

UNCLASSIFIED

AD NUMBER	
AD379231	
CLASSIFICATION CHANGES	
TO:	unclassified
FROM:	confidential
LIMITATION CHANGES	
TO:	Approved for public release, distribution unlimited
FROM:	Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; JUN 1965. Other requests shall be referred to Director, Federal Aviation Administration , Washington, DC.
AUTHORITY	
FAA ltr, 10 Oct 1972; FAA ltr, 10 Oct 1972	

THIS PAGE IS UNCLASSIFIED

AD-379 231

Best Available Copy

Best Available Copy

**Best
Available
Copy**

NOTICES

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

Copies have been placed in the DDC collection. U.S. Government agencies may obtain copies from DDC. Other qualified DDC users may request, by submission of a DDC Form 1, through:

Director of Supersonic Transport Development
Federal Aviation Agency
Washington, D.C. 20553

Defense Documentation Center release to the Clearinghouse for Federal, Scientific, and Technical Information (CFSTI) and foreign announcement and dissemination are not authorized. The distribution of this report is limited because it contains technology identifiable with items excluded from export by the Department of State (U.S. Export Control Act of 1949 as amended).

CONFIDENTIAL

Report No. PWA-2600
Date: 30 June 1965

Volume 1

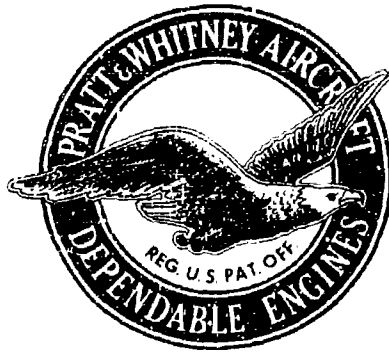
SUPERSONIC TRANSPORT AIRCRAFT ENGINE

PHASE D-B DEVELOPMENT PROGRAM

FINAL REPORT (U)

Prepared Under Contract FA-SS-65-18

Period Covered 1 January through 30 June 1965



PWA-E. H. Document Control
Eng. Sub-Control Station

JUN 25 1965

Number C63-5-C271-A

This document contains information affecting the national defense of the United States within the meaning of the Espionage Laws, Title 18 U.S.C., Sections 793 and 794. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

Reproduction, distribution and use of this document are limited as specified in the contract under which it was prepared. Use for other than U. S. Governmental purposes is subject to prior approval of United Aircraft Corporation.

Approved by

C. W. Bristol, Jr.
Project Engineer

Pratt & Whitney Aircraft DIVISION OF UNITED AIRCRAFT CORPORATION

EAST HARTFORD

CONNECTICUT

COPY NO. _____

CONFIDENTIAL

FAA SECURITY CONTROL
NO. 67557 A-7

DOWNGRADED AT 3 YEAR INTERVALS.
DECLASSIFIED AFTER 12 YEARS
DDO DIB 520510

PAGES NOT FILMED ARE BLANK.

PRATT & WHITNEY AIRCRAFT

PWA-2600

FOREWORD

This report describes the work that was accomplished by Pratt & Whitney Aircraft during the period 1 January 1965 through 30 June 1965 in accordance with the requirements of contract FA-SS-65-18 entitled "Development of Supersonic Transport Engine - Phase II-B". The report is submitted to fulfill the requirements of Item 7, Section D of the contract work statement.

This report is classified as CONFIDENTIAL in compliance with the provisions of DD Form 254 dated 1 January 1965 provided for this contract.

TABLE OF CONTENTS

FOREWORD

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

INTRODUCTION

Item 1 - INSTALLATION COORDINATION

A. Installation Coordination Meetings with Airplane Companies	1-1
1. Installation Coordination with the Boeing Airplane Company	1-1
2. Installation Coordination with the Lockheed California Corporation	1-11
B. Engine Cycle Studies	1-17
1. Introduction	1-17
2. Augmentation System Studies	1-21
3. Inflight Performance Measurement System	1-23

Item 2 - DESIGN

A. STF219 Engine	2A-1
1. Overall Engine Design	2A-1
2. Fan	2A-2
3. Intermediate Section	2A-15
4. Aerodynamic Brake	2A-21
5. High Pressure Compressor	2A-23
6. Diffuser Case	2A-34
7. Main Combustion Chamber	2A-38
8. Turbine Design	2A-46
9. Turbine Cooling Analysis	2A-65
10. Turbine Exhaust	2A-113
11. Duct Heater Diffuser	2A-115
12. Duct Heater	2A-120
13. Duct Heater Nozzle	2A-127
14. Ejector-Reverser	2A-130
15. Assembly of Major Components	2A-143
16. Critical Speeds	2A-150

TABLE OF CONTENTS (Cont'd)

17. Bearing Design	2A-151
18. Bearing Thrust Balance System	2A-162
19. Oil Heat Rejection	2A-168
20. Tubing System Design	2A-171
21. Weight Analysis	2A-176
 B. STF227 Turbojet Engine	 2B-1
I. General Description	2B-1
II. Aerodynamic Turbojet Studies	2B-3
1. Compressor	2B-3
2. Main Burner	2B-4
3. Turbine	2B-7
4. Afterburner	2B-16
5. Ejector-Reverser	2B-19
III. Mechanical Turbojet Studies	2B-25
1. Variable Inlet Guide Vanes and Case	2B-25
2. Compressor	2B-29
3. Diffuser	2B-39
4. Primary Combustion Chamber	2B-41
5. Turbine	2B-48
6. Turbine Exhaust Section	2B-60
7. Bearing and Seals	2B-61
8. Afterburner and Primary Nozzle	2B-65
9. Octagonal Blow-In-Door Ejector-Reverser	2B-71
10. Canted Engine Configurations	2B-80
11. External Plumbing, Brackets and Hardware	2B-82
12. Main Gearbox Accessory Drive	2B-92
13. Control System	2B-96
14. Engine Mounts	2B-110
15. Bleed and Vent Systems	2B-113
16. Thrust Balance	2B-115
17. Engine Heat Rejection	2B-121
18. Secondary Airflow	2B-124
19. Weight Analysis	2B-126
20. Assembly Procedure	2B-127

TABLE OF CONTENTS (Cont'd)

Item 3 - COMPRESSOR DEVELOPMENT

A. Introduction	3-1
B. Development Program	3-1
1. Two-Stage Fan Rig	3-1
2. Aeroelasticity Investigations	3-3
3. Multistage Compressor	3-5

Item 4 - PRIMARY COMBUSTION

A. Introduction	4-1
B. Description of Burner Section	4-3
C. Description of Rigs	4-3
1. Two-Dimensional Annular Segmental Rig	4-3
2. Large-Scale Full-Annular Burner Rig	4-4
3. Ram Induction Primary Burner Rigs	4-4
D. Test Description and Results	4-5
1. Two-Dimensional Annular Segmental Burner Rig Testing	4-5
2. Large-Scale Full-Annular Burner Rig Testing	4-7
3. Ram Induction Primary Burner Rigs	4-10

Item 5 - TURBINE DEVELOPMENT

A. Introduction	5-1
1. Investigation of the Heat Transfer Characteristics of Air Cooled Vanes	5-1
2. Investigation of New Fabrication Techniques	5-2
3. Thermal Fatigue Testing of Air Cooled Vanes in a Hot Chocked Cascade Rig	5-2
4. Evaluation of Aircooled Turbine Blades and Vanes in a Full-Scale Turbine Development Engine	5-2
5. Performance Testing of Film Cooled Vanes and Blades	5-2
6. Laboratory Testing of Advanced Vane and Blade Materials	5-3

TABLE OF CONTENTS (Cont'd)

B.	Basic Investigation of the Effectiveness of Film Cooling	5-3
1.	Introduction	5-3
2.	Description of Test Rig	5-4
3.	Description of Parts Tested	5-4
4.	Test Procedure	5-5
5.	Test Program and Results	5-5
C.	Investigation of the Heat Transfer Characteristics of Air Cooled Vanes in a High Pressure Cascade Rig	5-7
1.	Description of Rig	5-7
2.	Test Program	5-9
3.	Test Results	5-10
D.	Development of a Choked Cascade Thermal Shock Rig for the Thermal Fatigue Testing of Full-Scale Turbine Vanes	5-10
1.	Description of Test Rig	5-10
2.	Test Program	5-11
E.	Evaluation of Full-Scale Air Cooled Blades and Vanes in the Turbine Development Engine	5-12
1.	Description of Turbine Development Engine	5-13
2.	Description of Blades and Vanes Tested in the Turbine Development Engine	5-13
3.	Test Procedure	5-14
4.	Test Program and Results	5-15
F.	Performance Testing of Film Cooled Vanes	5-18
1.	Description of Test Rig	5-18
2.	Description of Parts Tested	5-19
3.	Method of Test	5-19
G.	Performance Testing of Film Cooled Blades	5-20
1.	Description of Test Rig	5-20
2.	Description of Parts Tested	5-20
3.	Test Program	5-21
4.	Test Results	5-22
H.	Laboratory Testing of Advanced Vane and Blade Materials	5-23
1.	Cyclic Stress Rupture Tests	5-23
2.	Low Cycle Fatigue Testing	5-24
3.	Testing to Determine Long Term Creep Properties of PWA 657 Cobalt Base Alloy	5-26

TABLE OF CONTENTS (Cont'd)

4. Creep Rupture Tests of PWA 658 Nickel Base Alloy	5-26
5. Combined Stress Rupture Fatigue Tests on Turbine Blade Materials	5-27
6. Strain Cycle Fatigue Tests	5-27
Item 6 - AUGMENTORS	
A. Aerodynamic Flameholder Tests in Small-Scale Duct Heater Rigs	6-1
1. Introduction and Summary	6-1
2. Detailed Program Description	6-2
B. Ram Induction Burner Tests in Small-Scale Duct Heater Rigs	6-6
1. Introduction	6-6
2. Phase IIB Program Summary	6-7
C. Full Scale Duct Heater Rig	6-8
1. Introduction	6-8
2. Test Results	6-8
3. Description of Rig and Test Stand	6-11
4. Procedure	6-13
5. Conclusions and Recommendations	6-13
Item 7 - EXHAUST SYSTEM	
A. Introduction	7-1
B. Discussion	7-1
1. Forward Flight Investigations	7-2
2. Thrust Reversers	7-4
C. Conclusions	7-5
Item 8 - NOISE REDUCTION	
A. Introduction	8-1
B. Inlet Duct Sound Propagation Characteristics	8-2
1. General	8-2
2. Conclusions	8-3
3. Recommendations	8-3

TABLE OF CONTENTS (Cont'd)

