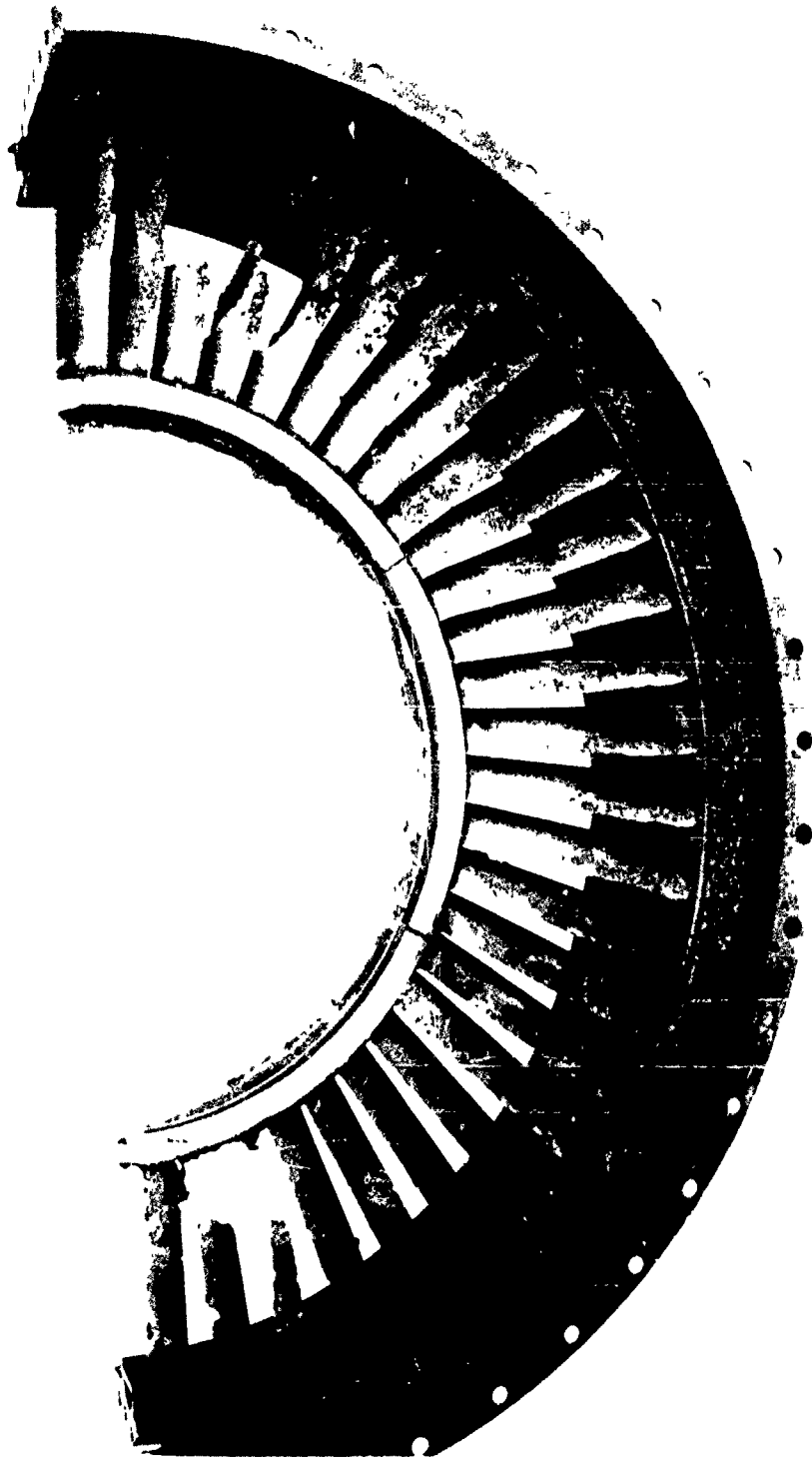


Figure 54 - Burned 2nd Stage Turbine Nozzle (185 Engine)

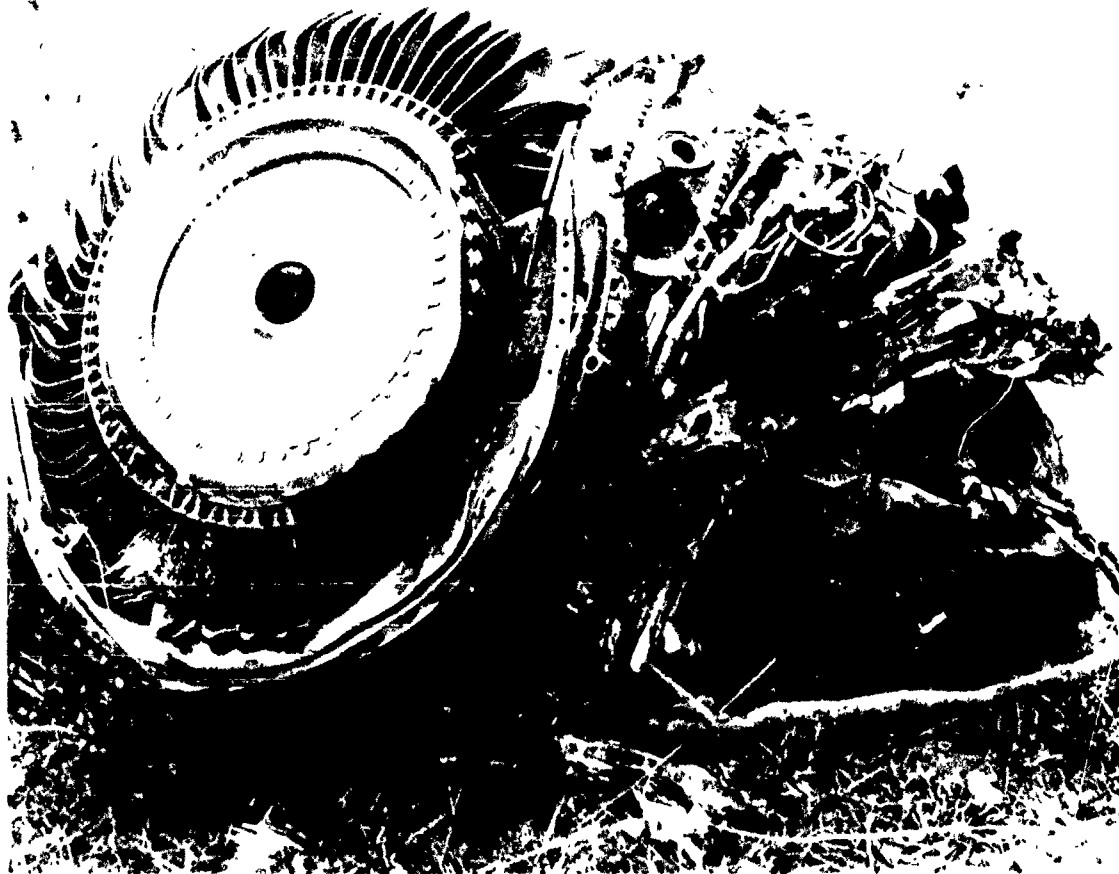


FIGURE 55 - TURBINE ROTOR THAT WAS OPERATING AT 83-85% RPM AT
TIME OF IMPACT. NOTE HIGH IMPACT DAMAGE.

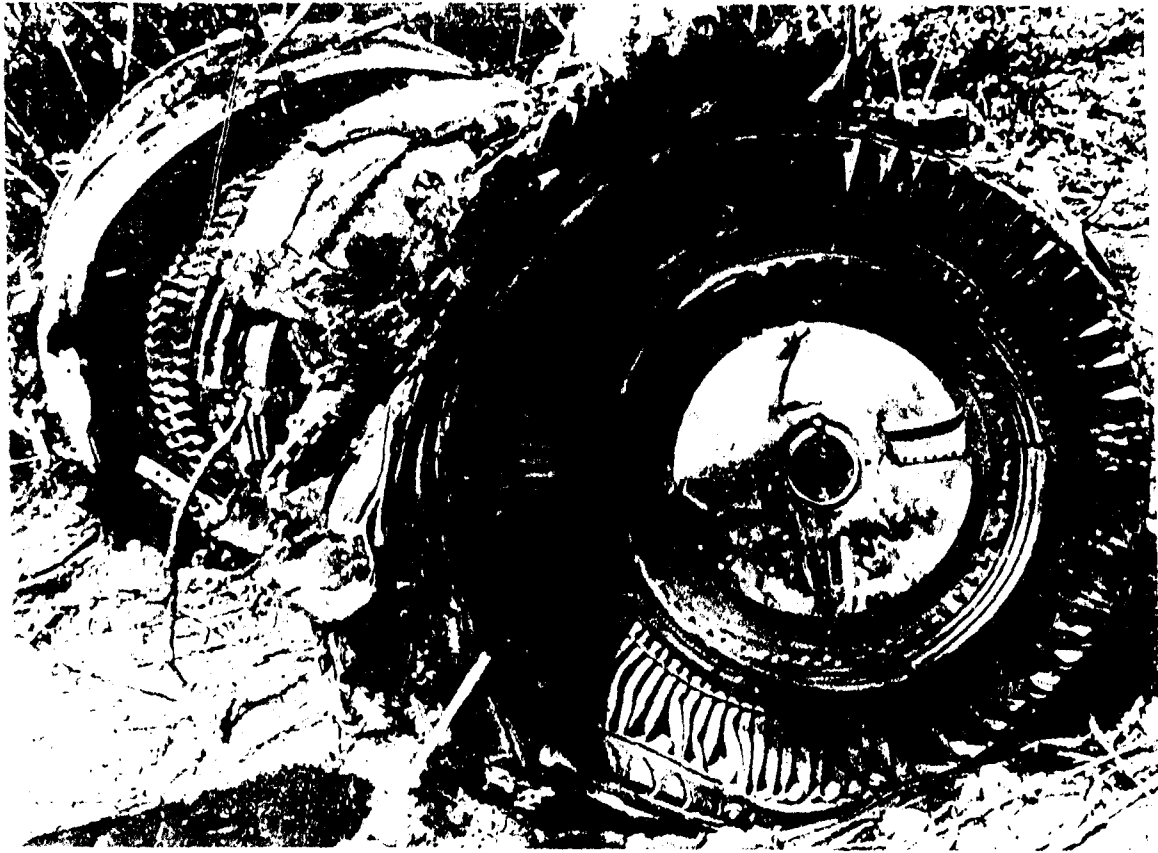


FIGURE 56 -- TURBINE ROTOR OPERATING AT 18-25% RPM AT TIME OF IMPACT . . . NOTE MEDIUM IMPACT DAMAGE

rupture are positive identification of an engine that had been running in an over temperature condition for at least several seconds. Over temperature can result from an engine stall. See Figures 57 and 58. An engine stall will not necessarily burn off blades, and the absence of this type of failure does not rule out the possibility of a stall.

Turbine blades that exhibit cracks on their leading edge, indicate that the engine may have been running at a higher than normal temperature, which reduces stall margin, resulting in tendency for the engine to stall. See Figure 59.

Moderate to severe impact normally results in shearing or breaking of the turbine blades at their roots. Therefore, shearing or breaking of the turbine blades at their roots is not indicative of engine RPM at time of impact. This type failure should not be used to estimate RPM unless no other means of making the estimate is available. See Figure 55.

6. VARIABLE EXHAUST NOZZLE.

The area of the variable exhaust nozzle varies with throttle angle in accordance with a fixed mechanical schedule until such time as the EGT exceeds the T_5 vs RPM reference schedule. When this happens the nozzle area may be larger than that illustrated in Figure 60. However, this figure should provide the investigator with a close approximation of the area for a given throttle angle. Determination of throttle angle was discussed in Section V. If the nozzle area is larger than it should be for any given engine operating condition, a loss of thrust will result.

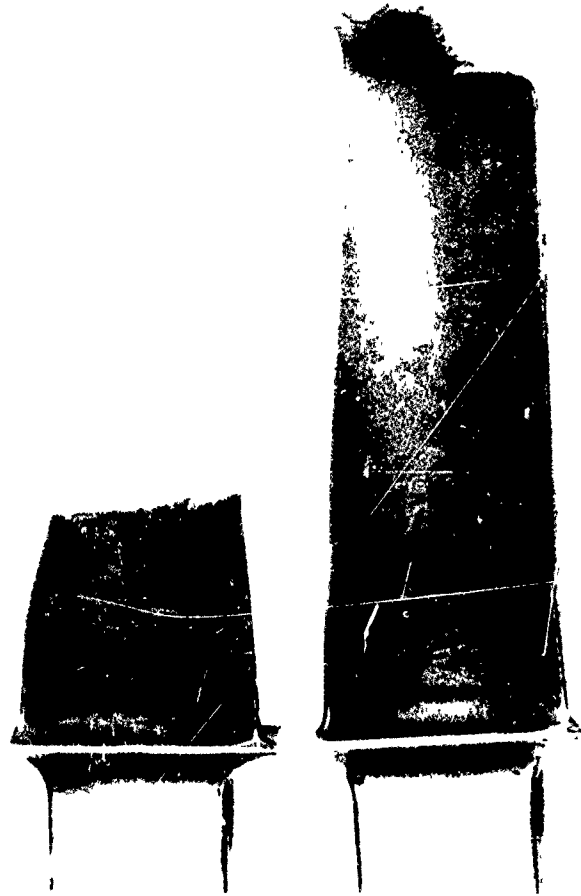


FIGURE 57 - MACROGRAPH OF 1ST STAGE TURBINE BLADES
THAT FAILED BY STRESS RUPTURE. NOTE THE
NECKING DOWN OF THE BLADE IN THE AREA
OF THE FAILURE

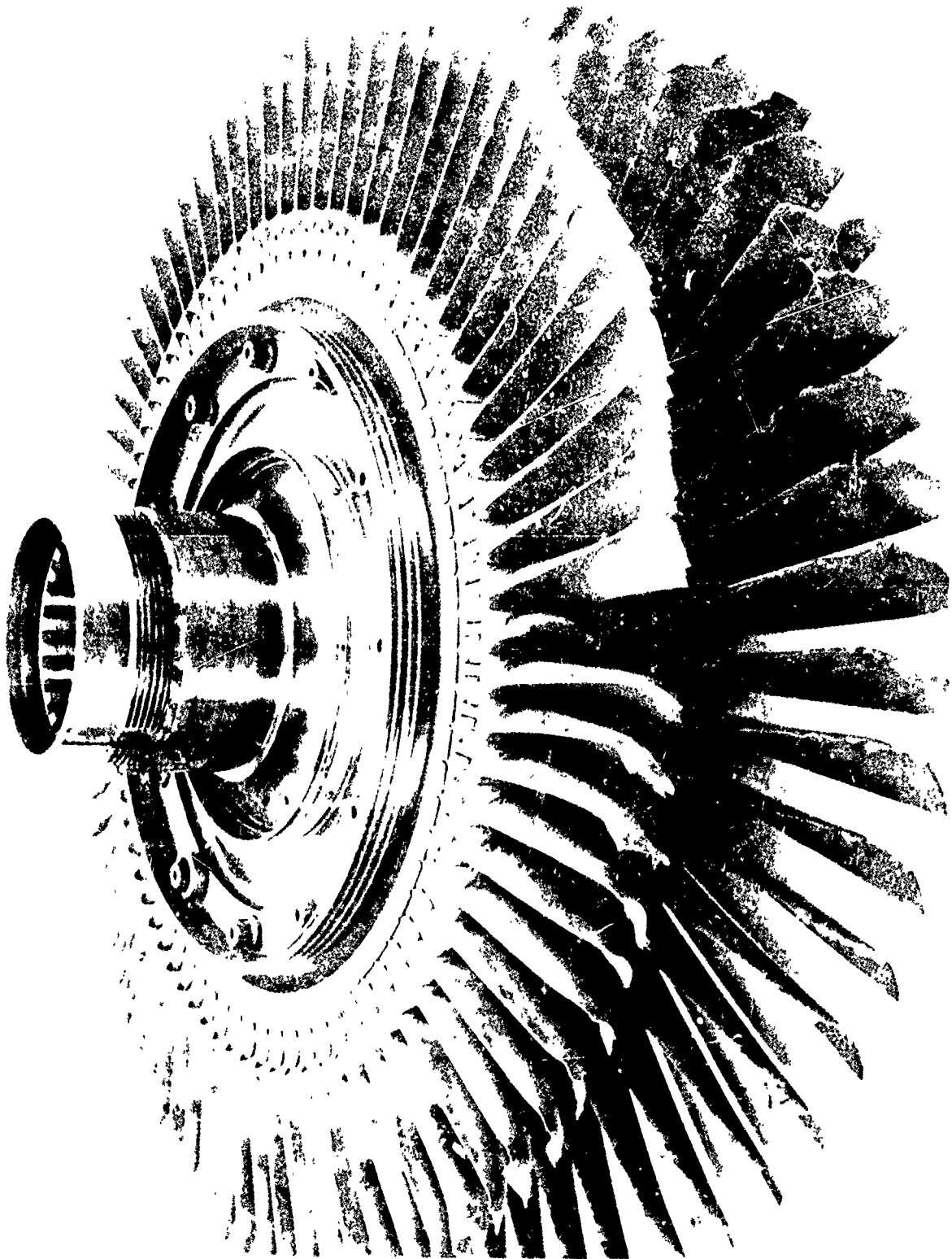


Figure 58 - Second Stage Turbine Blade that Failed From Stress Rupture At 100% RPM Power During Pre-Overtemperature Operation (185 Jet Engines)



FIGURE 59 - TURBINE BLADE SHOWING THERMAL STRESS
CRACKS ON LEADING EDGE

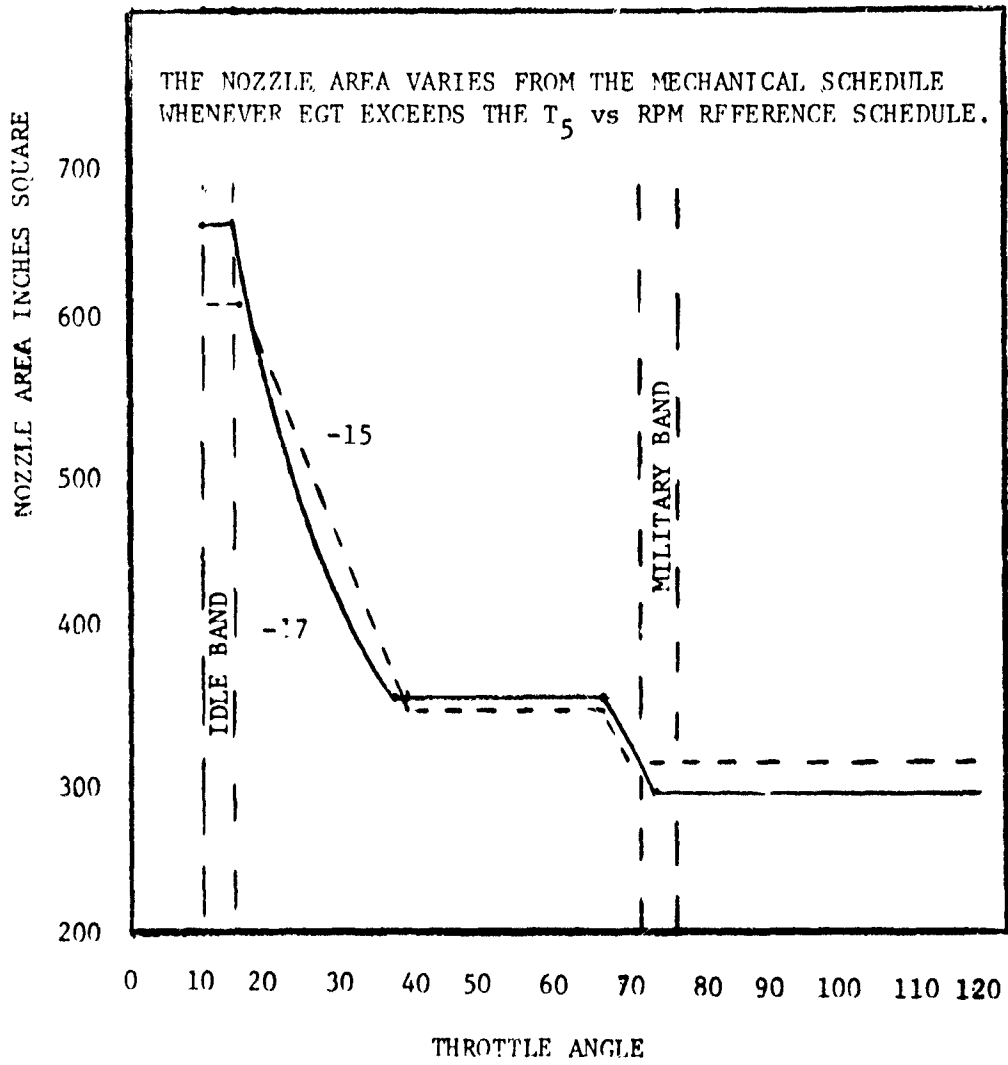


Figure 60 - Nozzle Area vs Throttle Angle

To properly interpret variable exhaust nozzle position, the accident investigator should be familiar with the operation of the variable exhaust system, and the way that the engine control system operates to position the nozzle. If the investigator is not familiar with the system, he can easily misinterpret the meaning of the nozzle position, and possibly be lead to improper conclusions regarding the accident. Methods of determining VEN area may be found in Section VII.

7. OPERATING PARAMETERS

The preceding chapters were devoted to means of determining such important parameters as RPM, fuel flow, exhaust gas temperature and variable exhaust nozzle area. It should be re-emphasized that all possible means of determining a parameter should be used. By cross checking the results, the most accurate determination can be established. It may be found that the easiest method might give erroneous results.

Having established two of the following parameters RPM, MACH number, fuel flow, or thrust, the other parameters can be determined through the use of Figures 61 through 68. It should be noted that the figures are drawn for both the -15 and -17 engines under sea level and 5000 foot altitudes. Interpolations of the figures must be made to more closely approximate the conditions at time of impact.

If it is found that the 3 "D" cam of the main fuel control or the torque booster show that a high throttle setting had been established but the fuel flow was low, it is a definite indication that the engine was flamed out.