4.	Discussion of Results	8-3
5.	Analysis of Results	8-6
C.	Sound Absorbing Materials	8-7
1.	Objectives of Tests	8-7
2.	Description of Facilities	8-7
3.	Method of Test	8-7
4.	Results	8-8
5.	Conclusions	8-9
D.	Model Exhaust System Noise-Tests	8-9
1.	Description of Facilities	8-9
2.	Method of Test	8-10
3.	Description of Model Nozzles	8-10
4.	Results	8-11
5.	Conclusions	8-17
E.	Full-Scale Blow-In-Door Ejector Noise Tests	8-17
1.	General	8-18
2.	Results	8-18
3.	Conclusions	8-20
4.	Recommendations	8-20
5.	Description of Test Equipment	8-20
6.	Method of Test	8-21
7.	Data Analysis	8-22
8.	Calibration of Acoustic Instrumentation	8-22
9.	Discussion	8-23
F.	Analysis of Total SST Airplane Noise	8-24
1.	General	8-24
2.	Discussion	8-25
3.	Conclusions	8-27
4.	Recommendations	8-27
Item 9 - CONTROLS AND ACCESSORIES		
A.	Controls	9-1
1.	Introduction	9-1
2.	Engine Simulation Studies	9-2
3.	Engine Control Studies	9-6
4.	Vendor Activity	9-14

TABLE OF CONTENTS (Cont'd)

B. Accessories - Ignition System	9-16
1. Introduction	9-16
2. Components	9-16
3. Facilities	9-17
4. Test Procedure	9-18
5. Discussion of Results	9-19
6. Recommendations	9-22
 Item 10 - BEARING AND SEAL DEVELOPMENT	
A. Bearing Development	10-1
1. Endurance and Calibration Tests	10-1
2. Single-Ball Material Tests	10-7
3. Oil Pump and Jet Tests	10-10
B. Seal Development	10-12
1. Face Seal Tests	10-12
2. Ring Seal Tests	10-16
3. Materials and Design Tests	10-20
 Item 11 - FUELS	
A. Introduction	11-1
B. Investigation of the Effect of Fuel Oxygen Content on Thermal Stability	11-1
 Item 12 - LUBRICANTS	
A. Introduction	12-1
B. Lubricant Environment	12-1
1. Temperature	12-2
2. Oil Tank Capacity	12-2
3. Air Contacting Oil	12-2
4. Bearings	12-2
5. Alloys in Lubricating System	12-3
6. Gears	12-3
7. Elastomers	12-3
8. Lubricant Volatility	12-3

TABLE OF CONTENTS (Cont'd)

C. Gear Pitting Tests	12-3
D. Discussion	12-4
Item 13 - ENGINE-INLET COMPATIBILITY	
A. Introduction	13-1
B. Discussion	13-1
C. Conclusions	13-2
Item 14 - MATERIALS AND MANUFACTURING TECHNIQUES	
A. Introduction	14-1
B. Creep Testing of IMI 679 Titanium Alloy	14-1
1. Mechanical Properties - IMI 679 Titanium Alloy	14-2
C. Metallographic Study of PWA 1007 (Waspaloy)	14-4
D. Containment Testing	14-5
1. Ballistic Tests	14-5
2. Spin Containment Tests	14-6
3. Analytical Investigation	14-6
4. Discussion of Results	14-6
E. Diffusion Bonding of Titanium Fan Disk	14-7
Item 15 - SUPPORTING DESIGN CONSIDERATIONS	
A. Reliability	15-1
1. Introduction	15-1
2. Reliability Support of the Design Effort	15-1
3. Numerical Reliability Analysis	15-2
B. Configuration Management	15-3
1. Objective	15-3
2. Discussion	15-3

TABLE OF CONTENTS (Cont'd)

C. Weight and Center of Gravity Control	15-5
1. Turbofan Engines	15-5
2. Turbojet Engines	15-5
D. Value Engineering	15-5
E. Maintainability	15-7
1. Introduction	15-7
2. Maintainability Review of Design Layouts	15-7
3. Maintenance Analysis	15-10
F. Safety	15-12
1. Support of Design for Flight Safety	15-12
2. Design Support for Maintenance Personnel Safety	15-13

LIST OF ILLUSTRATIONS

Item 1

Figure
No.

- 1-1 Boeing Installation - SST
- 1-2 Effect of Moving the Cant Point Aft
- 1-3 STF219B 600 Lbs./Sec. Turbofan
- 1-4 STF219B 600 Lbs./Sec. Turbofan
- 1-5 STF219B 600 Lbs./Sec. Turbofan
- 1-6 STF219B Lbs./Sec. Turbofan
- 1-7 STF219B 600 Lbs./Sec. Turbofan
- 1-8 STF219B 600 Lbs./Sec. Turbofan
- 1-9 STF219B 600 Lbs./Sec. Turbofan
- 1-10 STF219B 600 Lbs./Sec. Turbofan
- 1-11 STF219 650 Lbs./Sec. Turbofan
- 1-12 STF219 650 Lbs./Sec. Turbofan
- 1-13 STF219 600 Lbs./Sec. Turbofan
- 1-14 Proposed Turbofan Accessory Arrangement
- 1-15 STJ227 500 Lbs./Sec. Turbojet
- 1-16 STJ227 500 Lbs./Sec. Turbojet
- 1-17 STJ227 500 Lbs./Sec. Turbojet
- 1-18 STJ227 500 Lbs./Sec. Turbojet
- 1-19 STJ227 525 Lbs./Sec. Turbojet
- 1-20 STJ227 525 Lbs./Sec. Turbojet
- 1-21 STJ227, 525 Lbs./Sec. (High Flow) Turbojet
- 1-22 STJ227, 525 Lbs./Sec. (Base Flow) Turbojet
- 1-23 STJ227, 525 Lbs./Sec. (Low Flow) Turbojet
- 1-24 STJ227, 525 Lbs./Sec. (High Flow) Turbojet
- 1-25 STJ227, 525 Lbs./Sec. (Base Flow) Turbojet
- 1-26 STJ227, 525 Lbs./Sec. (Low Flow) Turbojet
- 1-27 Proposed Turbojet Accessory Arrangement
- 1-28 Revised Accessory Arrangement Showing Fuel Components
Removed from Bottom of Engine
- 1-29 Revised Accessory Arrangement Making Room for Mount
Structure at Top of Engine
- 1-30 Proposed Turbofan Accessory Arrangement
- 1-31 Schematic of Octagonal Ejector
- 1-32 STF219L 700 Lb/Sec Turbofan Outboard Engine
- 1-33 STF219L 700 Lb/Sec Turbofan Inboard Engine
- 1-34 STF219L 650 Lb/Sec Turbofan

LIST OF ILLUSTRATIONS (Cont'd)

Item 1

Figure
No.

- 1-35 STF219L 650 Lb/Sec Turbofan Showing Engine/Ejector Relationship
- 1-36 Inlet Pressure and Temperature Probes for Biasing Engine Fuel Control
- 1-37 Arrangement of Flame Arrestors in Secondary Air Passage
- 1-38 Secondary Airflow Passage General Arrangement
- 1-39 Details of Secondary Airflow Passage
- 1-40 STJ227 525 Lb/Sec Turbojet
- 1-41 STJ227 525 Lb/Sec Turbojet
- 1-42 Typical Turbojet Engine. Full Afterburning.
- 1-43 Proposed Turbojet Accessory Arrangement
- 1-44 Typical Turbojet Installation Showing Effect of Increased Cant Angle
- 1-45 Estimated Performance of STF219 and STJ227 Engines at Mach 2.7 at 65,000 Feet
- 1-46 Estimated Performance of STF219 and STJ227 Engines at Mach 0.9 at 36,150 Feet
- 1-47 Estimated Performance of STF219 and STJ227 Engines at Mach 0.6 at 15,000 Feet
- 1-48 Estimated Performance of STF219 Engine After Phase IIA and After Phase IIB at Mach 2.7 at 65,000 Feet
- 1-49 Estimated Performance of STF219 Engine After Phase IIA and After Phase IIB at Mach 0.9 at 36,150 Feet
- 1-50 Estimated Performance of STF219 Engine After Phase IIA and After Phase IIB at Mach 0.6 at 15,000 Feet
- 1-51 Estimated Performance of STJ227 Engine After Phase IIA and After Phase IIB at Mach 2.7 at 65,000 Feet
- 1-52 Estimated Performance of STJ227 Engine After Phase IIA and After Phase IIB at Mach 0.9 at 36,150 Feet
- 1-53 Estimated Performance of STJ227 Engine After Phase IIA and After Phase IIB at Mach 0.6 at 15,000 Feet
- 1-54 Loading and Thrust Increment Produced by Lighting Duct Heater of STF219 Engine
- 1-55 Effect of Turbine Inlet Temperature on Engine Weight and Maximum Diameter
- 1-56 Effect of Engine Supersonic Cruise Inlet Airflow on Engine Weight and Maximum Diameter

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A

Figure
No.

- 2A-1 STF219 Fan, Intermediate Case, and High Pressure Compressor Sections
- 2A-2 STF219 Fan Aerodynamic Design Rotor One Pressure Ratio
- 2A-3 STF219 Fan Aerodynamic Design Rotor Two Pressure Ratio
- 2A-4 STF219 Fan Aerodynamic Overall Pressure Ratio
- 2A-5 STF219 Fan Aerodynamic Design Stage One Rotor Inlet Mach Number
- 2A-6 STF219 Fan Aerodynamic Design Stage One Rotor Relative Gas Angles
- 2A-7 STF219 Fan Aerodynamic Design Stage One Rotor Meridional Velocities
- 2A-8 STF219 Fan Aerodynamic Design Stage One Rotor Aerodynamic Loading ($\Delta P/q$)
- 2A-9 STF219 Fan Aerodynamic Design Stage One Rotor "D" Factor
- 2A-10 STF219 Fan Aerodynamic Design Stage One Stator Inlet Mach Number
- 2A-11 STF219 Fan Aerodynamic Design Stage One Stator Relative Gas Angles
- 2A-12 STF219 Fan Aerodynamic Design Stage One Stator Meridional Velocities
- 2A-13 STF219 Fan Aerodynamic Design Stage One Stator Aerodynamic Loading ($\Delta P/q$)
- 2A-14 STF219 Fan Aerodynamic Design Stage One Stator "D" Factor
- 2A-15 STF219 Fan Aerodynamic Design Stage Two Rotor Inlet Mach Number
- 2A-16 STF219 Fan Aerodynamic Design Stage Two Rotor Relative Gas Angles
- 2A-17 STF219 Fan Aerodynamic Design Stage Two Rotor Meridional Velocities
- 2A-18 STF219 Fan Aerodynamic Design Stage Two Rotor Aerodynamic Loading $\Delta P/q$
- 2A-19 STF219 Fan Aerodynamic Design Stage Two Rotor "D" Factors
- 2A-20 STF219 Fan Aerodynamic Design Stage Two Stator and Degv Inlet Mach Number
- 2A-21 STF219 Fan Aerodynamic Design Stage Two Stator and Degv Relative Gas Angles

Item 2A (Cont'd)

2A-22	STF219 Fan Aerodynamic Design Stage Two Stator and Degv Meridional Velocities
2A-23	STF219 Fan Aerodynamic Design Stage Two Stator and Degv Aerodynamic Loading
2A-24	STF219 Fan Aerodynamic Design Stage Two Stator and Degv "D" Factor
2A-25	STF219 Fan Aerodynamic Design Rotor Loss Coefficient Vs. Inlet Relative Mach Number
2A-26	STF219 Fan Aerodynamic Design Stator Loss Coefficient Vs. Stator Inlet Mach Number
2A-27	STF219 Fan Aerodynamic Design Rotor Incidence
2A-28	STF219 Fan Aerodynamic Stator One Incidence - Difference Between Gas Angle and Minimum Loss Angle
2A-29	STF219 Fan Aerodynamic Deviation in Excess of Carter's Rule Vs. Blade Camber*
2A-30	STF219 Fan Resonance Diagrams
2A-31	Blade Ingestion Parameter Vs. Blade Length
2A-32	Intermediate Case and Accessory Drive Arrangement (Looking Aft)
2A-33	No. 1 and No. 2 Bearing Compartment Oiling and Scavenge Schematic
2A-34	Intermediate Case Loading
2A-35	STF219 Windmilling Predicted Fan Speed When Compressor Rotates at Constant 2000 RPM
2A-36	STF219 1900°F Turbofan at Mn = 2.7 at 65,000 Feet - Estimated Windmilling Speed
2A-37	STF219 1900°F Turbofan Corrected Total Airflow Ratio for a Compressor Inlet Area Which Varies to Maintain 2000 RPM
2A-38	STF219 Variable Inlet Guide Vane and Aerodynamic Brake
2A-39	STF219 High Pressure Compressor Stage Pressure Ratio Distribution
2A-40	STF219 High Pressure Compressor Exit Profiles
2A-41	STF219 Mid-Span Incidences at Part Speed and Full Speed
2A-42	STF219 Mid-Span Incidences at Part Speed and Full Speed

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A (Cont'd)

Figure
No.

- 2A-43 STF219 Compressor Final Cord Selector
- 2A-44 STF219 Stage Reaction Distribution
- 2A-45 STF219 High Pressure Compressor Choke Parameters
- 2A-46 STF219 High Pressure Compressor Flow Path
- 2A-47 STF219 High Pressure Compressor Aerodynamic Loading
($\Delta P/q$)
- 2A-48 STF219 High Pressure Compressor "D" Factor
- 2A-49 STF219 High Pressure Compressor Rotor Stresses
- 2A-50 STF219 High Pressure Compressor Stator Stresses
- 2A-51 STF219 High Pressure Compressor Stage One Bladed Disk
Frequency at 6400 RPM
- 2A-52 STF219 High Pressure Compressor Stage Three Coupled
Blade-Disk Frequency
- 2A-53 STF219 High Pressure Compressor Stage Five Coupled
Blade-Disk Frequency
- 2A-54 STF219 High Pressure Compressor Stage Seven Coupled
Blade-Disk Frequency
- 2A-55 STF219 High Pressure Compressor Stage Four Average
Tangential Stress Level Vs. Disk Thickness
- 2A-56 STF219 Initial Design Study - Compressor Case and Stator
Construction
- 2A-57 STF219 Initial Design Study - High Pressure Compressor
Case and Stator Construction
- 2A-58 STF219 Initial Design Study - Diffuser Case Strut Location
- 2A-59 STF219 Diffuser Case Fabrication Techniques Considered in
Early Design Studies
- 2A-60 STF219 Diffuser Case Fabrication Techniques Considered
in Early Design Studies
- 2A-61 STF219 Main Combustion Chamber Section
- 2A-62 STF219 Main Combustion Chamber Section - Detailed
Section of Scoop with Film-Cooled Louvers
- 2A-63 Seals Studied for Use in STF219 Annular Combustion
Chamber
- 2A-64 STF219 Estimated Pressure Through Combustion Chamber -
Sea Level Take-off
- 2A-65 STF219 Estimated Pressures Through Combustion Chamber -
Altitude Cruise

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A (Cont'd)

Figure
No.

- 2A-66 STF219 Estimated Combustion Chamber Mach Numbers and Flow Areas
- 2A-67 STF219 Turbine Flowpath
- 2A-68 Turbine Mean Line Aerodynamics at 1900°F Cruise Condition
- 2A-69 STF219 Turbine Blade Cooling Scheme
- 2A-70 STF219 Turbine Blade Cooling Scheme Evolved from J58 Engine Experience
- 2A-71 STF219 First Stage Vane Attachment and Cooling Scheme
- 2A-72 STF219 First Stage Blade and Disk Coupled Vibrations Mode
- 2A-73 STF219 Second Stage Disk Blade Coupled Frequencies
- 2A-74 STF219 Third Stage Disk Blade Coupled Frequencies
- 2A-75 Burner Temperature Profiles Vs Average Radial Profile
- 2A-76 Vane Design - Initial (2000°F) Rating
- 2A-77 Spanwise Gradient for the Initial Engine
- 2A-78 Growth Capability for the Initial Engine
- 2A-79 Vane Design - Basic (2300°F) Rating
- 2A-80 Spanwise Gradient for the Basic Rating
- 2A-81 Growth Capability for the Basic Rating
- 2A-82 Alternate Basic Rating Engine Vane Design
- 2A-83 Spanwise Gradient for the Alternate Vane
- 2A-84 Basic Rating Capability for the Alternate Vane
- 2A-85 Blade Design for the Initial Rating
- 2A-86 Basic Rating Engine Blade Design
- 2A-87 Spanwise Gradient First STG Blade - Initial Rating
- 2A-88 Growth Capability of the Initial Rating Configuration - First Stage Blade
- 2A-89 Alternate Initial Rating Engine Blade Design - 1st Stage
- 2A-90 Spanwise Gradient for the Initial Rating Engine - First Stage Blade
- 2A-91 Growth Capability for the Initial Rating Engine - First Stage Blade
- 2A-92 Spanwise Gradient for the Basic Rating - First Stage Blade
- 2A-93 Chordwise Gradient for the Basic Rating - First Stage Blade
- 2A-94 Transient Thermal Gradients on the Turbine Development Engine

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A (Cont'd)

Figure
No.

- 2A-95 This page intentionally blank
2A-96 This page intentionally blank
2A-97 Comparison of Radial Temperature Profiles
2A-98 First Blade Stress Distribution - Initial Rating
2A-99 Second Blade Stress Distribution - Initial Rating
2A-100 Third Blade Stress Distribution - Initial Rating
2A-101 First Stage Vane Stress Distribution - Initial Rating
2A-102 Second Stage Vane Stress Distribution - Initial Rating
2A-103 Third Stage Vane Stress Distribution - Initial Rating
2A-104 Super Alloy Relaxation Curve
2A-105 Goodman Diagram, Typical Turbine Blade Material
2A-106 Goodman Diagram, Unnotched U-700, Showing Temperature
Dependence
2A-107 Midspan Centrifugal Stress
2A-108 Comparison of Creep Strength Temperature Margin
2A-109 Trailing Edge Bow Predictions and Test Values
2A-110 Weight Loss Plot for Coated B-1900 Erosion Test
2A-111 Temperature Dependence of Rate of Erosion
2A-112 Erosion Limits for Coated Turbine Materials
2A-113 First Stage Turbine Bladed-Disk Frequency
2A-114 Turbine Blade Vibratory Stress
2A-115 Stress Levels JT3C Shrouded and Unshrouded Blades
2A-116 Transient Temperatures on Convectively Cooled Blades
Showing Predicted and Actual Values
2A-117 First Stage Turbine Blade Strain Range
2A-118 First Stage Turbine Vane Strain Range
2A-119 Engine Substantiation of Fatigue Strength of Turbine Blade
Alloys
2A-120 First Stage Blade Tip Section
2A-121 First Stage Vane Tip Section
2A-122 First Turbine Vane Geometry - Initial (2000° F) Rating
2A-123 First Turbine Vane Spanwise Temperature Distribution
2A-124 First Turbine Vane Spanwise Temperature Distribution
2A-125 First Turbine Vane Midspan Local Metal Temperature
2A-126 First Vane Typical Transient Response
2A-127 First Turbine, Three Cavity Blade Section - Initial
(2000° F) Design

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A (Cont'd)

Figure
No.

- 2A-128 First Turbine, Three Cavity Blade Spanwise Temperature Distribution
2A-129 First Turbine, Three Cavity Blade Spanwise Temperature Distribution
2A-130 First Turbine, Three Cavity Blade Spanwise Temperature Distribution
2A-131 Convectively Cooled First Turbine Blade - Transient Response
2A-132 Eleven Hole Blade Midspan Cross Section
2A-133 0.020 Sawtooth Blade
2A-134 0.030 Sawtooth Blade
2A-135 Typical Cycling Test Data - High Temp. Turbine Development Test Engine
2A-136 Instrumented Blade
2A-137 Eleven Hole Blade Test Data
2A-138 0.020 Sawtooth Blade Test Data
2A-139 0.030 Sawtooth Blade Test Data
2A-140 Convectively Cooled Blades Life Limits
2A-141 Mar M200 Stress Rupture Data
2A-142 Creep Limit Temperature Margin, First Stage Vane
2A-143 Creep Limit Temperature Margin, First Stage Blade
2A-144 Creep Limit Temperature Margin, Second Stage Vane
2A-145 Creep Limit Temperature Margin, Second Stage Blade
2A-146 Creep Limit Temperature Margin, Third Stage Vane
2A-147 Creep Limit Temperature Margin, Third Stage Blade
2A-148 Transient Response First Stage Vane, 2300° F TIT
2A-149 This page intentionally blank
2A-150 Transient Response First Stage Blade, 2300° F TIT
2A-151 Strain S-N Curve for Turbine Super Alloys
2A-152 Runout Fatigue Strengths of Notched and Smooth Turbine Materials
2A-153 JT4 Vibration Surveys
2A-154 First Stage Basic Engine Vane Geometry
2A-155 Average Vane Temperature
2A-156 Local Metal Temperature (75% Span)
2A-157 First Stage Vane Transient Temperature, 2300° F TIT-SLTO

Item 2A (Cont'd)