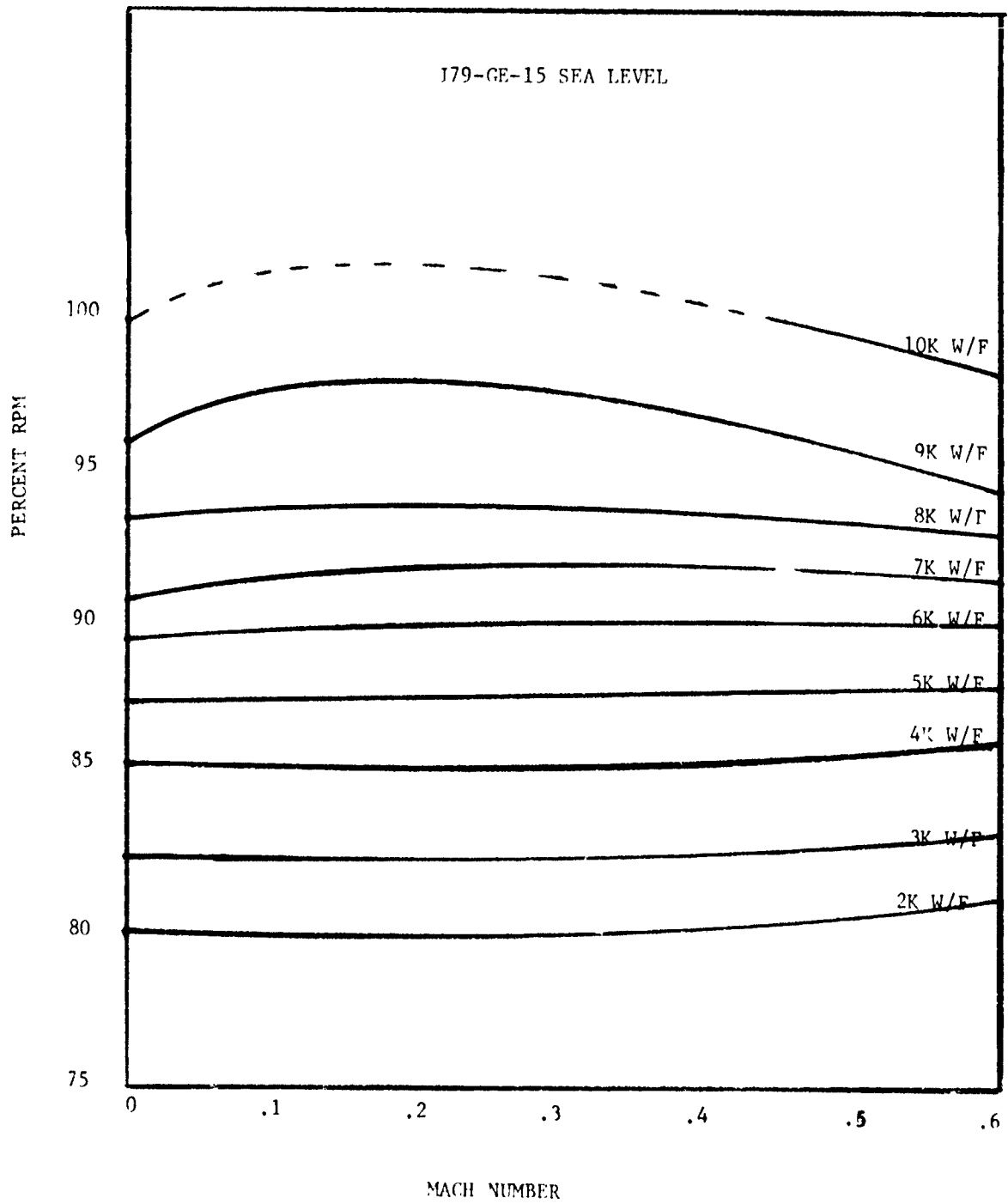


Figure 61 - J79-15 Fuel Flow for various RPMs and Mach Numbers (Sea Level)

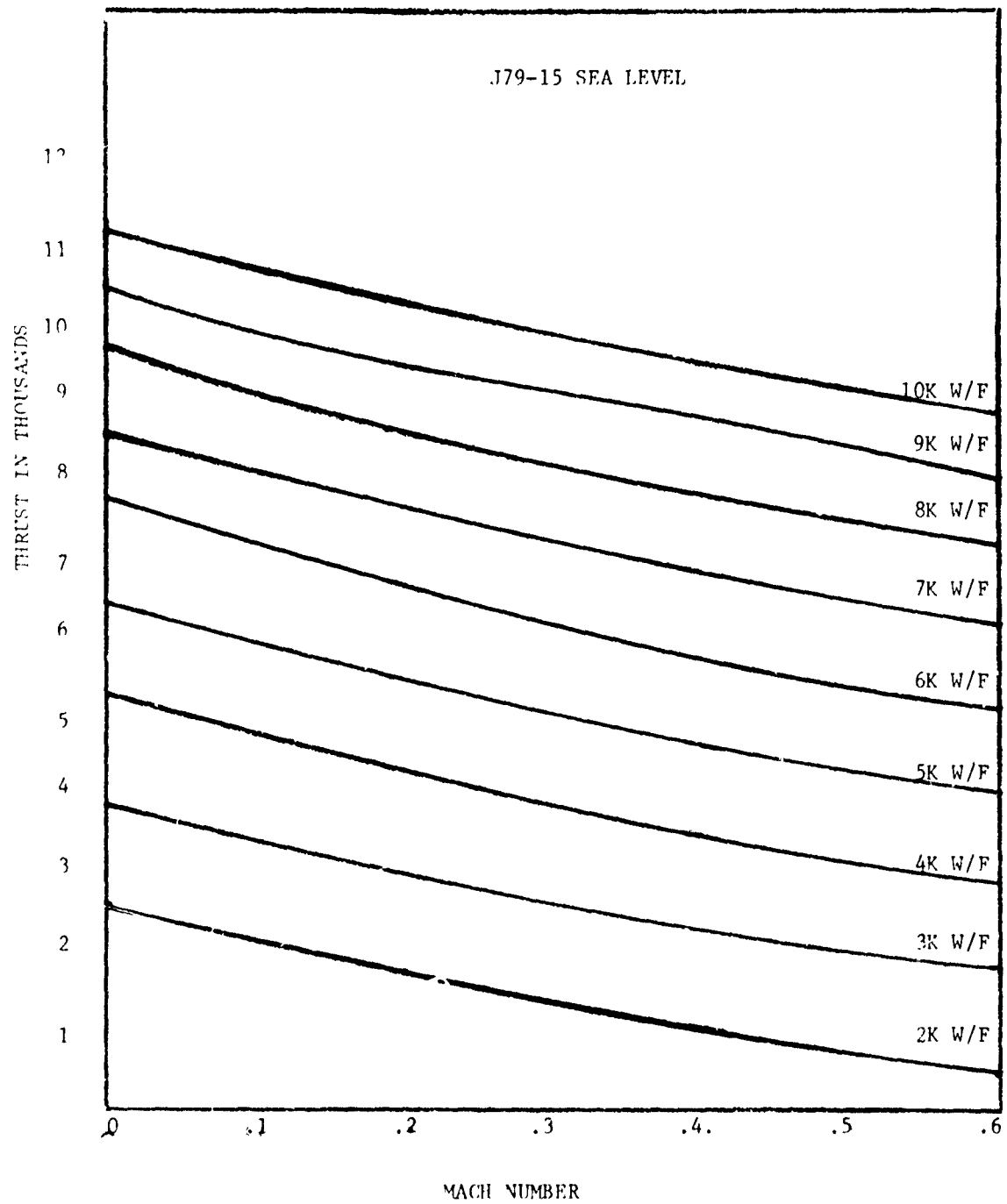


Figure 62 - J79-15 Fuel Flow at Various Thrust Settings and Mach Numbers (Sea Level)

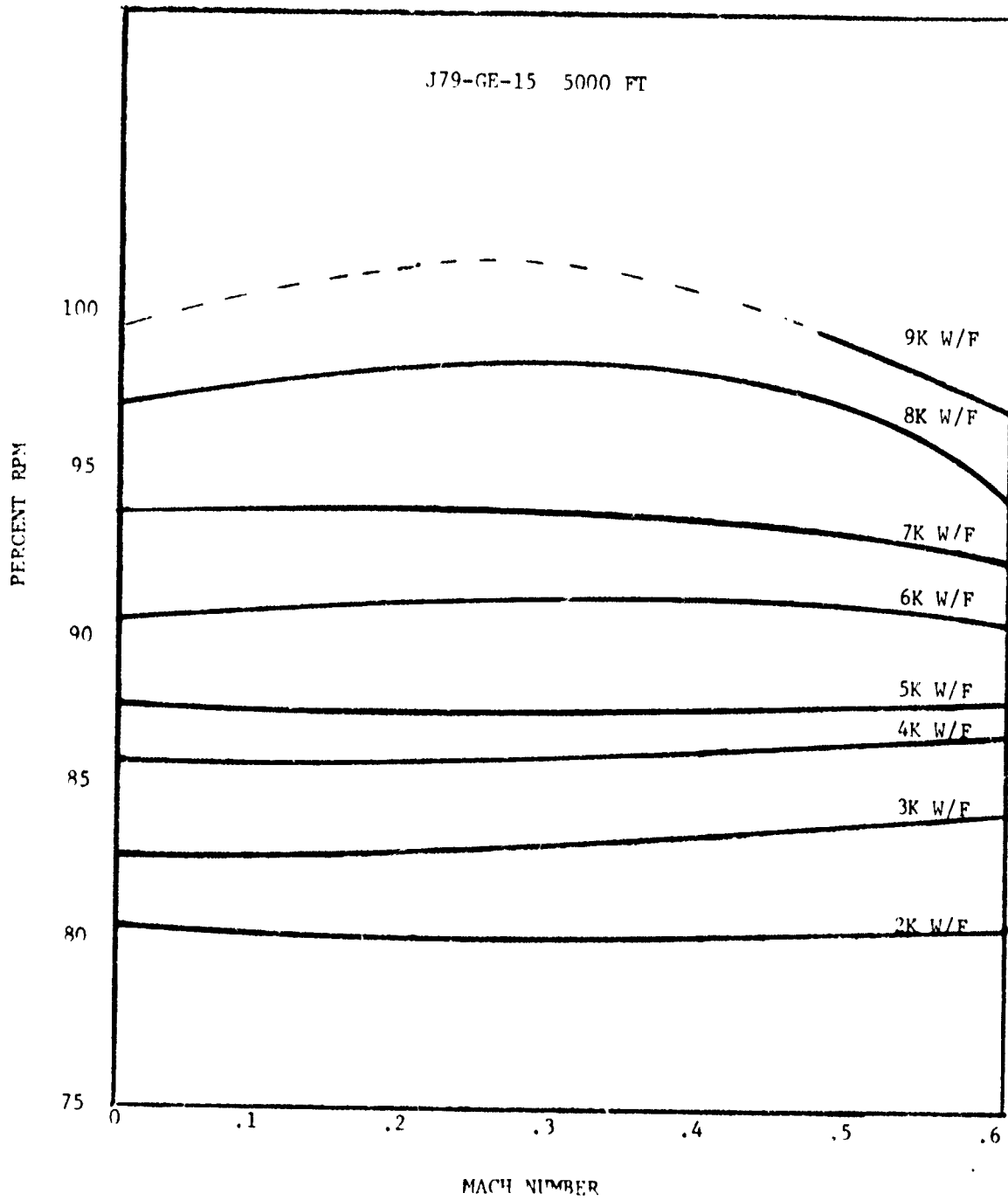


Figure 63 - J79-15 Fuel Flow at Various RPMs and Mach Numbers (5000 Feet)

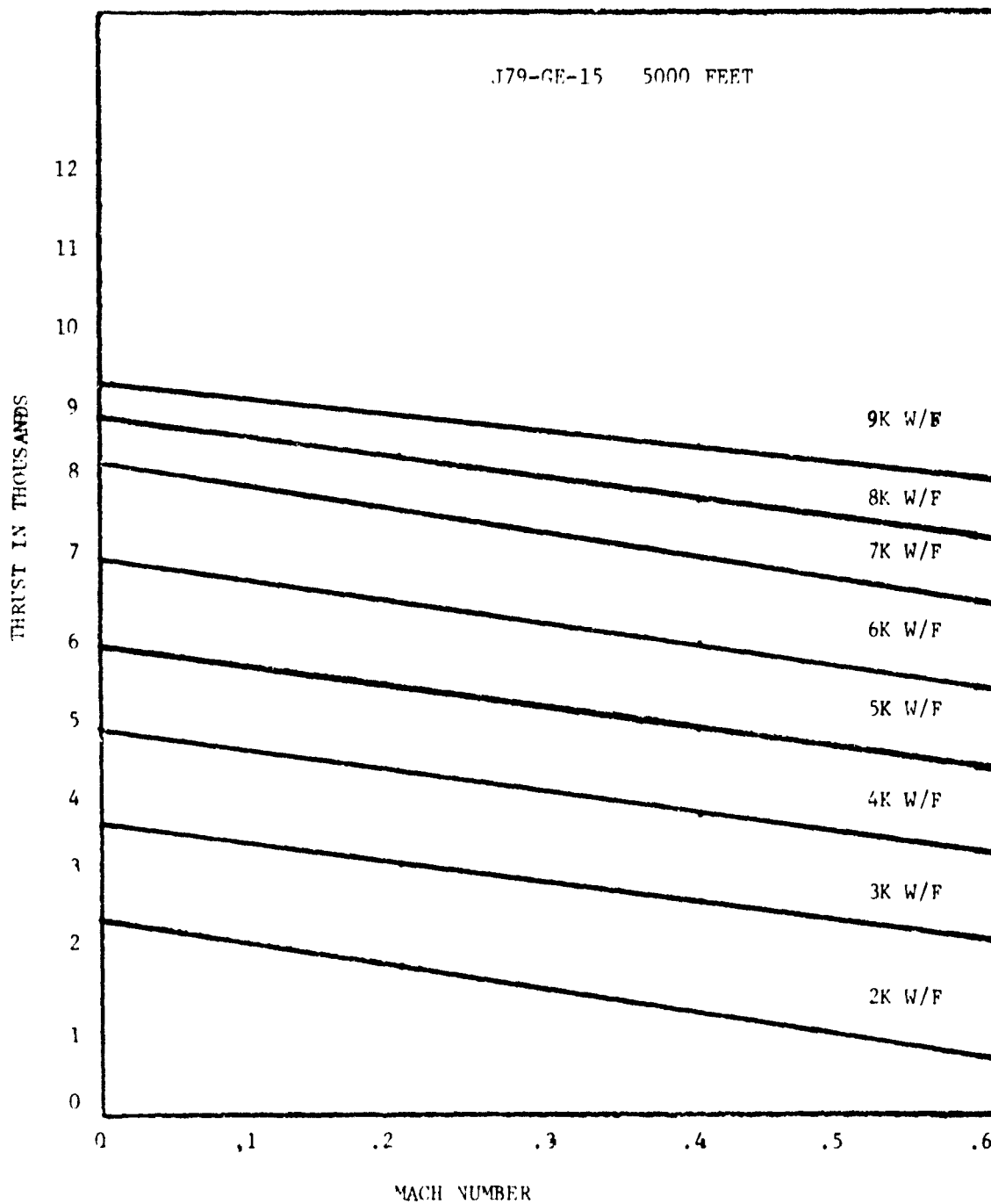


Figure 64 - J79-15 Fuel Flow at Various Thrust Settings and Mach Numbers (5000 Feet)

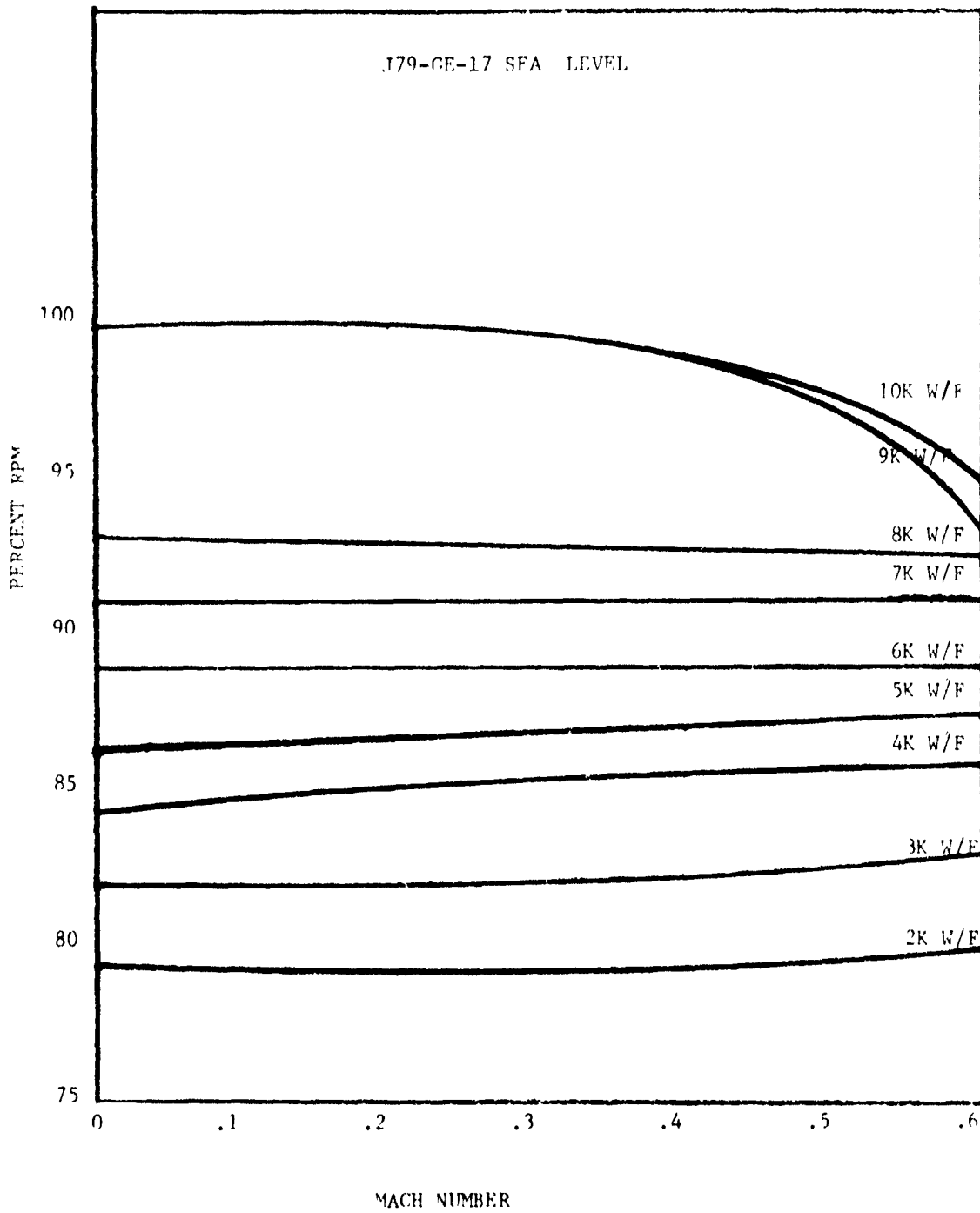


Figure 65 - J79-17 Fuel Flow at Various RPMs and Mach Numbers (Sea Level)

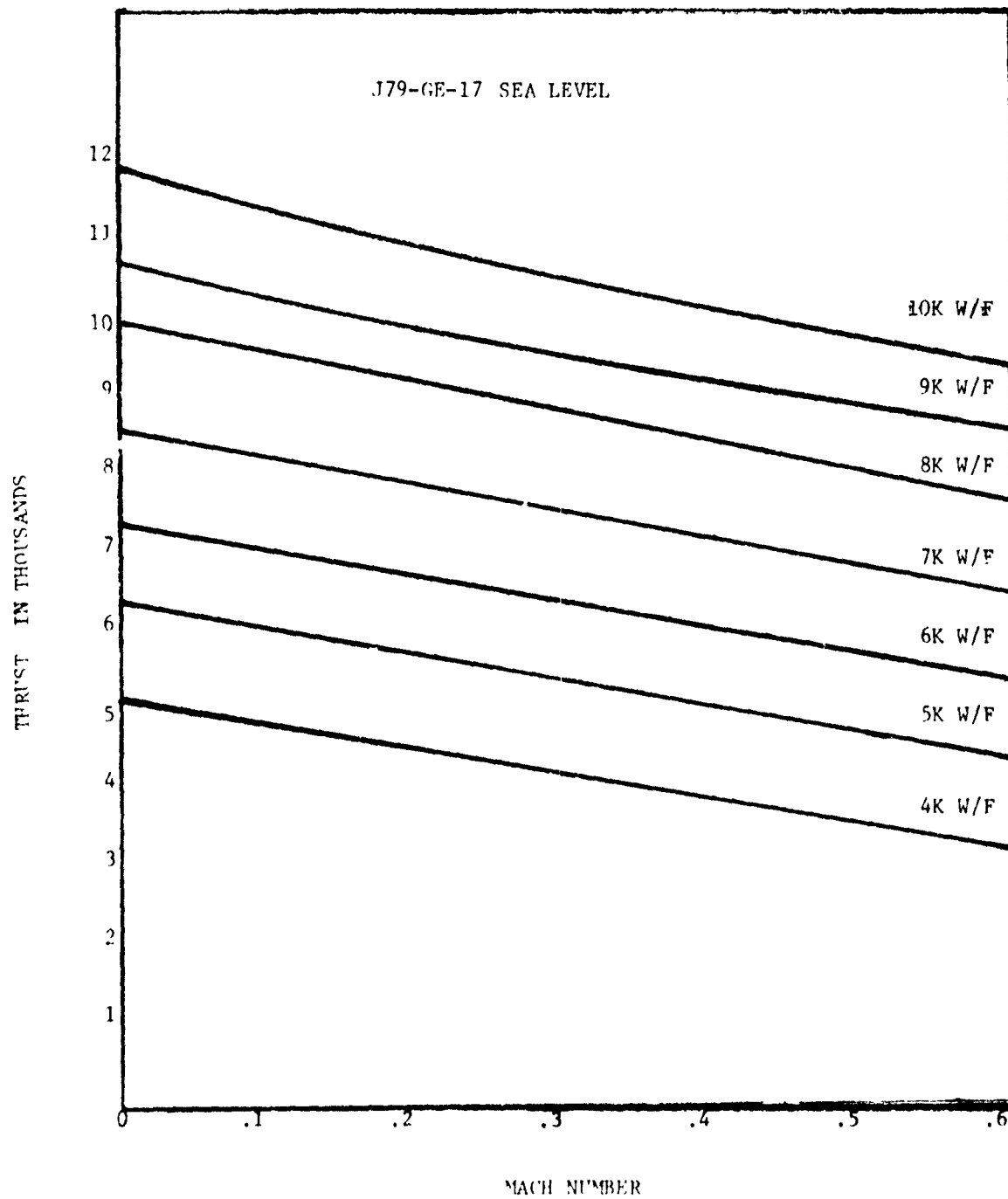


Figure 66 - J79-17 Fuel Flow at Various Thrust Settings and Mach Numbers (Sea Level)

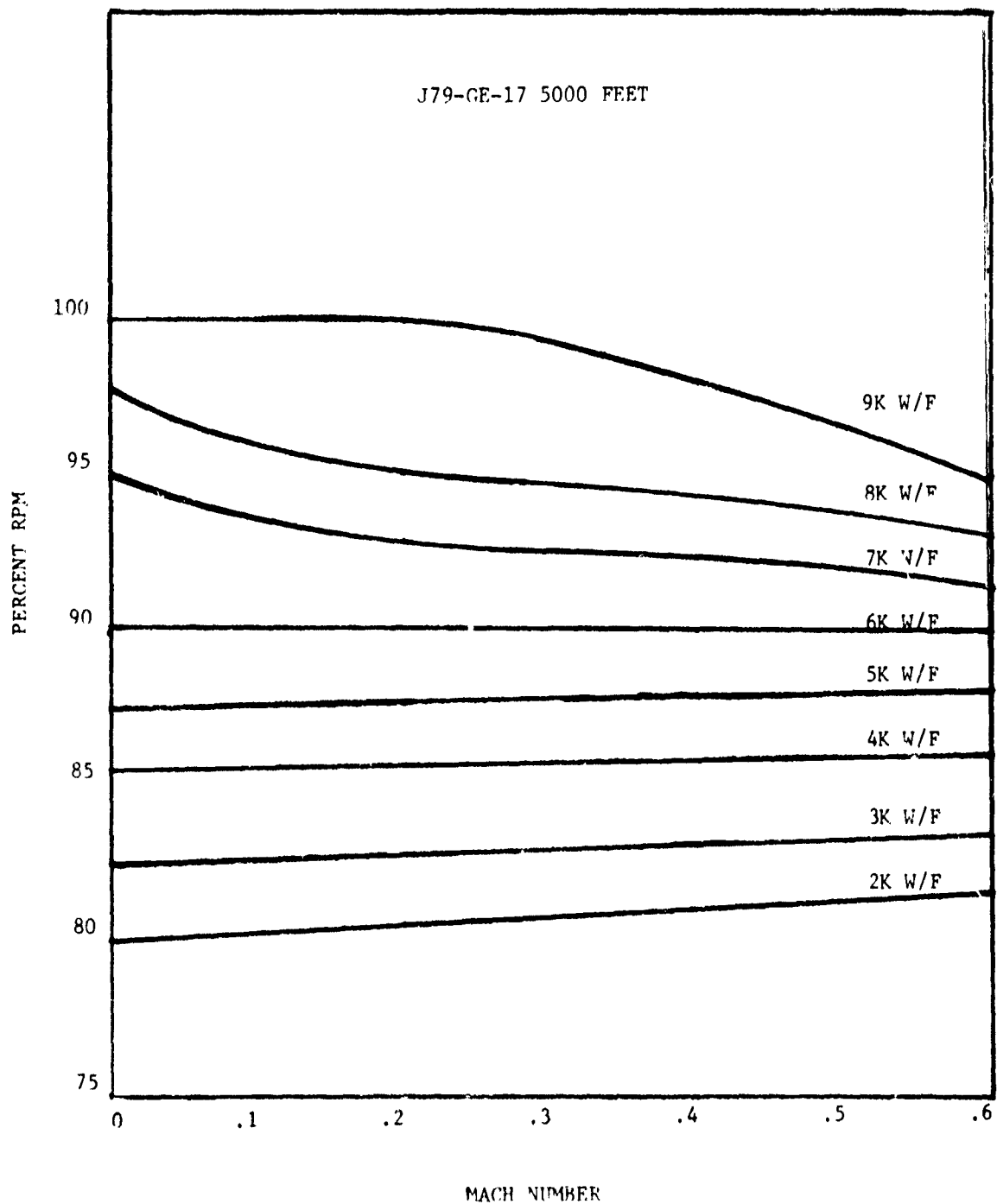


Figure 67 - J79-17 Fuel Flow at Various RPMs and Mach Numbers (5000 Feet)