2A-158	First Stage Basic Engine Vane Geometry
2A-159	Average Vane Temperature
2A-160	Local Metal Temperatures (75% Span) - MN 2.7
2A-161	Vane Temperature Response - Transient Accell. SLTO - 2300° F Rating
2A-162	First Stage Blade Geometry - 2300° F Rating
2A-163	Average Blade Temperatures - First Stage Blade
2A-164	Local Blade Temperatures (50% Span) - First Stage Blade
2A-165	Blade Temperature Response Transient Accell. SLTO - 2300° F
2A-166	Second Turbine Vane Geometry - 2300° F Rating
2A-167	Average Second Stage Vane Temperatures
2A-168	Second Stage Vane Maximum Metal Temperature
2A-169	Typical Transient Gradients, Second Stage Vane - Accell. SLTO - 2300° F
2A-170	Vane Design, Impingement Cooled Leading Edge and Con- vectively Cooled Trailing Edge
2A-171	Vane Design, Radial Hole Convectively Cooled Leading Edge and Convectively Cooled Trailing Edge
2A-172	Vane Design, Showerhead Leading Edge and Film Cooled Trailing Edge
2A-173	Vane Design, Detached Leading Edge and Film Cooled Trailing Edge
2A-174	Vane Design, Film Cooled Trailing Edge
2A-175	Typical Thermocoupled Vane
2A-176	STD-20 First Turbine Vane Metal Temperature
2A-176A	Podestal T. E. Vane Data - 2300° F Gas Temp
2A-177	Showerhead Vane Metal Temperature Vs Chord - 2300° F Gas Temp
2A-178	Film Cooled Airfoil Test Results
2A-179	Film Cooled Airfoil Test Results, Vane Trailing Edge Film Cooling
2A-180	Flat Plate Test Rig - Schematic
2A-181	Comparison of Metal Temperatures Basic Film Cooling Rig
2A-182	Film Cooled Blade, .008 Diameter Holes
2A-183	Film Cooled Blade, .013 Diameter Holes, 30°

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A (Cont'd)

Figure
No.

- 2A-184 Film Cooled Blade, .013 Diameter Holes, 90°
- 2A-185 Film Cooled Blade, .030 Diameter Holes, 30°
- 2A-186 JT4 Turbine Inlet Temperature Variation - High
Temperature Development Engine
- 2A-187 Convectively Cooled Turbine Blade Test Results - High
Temp Development Engine
- 2A-188 Film Cooled Turbine Blade Test Results - High Temp
Development Engine
- 2A-189 Film Cooled Blade Test Results - High Temp Development
Engine
- 2A-190 Turbine Exhaust Section
- 2A-191 Duct Heater Diffuser Schematic
- 2A-192 Duct Heater Access Panel
- 2A-193 Duct Heater Schematic
- 2A-194 Fuel-Air Ratio Vs. Power Lever Angle
- 2A-195 Balanced Flap System Schematic
- 2A-196 Secondary Air to Primary Nozzle Schematic
- 2A-197 Bellcrank Flap Synchronization Schematic
- 2A-198 Hinged Flap Seal Arrangement Looking Forward
- 2A-199 Sixteen Sided Nozzle Flap Arrangement Looking Forward
- 2A-200 Duct Heater Nozzle Linkage Longitudinal Section
- 2A-201 Duct Heater Nozzle Assembly Looking Forward
- 2A-202 Octagonal Ejector For Boeing
- 2A-203 Octagonal Ejector-Translating Shroud
- 2A-204 Cascade Ejector-Translating Shroud
- 2A-205 Rack and Pinion Details-Translating Shroud
- 2A-206 Actuation System Details-Translating Shroud
- 2A-207 Model of Translating Cascade - Cruise Position
- 2A-208 Model of Translating Cascade - Loiter Position
- 2A-209 Model of Translating Cascade - Reverse Position
- 2A-210 Model of Translating Cascade - Reverse Position
- 2A-211 Assembly of Major Components
- 2A-212 Stiff Bearing - First Mode - 124% of Rotor Speed
- 2A-213 Stiff Bearing - First Mode - 120% of Rotor Speed
- 2A-214 Number 3 Bearing Compartment
- 2A-215 Compartment Air Cooling - Number 3 Bearing
- 2A-216 Compartment Oil Flow - Number 3 Bearing

LIST OF ILLUSTRATIONS (Cont'd)

Item 2A (Cont'd)

Figure
No.

2A-217	Number 4 Bearing Compartment
2A-218	Hydrostatic Seal Schematic
2A-219	Summary of Proposed Thrust Balance Systems
2A-220	Secondary Flow Schematic - System E
2A-221	Thrust Balance Schematic - System E
2A-222	High Rotor Thrust Vs. Bias Pressure - System E
2A-223	Low Rotor Thrust Vs. Bias Pressure - System E
2A-224	High Rotor Bearing Thrust Vs. Mission Time
2A-225	Low Rotor Bearing Thrust Vs. Mission Time
2A-226	Fundamental Thrust Vs. Climb Characteristic
2A-227	Thrust Balance and Flow Schematic - System A
2A-228	Thrust Balance and Flow Schematic - System B
2A-229	Thrust Balance and Flow Schematic - System C
2A-230	Thrust Balance and Flow Schematic - System D
2A-231	Flow System Schematic - Fan Duct Diffuser Struts
2A-232	Air Bleed From Diffuser Case to Outer Case
2A-233	Typical Strut Connections for Lines
2A-234	Typical Fuel Manifold Quadrant

Item 2B

2B-1	Compressor Polytropic Efficiency Vs. Specific Flow
2B-2	Compressor Preliminary Design Loading Estimate
2B-3	Ideal Work Coefficient vs. Flow Coefficient
2B-4	STJ227 Nine Stage Compressor
2B-5	STJ227 Eight Stage Compressor
2B-6	Primary Combustion Chamber
2B-7	Burner System Pressure Loss Estimates
2B-8	Compressor Exit Velocity Profile
2B-9	Compressor Exit Velocity Profile
2B-10	Effect of Inlet Velocity on Outlet Temperature
2B-11	Average Combustor Exit Temperature Profile
2B-12	Combustor Exit Temperature Profile
2B-13	Selected Combustor Air Distribution
2B-14	Total Turbine Cooling Air Requirements

LIST OF ILLUSTRATIONS (Cont'd)

Item 2B (Cont'd)

Figure
No.

- 2B-15 First Stage Turbine Vane Heat Balance Results
- 2B-16 First Stage Turbine Vane Configuration
- 2B-17 Static Heat Transfer Results
- 2B-18 First Stage Estimated Metal Temperatures
- 2B-19 Baffle Blade Flow Characteristics
- 2B-20 Advanced Cooling Blade Scheme (Fraenkl Surface)
- 2B-21 Effect of Vortex Design on Turbine Efficiency
- 2B-22 Afterburner Cold Flow Pressure Losses
- 2B-23 Effect of Fuel/Air Ratio on Afterburner Efficiency
- 2B-24 STJ227 Cross-Section Drawing
- 2B-25 Typical Full Afterburning Turbojet Ejector
- 2B-26 Comparison of Scale Model Blow-in Door Ejector With
Goal Values
- 2B-27 Comparison of Scale Model Blow-in Door Ejector With
Goal Values
- 2B-28 Effect of Primary Nozzle Spacing on Ejector Performance
- 2B-29 Effect of Primary Nozzle Spacing on Ejector Performance
- 2B-30 Effect of Reduced Tertiary Airflow
- 2B-31 Variable Inlet Guide Diagram
- 2B-32 General Inlet Case Construction and Assembly
- 2B-33 Anti-Icing Air Flow Through Inlet Guide Vane
- 2B-34 STJ227 Compressor Section Assembly
- 2B-35 Extended Root Blade With Mechanical Damping
- 2B-36 First Rotor Resonance Diagram
- 2B-37 Third Rotor Resonance Diagram
- 2B-38 STJ227 Rotor Stiff Bearing Mode Shape
- 2B-39 JT11D-20 Rotor Stiff Bearing Mode Shape
- 2B-40 Typical JT11D-20 Descent Disk Temperatures
- 2B-41 Typical JT11D-20 Ascent Disk Temperatures
- 2B-42 Effect of Thermal Gradients on LCF Life
- 2B-43 Typical Compressor Disk Thermal Gradient (250°F ΔT)
- 2B-44 Typical Compressor Disk Thermal Gradient (130°F ΔT)
- 2B-45 Diffuser Case Construction
- 2B-46 Front Combustor Section
- 2B-47 Spark Ignitors
- 2B-48 Second Stage Turbine Vane

LIST OF ILLUSTRATIONS (Cont'd)

Item 2B (Cont'd)

Figure
No.

2B-49	First Stage Blade and Disk Cooling Passages
2B-50	First Stage Blade Cooling Scheme
2B-51	JT11D-20 First Stage Blade Airfoil Stress vs. Damper Force
2B-52	STJ227 First Stage Bladed-Disk Coupled Mode
2B-53	Turbine Exhaust Section
2B-54	Turbine Exit Guide Vane Outer Attachment
2B-55	Turbine Exit Guide Vane Attachment to Inner Case
2B-56	Number 1 Bearing Compartment Seal
2B-57	Number 2 Bearing Compartment Seal
2B-58	Number 3 Bearing Compartment Seal
2B-59	Bearing Compartments
2B-60	Hydrostatic Seal Arrangement
2B-61	Primary Nozzle Actuation System
2B-62	3TJ227 Afterburner System
2B-63	Flap Load Calculation
2B-64	Octagonal Blow-In Door Ejector-Reverser
2B-65	Cross Section Through Reverser
2B-66	Ejector Actuation System
2B-67	Reverser Targeting Patterns
2B-68	Exhaust Nozzle Configurations
2B-69	Exhaust Nozzle Hydraulic Analogy
2B-70	Sliding Bracket Assembly
2B-71	Fixed Bracket Assembly
2B-72	Hydraulic System
2B-73	Boss-to-Adapter Joint
2B-74	Integral Ferrule Joint With Sealing Gasket
2B-75	Afterburner Fuel System
2B-76	Main Fuel System
2B-77	Metallic Static Seal for Joints Subject to Moderate Thermal Shock
2B-78	Metallic Static Seal for Joints Subject to High Thermal Shock
2B-79	STJ227 Fuel, Hydraulic, and Lubrication System Schematic
2B-80	Bellows Expansion Joint
2B-81	Tube Support Clamp Stand-Off

LIST OF ILLUSTRATIONS (Cont'd)

Item 2B (Cont'd)

Figure
No.

2B-82	STJ227 Main Gearbox
2B-83	Power Lever Operation
2B-84	Main Burner Fuel Schedule
2B-85	Afterburner Fuel Schedule
2B-86	Main Fuel Pump Flow Passages
2B-87	Main Fuel Control
2B-88	Afterburner Fuel Control and Pump
2B-89	Hydraulic Pump Characteristics
2B-90	STJ227L Engine Installation (Lockheed)
2B-91	STJ227B Engine Installation (Boeing)
2B-92	Lockheed Engine Mounting System
2B-93	Boeing Engine Mounting System
2B-94	Engine Mounting System (Lockheed)
2B-95	Engine Mounting System (Boeing)
2B-96	Overhung Front Mount System (Boeing)
2B-97	STJ227 Bleed System Requirements
2B-98	Airframe Bleed Flow Requirements
2B-99	Engine Bleed Flow Requirements
2B-100	Typical Forces Acting on a Turbojet Rotor
2B-101	Load Action on a Bearing
2B-102	STJ227 Flow Schematic for Engine Thrust Balance and Cooling
2B-103	STJ227 Oil Flows
2B-104	Characteristic Heat Flux Trends
2B-105	Heat Transfer Calculations
2B-106	Metal Temperature Heat Transfer Results
2B-107	STJ227 Weight Breakdown
2B-108	Specific Installation Weight

LIST OF ILLUSTRATIONS (Cont'd)

Item 3

Figure
No.

- 3-1 Two-Stage Fan Design Specifications
- 3-2 Design Layout of Two-Stage Fan Rig
- 3-3 Detail of Single-Stage Fan Rotor Assembly with Welded
Mid-Span Shrouds (Phase II-A)
- 3-4 Rubber Model of Typical Shrouded Compressor Rotor
Operating in Severe Resonance
- 3-5 Test of Rubber Model of Compressor Rotor Incorporating
Segmented Shroud
- 3-6 Detail of Single-Stage Fan Rotor Assembly with Paired-
Welded Mid-Span Shrouds (Phase II-B)
- 3-7 Performance Comparison of Third and Fourth Builds of the
Multistage Compressor
- 3-8 Performance Comparison of Fourth and Fifth Builds of the
Multistage Compressors

Item 4

- 4-1 Sketch of Annular Burner Configuration
- 4-2 Two-Dimensional Annular Segmental Burner Rig Assembly
- 4-3 SST Two-Dimensional Annular Burner Rig
- 4-4 Large-Scale Full-Annular Burner Rig Mounted in Test Stand
- 4-5 Large-Scale Full-Annular Burner Rig with Exit Duct
Removed
- 4-6 Large-Scale Full-Annular Burner Rig
- 4-7 Schematic of Large-Scale Full-Annular Burner Rig
- 4-8 Typical Burner for Ram Induction 72° Straight Sector
Burner Rig
- 4-9 72-Degree Straight Sector Test Rig Mounted in Test Stand
- 4-10 Minimum Lighting Curve for the 2-D Burner
- 4-11 Partial Annular Burner Schematic
- 4-12 Partial Annular Burner
- 4-13 Cross-Section of Hot-Cone Burner
- 4-14 Location of Inlet Trips on Full-Annular Burner Rig
- 4-15 Diffuser Inlet Pressure Profile on Full-Annular Burner Rig
- 4-16 Effect of Inlet Profile on Δ TVR
- 4-17 Variation of Burner Exit Temperature Profile with
Temperature Profile with Temperature Rise

LIST OF ILLUSTRATIONS (Cont'd)

Item 4

Figure
No.

- 4-18 Effect of Inlet Pressure Profile on Burner Exit Temperature Profile
- 4-19 Full-Annular Burner Rig Igniter Location
- 4-20 Rich and Lean Operating Limits
- 4-21 Minimum Lighting Curve - JP-5 Fuel
- 4-22 Model 2-1 Ram Induction Burner
- 4-23 Burner Outlet Temperature Distribution of the Model 2-7 RPM Induction Burner
- 4-24 Temperature Profile of the Model 2-7 Ram Induction Burner
- 4-25 Burner Outlet Temperature Distribution of the Model 2-7 Ram Induction Burner
- 4-26 Temperature Profile of the Model 2-7 Ram Induction Burner
- 4-27 Burner Outlet Temperature Distribution of the Model 2-7 Ram Induction Burner
- 4-28 Temperature Profile of the Model 2-7 Ram Induction Burner
- 4-29 Percent of Average Velocity Head at Burner Diffuser Inlet Versus Distance from I. D. Wall
- 4-30 Effect of Inlet Velocity Profile on Outlet Temperature Profile of Model 2-7 Burner
- 4-31 Model 3-1 Ram Induction Burner
- 4-32 Model 3-2 Ram Induction Burner
- 4-33 Test Stand for Model 4 120-Degree Segment Ram Induction Burner
- 4-34 Model 4 120-Degree Segment Ram Induction Burner
- 4-35 Model 4 120-Degree Segment Ram Induction Burner

Item 5

- 5-1 Overall View of Basic Flow Rig Test Installation
- 5-2 Schematic Diagram of Basic Flow Rig
- 5-3 Basic Flow Rig - Test Plate Holder and Test Plate
- 5-4 Film Cooled Plates Tested in Basic Flow Rig
- 5-5 Cooling Effectiveness Test Results Obtained in Basic Flow Rig Using Wooden Plate - Low Temp.

PAGE NO. XXX

CONFIDENTIAL

UNCLASSIFIED BY 3000 HRS/MS/MS
DECLASSIFIED DATE 12-15-2000
BY 3000 HRS/MS/MSTHIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
DEFENSE OF THE UNITED STATES OF AMERICA AND THE
SECURITY OF THE UNITED STATES. IT IS THE POLICY OF THE
GOVERNMENT OF THE UNITED STATES OF AMERICA TO
REPRODUCE AND TRANSMIT THIS INFORMATION IN THE
FORM OF A MICROFILME OR MICROFICHE.

LIST OF ILLUSTRATIONS (Cont'd)

Item 5 (Cont'd)

Figure No.	
5-6	Comparison of Measured Temperature Distribution with Analytically Predicted Data - High Temp.
5-7	Metal Temperatures Measured in Typical Test with Film Cooled Plate - Uncorrected for Radiation Effects
5-8	Cooling Effectiveness Data for Plate #3
5-9	Cooling Effectiveness Data for Plate #4
5-10	Cooling Effectiveness Data for Plate #5
5-11	Overall View of High Pressure Cascade Rig Test Installation
5-12	Schematic Diagram of Heat Transfer Cascade Rig
5-13	Instrumented Pressure Distribution Vane #1 Used for Measuring Airfoil Pressure Profiles in High Pressure Cascade Rig
5-14	Pressure Distribution Vane #2 Used for Measuring Airfoil Profiles in High Pressure Cascade Rig
5-15	Instrumented Showerhead Vane After Testing in High Pressure Cascade Rig at 30, 45, 175 psia, and 2300° F Main Gas Stream Conditions
5-16	Schematic of Showerhead Vane
5-17	Instrumented TD Nickel Vane Tested in High Pressure Cascade Rig
5-18	Schematic of TD Nickel Vane Tested in High Pressure Cascade Rig
5-19	Instrumented Wafer #1 Vane Tested in High Pressure Cascade Rig
5-20	Front and Rear Views of Segments Used in Assembly of Wafer #1
5-21	Instrumented Pedestal Trailing Edge Vane Tested in High Pressure Cascade Rig at 30, 45, 175 psia, 2300° F Gasstream Conditions
5-22	Schematic of Pedestal Trailing Edge Vane
5-23	Wafer #2 Vane Tested in High Pressure Cascade Rig at 45 psia, 2300° F Gasstream Conditions
5-24	Front and Rear Views of Segments used in Wafer Vane #2
5-25	Pierced Sheet Leading Edge Vane Tested in High Pressure Cascade Rig at 30, 45 Psia, 2300° F Gasstream Conditions

LIST OF ILLUSTRATIONS (Cont'd)

Item 5 (Cont'd)

Figure
No.

- 5-26 Schematic of Pierced Sheet Vane Tested in High Pressures Cascade Rig
- 5-27 Segments Used in Assembly of Wafer Vane #3
- 5-28 Segments Used in Assembly of Wafer Vane #4
- 5-29 Pressure Profile Data Derived from Tests
- 5-30 Metal Temperatures Measured on Showerhead Vane in High Pressure Cascade Rig. Main Gasstream Temp 2300° F
- 5-31 Metal Temperatures Measured on Pedestal Trailing Edge Vane in High Pressure Cascade Rig. Main Stream Temp. 2300° F - All Cases
- 5-32 Metal Temperatures Measured on Wafer #1 Vane in High Pressure Cascade Rig. Main Stream Temp. 2300° F - All Cases
- 5-33 Metal Temperatures Measured on Wafer #2 Vane in High Pressure Cascade Rig. Mainstream Temp. 2300° F - All Cases
- 5-34 Metal Temperatures Measured on Wafer Vane #3. Main-stream Temp. 2300° F
- 5-35 Metal Temperatures Measured on Wafer #4. Mainstream Temp. 2300° F
- 5-36 Metal Temperatures Measured on Wafer #4. Mainstream Temp. 2300° F
- 5-37 Metal Temperatures Measured on Wafer #4. Mainstream Temp. 2300° F
- 5-38 Front View of Choked Cascade - Thermal Shock Rig
- 5-39 Schematic Diagram of Choked Cascade Thermal Shock Rig
- 5-40 Side View of Test Installation for High Temp. Turbine Development Engine Showing Heated Inlet
- 5-41 Schematic Diagram of High Temp. Turbine Development Engine Installation
- 5-42 JT12 Engine Used to Supply Blade and Vane Cooling Air to High Temp Turbine Development Test Engine
- 5-43 Overall View of High Temp. Turbine Development Engine Test Installation
- 5-44 Film Cooled Turbine Blade with Three Rows of .013" Diameter Holes at 90° to L. E.

PAGE NO XXXII

CONFIDENTIAL

REPRODUCTION AT 5 YEAR INTERVAL
 DELAYED AFTER 10 YEARS
 500 PER CENT

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
 NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
 MEANING OF THE ESPIONAGE LAWS, TITLE 18, U. S. C.
 SECTION 793 AND 794, AND IS HEREBY REPRODUCED IN
 FULL AT THE DISCRETION OF THE NATIONAL ARCHIVE TO
 BE AVAILABLE FOR REPRODUCTION AT 5 YEAR

LIST OF ILLUSTRATIONS (Cont'd)

Item 5 (Cont'd)

Figure
No.