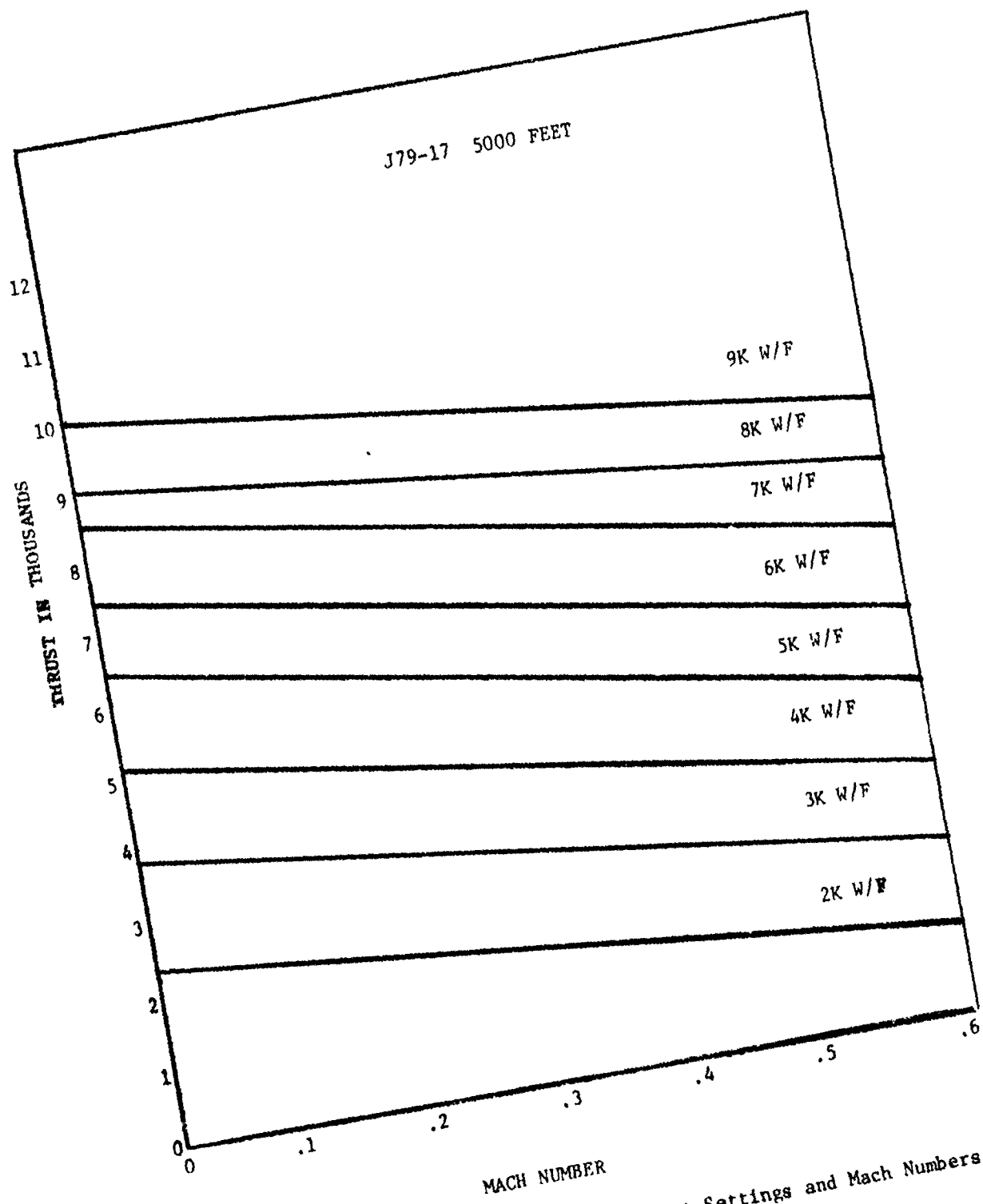


Figure 68 - J79-17 Fuel Flow at Various Thrust Settings and Mach Numbers (5000 Feet)

INTRODUCTION TO APPENDICES

The following Appendices are provided to assist the propulsion accident investigator to accomplish his assigned task in an orderly and systematic manner. Appendix A provides a listing of information to be obtained at the preliminary meeting.

Appendix B is a listing of things to do and items to be checked at the crash site. Some of the information listed in Appendices A and B may not be obtainable. However, it is suggested that as much of the information as possible be recorded. It is also suggested that these two appendices be locally reproduced for the accident investigator's use.

Appendix C lists the items to be checked during disassembly in the engine shop. The condition of the various parts should be described in notes and photographs should be taken as appropriate.

Often metallic objects are found in the lubrication system. Appendix D is provided to help determine the origin of the object.

Appendix E is a listing of miscellaneous aircraft and engine information that will be beneficial to the accident investigator in conducting his investigation and in writing his report of his findings.

APPENDIX A
INFORMATION TO BE OBTAINED AT PRELIMINARY MEETING

- A. Aircraft Serial Number _____
- B. Engine's Serial Number LH _____ RH _____
- C. Pilot's name and rank. _____
- D. Pilot's flying experience _____
- E. Altitude, attitude and speed of aircraft when problem was first encountered _____
- F. Length of flight _____
- G. What were the engine's instrumentation readings?
- | | LH | RH |
|------------------|-------|-------|
| (1) EGT | _____ | _____ |
| (2) RPM | _____ | _____ |
| (3) NPI | _____ | _____ |
| (4) Oil Pressure | _____ | _____ |
| (5) Fuel Flow | _____ | _____ |
- H. Were any warning lights illuminated _____
- I. What was the extent of the crash fire _____
- J. What was the weather conditions
- (1) Temperature _____
 - (2) Wind Velocity _____
 - (3) Freezing level _____
 - (4) Visibility _____
 - (5) Turbulence _____
 - (6) Dew Point _____
 - (7) Wind Direction _____
- K. What was the type of mission _____
- L. Were there radio transmissions indicating a difficulty being encountered
- M. What was the power setting
- N. Were there throttle movement
- O. Are engine records available
- P. Review the records for the following:
- | | LH | RH |
|------------------------|-------|-------|
| Engine's serial number | _____ | _____ |

LH

RH

ABC serial number and part number _____
ABP serial number and part number _____
MFC serial number and part number _____
MFP serial number and part number _____
T₅ Amplifier serial number and part number _____
Engine EGT _____
Engine Time since P.E. _____
Engine Time since O/H _____
Recent Maintenance _____
Outstanding TCTO's _____ - _____
R. Are last test cell run sheets available _____
S. What is the base maintenance capability _____
T. Have fuel and oil samples been taken _____
U. Are plastic bags and tags available for exhibits _____

APPENDIX B

THINGS TO DO AND ITEMS TO BE CHECKED AT THE CRASH SITE

Assist in making a chart or diagram of where parts are found at crash site. Identify parts and record serial numbers. All components and parts should be found if at all possible.

- A. Record the throttle Settings. LH _____ RH _____
- B. Are the engines aligned with the intake ducts. LH _____ RH _____
- C. Are there any indications of mounting failures. LH _____ RH _____
- D. Record the variable exhaust nozzle positions LH _____ RH _____
- E. Is there any foreign material in the afterburner LH _____ RH _____
- F. Record the distance from the variable stator actuator body to the retract stop LH _____ RH _____
- G. Are the throttle linkages connected. LH _____ RH _____
- H. Record the variable stator positions (VPI). LH _____ RH _____
- I. Record the variable vane feedback linkage extension with respect to swivel assembly. LH _____ RH _____
- J. Record the exhaust nozzle diameter LH _____ RH _____
- K. Record distance from actuator body to the stops. LH _____ RH _____
- l. Is there evidence of failure in the fuel or oil lines. LH _____
RH _____
- M. Are there indications of fire. LH _____ RH _____
- N. Record the A₈ feed back mechanism position LH _____ RH _____
- O. Obtain fuel and oil samples. A minimum of 1 gallon of jet fuel and a quart of lubrication oil is desirable. If lesser amounts are all that are obtainable, they should be extracted for analysis.
- P. The fuel and oil filters should be removed and placed in a clean container for future analysis.
- O. Check the filaments of the panel lights to determine whether they were burning at the time of the crash. (If filaments were hot they would be stretched. If the filaments were cold, they would be shattered.)

R. Record the following instrumentation readings and switch positions in the cockpit.

A. RPM. LH _____ RH _____

B. EGT. LH _____ RH _____

C. NPT. LH _____ RH _____

D. Oil Pressure. LH _____ RH _____

E. Fuel quantity _____

F. Fuel Flow. LH _____ RH _____

G. Anti-Icing switch positions. LH _____ RH _____

H. Circuit breaker positions _____

I. Boost pump switches positions _____

NOTE: While it is recommended that the engine instrument gage readings be observed and recorded, their indications should not be used to determine operating conditions of the engine at time of impact. The mechanical construction of the gages is such that their indicators could deflect in either direction upon impact of the aircraft. Therefore, their indications should be used to verify the engine's operating conditions as determined by other means outlined in this manual. Often the gage reading at time of impact can be determined by the use of an ultraviolet light examination of the gage's face and/or its glass.

APPENDIX C

ITEMS TO BE CHECKED DURING DISASSEMBLY IN THE ENGINE SHOP

A. General

1. Inspect engines for broken mounts for the accessories which might indicate high vibration.
2. Conduct a complete rigging check,
3. Record the variable vane positions.
4. Inspect the 1st stage compressor blades for evidence of contacting the inlet guide vanes.
5. Inspect the inlet guide vanes for missing vanes.
6. Inspect the condition of the master rod.
7. Inspect the bullet nose for general condition.
8. Inspect for engine case penetration or burn through.

B. Front Frame

1. Inspect the condition of the number one bearing.
2. Inspect the condition of the number one bearing scavenge tube.
3. Inspect the condition of the number one bearing oil supply tube.
4. Inspect the number one bearing cavity for evidence of oil.
5. Inspect the front frame for general condition and security.
6. Inspect the lubrication nozzles for clogging.

C. Compressor

1. Record the position of the variable vanes.
2. Inspect compressor section for evidence of foreign object damage.
3. Inspect the condition of the compressor rotor.
4. Inspect the stator vanes for possible backward installation.
5. Inspect the condition of the stator vanes.
6. Inspect the stator vanes for evidence of rotation within the compressor case.
7. Record amount and kind of foreign material in the compressor.
8. Inspect the number one bearing area for oil leaks.
9. Inspect the compressor rotor for evidence of shifting.
10. Inspect compressor blades and case for evidence of rubbing.
11. Inspect general condition of compressor casing.
12. Have the compressor casing checked for runout.
13. Inspect the condition of the compressor rotor shaft.

D. Combustor

1. Inspect for evidence of hot spots or streaks.
2. Inspect for missing pieces from the combustors.
3. Inspect for overall condition of the combustors.
4. Record amount and type debris in the combustors.
5. Inspect the combustors for evidence of fusion and metal spray.
6. Inspect the combustors for proper assembly.
7. Inspect the fuel nozzles for broken and missing pieces.
8. Flow check the fuel nozzles.

E. Mainframe.

1. Inspect the 17th stage poppet valve.
2. Inspect the condition of the 17th stage air seal.
3. Inspect the struts for cracks or breaks.
4. Inspect the general condition of the mainframe.
5. Inspect the number two bearing for failure.
6. Inspect the number two bearing lubrication.
7. Inspect the number two bearing lube nozzle for clogging.
8. Inspect the condition of the 17th stage exist guide vanes.

F. Turbine Section.

1. Inspect for evidence of hot streaks.
2. Record amount and type of debris in the cooling air passages.
3. Inspect for evidence of overheating.
4. Inspect for evidence of foreign object damage.
5. Inspect the number three bearing for failure.
6. Inspect the lubrication nozzles for clogging.
7. Inspect for oil leakage around the number three bearing.
8. Inspect the turbine blades for evidence of rub.
9. Record the amount and type of debris in the turbine.
10. Inspect the turbines for evidence of metal fusion or metal spray.
11. Inspect the overall condition of the turbine casing.
12. Inspect the interstage seals for general condition.
13. Inspect the torque ring for general condition.
14. Inspect the condition of the turbine nozzles.

G. Afterburner Section.

1. Inspect for evidence of hot spots or streaks.
2. Inspect the overall condition of the A/B liner.
3. Inspect the condition of the A/B Housing.
4. Inspect for evidence of fusion or metal spray in the A/B.

H. Variable Exhaust Nozzle.

1. Inspect the overall condition of the nozzle housing.
2. Determine the variable exhaust nozzle area by the following methods: (See Appendix i).

- a. Actuators.
- b. Measured.
- c. A_8 feed back cable.
- d. Nozzle position transmitter.
3. Check the actuators for binding.
4. Inspect the overall condition of the nozzle actuator ring.
5. Inspect the condition of the VEN leaves.
6. Inspect the VEN for evidence of fusion or metal spray.
7. Inspect the condition of the VEN rollers.

I. Diffuser Section.

1. Inspect the general condition of the diffuser section.
2. Bench test the pilot and main spraybars for correct flow rates.
3. Inspect the condition of the flameholder.
4. Inspect the evidence of fusion or metal spray.
5. Check the A/B igniter on the bench.
6. Check the A/B igniter for proper immersion depth.

J. Accessories

1. Gear Boxes.
 - A. Inspect the general condition of gear boxes (front, transfer and rear.).
 - B. Inspect the condition of splines (both male and female).
 1. CSD
 2. MFP
 3. ABP
 4. Hydraulic Pumps.
 5. High speed generator.

6. Starter
7. Tach generator.
8. Control alternator.
9. MFC
10. Main lube pump.
11. Nozzle actuator pump.
12. Scavenge pump.

C. Determine if the spline drives will rotate by applying torque to the radial drive shaft.

D. Inspect for evidence of lubrication leaks.

NOTE; See Figures D-1, D-2, and D-3 for diagrams of the various pads and RPMs.

K. Main Fuel Control

1. Inspect the control's general condition.
2. Inspect the condition of the mating fuel control gear.
3. Check for evidence of external leakage.
4. Check throttle to determine if shaft is sheared.
5. Record the throttle setting.
6. Record the fuel density settings.
7. Inspect the condition of the T₂ aspirator tube.

L. Main Fuel Pump.

1. Inspect for evidence of improper assembly or leakage between aircraft and engine inlet fuel manifold and the main pump.
2. Inspect the impeller shaft to see if it had sheared.
3. Inspect for evidence of sheared metal in the impeller housing.
4. Inspect the main pump for proper assembly.
5. Record the pump's break away torque.
6. Check to determine whether the impeller turns when drive shaft is turned.
7. Check condition of fuel filter and retain any foreign objects.

M. Afterburner Fuel Control

1. Inspect the control's general condition.
2. Record the throttle setting.
3. Inspect for evidence of external fuel leakage.
4. Inspect for evidence of water in the control.
5. Record the fuel density settings.

6. Inspect to determine if the P_3 lines to main and A/B controls are in tact.

N. Afterburner Pump.

1. Inspect the condition of the afterburner pump male splines and mating internal female spline.
2. Check the ABP drive shaft to determine if it turns freely.
3. Inspect the condition of the "O" rings between the ABP and ABC.

O..Lubrication Pump and Tank.

1. Determine whether the drive shaft for the pump is free to rotate.
2. Determine whehter the pendulum in the tank is free to rotate.
3. Determine whether there is evidence that the pendulum contacted the tank,
4. Determine whether the pump operates correctly.
5. Inspect the lube tank for evidence of bulging.

P. Lubrication Cooler.

1. Inspect the general condition of the cooler.
2. Determine whehter there is evidence of fuel oil contamination.

Q. Oil Pressure Transmitter.

1. Determine whether the transmitter operates correctly.

R. Electrical

1. Inspect the general condition of the alternator tachometer generator.
2. Inspect the condition of the connectors and pins.
3. Determine whether the drive shaft of the tach generator is free to rotate.
4. Subject the alternator tachometer generator to a functional bench test.

S. T_5 Amplifier.

1. Inspect the general condition of the T_5 amplifier.
2. Inspect the condition of the T_5 amplifier mountings.
3. Inspect the condition of the connectors and pins.
4. Subject the T_5 amplifier to the system functional test.
5. Subject the T_5 amplifier to the T_5 bench test.

T. Anti-Icing Valve.

1. Inspect the general condition of the anti-icing valve.

2. Subject the valve to the operational bench test.
 3. Inspect the condition of the electrical connections.
- U. Ignition Exciters
1. Inspect the general condition of the ignition exciters.
 2. Subject the ignition exciter to a functional bench test.
- V. Ignition Igniters and Leads.
1. Inspect the general condition of the ignition igniters and leads.
 2. Subject the ignition igniters and leads to a functional bench test.
 3. Check the immersion of the igniter.
- W. Junction Box and Harness Assembly.
1. Inspect the general condition of the junction box and harness assembly.
 2. Subject the junction box and harness assembly to a functional bench test.
- X. Exhaust Gas Thermocouples and Leads.
1. Inspect the general condition of the exhaust gas thermocouples and leads.
 2. Subject the parts to a functional check prior to removal from the engine.
 3. Subject the parts to a functional bench check in accordance with applicable T.O.s.
- Y. Nozzle Position Transmitter
1. Inspect the general condition of this part.
 2. Subject the unit to the functional bench test.
- Z. Inlet Guide Vane Actuators (Fuel System).
1. Inspect the general condition of this part.
 2. Subject this part to a functional bench test.
 3. Determine the IGV Position as determined by internal impact marks.
- AA. Drain Valves.
1. Inspect the general condition of the pressurizing and drain valves and A/B drain valves.
- BB. Fuel Flow Transmitter
1. Inspect this part for general condition.
 2. Subject this part to a functional bench test.

3. Inspect the condition of mating surfaces between the transmitter and mounting brackets.

CC. T₂ Sensor.

1. Inspect the general condition of this part.
2. Subject this unit to a functional test.

DD. Variable Exhaust Nozzle Actuators.

1. Inspect the general condition of these parts.
2. Subject the actuators to a functional bench test.

APPENDIX D
METALLIC COMPOSITION OF ENGINE OIL WETTED COMPONENTS

Item No.	Silver Ag	Aluminum Al	Iron Fe	Chromium Cr	Copper Cu	Tin Sn	Magnesium Mg	Lead Pb	Nickel Ni	Silicon Si
1. Constant Speed Drive Components Upstream of Transfer Gearbox Scavenge Pump Inlet Screen		X	M		M		M			
2. Front Gearbox Front and Transfer Gearbox Bearings	X		M	X	M					X
Gearbox Castings		M					M			
Gears, Shafts, Splines			M	X					X	
3. No. 1 Bearing Sump Area No. 1 Bearing Cages	X		X		X					X
No. 1 Bearing Rollers and Races			M	X					X	
No. 1 Bearing Seal Race			X	M					X	
Carbon Seal Runners			M	X					X	
Bearing Housings			M						X	
4. Horizontal Shaft Support Bearing			M							
Bearing Housing		M	M				M			
Bearings	X			X	M					X
Shafts and Splines				X					X	
5. Afterburner Fuel Pump (Gear and Bearing)			M	X						
Components Upstream of Rear Gearbox Scavenge Pump Inlet Screen										
6. Transfer Gearbox Gearbox Bearings	X		M	X	M					X
Gearbox Castings		M					M			

METALLIC COMPOSITION OF ENGINE OIL WETTED COMPONENTS (Cont.)

Item No.	Silver Ag	Aluminum Al	Iron Fe	Chromium Cr	Copper Cu	Tin Sn	Magnesium Mg	Lead Pb	Nickel Ni	Silicon Si
Gears, Shafts, Splines			M	X					X	
7. No. 2 Bearing Area Sump										
No. 2 Bearing Cages	X		X		M					X
No. 2 Bearing Rollers and Races			M	X						
Bearing Housings			M						X	
Carbon Seal Runners			M	X					X	
Components Upstream of No. 3 Bearing Scavenge Pump Inlet Screen										
8. No. 3 Bearing Sump Area										
No. 3 Bearing Cages	X		X		M					X
No. 3 Bearing Roller and Cages		M	M	X						
No. 3 Bearing Seal Races			X	M					X	
Carbon Seal Runners			M	X					X	
Bearing Housings			M						X	
Components Upstream of Main Lube and Hydraulic Filter										
9. Main Lube and Hydraulic Pump										
(Gear and Bearing)		X	M		M					X
Components Upstream of Scavenge Oil Filter (Plus Item No. 2 thru Item No. 9)										
10. Transfer Gearbox Scavenge Pump		X	M		M				X	
11. Rear Gearbox Scavenge Pump		X	M		M				X	
12. No. 3 Bearing Scavenge Pump		X	M		M				X	
13. Variable Nozzle Pump	M*	X	M	X	M	X		X	X	X
14. Variable Nozzle Actuator			M	X	X				X	

METALLIC COMPOSITION OF ENGINE OIL WETTED COMPONENTS (Cont.)