- 5-45 Film Cooled Turbine Blade with Three Rows of .013" Diameter Holes at 30° to L. E.
- 5-46 Film Cooled Turbine Blade with Three Rows of .008 Diameter Hole at 90° to L. E.
- 5-47 Film Cooled Turbine Blade with Five Rows of .008 Diameter Holes at 90° to L. E.
- 5-48 Film Cooled Turbine Blade with Five Rows of .008 Diameter Holes at 30° to L. E.
- 5-49 Film Cooled Turbine Blade with Single Row of .030 Diameter Holes at 30° to L. E.
- 5-50 Film Cooled Turbine Blade with Three Rows of .030 Diameter Hole at 30° to L. E.
- 5-51 Cutaway Sections of Convectively-Cooled Blades Showing Configuration of Cooling Passages Showing Left to Right - .020 Sawtooth Blades, .030" Sawtooth Blade, Ribbed Core Blade, and Eleven Hole Blade
- 5-52 Instrumented Turbine Vane with Shielded Thermocouples Installed on Leading Edge
- 5-53 Instrumented Turbine Vane with Mid Chord Leading and Trailing Edge Thermocouples for Measuring Metal Temp.
- 5-54 Typical Temperature Readings Recorded on Convectively Cooled Sawtooth Blade During Cyclic Endurance Test
- 5-55 Typical Temperature Readings Recorded on Film Cooled Blade During Cyclic Endurance Test
- 5-56 Blade Growth Measurement Recorded During 200 Hour Steady State Endurance Test at 2300° F Turbine Inlet Temperature and 1200° F Cooling Air Temperature
- 5-57 PWA 663 Ribbed Core Blade with Stress Rupture Cracks Following 150 Hours of Steady State Endurance Testing at 2300° F T. I. T.
- 5-58 Cascade Rig Used for Vane Performance Testing
- 5-59 Close Up View of Vane Cascade Test Section
- 5-60 Schematic Diagram of Film Cooled Vane Performance Cascade Rig
- 5-61 Basic Vane Used for Evaluating Performance of Film Cooled Vanes
- 5-62 Front View of Vane Pack Showing Pressure Surface Cooling Air Slots

PAGE NO XXXiii

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS

DUC OR 92809

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U. S. C.
SECTION 793 AND 794. THE TRANSMISSION OR THE
DISCLOSURE OF ITS CONTENTS IN ANY MANNER TO
AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

LIST OF ILLUSTRATIONS (Cont'd)

Item 5 (Cont'd)

Figure
No.

- 5-63 Rear View of Vane Pack Showing Pressure Surface Cooling Air Slots
- 5-64 Cross-Section of Vanes with Leading Edge Holes Only Open Used in Vane Performance Tests
- 5-65 Schematic of Film Cooled Turbine Blade Performance Rig Installation
- 5-66 Schematic of Turbine Performance Rig Test Section Showing Path of Main Gastream, Cooling and Leakage Air
- 5-67 Schematic of Turbine Performance Rig Instrumentation
- 5-68 Overall View of Turbine Blade Performance Rig
- 5-69 Cross-Section of Initial Blades Tested During Phase II-B Performance Tests
- 5-70 Typical Blade with Leading Edge Slots and Holes Near Trailing Edge on Suction Surface Tested in Performance Rig. Holes on Pressure Surface Near T. E. are Plugged
- 5-71 Typical Blade with Holes Near Trailing Edge on Suction Surface Tested in Performance Rig. Holes on Pressure Surface Near T. E. are Plugged
- 5-72 Typical Blade with Holes Near Trailing Edge on Pressure Surface Tested in Performance Rig. Holes on Suction Surface Near T. E. are Plugged
- 5-73 Typical Blade with Leading Edge Slots only Open Tested in Performance Rig. Holes on Both Sides Near T. E. are Plugged
- 5-74 Cross-Section of Blade with Suction and Pressure Surface Slots at Leading and Trailing Edge
- 5-75 Typical Blade with Suction and Pressure Surface Slots at Leading and Trailing Edges
- 5-76 Change in Turbine Efficiency Due to Leakage Only
- 5-77 Change in Turbine Efficiency with Various Blade Cooling Schemes as a Function of Cooling Flow
- 5-78 Effect of Cooling Hole Area on Cooling Air Flow and Pressure
- 5-79 Creep Machine Used For Cyclic Stress Rupture Tests
- 5-80 Cyclic Stress Rupture Specimen
- 5-81 Cyclic Stress Rupture Data on PWA 663, PWA 659, and PWA 657 Alloys
- 5-82 Low Cycle Fatigue Specimen

LIST OF ILLUSTRATIONS (Cont'd)

Item 5 (Cont'd)

Figure No.	
5-83	Comparison of LCF Strength of Smooth and Notched Cast Nickel Base Alloys at 1400° F
5-84	Low Cycle Fatigue Test Data
5-85	Creep and Room Temperature Tensile Properties for As-Cast PWA 657 Alloy
5-86	Creep Rupture Properties of Cast PWA 658 Nickel Base Alloy - PWA 47 Coated and Aged at 1600° F
5-87	Stress Versus Time to Produce One Percent Creep for PWA 658 Jo Coated and Aged at 1600° F (12)
5-88	Stress Versus Rupture Life for PWA 658 Jo Coated and Aged at 1600° F (12)
5-89	Combined Stress-Fatigue Properties of PWA 659 and PWA 663 Alloys at 1700° F
5-90	Modified Goodman Diagram for Smooth PWA 663 Alloy Tested at 1700° F
5-91	Hrain Cycle Data on PWA 657, PWA 659 and PWA 663 Materials

Item 6

6-1	Efficiency of Small Scale Duct Heater Rig
6-2	Stability Limits of Cross Jet Flameholder
6-3	Cold Flow Pressure Losses of Jet Flameholder System
6-4	Jet Flameholder with Primary and Secondary Spraybars Installed in Test Rig
6-5	Schematic Diagram of Cross Jet Flameholder Installed in 12-inch by 16-inch Test Rig
6-6	Ignition Characteristics of 20-Joule and 4-Joule Spark Ignition Systems
6-7	Effect of Location of Primary Spraying on Combustion Efficiency
6-8	Effect of Primary Spraying Misalignment
6-9	Secondary Fuel Injection Patterns
6-10	Combustion Efficiency Obtained with Several Secondary Fuel Spray Patterns

LIST OF ILLUSTRATIONS (Cont'd)

Item 6 (Cont'd)

Figure
No.

- 6-11 Combustion Efficiency Obtained with the Addition of
Secondary Fuel in Region of Primary System and Uniform
Flameholder Coverage
- 6-12 Effect of Fuel Mixing Length on Combustion Efficiency
- 6-13 Effect of Splash Plate Gap Width on Combustion Efficiency
- 6-14 Effect of Combustion Chamber Length on Combustion
Efficiency
- 6-15 Effect of Bleed Flow Varied by Changing the Pressure Ratio
on Combustion Efficiency
- 6-16 Effect of Bleed Flow Varied by Changing Hole Size on
Combustion Efficiency
- 6-17 Single-Bar Jet Flameholder Perpendicular to Conventional
Vee Gutter
- 6-18 Performance of Single-Bar Jet Flameholder with and without
Perpendicular Vee Gutter
- 6-19 Porous Cooling Liner Constructed of 0, 120, and 230 SCFM
N-155 Material and Installed in 12-Inch by 16-Inch Rig
- 6-20 Combustion System Used for Testing Porous Cooling Liner
- 6-21 Water Table Used to Determine Jet Flameholder Flow
- 6-22 Water Table Stream Lines Around Flameholders
- 6-23 Effect of Trailing Edge Geometry on Combustion Efficiency
- 6-24 Model E Ram Induction Duct Burner
- 6-25 Blow-Out Characteristics of Model E Ram Induction Duct
Burner
- 6-26 Effect of Mach Number on Blow-Out Characteristics of
Model E Ram Induction Duct Burner
- 6-27 Model D Ram Induction Duct Burner
- 6-28 Blow-Out Characteristics of Model D Ram Induction Duct
Burner
- 6-29 Combustion Efficiency of Ram Induction Duct Burners
- 6-30 Cross-Jet Flameholder Tested in Full-Scale Duct Heater Rig
- 6-31 Full-Scale Duct Heater Rig Test Results with Primary Fuel
Injection
- 6-32 Full-Scale Duct Heater Rig Test Results with Primary and
Secondary Fuel Injection
- 6-33 Cold Pressure Loss of Full-Scale Duct Heater Rig
- 6-34 Effect of Bleed Flow on Efficiency of Full-Scale Duct Heater
Rig

PAGE NO. XXXVI

CONFIDENTIAL

DOWNGRADED AT 2 YEAR INTERVALS
UNCLASSIFIED AFTER 16 YEARS
DDC: DM 188010THIS DOCUMENT CONTAINS INFORMATION RELATING TO
THE NATIONAL DEFENSE OF THE UNITED STATES OF AMERICA
WHICH IS UNCLASSIFIED EXCEPT WHERE SHOWN
OTHERWISE BY THE NATIONAL DEFENSE AGENCY
OR ITS SUCCESSORS. IT IS NOT TO BE RELEASED TO
THE PUBLIC OR TO ANY OTHER PERSON OR ENTITY
EXCEPT AS AUTHORIZED BY THE NATIONAL DEFENSE AGENCY.

LIST OF ILLUSTRATIONS (Cont'd)

Item 6 (Cont'd)

Figure
No.

- 6-35 Duct Heater Test Results Obtained with Full-Scale and Small-Scale Rigs
- 6-36 Full-Scale Duct Heater Rig
- 6-37 Full-Scale Duct Heater Rig Inlet Diffuser Assembly
- 6-38 Full-Scale Duct Heater Rig Showing Jet Flameholder
- 6-39 Jet Flameholder and Spray Rings in Full-Scale Duct Heater Rig

Item 7

- 7-1 STF-219 Engine Operating Conditions Used for Exhaust Nozzle Models
- 7-2 Co-Annular Blow-In-Door Ejector
- 7-3 Effect of Internal Shroud Geometry at Supersonic Cruise
- 7-4 Effect of Internal Shroud Geometry at Sea Level Static with Maximum Duct Heat
- 7-5 Typical Effects of Shroud Spacing Ratio for Subsonic Conditions
- 7-6 Effect of Internal Shroud Geometry at Subsonic Cruise (Mach 0.9)
- 7-7 Effect of Blocking Two of Twelve Blow-In-Doors
- 7-8 Effect of Internal Shroud Geometry at Transonic Acceleration (Mach 1.2)
- 7-9 Scale Model STF219 Blow-In-Door Ejector Performance
- 7-10 Schematic Thrust Reverser
- 7-11 Thrust Reverser Blow-In-Door Blockage
- 7-12 Effect of Blow-In-Door Blockage
- 7-13 Effect of Reverser Bleed Flow
- 7-14 Typical Effect of Flight Mach Number with Four Doors Blocked
- 7-15 Typical Effect of Flight Mach Number with Five Doors Blocked
- 7-16 Influence of Trailing Edge Flap Position on Mach Number with Spoiled Flow
- 7-17 Effect of Trailing Edge Flap Position on Mach Number for 20% and 30% Bleed
- 7-18 Simulated Free Flap Nozzle Performance

LIST OF ILLUSTRATIONS (Cont'd)

Item 8

Figure No.

- 8-1 Airplane Engine Noise Power (Watts) Generated at Maximum Static Thrust
- 8-2 Lockheed Two-Dimensional Inlet
- 8-3 Boeing Axisymmetric Inlet with Partially Expanded Center-body
- 8-4 Results of Noise Tests
- 8-5 Effect of Duct Flow Mach Number on Inlet Noise for Axisymmetric Ducts
- 8-6 Inlet Noise and Throat Mach Number as a Function of Duct Airflow
- 8-7 Effect of Duct Flow Mach Number on Inlet Noise for Two-Dimensional Inlets
- 8-8 Effect of Feltmetal Acoustic Lining Length on Inlet Noise Reduction
- 8-9 Effect of Inlet Duct Airflow Mach Number on the Acoustic Effectiveness of Feltmetal Duct Lining
- 8-10 Comparison of Inlet Noise Spectra for Feltmetal Liners and Solid-Walled Liners
- 8-11 Noise Attenuation vs. Frequency for a Feltmetal Lined Inlet Duct
- 8-12 Reverberation Chamber Test Facility
- 8-13 Fan Discharge Duct Model for Testing Sound Absorption Materials
- 8-14 Specifications for Noise Attenuation Treatments Tested in Fan Discharge Duct Model
- 8-15 Insertion Loss with Fan Duct Treatments A, B and C
- 8-16 Insertion Loss with Fan Duct Treatments C, D and E
- 8-17 Overall Diagram of Anechoic Chamber Test Facility
- 8-18 Blow-In-Door Ejector Nozzle Models for Noise Tests without Duct Heating
- 8-19 Blow-In-Door Ejector Nozzle Models for Noise Tests with Duct Heating
- 8-20 Scalloped Primary Nozzle
- 8-21 Scalloped Primary Nozzle and Duct Nozzle Assembly
- 8-22 Tantalum Alloy Turbojet Nozzle Model
- 8-23 Turbojet Nozzle Model with Water-Cooled Jacket Simulating Blow-In Doors
- 8-24 Turbojet Nozzle with Model of Ejector Used on YF-12A Aircraft

CONFIDENTIAL

PWA-2600

LIST OF ILLUSTRATIONS (Cont'd)

Item 8 (Cont'd)

Figure
No.

- 8-25 Test Stand Installation of Turbojet Nozzle with Model of Ejector Used on YF-12A Aircraft
- 8-26 Test Stand Installation of Turbojet Nozzle with Ejector Geometrically Similar to STF219 But Scaled to the Turbo-Jet Model
- 8-27 Water-Cooled Channel Ejectors
- 8-28 STF219 Tantalum-Alloy Primary Nozzle Model
- 8-29 STF219 Tantalum-Alloy Duct Nozzle Model
- 8-30 STF219 Nozzle Model with Water Jacket Simulating Blow-In Doors
- 8-31 STF219 Nozzle Model Assembly with Ejector
- 8-32 Model Turbofan Nozzle Noise Test Results without Duct Heating for Configurations (1, 6, 0) and (1, 6, 1)
- 8-33 Model Turbofan Nozzle Noise Test Results without Duct Heating for Configurations (1, 6, 1), (1, 6, 3) and (1, 6, 4)
- 8-34 Model Turbofan Nozzle Noise Test Results without Duct Heating for Configurations (1, 6, 1), (1, 6, 5) and (1, 6, 6)
- 8-35 Model Turbofan Nozzle Noise Test Results without Duct Heating for Configurations (3, 5, 0) and (3, 5, 1)
- 8-36 Model Turbofan Nozzle Noise Test Results without Duct Heating for Configurations (1, 6, 0) and (3, 5, 0)
- 8-37 Model Turbofan Nozzle Noiser Test Results without Duct Heating for Configurations (1, 6, 1) and (3, 5, 1)
- 8-38 Model Turbojet Nozzle Noise Test Results for Configurations (8, 0, 0), (8, 0, 7) and (8, 0, 8)
- 8-39 Model Turbofan Nozzle Noise Test Results with Duct Heating for Configurations (2, 3, 0) and (2, 3, 1)
- 8-40 Model Turbofan Nozzle Noise Test Results with Duct Heating for Configurations (2, 3, 0), (2, 3, 3) and (2, 3, 4)
- 8-41 Model Turbofan Nozzle Noise Test Results with Duct Heating for Configurations (2, 3, 1), (2, 3, 3) and (2, 3, 4)
- 8-42 J57 Afterburning Engine Used for Baseline Tests
- 8-43 Results of J57 Afterburning Engine Noise Tests; Baseline Versus Predicted Noise Levels
- 8-44 Full-Scale J57 Afterburning Engine with Basic STF219-Type Blow-In-Door Ejector
- 8-45 Results of Full-Scale Blow-In-Door Ejector Noise Tests; Basic STF219-Type Ejector

PAGE NO. XXXIX

DOWNGRADED AT 3 YEAR INTERVALS
DECLASSIFIED AFTER 10 YEARS
DDO ON 12-14-90

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE SPYING LAWS, TITLE 18, U.S.C. IT IS EXTENDED TO AND FOR THE TRANSMISSION OF THE INFORMATION OR TO DISCLOSE IT TO ANY PERSON IN AN UNAUTHORIZED MANNER IS PROHIBITED BY LAW.

CONFIDENTIAL

LIST OF ILLUSTRATIONS (Cont'd)

Item 8 (Cont'd)

Figure No.	
8-46	Full-Scale Blow-In-Door Ejector with Air-Scoops in Ejector Shroud
8-47	Results of Full-Scale Blow-In-Door Ejector Noise Tests; Air-Scoops in Ejector Shroud
8-48	Full-Scale Blow-In-Door Ejector with Mixers Fully Extended
8-49	Results of Full-Scale Blow-In-Door Ejector Noise Tests; Ejector with Mixers Extended
8-50	Full-Scale Blow-In-Door Ejector Relocated Eight Inches Forward
8-51	Results of Full-Scale Blow-In-Door Ejector Noise Tests; Ejector Relocated Eight Inches Forward
8-52	Results of Full-Scale Blow-In-Door Ejector Noise Tests; Ejector Relocated Five Inches Forward
8-53	Results of Full-Scale Blow-In-Door Ejector Noise Tests; Ejector with One-Half Blockage or Blow-In-Door Area
8-54	Full-Scale Blow-In-Door Ejector with One-Quarter Blockage of Blow-In-Door Area
8-55	Results of Full-Scale Blow-In-Door Ejector Noise Tests; Ejector with One-Quarter Blockage of Blow-In-Door Area
8-56	J57-P-55 Engine Performance Parameters
8-57	Full-Scale Blow-In-Door Ejector Noise Tests; Typical Exhaust Noise Spectra at Maximum A/B Power
8-58	Blow-In-Door Airflow as a Function of Engine Thrust
8-59	Effect of Blow-In-Door Ejector on Near-Field Noise
8-60	Aerial View of Bradley Field Full-Scale Engine Noise Test Facility
8-61	Microphone Locations at Bradley Field Full-Scale Engine Noise Test Facility
8-62	Sound Recording Console at Bradley Field Full-Scale Engine Noise Test Facility
8-63	Automatic Sound Spectrum Analyzer
8-64	Primary Acoustic Calibration Equipment
8-65	Suppression of Take-Off Noise by Blow-In-Door Ejector; YF-12A Fly-Over Measurements
8-66	"Flapped" Nozzle and Ejector Assembly Tested by NACA
8-67	Comparison of 1500-Foot Sideline Noise From SST Engines
8-68	Maximum Power Take-Off Noise From SST Airplanes at the Three-Mile Point Before Power Cut-Back

LIST OF ILLUSTRATIONS (Cont'd)

Item 8 (Cont'd)

Figure
No.

- 8-69 Reduced Power Take-Off Noise From SST Airplanes at the
Three-Mile Point Before Power Cut-Back
- 8-70 Comparison of Take-Off Noise From SST Airplanes and
Subsonic Jets at the Three-Mile Point After Power
Cut-Back
- 8-71 SST Landing Approach Noise Levels by SAE-A-21 Calculation
Procedure for 327 Feet Altitude
- 8-72 Comparison of SST Take-Off and Approach Noise Levels

Item 9

- 9-1 Airflow Balance Downstream of Shock - Aircraft Inlet
Program
- 9-2 Calculation of Shock Position - Aircraft Inlet Program
- 9-3 Calculation of Subsonic Mach Number After Shock - Aircraft
Inlet Program
- 9-4 Calculation of Total Pressure Losses and Bypass Area
Control - Aircraft Inlet Program
- 9-5 Block Diagram of Boeing Inlet Shock Position Control
- 9-6 Block Diagram of Lockheed Approach Control Study
- 9-7 STF219 SST Engine Operation with Approach Control
- 9-8 STF219 SST Engine Approach Control
- 9-9 Estimated θ_{T2} Biasing of Zone I Minimum Fuel Flow Ratio
- 9-10 Estimated θ_{T2} Biasing of Zone I Maximum Fuel Ratio
- 9-11 Estimated θ_{T2} Biasing of Zone II Fuel Flow Ratio
- 9-12 Estimated Fuel Flow Requirements for the Duct Burner
- 9-13 Estimated θ_{T2} and δ_{T2} Biasing of N_2 for Maximum Rated
and Cruise Power
- 9-14 Estimated Control of T_{t5} During Aircraft Acceleration
- 9-15 Percent Thrust Deviation From Desired Normal Rated Value
vs. Altitude at $M_h = 0$
- 9-16 Percent Thrust Deviation From Desired Normal Rated Value
vs. Altitude at $M_h = 0.3$
- 9-17 Percent Thrust Deviation From Desired Normal Rated Value
vs. Altitude at $M_h = 0.6$
- 9-18 Descriptive Explanation of Flat Rating Control During Part
Power

CONFIDENTIAL

PWA-2600

LIST OF ILLUSTRATIONS (Cont'd)

Item 9 (Cont'd)

Figure
No.