Item No.	Silver Ag	Aluminum Al	Iron Fe	Chromium Cr	Copper Cu	Tin Sn	Magnesium Mg	Lead Pb	Nickel Ni	Silicon Si
15. Rear Gearbox										
Gearbox Bearings	X		M	X	M					X
Gearbox Castings								M		
Gearbox Bearing Housing			M						X	
Gears, Shafts, and Splines			M	X					X	

M = Major Elements Present

X = Minor Elements Present

*** = Only engines which have -1P, 1M or -1N model pumps with silver plated pistons**

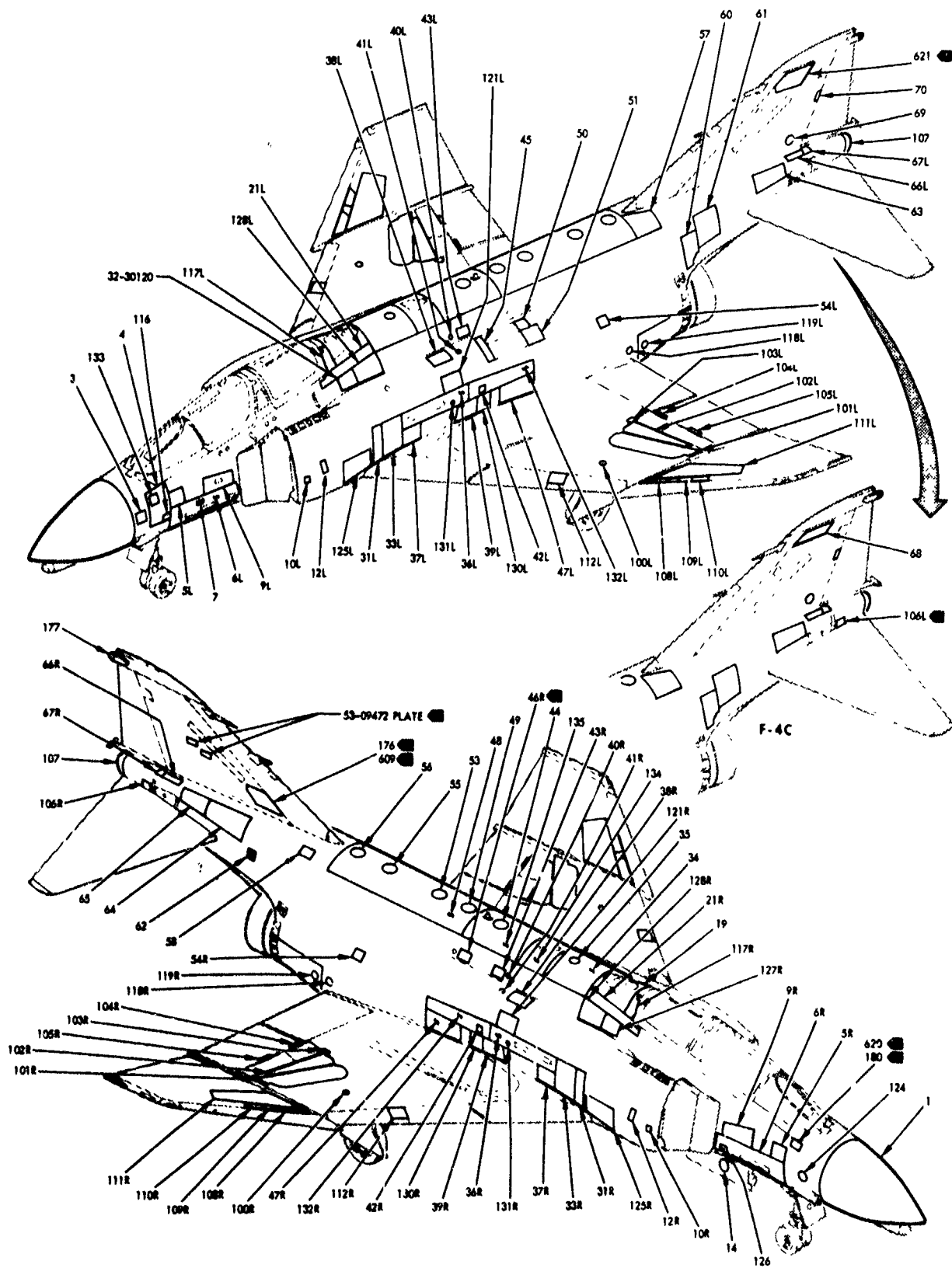
The above listings do not represent all the elements in each component; however, they do represent the significant elements of those components which are expected to wear.

APPENDIX E

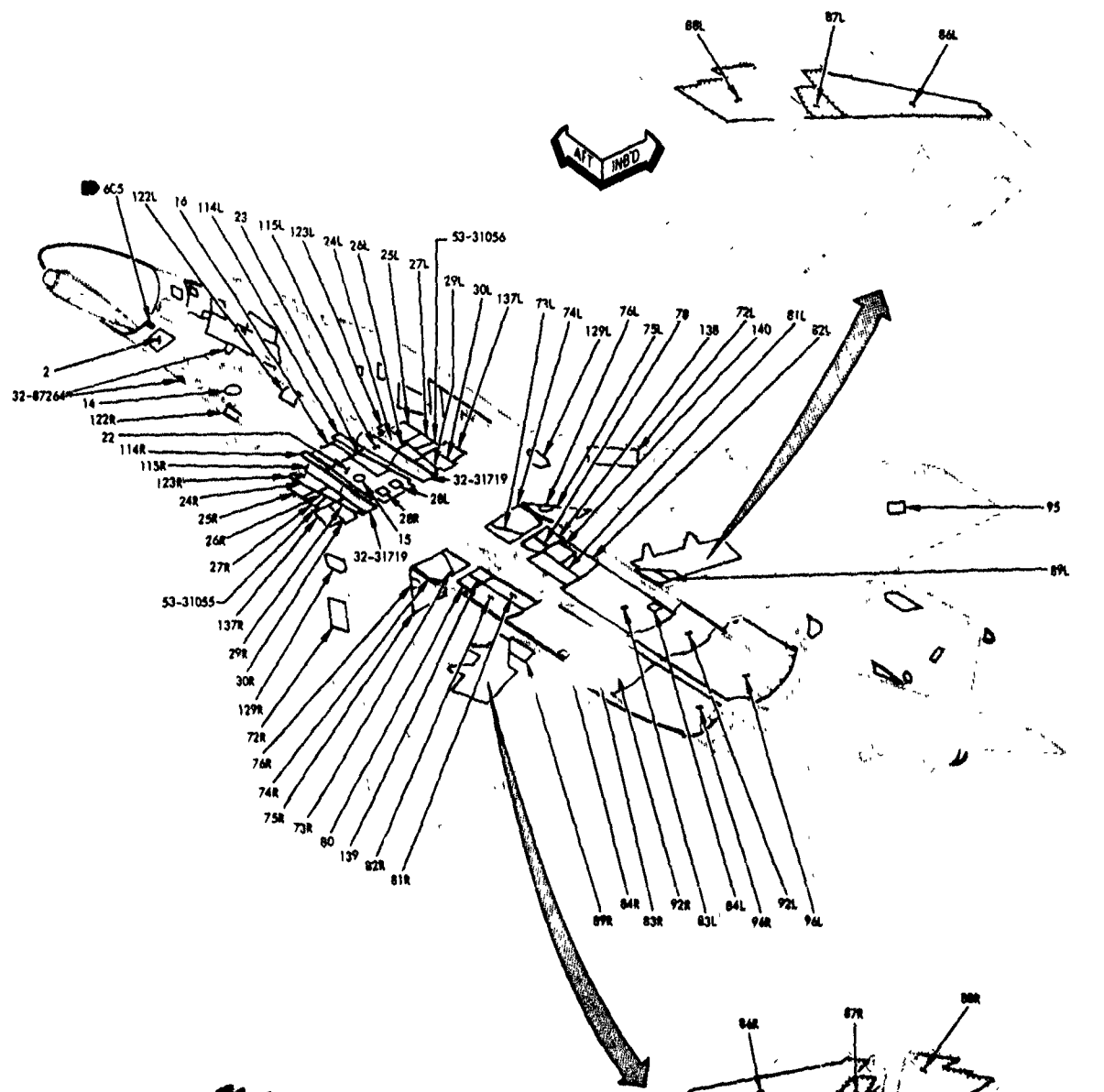
MISCELLANEOUS AIRCRAFT AND ENGINE INFORMATION.

This appendix was prepared to provide the propulsion accident investigator with a ready reference to information that could be useful when he writes his report. Listed below is a tabulation of the information provided:

- E-1 Upper External Access Openings for F-4C/D Aircraft
- E-2 Lower External Access Openings for F-4C/D Aircraft
- E-3 Upper External Access Openings for F-4E Aircraft (After Leading Edge Slats)
- E-4 Lower External Access Openings for F-4E Aircraft (After
E-4E Leading Edge Slats)
- E-5 External Access Openings for F-4E Aircraft (Before Leading Edge Slats)
- E-6 Upper External Access Openings for RF-4C Aircraft
- E-7 Lower External Access Openings for RF-4C Aircraft
- E-8 Station Diagram F-4C and F-4D Aircraft.
- E-9 Station Diagram RF-4C and F-4E Aircraft.
- E-10 Center Fuselage Structure.
- E-11 Aft Fuselage
- E-12 RF-4E Interior Arrangement.
- E-13 Engine Controls
- E-14 Engine Bleed Air and B.L.C. System
- E-15 Portion of Engine Bleed Air System Pressurized for All Flight Conditions
- E-16 Wire Bundle Splices and Disconnects
- E-17 F-4C Fuel System
- E-18 F-4D/RF-4C Fuel System
- E-19 F-4E Fuel System
- E-20 Front Gearbox Casing and Shaft Gear (J79-15/-17)
- E-21 Transfer Gearbox (J79-15/17)
- E-22 Rear Gearbox (J79-15/-17)
- E-23 F-4 Stall Characteristics

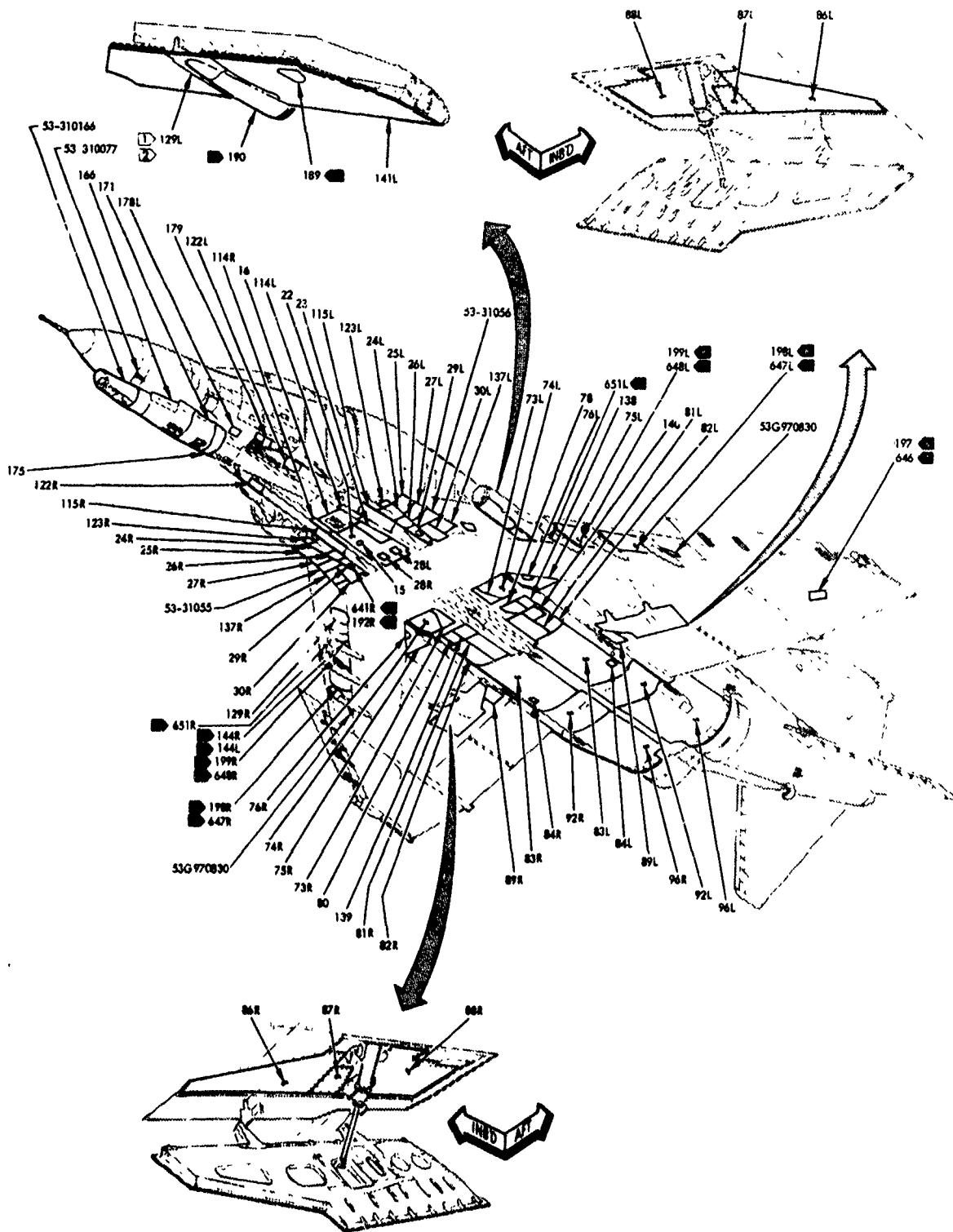


F-1 Upper External Access Openings for F-4C/D Aircraft



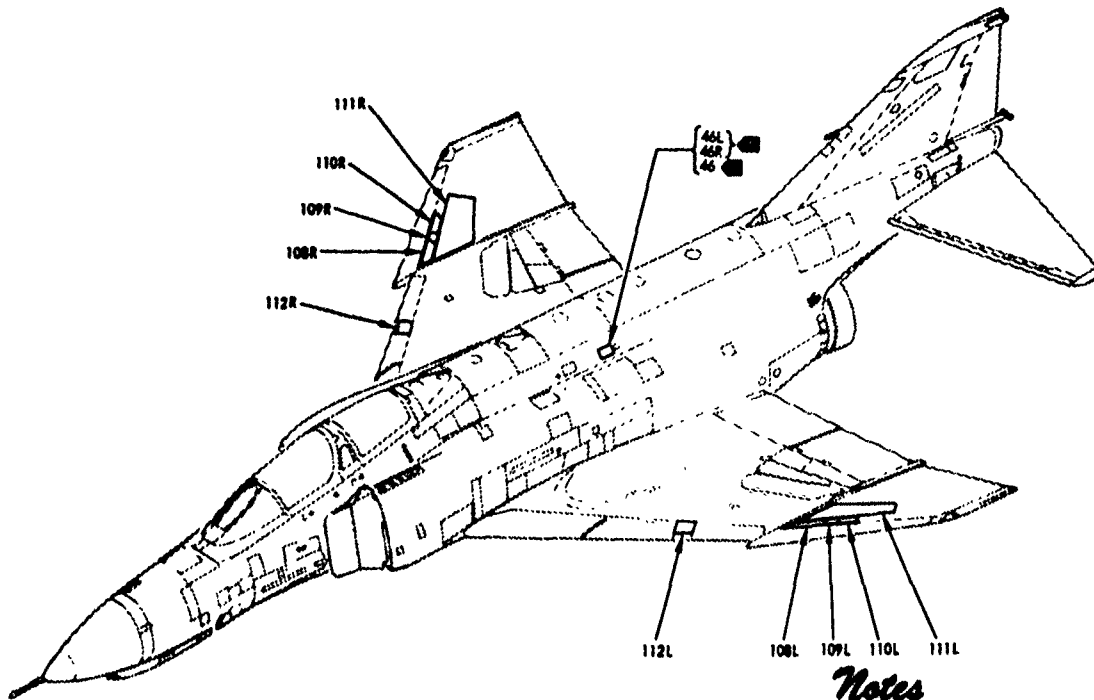
Notes

- F-4C 63 7407 THRU 64-687.
- F-4C 64-781, 64-787, 64-790, AND 64-815, AND F-4C 63-7407 THRU 64-780, 64-782 THRU 64-786, 64-788, 64-791 THRU 64-814, AND 64-816 THRU 64-928. ALSO F-4D 64-929 THRU 8499.
- F-4D 64-8499 THRU 64-8823.
- F-4D, RIGHT SIDE ONLY.
- F-4C.
- F-4D.
- F-4D 65-657 AND 65-66C AND F-4D 64-929 THRU 65-656, 65-658, 65-659, AND 65-661 THRU 66-8699.



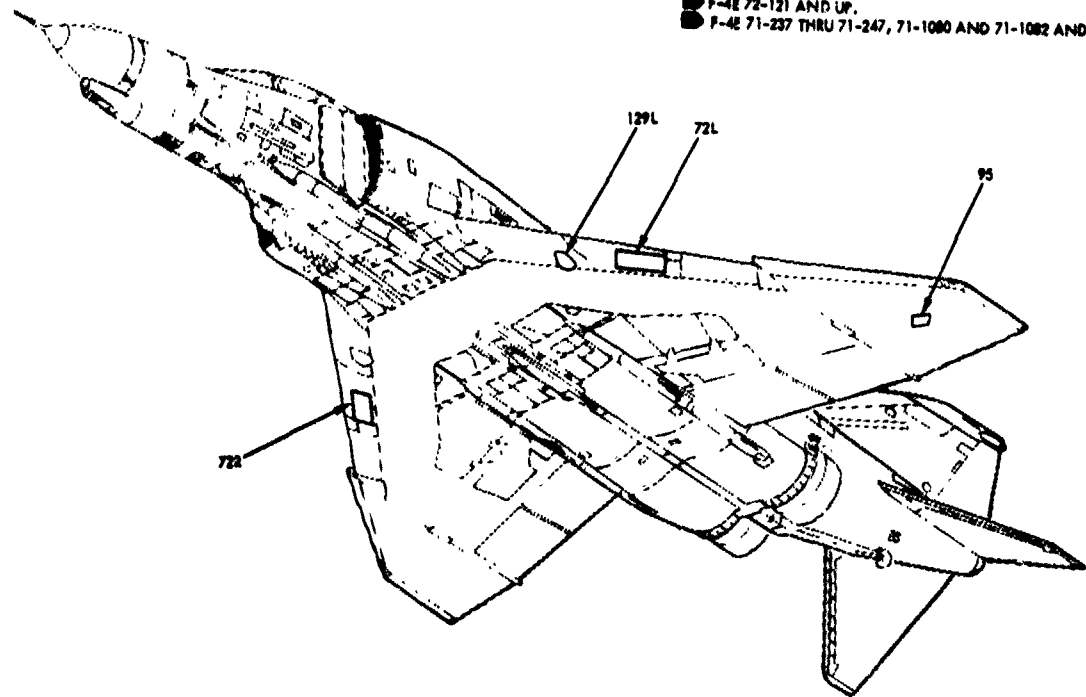
F-4E 72-121 AND UP, F-4E 66-287 AND 66-296 THRU 71-1402 AFTER T.O. 1F-4E-566

E-4 Lower External Access Openings for F-4E Aircraft (After Leading Edge Slats)
 -147-



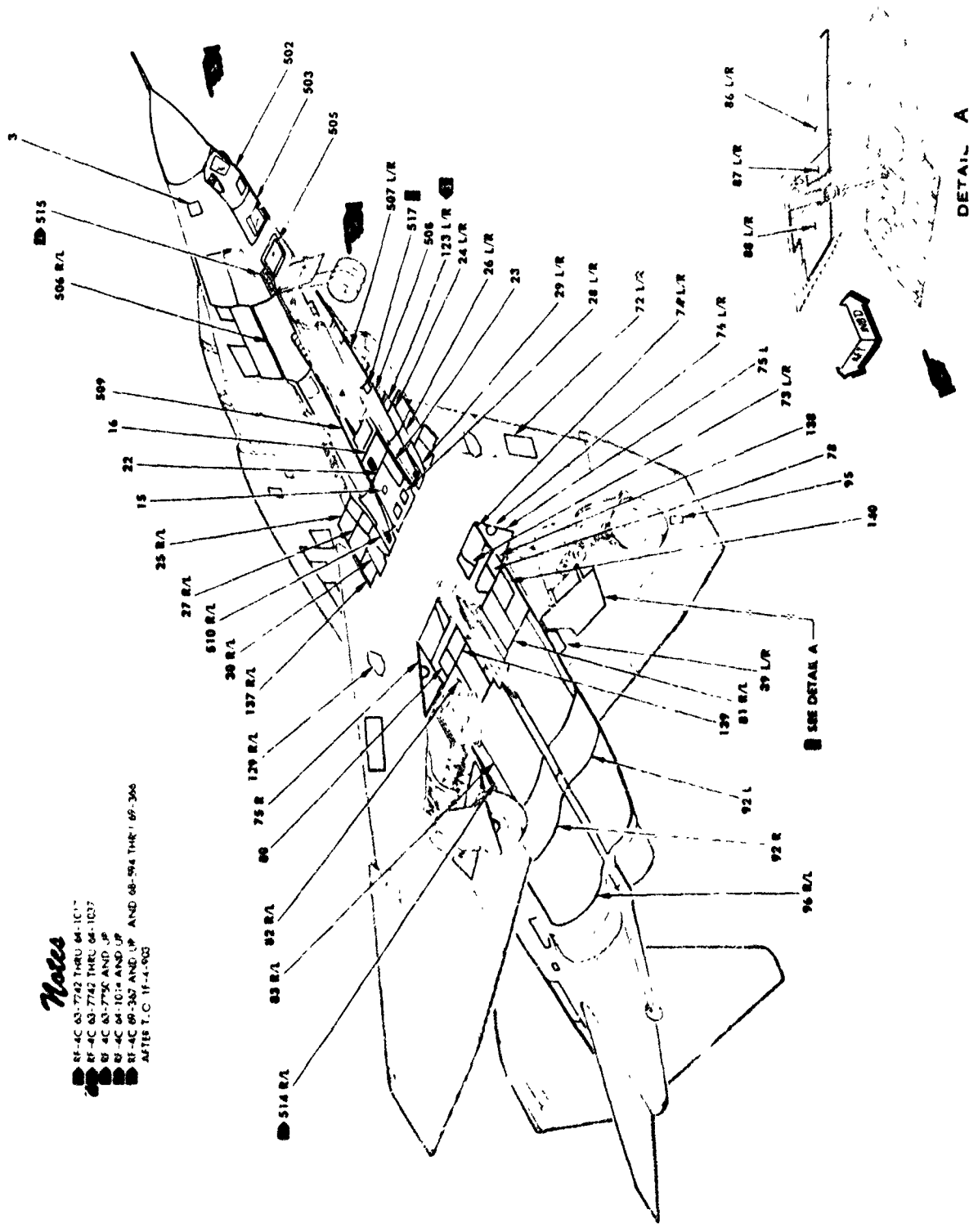
Notes

- ① S3-110110 DOOR IS INSTALLED WHEN PYLON IS NOT INSTALLED.
- ② S3-1-0114 DOOR IS INSTALLED WHEN PYLON IS INSTALLED.
- F-4E 66-268 AND UP.
- F-4E 66-284 THRU 66-367.
- F-4E 68-432 THRU 69-303 AFTER T.O. 1F-4E-903 AND F-4E 69-304 AND UP.
- DOOR 46 RIGHT SIDE ONLY F-4E 66-268 AND UP.
- F-4E 66-287 AND 66-296 THRU 71-1402 AFTER T.O. 1F-4E-566.
- F-4E 72-121 AND UP.
- F-4E 71-237 THRU 71-247, 71-1080 AND 71-1082 AND UP.



F-4E 66-284 THRU 66-286 AND 66-288 THRU 66-295;
 F-4E 66-287 AND 66-296 THRU 71-1402 BEFORE T.O. 1F-4E-566.

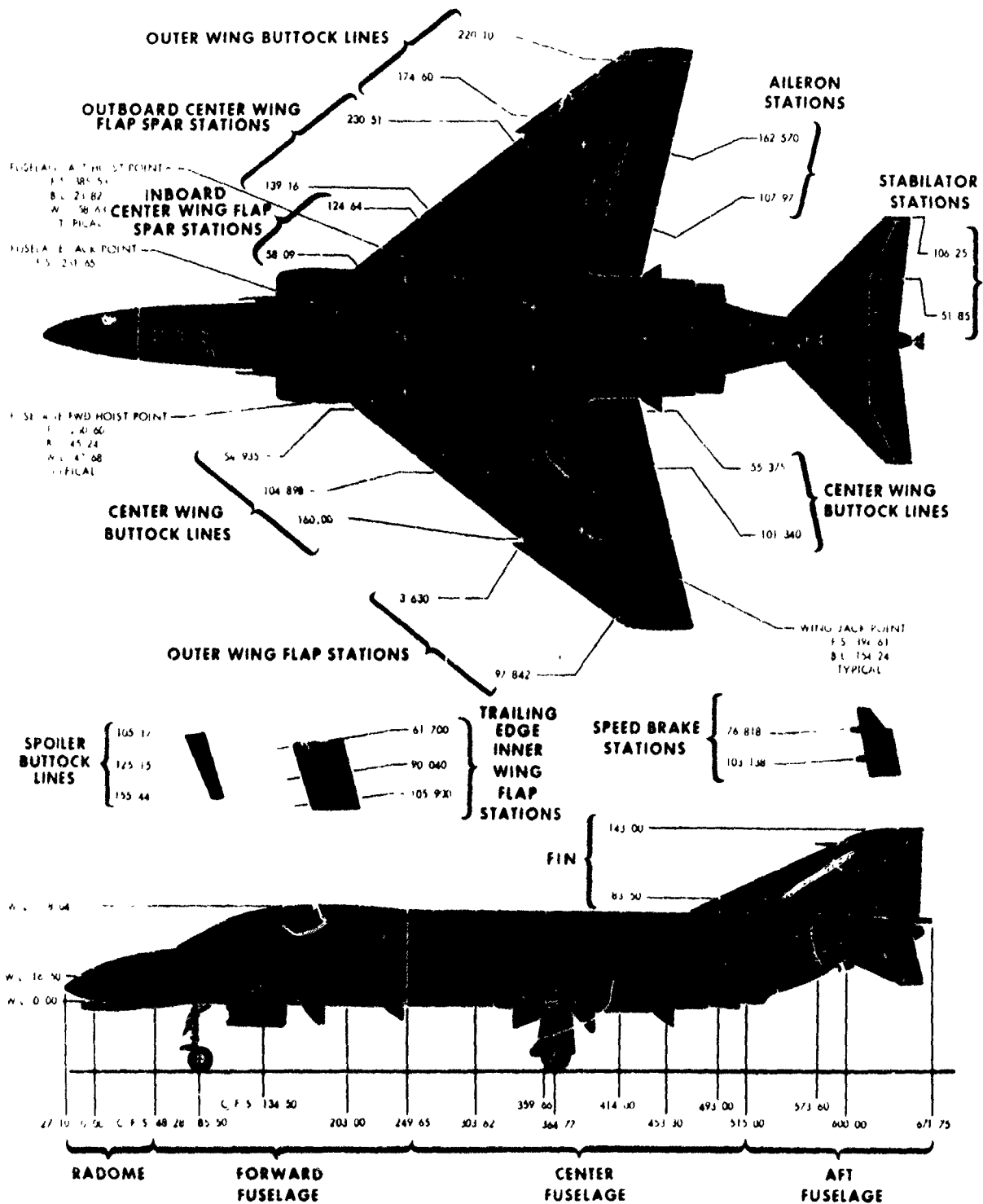
E-5 External Access Openings for F-4E Aircraft (Before Leading Edge Slats)



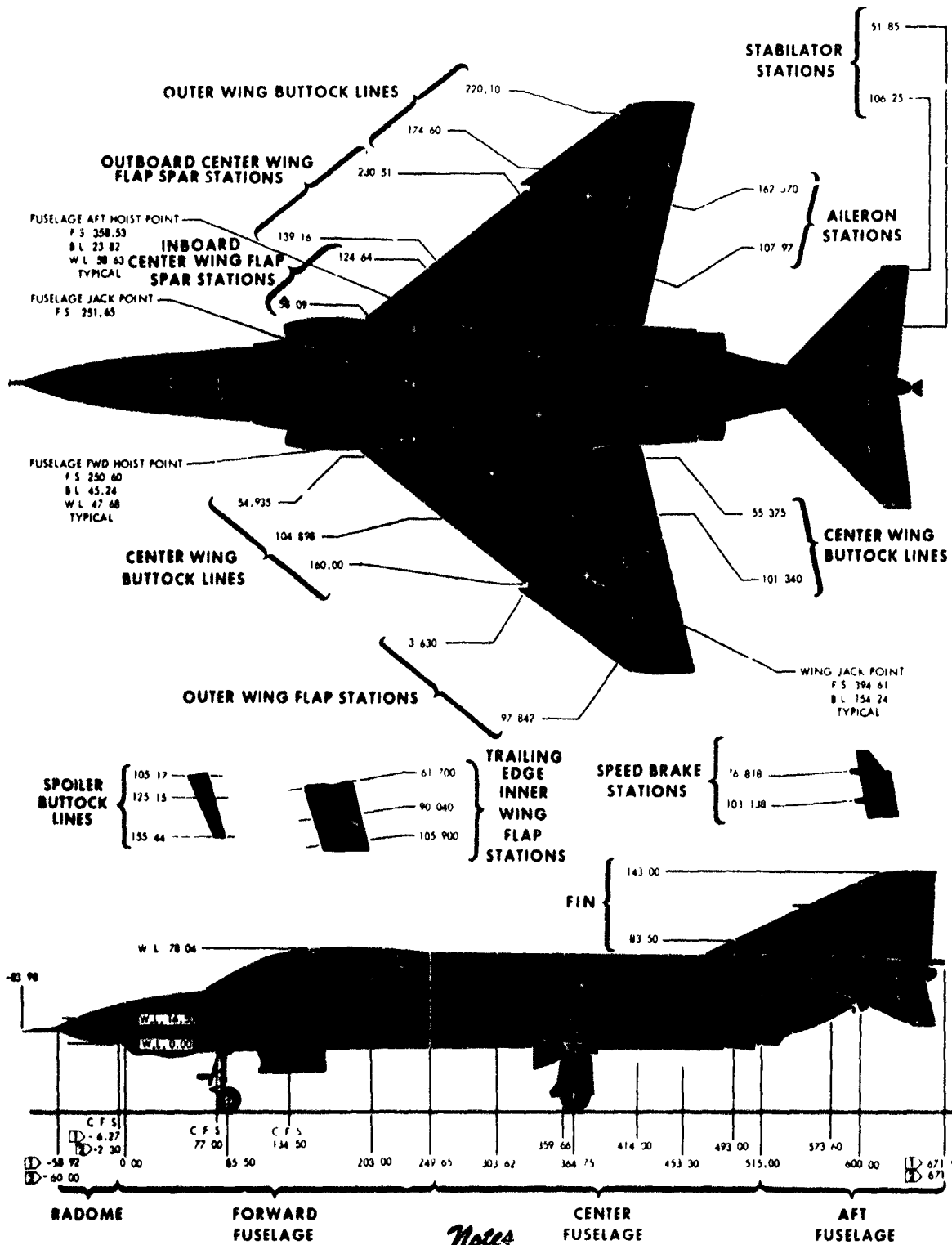
Notes

- RF-4C 63-7742 THRU 64-1017
- RF-4C 63-7742 THRU 64-1037
- RF-4C 63-775X AND UP
- RF-4C 64-1017 AND UP
- RF-4C 69-367 AND UP AND 68-594 THRU 69-366
- AFTER T.O. 1F-4-903

E-7 Lower External Access Openings for RF-4C Aircraft
-150



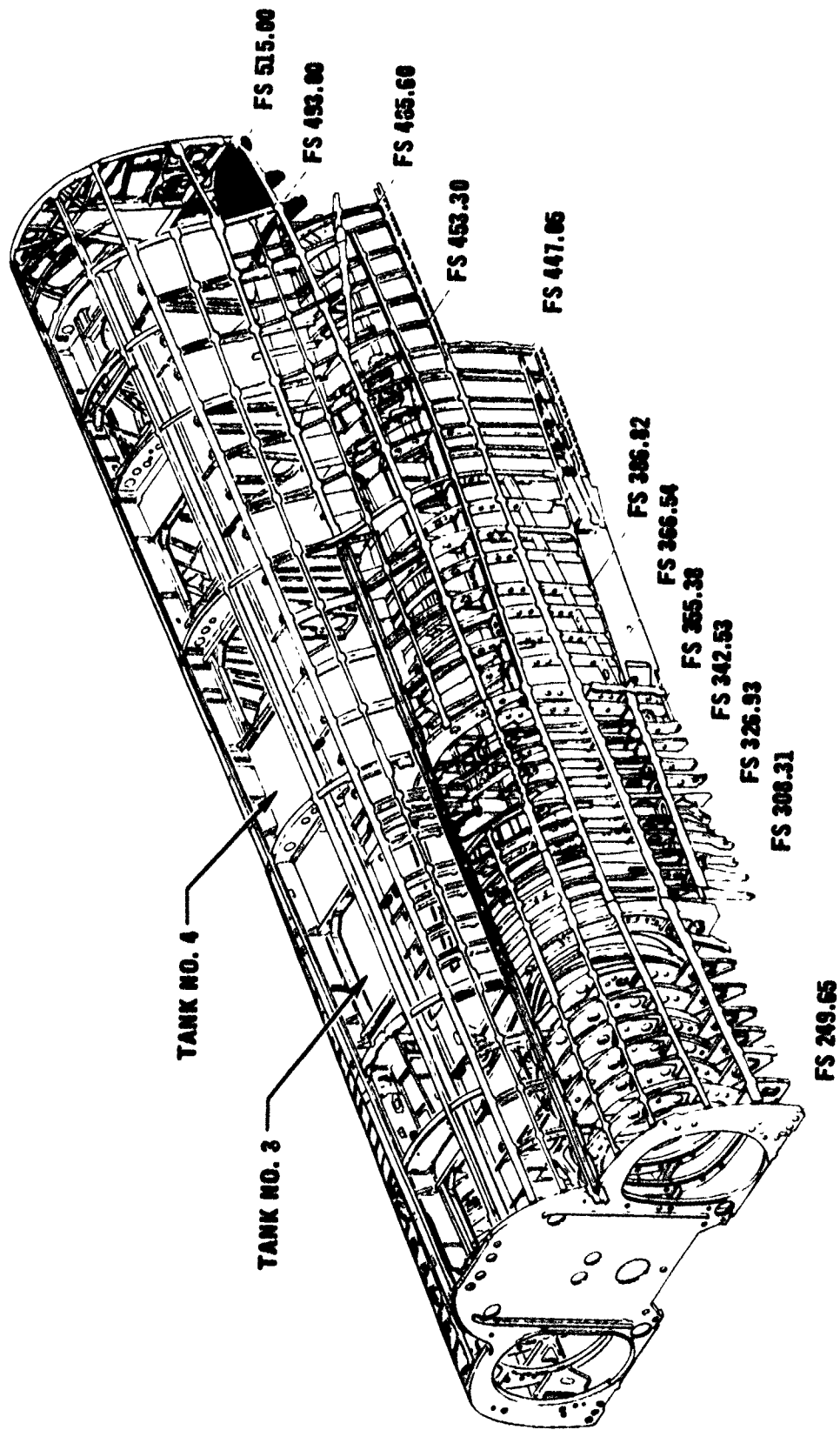
E-8 Station Diagram F-4C and F-4D Aircraft



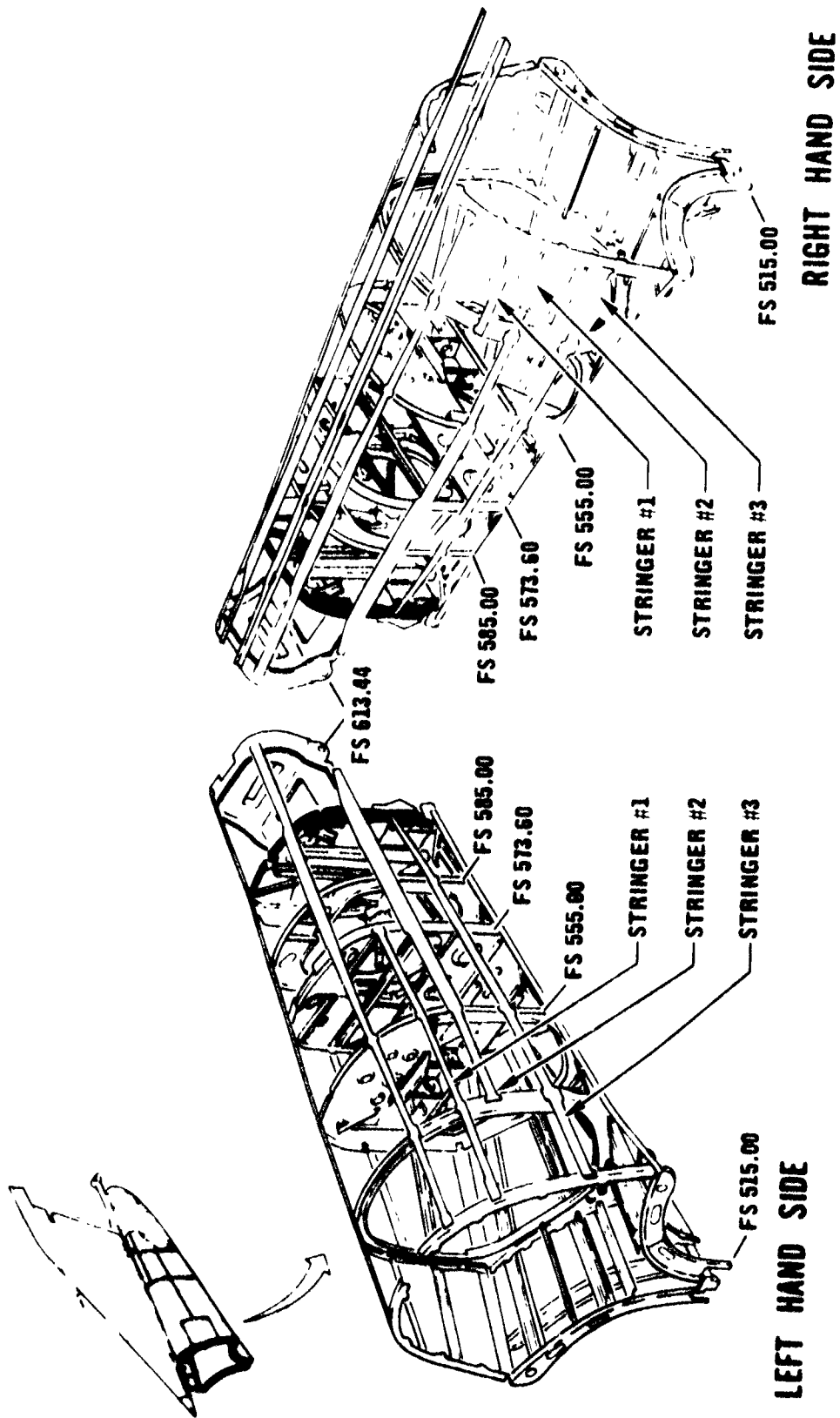
Notes

- Ⓛ RF-4C AIRCRAFT
- Ⓧ F-4E AIRCRAFT

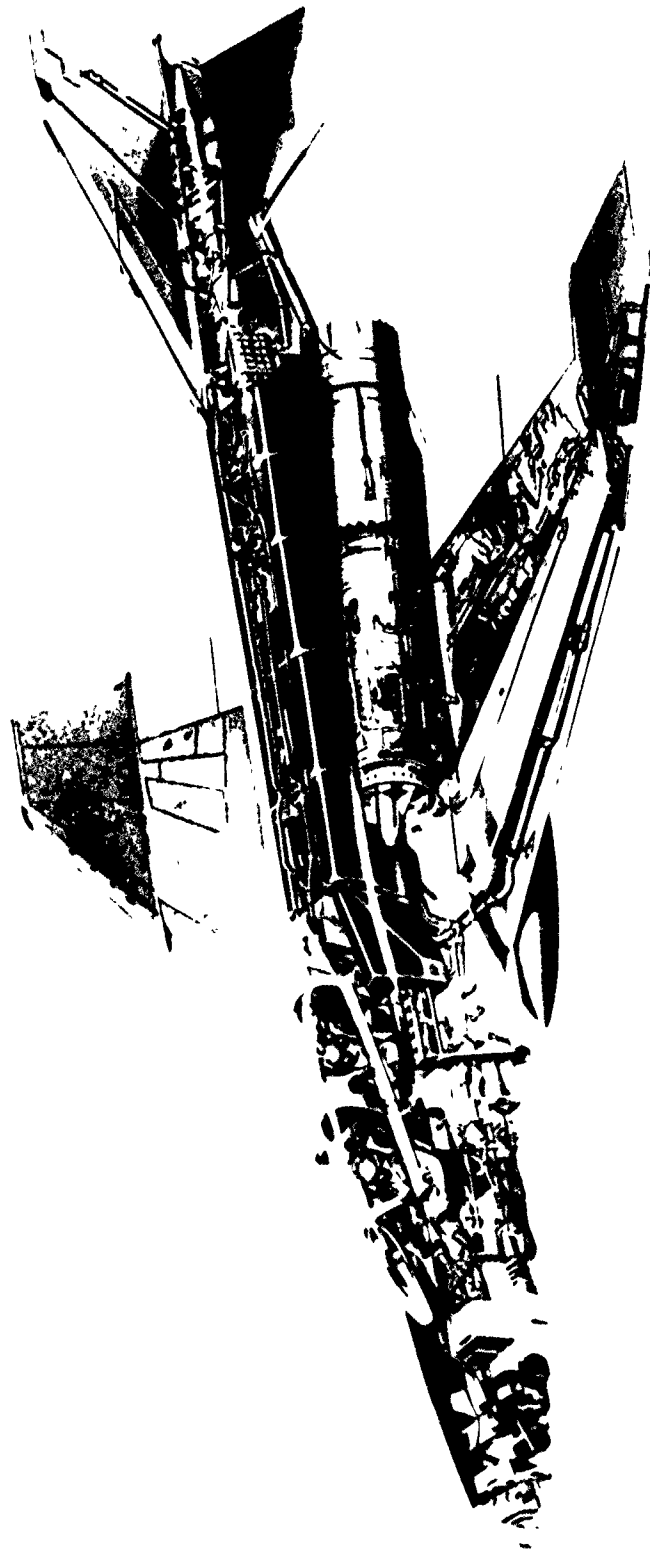
E-9 Station Diagram RF-4C and F-4E Aircraft



E-10 CENTER FUSELAGE STRUCTURE



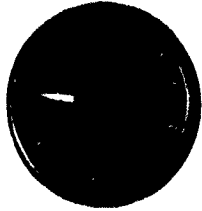
E-11 AFT FUSELAGE



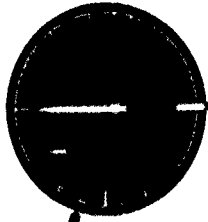
E-12 INTERIOR ARRANGEMENT

**FUEL FLOW INDICATOR
TYPICAL**

POUNDS PER HOUR
DOES NOT INCLUDE
FUEL FLOW TO
AFTERBURNERS



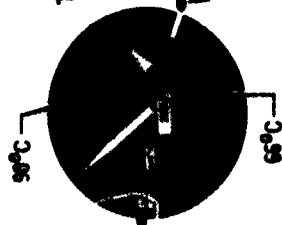
**TACHOMETER INDICATOR
TYPICAL**



102% RPM

GLASS SLIPPAGE INDEX

**EXHAUST GAS
TEMPERATURE INDICATOR
TYPICAL**

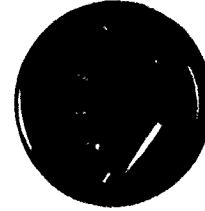


90°C

630°C

GLASS
SLIPPAGE
INDEX

**EXHAUST NOZZLE
POSITION INDICATOR
TYPICAL**



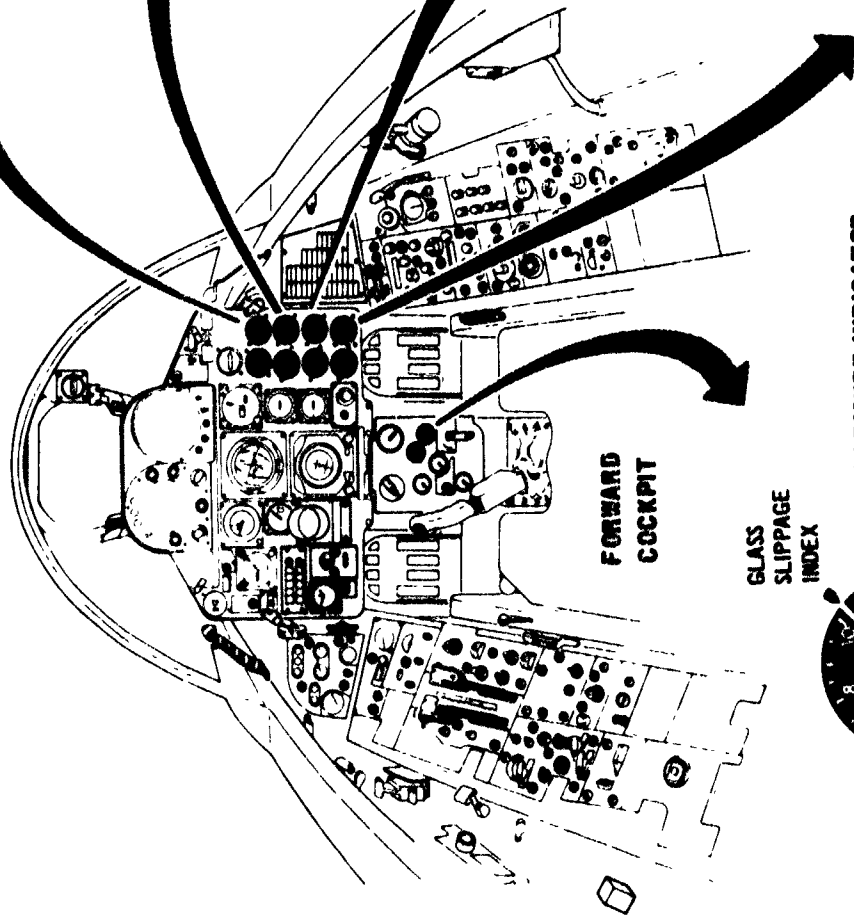
**OIL PRESSURE INDICATOR
TYPICAL**



GLASS
SLIPPAGE
INDEX

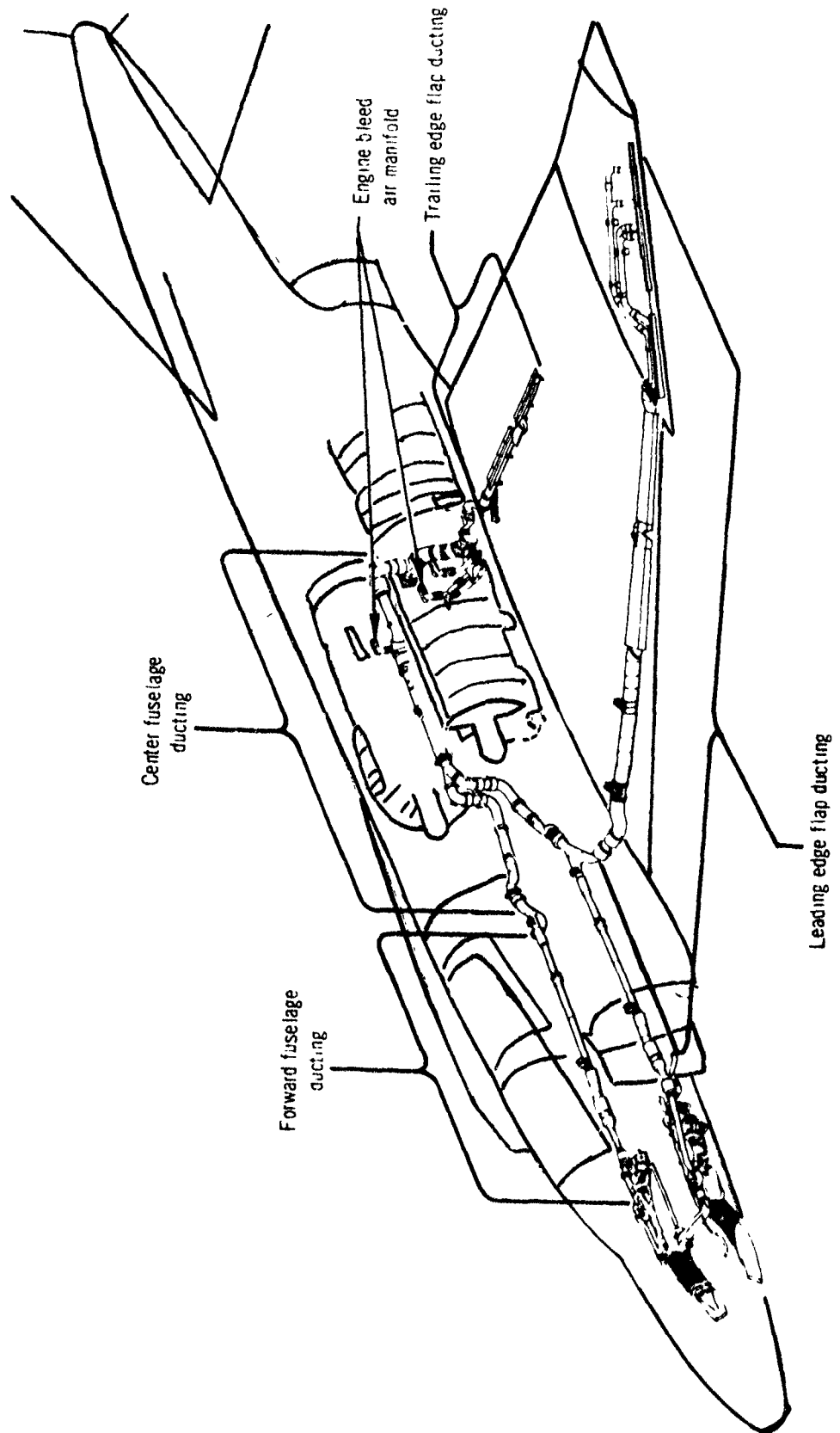
60 P.S.I.

12 P.S.I.

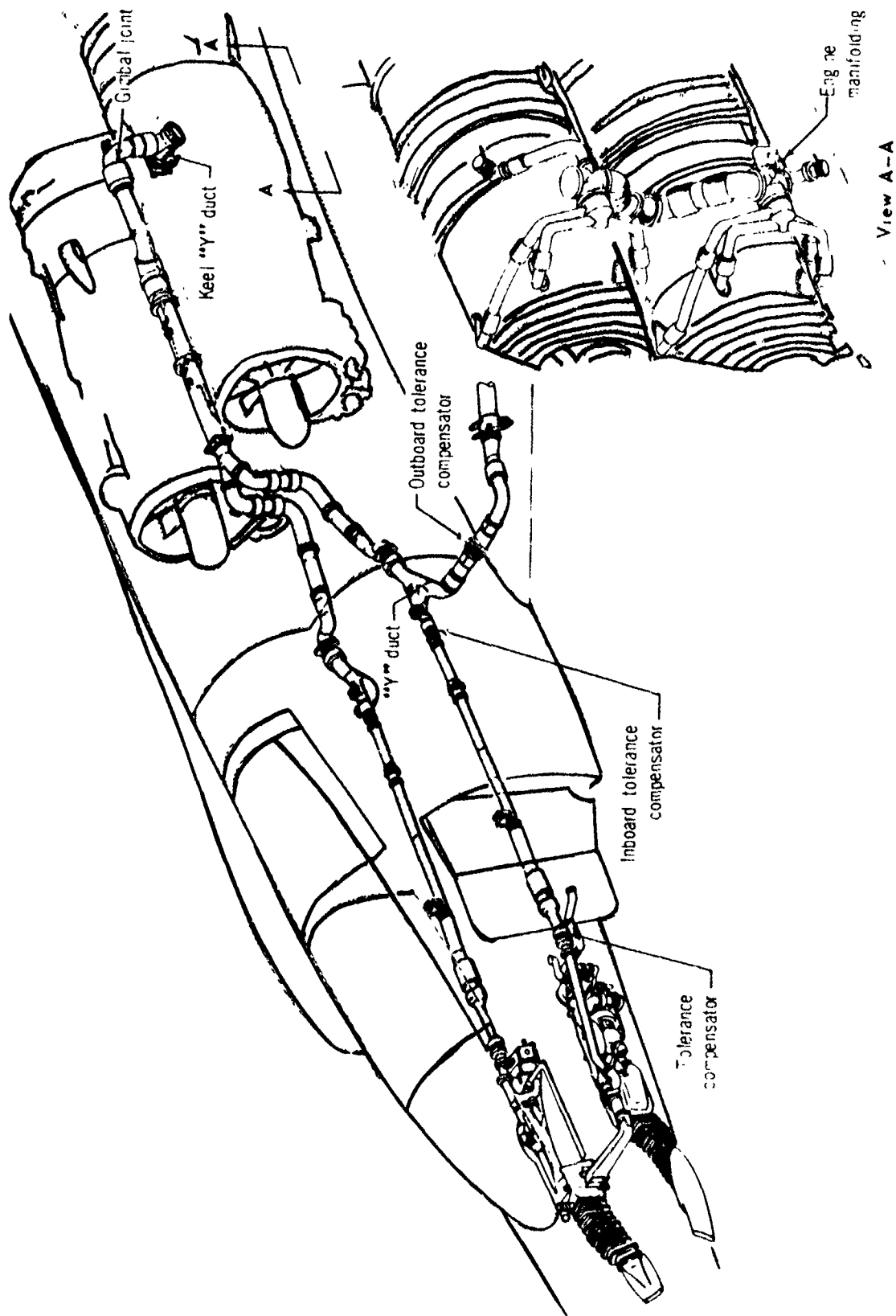


**FORWARD
COCKPIT**

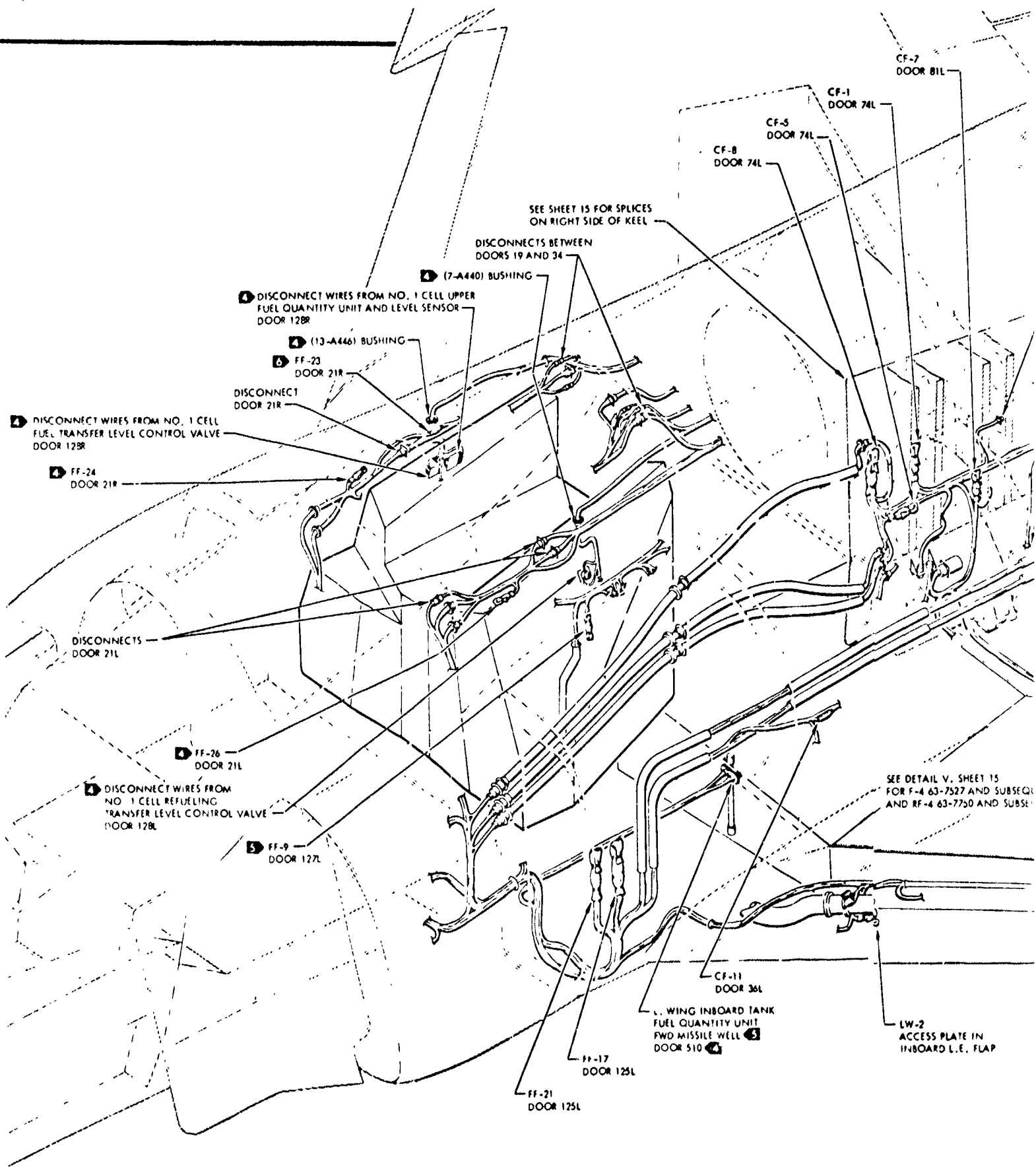
E-13 ENGINE CONTROLS



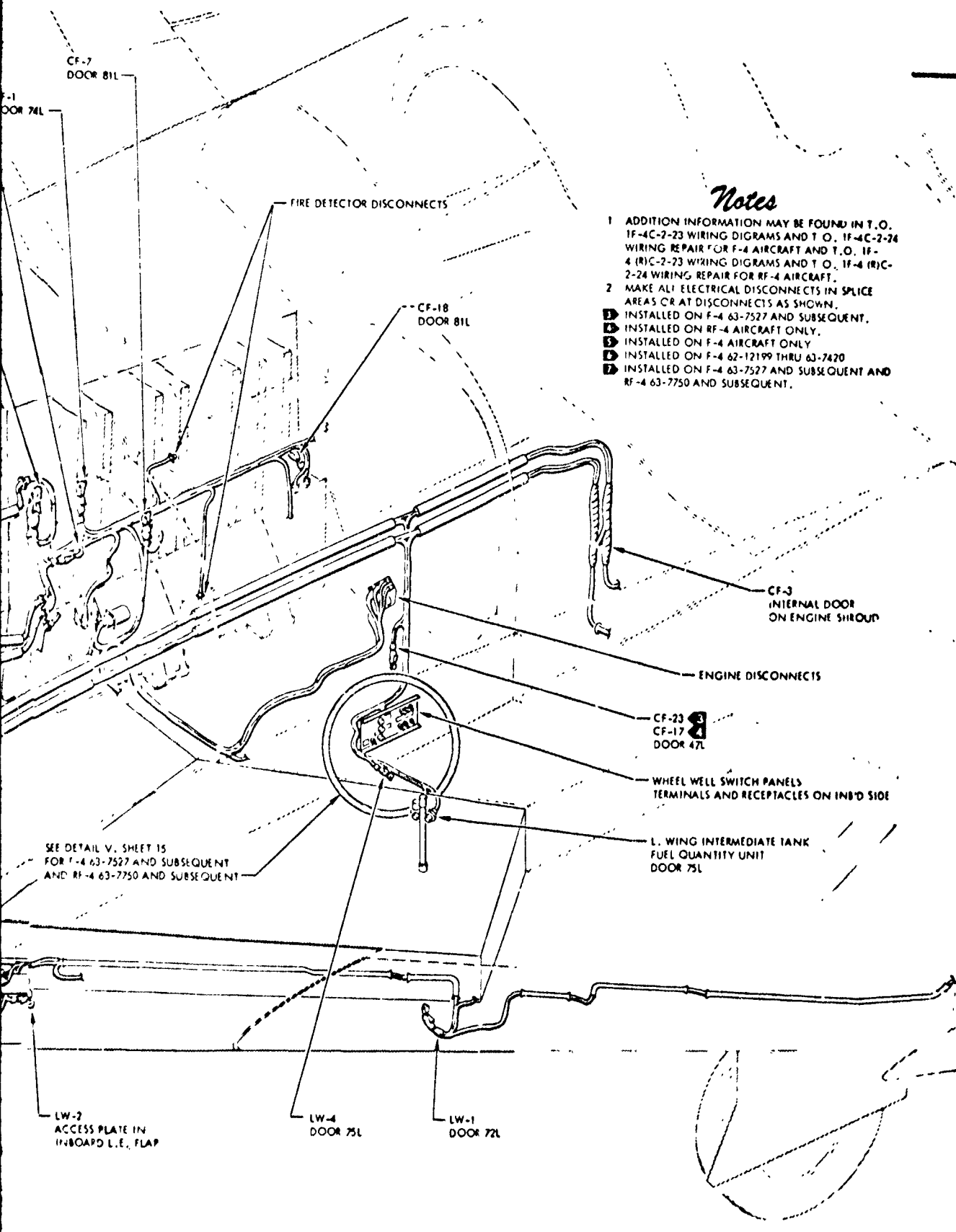
F-14 ENGINE BLEED AIR AND B.L.C. SYSTEM



E-15 PORTION OF ENGINE BLEED AIR SYSTEM PRESSURIZED FOR ALL FLIGHT CONDITIONS



E-16 WIRE BUNDLE SPLICES AND DISCONNECTS

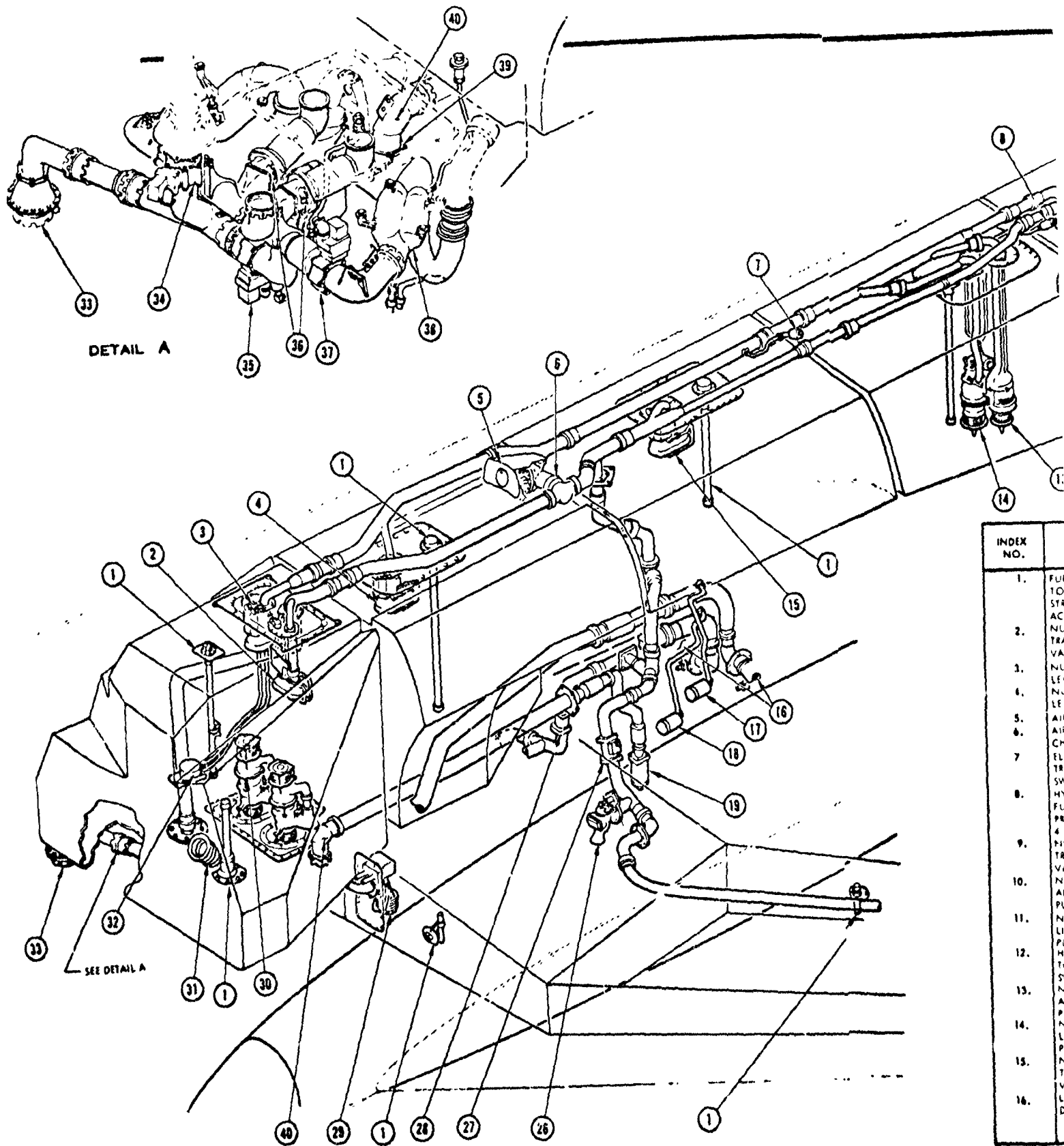


Notes

- 1 ADDITION INFORMATION MAY BE FOUND IN T.O. 1F-4C-2-23 WIRING DIAGRAMS AND T.O. 1F-4C-2-24 WIRING REPAIR FOR F-4 AIRCRAFT AND T.O. 1F-4 (RIC-2-23 WIRING DIAGRAMS AND T.O. 1F-4 (RIC-2-24 WIRING REPAIR FOR RF-4 AIRCRAFT.
- 2 MAKE ALL ELECTRICAL DISCONNECTS IN SPLICE AREAS OR AT DISCONNECTS AS SHOWN.
 - D** INSTALLED ON F-4 63-7527 AND SUBSEQUENT.
 - A** INSTALLED ON RF-4 AIRCRAFT ONLY.
 - S** INSTALLED ON F-4 AIRCRAFT ONLY.
 - D** INSTALLED ON F-4 62-12199 THRU 63-7420
 - D** INSTALLED ON F-4 63-7527 AND SUBSEQUENT AND RF-4 63-7750 AND SUBSEQUENT.

ECTS

NOTE: For F-4C. For F-4E see TO 1F-4E-2-23).



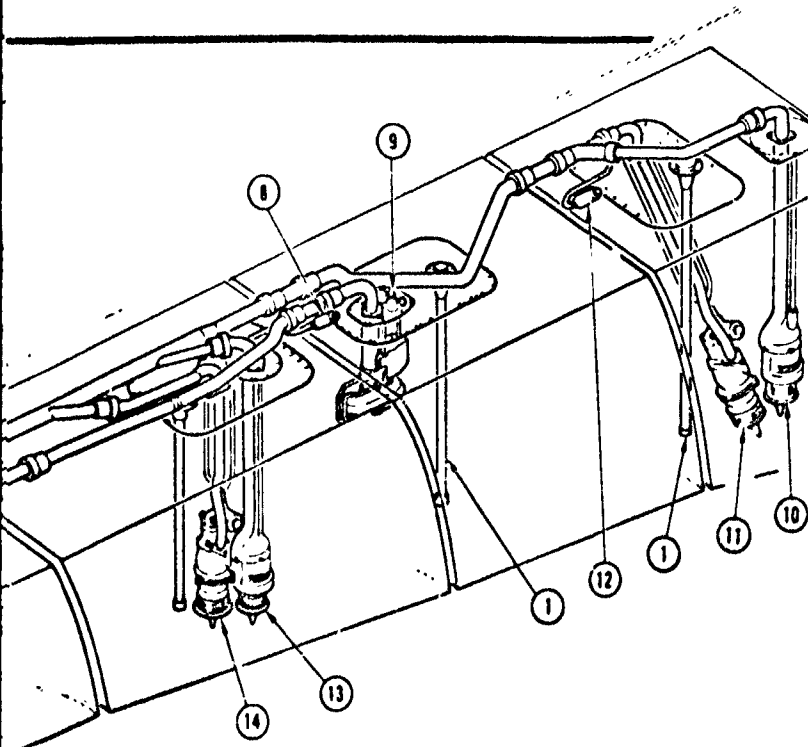
DETAIL A

SEE DETAIL A

INDEX NO.	
1.	FUE TO STR ACC NU; TRA VAL NU; LEV NU; LEV
2.	NU; TRA VAL NU; LEV NU; LEV
3.	NU; LEV NU; LEV
4.	NU; LEV
5.	AIR CH; ELE; TRA SW; HY; FUE; PRE; 4 C
6.	NU; TRA VAL NU; ALL; PUA; NU; LIC; PUA; HY; TRA SW; NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
7.	TRA SW; HY; FUE; PRE; 4 C
8.	NU; TRA VAL NU; ALL; PUA; NU; LIC; PUA; HY; TRA SW; NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
9.	NU; TRA VAL NU; ALL; PUA; NU; LIC; PUA; HY; TRA SW; NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
10.	NU; TRA VAL NU; ALL; PUA; NU; LIC; PUA; HY; TRA SW; NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
11.	NU; LIC; PUA; HY; TRA SW; NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
12.	NU; LIC; PUA; HY; TRA SW; NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
13.	NU; ALL; PUA; NU; LIC; PUA; NU; TRA VAL; LEFT DISC
14.	NU; LIC; PUA; NU; TRA VAL; LEFT DISC
15.	NU; LIC; PUA; NU; TRA VAL; LEFT DISC
16.	NU; LIC; PUA; NU; TRA VAL; LEFT DISC

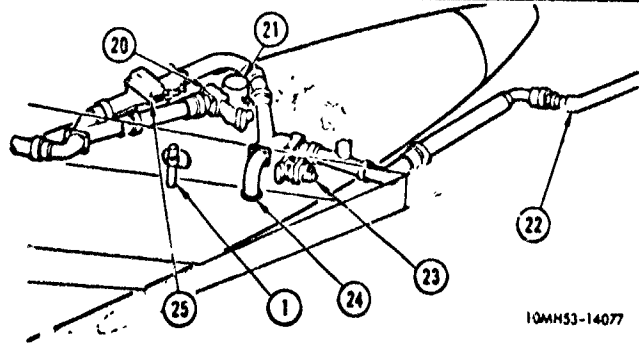
E-17 F-4C FUEL SYSTEM

2



INDEX NO.	NOMENCLATURE	ACCESS DOOR OR LOCATION
1.	FUEL QUANTITY UNIT, REFER TO T.O. 1F-4C-2-11, INSTRUMENT SYSTEMS FOR ACCESS DOOR LOCATION.	
2.	NUMBER 1 CELL REFUELING TRANSFER LEVEL CONTROL VALVE	19
3.	NUMBER 1 CELL TRANSFER LEVEL CONTROL VALVE.	19
4.	NUMBER 2 CELL TRANSFER LEVEL CONTROL VALVE.	34
5.	AIR REFUELING RECEPTACLE	34
6.	AIR REFUELING RECEPTACLE CHECK VALVE	34
7.	ELECTRICALLY DRIVEN FUEL TRANSFER PUMP PRESSURE SWITCH	135
8.	HYDRAULICALLY DRIVEN FUEL TRANSFER PUMP PRESSURE SWITCH (NUMBER 4 CELL)	34
9.	NUMBER 3 CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	48
10.	NUMBER 6 CELL ELECTRICALLY DRIVEN TRANSFER PUMP	57
11.	NUMBER 6 CELL HYDRAULICALLY DRIVEN TRANSFER PUMP	48
12.	HYDRAULICALLY DRIVEN TRANSFER PUMP PRESSURE SWITCH - NUMBER 6 CELL.	48
13.	NUMBER 4 CELL ELECTRICALLY DRIVEN TRANSFER PUMP	48
14.	NUMBER 4 CELL HYDRAULICALLY DRIVEN TRANSFER PUMP.	48
15.	NUMBER 3 CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	34
16.	LEFT AND RIGHT ENGINE DISCONNECTS.	81L AND R

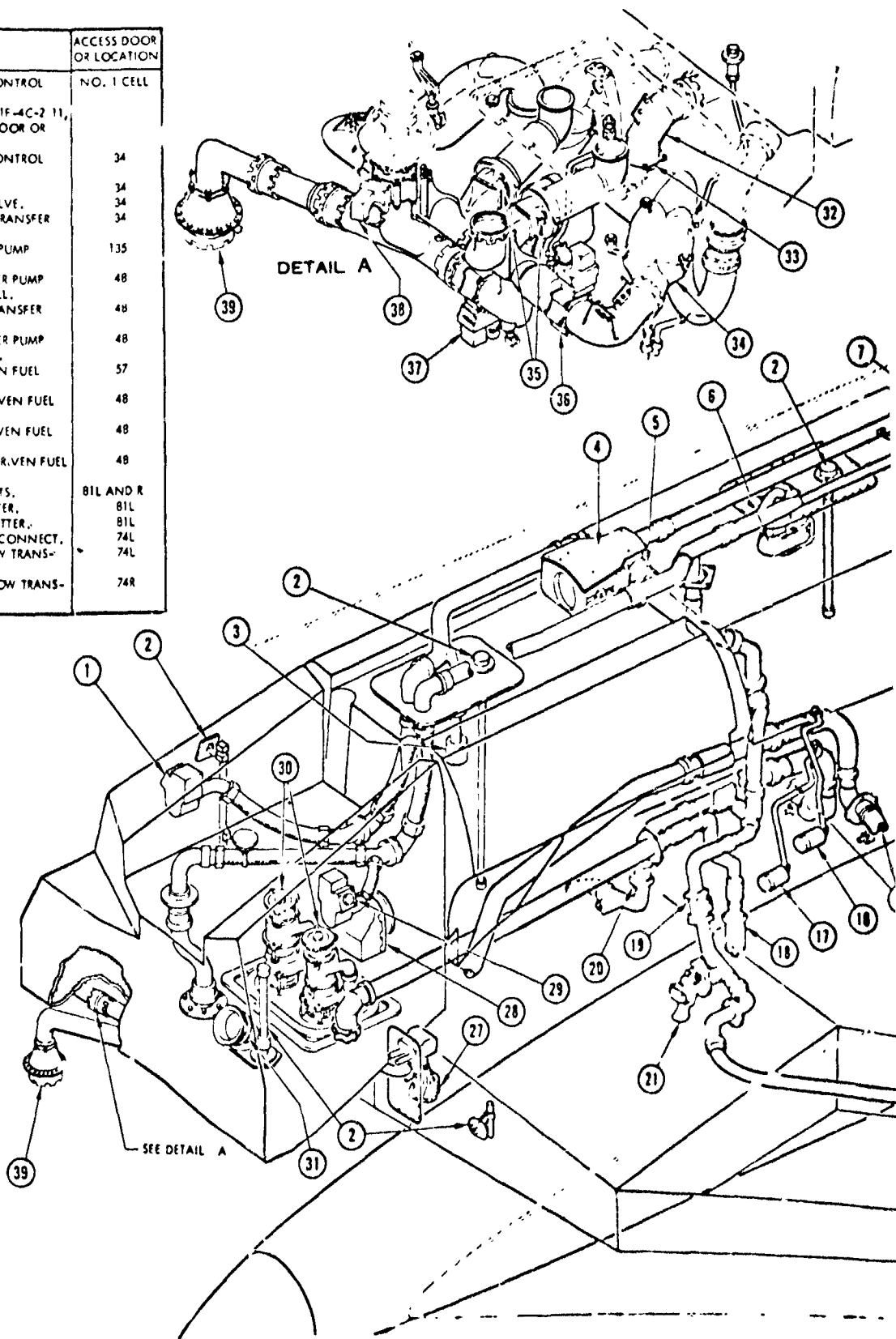
INDEX NO.	NOMENCLATURE	ACCESS DOOR OR LOCATION
17.	LEFT FUEL BOOST PUMP TRANSMITTER	81L
18.	RIGHT FUEL BOOST PUMP TRANSMITTER	81R
19.	EXTERNAL CENTERLINE TANK FUEL DISCONNECT.	74L
20.	EXTERNAL WING TANK SHUT-OFF VALVE, RIGHT	101R
	LEFT	101L
21.	EXTERNAL WING TANK FUEL DISCONNECT, RIGHT	101R
	LEFT	101L
22.	INTERNAL WING TANK DUMP MAST	
23.	INTERNAL WING FUEL DUMP SHUT-OFF VALVE, LEFT	101L
	RIGHT	101R
24.	INTERNAL WING FUEL OUTLET BELLMOUTH, LEFT	
	RIGHT	L. INTERNAL WING R. INTERNAL WING
25.	INTERNAL WING TANK TRANSFER-LOW LEVEL SHUT-OFF VALVE (OUTBOARD), LEFT	101L
	RIGHT	101R
26.	INTERNAL WING TANK TRANSFER-LOW LEVEL SHUT-OFF VALVE (INBOARD), LEFT	73L, 74L 73R, 74R
	RIGHT	74L
27.	LEFT EXTERNAL WING TANK FUEL FLOW TRANSMITTER.	74L
28.	RIGHT EXTERNAL WING TANK FUEL FLOW TRANSMITTER.	74R
29.	INTERNAL WING TANK REFUELING LEVEL CONTROL VALVE, LEFT	22
	RIGHT	22
30.	FUEL BOOST PUMPS, LEFT	22
	RIGHT	22
31.	BOOST PUMP BYPASS CHECK VALVE.	22
32.	NUMBER 1 CELL FUEL TRANSFER LEVEL CONTROL PILOT VALVE	NO 1 CELL
33.	GROUND FUELING ADAPTER, REFUELING SHUTOFF VALVE.	26R
34.	DEFUELING SHUTOFF VALVE.	22
35.	FUEL BOOST PUMP CHECK VALVES, LEFT	22
	RIGHT	22
37.	ENGINE MANIFOLD SHUT-OFF VALVE, LEFT	22
	RIGHT	22
38.	ENGINE MANIFOLD FUEL-HYDRAULIC RADIATOR, LEFT	22
	RIGHT	22
39.	EXTERNAL CENTERLINE TANK FUEL SHUT-OFF VALVE.	22
40.	EXTERNAL CENTERLINE TANK FUEL FLOW TRANSMITTER.	22



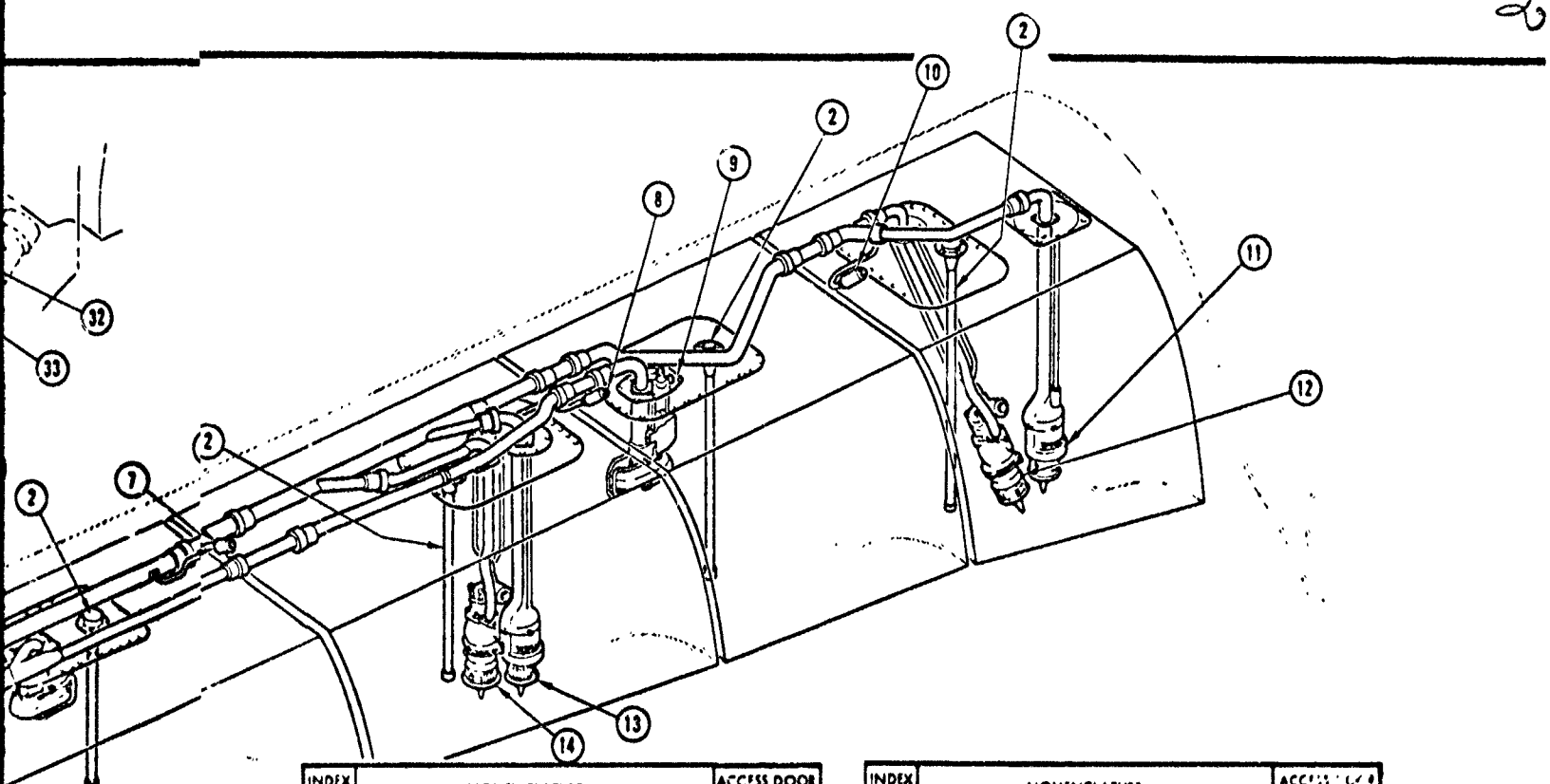
10MM53-14077

INDEX NO.	NOMENCLATURE	ACCESS DOOR OR LOCATION
1.	NUMBER ONE CELL TRANSFER LEVEL CONTROL VALVE	NO. 1 CELL
2.	FUEL QUANTITY UNIT. REFER TO T.O 1F-4C-2 11, INSTRUMENT SYSTEMS FOR ACCESS DOOR OR LOCATION.	
3.	NUMBER TWO CELL TRANSFER LEVEL CONTROL VALVE	34
4.	AIR REFUELING RECEPTACLE.	34
5.	AIR REFUELING RECEPTACLE CHECK VALVE.	34
6.	NUMBER THREE CELL REFUELING AND TRANSFER LEVEL CONTROL VALVE	34
7.	ELECTRICALLY DRIVEN FUEL TRANSFER PUMP PRESSURE SWITCH	135
8.	HYDRAULICALLY DRIVEN FUEL TRANSFER PUMP PRESSURE SWITCH, NUMBER FOUR CELL.	48
9.	NUMBER FIVE CELL REFUELING AND TRANSFER LEVEL CONTROL VALVE	48
10.	HYDRAULICALLY DRIVEN FUEL TRANSFER PUMP PRESSURE SWITCH, NUMBER SIX CELL.	48
11.	NUMBER SIX CELL ELECTRICALLY DRIVEN FUEL TRANSFER PUMP	57
12.	NUMBER SIX CELL HYDRAULICALLY DRIVEN FUEL TRANSFER PUMP.	48
13.	NUMBER FOUR CELL ELECTRICALLY DRIVEN FUEL TRANSFER PUMP	48
14.	NUMBER FOUR CELL HYDRAULICALLY DRIVEN FUEL TRANSFER PUMP.	48
15.	LEFT AND RIGHT ENGINE DISCONNECTS.	B1L AND R
16.	LEFT BOOST PUMP PRESSURE TRANSMITTER.	B1L
17.	RIGHT BOOST PUMP PRESSURE TRANSMITTER.	B1R
18.	EXTERNAL CENTERLINE TANK FUEL DISCONNECT.	74L
19.	LEFT EXTERNAL WING TANK FUEL FLOW TRANSMITTER.	74L
20.	RIGHT EXTERNAL WING TANK FUEL FLOW TRANSMITTER	74R

34

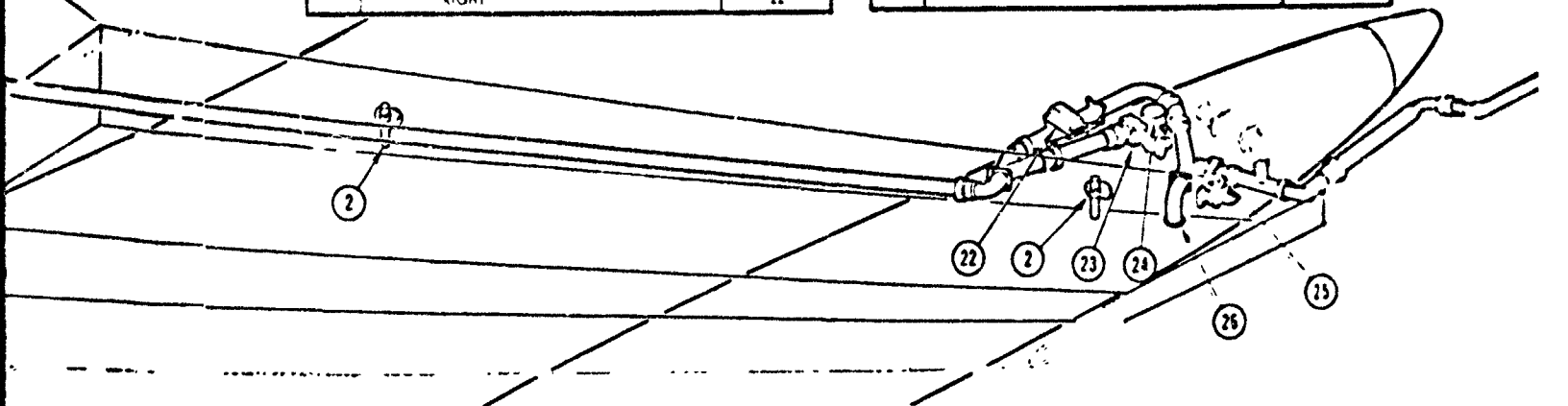
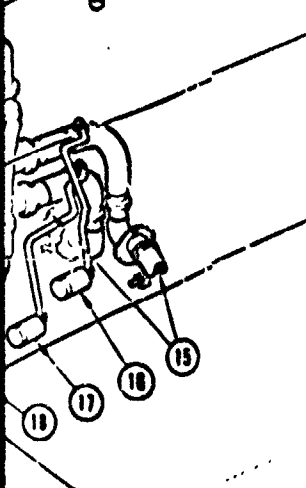


E-18 F-4D/RF-4C FUEL SYSTEM

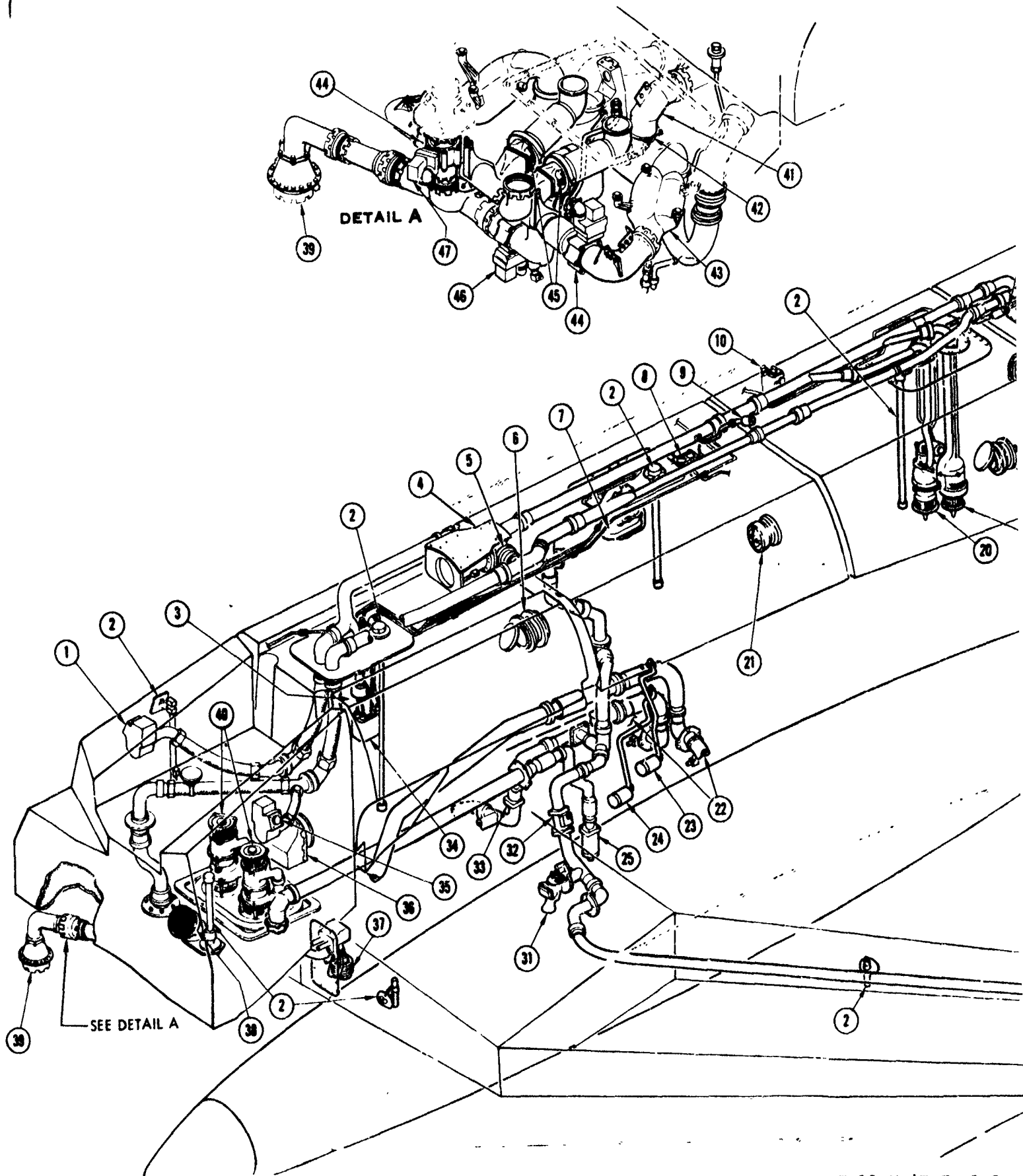


INDEX NO.	NOMENCLATURE	ACCESS DOOR OR LOCATION
21.	INTERNAL WING TANK TRANSFER-LOW LEVEL SHUTOFF VALVE. (INBOARD) LEFT RIGHT	73L, 74L 73R, 74R
22.	INTERNAL WING TANK TRANSFER-LOW LEVEL SHUTOFF VALVE. (OUTBOARD) LEFT RIGHT	101L 101R
23.	EXTERNAL WING TANK FUEL SHUTOFF VALVE. LEFT RIGHT	101L 101R
24.	EXTERNAL WING TANK FUEL DISCONNECT. LEFT RIGHT	101L 101R
25.	INTERNAL WING TANK DUMP SHUTOFF VALVE. LEFT RIGHT	101L 101R
26.	INTERNAL WING TANK FUEL OUTLET BELLMOUTH. LEFT RIGHT	L, INT WING R, INT WING
27.	INTERNAL WING TANK REFUELING LEVEL CONTROL VALVE. LEFT RIGHT	22 22

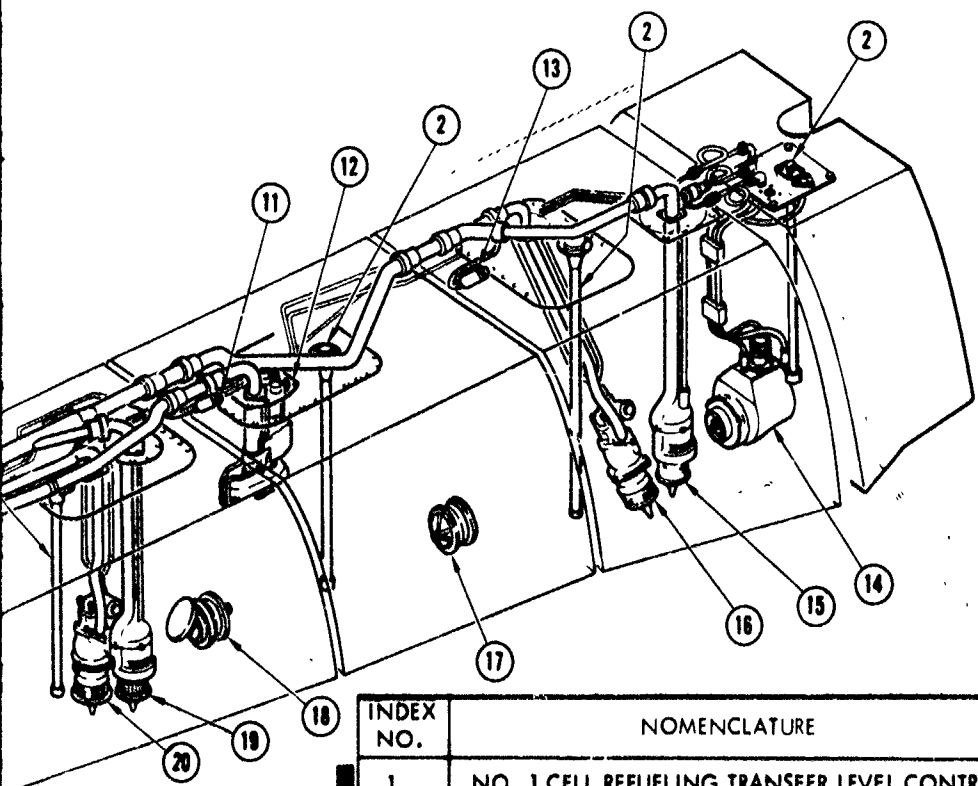
INDEX NO.	NOMENCLATURE	ACCESS DOOR OR LOCATION
28.	INTERCONNECTOR CHECK VALVE; CELL ONE - CELL TWO	NO. 1 CELL
29.	NUMBER ONE CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	NO. 1 CELL
30.	FUEL BOOST PUMPS LEFT RIGHT	22 22
31.	BOOST PUMP BYPASS CHECK VALVE.	22
32.	EXTERNAL CENTERLINE TANK FUEL FLOW TRANSMITTER.	22
33.	EXT CENTERLINE TANK FUEL SHUTOFF VALVE.	22
34.	ENGINE MANIFOLD FUEL-HYDRAULIC RADIATOR. LEFT RIGHT	22 22
35.	BOOST PUMP CHECK VALVES: LEFT RIGHT	22 22
36.	ENGINE MANIFOLD SHUTOFF VALVE. LEFT RIGHT	22 22
37.	DEFUELING SHUTOFF VALVE.	22
38.	REFUELING SHUTOFF VALVE.	22
39.	GROUND FUELING ADAPTER.	210



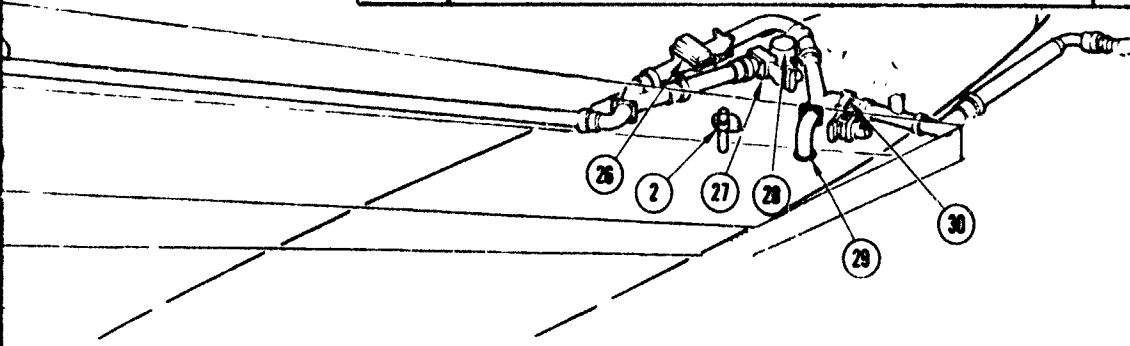
FUEL SYSTEM



2



INDEX NO.	NOMENCLATURE	REFERENCE DESIGNATION	DOOR OR LOCATION
1.	NO. 1 CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	8L436	128R
2.	FUEL QUANTITY UNIT. REFER TO T.O. 1F-4E-2-11. INSTRUMENT SYSTEMS FOR ACCESS DOOR OR LOCATION.		
3.	NO. 2 CELL FUEL TRANSFER LEVEL CONTROL VALVE.	8L537	34
4.	AIR REFUELING RECEPTACLE.	7PA5M16	34
5.	REFUELING RECEPTACLE CHECK VALVE.	7VA5M33	34
6.	INTERCONNECTOR CHECK VALVE.	7VA5M24	NO. 2 CELL
7.	NO. 3 CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	7L511	34
8.	NO. 7 CELL TRANSFER CONTROL PRESSURE REGULATOR AND RELIEF VALVE.	7VA5M36	34
9.	ELECTRICAL FUEL TRANSFER PUMP PRESSURE SWITCH.	8S527	135
10.	HYDRAULIC FUEL TRANSFER PUMP CONTROL VALVE.	8L538	48
11.	HYDRAULIC FUEL TRANSFER PUMP PRESSURE SWITCH, NO. 4 CELL.	8S528	48
12.	NO. 5 CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	7L510	48
13.	HYDRAULIC FUEL TRANSFER PUMP PRESSURE SWITCH NO. 6 CELL.	8S529	48



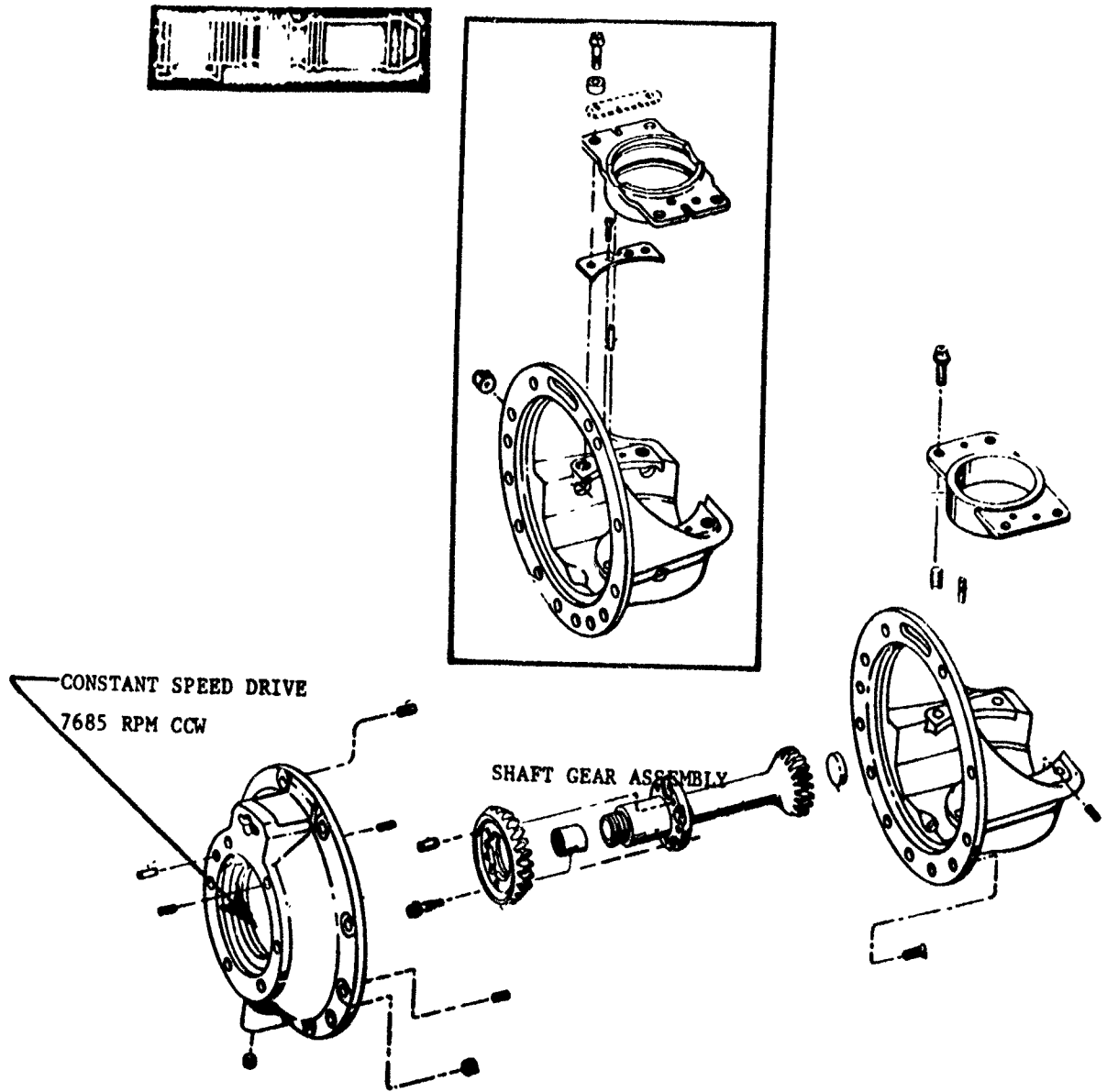
- 14. NO. 7 CELL TRA
- 15. NO. 6 CELL ELE
- 16. NO. 6 CELL HYI
- 17. INTERCONNECT
- 18. INTERCONNECT
- 19. NO. 4 CELL ELE
- 20. NO. 4 CELL HYI
- 21. INTERCONNECT
- 22. LEFT AND RIGH
- 23. LEFT BOOST PUA
- 24. RIGHT BOOST PI
- 25. EXTERNAL CENT
- 26. INTERNAL WINC
TRANSFER LOW
LEFT
RIGHT
- 27. EXTERNAL WINC
LEFT
RIGHT
- 28. EXTERNAL WINC
LEFT
RIGHT
- 29. INTERNAL WINC
LEFT
RIGHT
- 30. INTERNAL WINC
LEFT
RIGHT
- 31. INTERNAL WINC
LOW LEVEL SHU'
LEFT
RIGHT
- 32. LEFT EXTERNAL
TRANSMITTER.
- 33. RIGHT EXTERNAL
TRANSMITTER.
- 34. NO. 7 CELL COI
- 35. NO. 1 CELL REFL
VALVE.
- 36. NO. 1 CELL INTI
- 37. INTERNAL WINC
LEFT
RIGHT
- 38. BOOST PUMP BYI
- 39. PRESSURE FUELI
- 40. FUEL BOOST PUA
LEFT
RIGHT
- 41. CENTERLINE EXT
- 42. EXTERNAL CENTI
VALVE.
- 43. ENGINE MANIF
LEFT
RIGHT
- 44. ENGINE FUEL SH
LEFT
RIGHT
- 45. BOOST PUMP CH
LEFT
RIGHT
- 46. DEFUELING SHU'
- 47. REFUELING SHUT

(NOTE: F-4 F-4Es through 68-494. For subsequent F-4Es see TO 1F-4E-2-10).

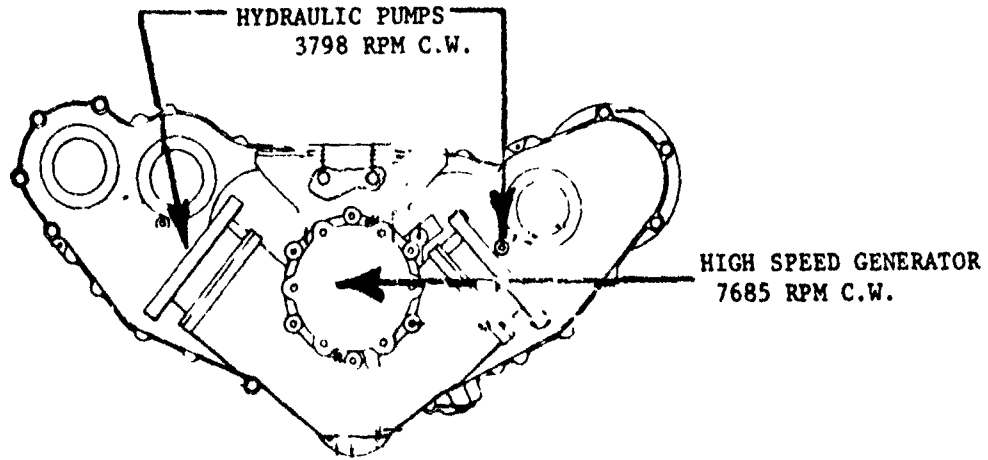
3

REFERENCE IDENTIFICATION	DOOR OR LOCATION
436	128R
437	34
4A5M16	34
4A5M33	34
4A5M24	NO. 2 CELL
4511	34
4A5M36	34
4527	135
4538	48
4528	48
4510	48
4529	48

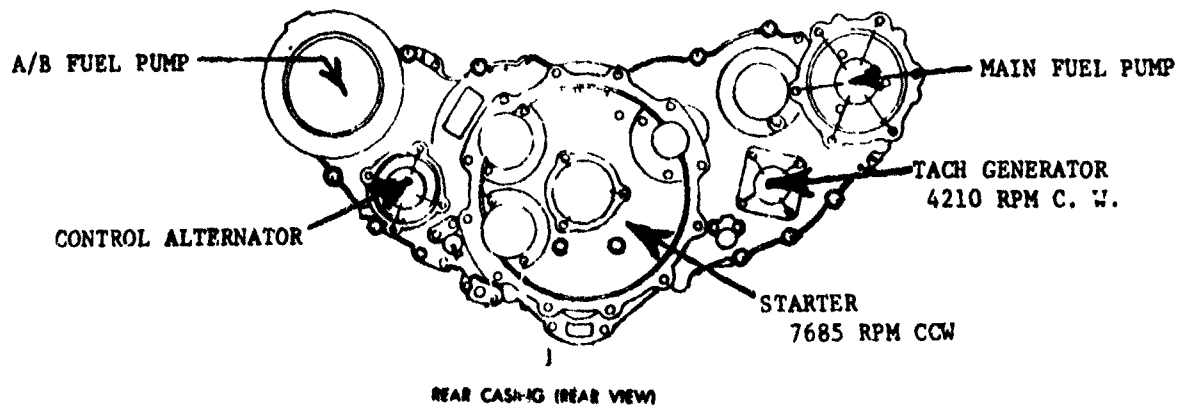
14.	NO. 7 CELL TRANSFER SHUTOFF VALVE.	7VA5M40	NO. 7 CELL
15.	NO. 6 CELL ELECTRICAL FUEL TRANSFER PUMP.	8B520	57
16.	NO. 6 CELL HYDRAULIC FUEL TRANSFER PUMP.	8BA452	48
17.	INTERCONNECTOR.		NO. 5 CELL
18.	INTERCONNECTOR CHECK VALVE.	7VA5M25	NO. 4 CELL
19.	NO. 4 CELL ELECTRICAL FUEL TRANSFER PUMP.	8B517	48
20.	NO. 4 CELL HYDRAULIC FUEL TRANSFER PUMP.	8BA5M1	48
21.	INTERCONNECTOR.		NO. 3 CELL
22.	LEFT AND RIGHT ENGINE DISCONNECTS.		81L AND R
23.	LEFT BOOST PUMP PRESSURE TRANSMITTER.	11TR601	74L
24.	RIGHT BOOST PUMP PRESSURE TRANSMITTER.	11TR602	74L
25.	EXTERNAL CENTERLINE TANK FUEL DISCONNECT.	7PA5M30	74L
26.	INTERNAL WING TANK OUTBOARD FUEL TRANSFER LOW LEVEL SHUTOFF VALVE:		
	LEFT	7L918	101L
	RIGHT	7L815	101R
27.	EXTERNAL WING TANK FUEL SHUTOFF VALVE:		
	LEFT	7B904	101L
	RIGHT	7B803	101R
28.	EXTERNAL WING TANK FUEL DISCONNECT:		
	LEFT	7PA9M6	101L
	RIGHT	7PA8M5	101R
29.	INTERNAL WING TANK FUEL OUTLET BELLMOUTH:		
	LEFT		L INT WING
	RIGHT		R INT WING
30.	INTERNAL WING TANK FUEL DUMP SHUTOFF VALVE:		
	LEFT	7B908	101L
	RIGHT	7B809	101R
31.	INTERNAL WING TANK INBOARD FUEL TRANSFER LOW LEVEL SHUTOFF VALVE:		
	LEFT	7L517	73L, 74L
	RIGHT	7L516	73R, 74R
32.	LEFT EXTERNAL WING TANK FUEL FLOW TRANSMITTER.	10TR603	74L
33.	RIGHT EXTERNAL WING TANK FUEL FLOW TRANSMITTER.	10TR604	74L
34.	NO. 7 CELL CONTROL VALVE.	7VA5M39	34
35.	NO. 1 CELL REFUELING TRANSFER LEVEL CONTROL VALVE.	7L412	128L
36.	NO. 1 CELL INTERCONNECTOR CHECK VALVE.	7VA5M23	NO. 1 CELL
37.	INTERNAL WING TANK FUEL LEVEL CONTROL VALVE:		
	LEFT	7L513	22
	RIGHT	7L514	22
38.	BOOST PUMP BYPASS CHECK VALVE.	8VA4M7	22
39.	PRESSURE FUELING AND DEFUELING ADAPTER .	7PA4M15	26R
40.	FUEL BOOST PUMPS:		
	LEFT	8B419	22
	RIGHT	8B418	22
41.	CENTERLINE EXTERNAL TANK FUEL FLOW TRANSMITTER	10TR402	22
42.	EXTERNAL CENTERLINE TANK FUEL TRANSFER SHUTOFF VALVE .	7B406	22
43.	ENGINE MANIFOLD FUEL - HYDRAULIC RADIATOR:		
	LEFT	14CA4M27	22
	RIGHT	14CA4M28	22
44.	ENGINE FUEL SHUTOFF VALVE:		
	LEFT	7B402	22
	RIGHT	7B401	22
45.	BOOST PUMP CHECK VALVES:		
	LEFT	8VA4M5	22
	RIGHT	8VA4M6	22
46.	DEFUELING SHUTOFF VALVE .	7B405	22
47.	REFUELING SHUTOFF VALVE .	7B407	22



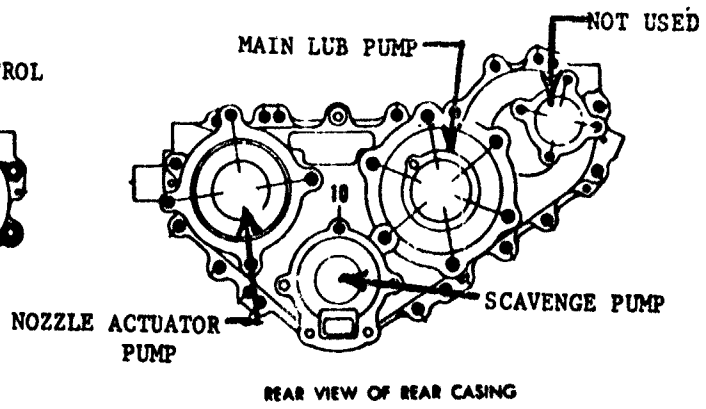
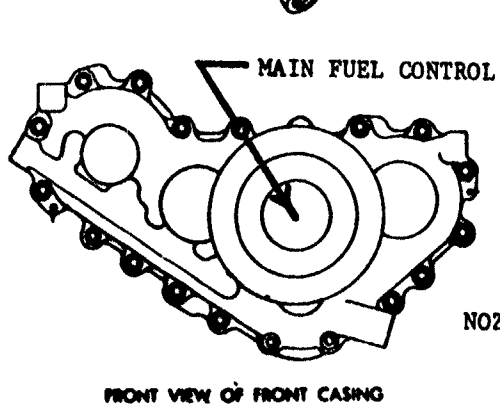
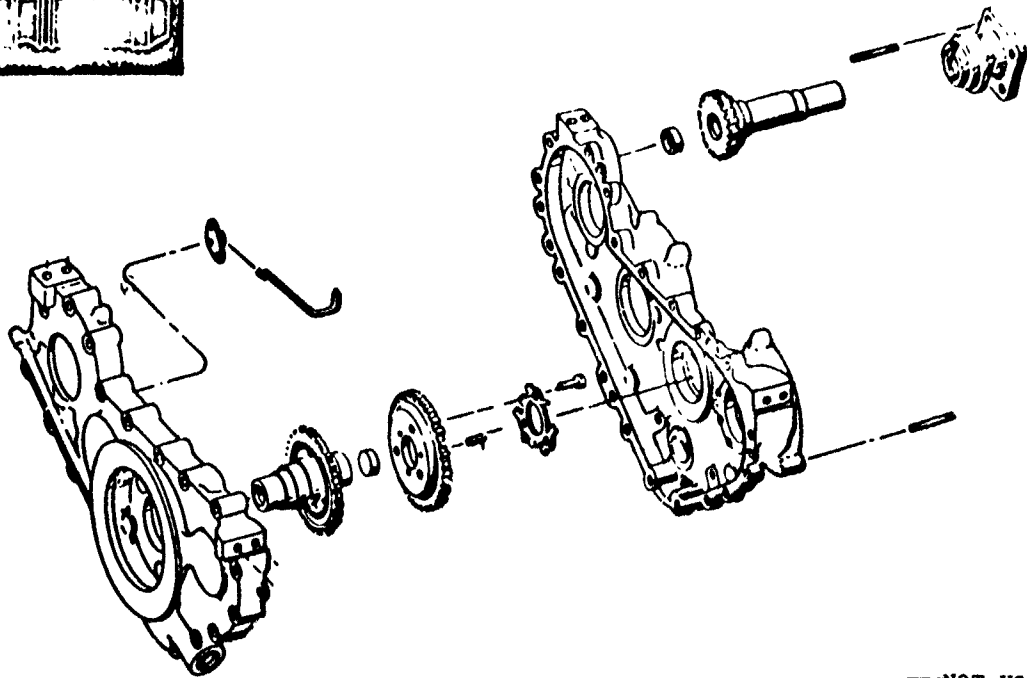
E-20 FRONT GEARBOX CASING AND SHAFT GEAR (J79-15-AND-17)



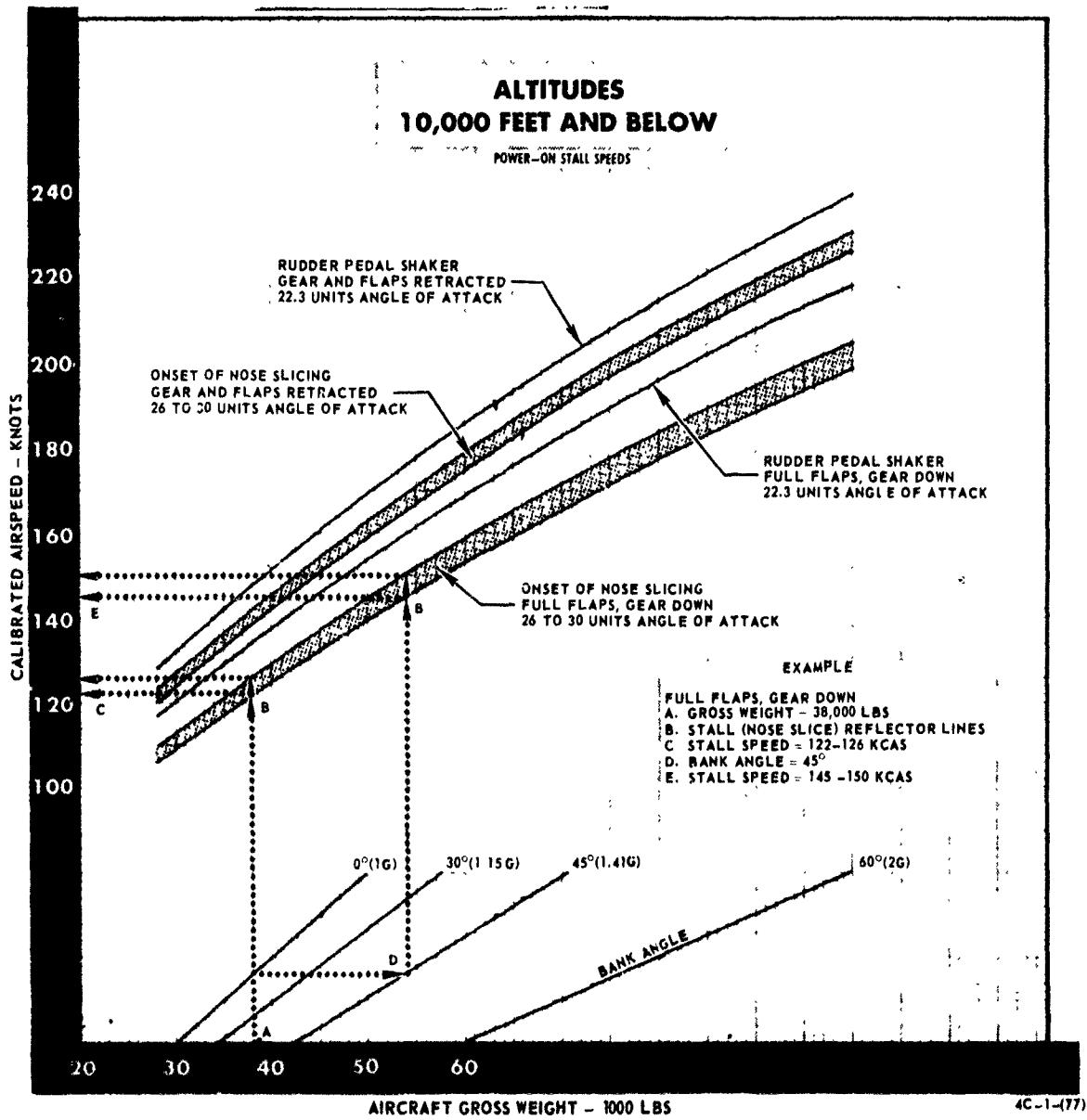
MAIN CASING (FRONT VIEW)



E-21 TRANSFER GEAR BOX (J79-15 -AND -17



E-22 REAR GEAR BOX (J79-15 AND -17)



E-23 F-4 Stall Characteristics for all Configurations

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