- 9-19 1ST Low Tension Spiral Wrapped Cooled Exciter
- 9-20 1ST Peripherally Cooled Exciter
- 9-21 SST High Tension Spiral Wrapped Cooled Exciter

Item 10

- 10-1 Weibull Distribution of Test Data From M-50 CVM Ball Bearing (125 MM Bore)
- 10-2 Weibull Distribution of Test Data From M-50 CVM Ball Bearing (125 MM Bore)
- 10-3 Weibull Distribution of Test Data From M-50 CVM Ball Bearing (178 MM Bore)
- 10-4 View of 125 MM Bore M-50 CVM Bearing After 150-Hour Endurance Using PWA-524 Oil at 500°F For Cooling
- 10-5 Speed Vs. Temperature Effects with Various Cage Plating In Three 178 MM Bore Bearings
- 10-6 Speed Vs. Temperature Effects with Various Cage Plating In Three 178 MM Bore Bearings
- 10-7 Load Vs. Temperature Effects With Various Cage Plating In Three 178 MM Bore Bearings
- 10-8 Oil Flow Vs. Temperature Effects With Three 178 MM Bore Bearings Containing Different Cage Plating Under Two Load Conditions
- 10-9 Load, Speed and Oil Flow Vs. Temperature Effects With Fafnir 125 MM Bore Bearings
- 10-10 Load, Speed and Oil Flow Vs. Temperature Effects With SKF 125 MM Bore Bearings
- 10-11 View of 178 MM Bore M-50 CVM Ball Bearing Tested in PWA-524 Oil at 500°F
- 10-12 Cage From 178 MM Bore M-50 CVM Ball Bearing Tested In PWA-524 Oil at 500°F
- 10-13 Ball-Fatigue Test Rig Including (A) Lower Race, (B) Upper Race in Up Position, and (C) Cage Holder
- 10-14 Schematic of Ball-Fatigue Test Rig
- 10-15 Half-Inch Diameter M-50 CVM Material Test Balls Showing Typical Spall Failure Obtained in Ball-Fatigue Test Rig

PAGE NO. xlii

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DDC OR 50000

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C. SECTION 793 AND 794. THE TRANSMISSION OR THE RECEIVING OF IT, OR CONTENTS OR INFORMATION THEREIN OR INFORMATION THEREON IS PROHIBITED BY LAW.

LIST OF ILLUSTRATIONS (Cont'd)

Item 10 (Cont'd)

Figure No.	
10-16	Weibull Distribution of Test Data From M-50 CVM Balls In Humble WS-5251 Oil
10-17	Weibull Distribution of Test Data From M-50 CVM Balls In PWA-524R Oil
10-18	Weibull Distribution of Test Data From M-50 CVM Balls In Humble 4040E Oil
10-19	Weibull Distribution of Test Data From M-50 CVM Balls In Humble 4040 Oil
10-20	Weibull Distribution of Test Data From M-50 CVM Balls In Cellutherm 2712C Oil
10-21	Weibull Distribution of Test Data From AISI 52100 CVM Balls in Cellutherm 2712C Oil
10-22	Weibull Distribution of Test Data From AISI 52100 CVM Balls In MIL-L-7808D (PWA 521A) Oil
10-23	Diagram of Experimental Oil Pump Featuring Roller Bearings
10-24	Oil Jet Test Configuration Showing (A) Pressure Taps (B) Inlet Jet, (C) O-Ring Seal, and (D) Orifice
10-25	Oil Jet Test Rig Showing (A) Scale for Flow Measurement, (B) Pressure Drop Gauge, (C) Pressure Gauge (D) Accumulator (E) Pump, (F) Motor, (G) Temperature Meter, (H) Oil Tank, (I) Shield and (J) Oil Jet
10-26	Component Parts of Oil Seal Testing Rig Including (A) Front Cover, (B) Rig Bearing, (C) Hub, (D) Oil Jet to Seal Plate, (E) Breather Manifold, and (F) Shaft
10-27	Face Seal High-Temperature Test Rig Installation
10-28	CDJ-83 Material Carbon Run for 57.5 Hours at 40 PSI, 400°F, and 12,500 RPM
10-29	Dry-Face Seal Plate Run for 57.5 Hours at 40 PSI, 400°F and 12,500 RPM
10-30	Time Vs. Leakage Effects with a PO3XHT Carbon (.200 LIP) and Wet-Face Plate Combination
10-31	Time Vs. Leakage Effects with a CDJ-83 Carbon (.150 LIP) and Wet-Face Plate Combination
10-32	Time Vs. Leakage Effects with a PO3XHT Carbon (.200 LIP) and Wet-Face Plate Combination
10-33	Time Vs. Leakage Effects with a CDJ-83 Carbon (.150 LIP) and Wet-Face Plate Combination

LIST OF ILLUSTRATIONS (Cont'd)

Item 10 (Cont'd)

- | Figure No. | |
|------------|--|
| 10-34 | Wet-Face Seal Plate Run for 45.5 Hours at 50 PSI, 400°F and 12,500 RPM |
| 10-35 | Time Vs. Leakage Effect with a CDJ-83 (.150 LIP) Carbon Seal and Wet-Face Seal Plate Combination at Varied Pressures |
| 10-36 | Wet-Face Seal Plate Run for 89.75 Hours at 10-50 PSI, 400°F and 12,500 RPM. |
| 10-37 | Time Vs. Leakage Effect with a CDJ-83 (.150 LIP) Carbon Seal and Wet-Face Plate Combination |
| 10-38 | Time Vs. Leakage Effect with a CDJ-8 (75% Unbalance) Carbon Seal and Wet Face Plate Combination |
| 10-39 | Time Vs. Leakage Effect with a CDJ-83 (55% Unbalance) Carbon Seal and Wet-Face Seal Plate Combination |
| 10-40 | Wet-Face Seal Plate Run For 129 Hours at 10-50 PSI, 400°F, and 12,500 RPM |
| 10-41 | Time Vs. Leakage Effects with a CDJ-83 (.250 LIP) Carbon Seal and Wet-Face Plate Combination |
| 10-42 | Time Vs. Leakage Effects with a CDJ-83 (.150 LIP) Carbon Seal and Wet-Face Plate Combination |
| 10-43 | Time Vs. Leakage Effect with a CDJ-83 (.200 LIP) Carbon Seal and Wet-Face Plate Combination |
| 10-44 | Radial Groove-Type Ring Seal Layout |
| 10-45 | Ring Seal Test Shaft and Details, Including (1) Seal Plate, (2) Test Carbon Ring Seal, (3) Spacer, (4) Back-Up Plate, and (5) Spacer |
| 10-46 | Ring Seal Test Shaft Assembled, Showing (1) Back-Up Plate, (2) Test Carbon Ring Seal, and (3) Seal Plate |
| 10-47 | Schematic Diagram of Seal Material Evaluation Rig |
| 10-48 | Seal Material Test Rig Installation Showing (A) Hot Air Supply Line, (B) Rig Breather Line, (C) Breather Settlement Tank, (D) Bleed Control Valve, (E) Bearing Oil Lines, and (F) Seal Air Temperature Probe |
| 10-49 | Seal Material Test Rig Installation Showing (A) Front Breather Thermocouple, (B) Rear Breather Thermocouple, (C) Front Breather Pressure, (D) Rear Breather Pressure, (E) Front Seal Oil, (F) Rear Seal Oil, (G) Seal Air Pressure, and (H) Oil Drains |

LIST OF ILLUSTRATIONS (Cont'd)

Item 10 (Cont'd)

Figure
No.

- 10-50 Ring Seal Rig Including (1) Air Heaters, (2) High Pressure Air Lines, (3) Stand Breather Line, (4) Thermocouple Conduit, (5) Rig Breather, (6) Drive Motors (Three 14 OP), (7) Gearbox Oil Drain, (8) Two-Speed Gearboxes, (9) Rig Drain, (10) Rig Oil Supply Manifold
- 10-51 Rig Test Control Stand Including (1) Air Heater, (2) Fire Control, (3) Power Station to Control Heater, (4) Lewis Switch, (5) Potentiometer, (6) Pressure Gauges, (7) Rotometers, (8) Pressure Gages, (9) Heater Temperature Controls
- 10-52 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Stellite 6B Plate Combination at 500°F
- 10-53 Air Pressure Vs. Wear Rate with a P-2239 Carbon-Stellite 6B Plate Combination at 500°F
- 10-54 Air Pressure Vs. Wear Rate of a P-2239 Carbon-Stellite 6B Plate Combination at 1000°F
- 10-55 Air Pressure Vs. Wear Rate of a Speer 9341 Carbon-Stellite 6K Plate Combination at 500°F
- 10-56 Air Pressure Vs. Wear Rate of a Speer 9341 Carbon-Stellite 6K Plate Combination at 1000°F
- 10-57 Air Pressure Vs. Wear Rate of a Speer 9341 Carbon-Stellite 6B Plate Combination at 1000°F
- 10-58 Air Pressure Vs. Wear Rate of a Speer 9341 Carbon-Stellite 6B Plate Combination at 500°F
- 10-59 Air Pressure Vs. Wear Rate of a PO3XHT Carbon-Linde LCIC Plate Combination at 1000°F
- 10-60 Air Pressure Vs. Wear Rate of a PO3XHT Carbon-Linde LCIC Plate Combination at 500°F
- 10-61 Air Pressure Vs. Wear Rate of a PO3XHT Carbon-Chrome-Plate AMS 6323 Plate Combination at 500°F
- 10-62 Air Pressure Vs. Wear Rate of a PO3XHT Carbon-Chrome-Plated AMS 6323 Plate Combination at 1000°F
- 10-63 Air Pressure Vs. Wear Rate of a PO3XHT Carbon-Stellite 6B Plate Combination at 500°F
- 10-64 Air Pressure Vs. Wear Rate of a PO3XHT Carbon-Stellite 6B Plate Combination at 1000°F
- 10-65 Air Pressure Vs. Wear Rate of a B-2936 Carbon-Stellite 6K Plate Combination at 500°F

PAGE NO. xlv

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS

DOO ON 820710

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S. CODE, SECTION 793 AND 794, TRANSMISSION OR THE REVELATION OF ITS CONTENTS TO ANY PERSON OR UNAUTHORIZED PERSON IS PROHIBITED BY LAW

LIST OF ILLUSTRATIONS (Cont'd)

Item 10 (Cont'd)

Figure
No.

- 10-66 Air Pressure Vs. Wear Rate of a B-2936 and CDJ-83
Carbon-Stellite 6K Plate Combination at 1000°F
- 10-67 Air Pressure Vs. Wear Rate of a B-2936 and CDJ-83
Carbon-Stellite 6K Plate Combination at 500°F
- 10-68 Air Pressure Vs. Wear Rate of a USG 2800 Carbon-Stellite
6K Plate Combination at 500°F
- 10-69 Air Pressure Vs. Wear Rate of USG 2800 Carbon-Stellite 6K
Plate Combination at 1000°F
- 10-70 Air Pressure Vs. Wear Rate of a USG 2803 Carbon-Stellite
6K Plate Combination at 1000°F
- 10-71 Air Pressure Vs. Wear Rate of USG 2803 Carbon-Stellite 6K
Plate Combination at 500°F
- 10-72 Air Pressure Vs. Wear Rate of a USG 2777 Carbon-Linde
LCIC Plate Combination at 500°F
- 10-73 Air Pressure Vs. Wear Rate of a USG 2777 Carbon-Linde
LCIC Plate Combination at 1000°F
- 10-74 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Brush
NiBe Plate Combination at 500°F
- 10-75 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Brush
NiBe Plate Combination at 1000°F
- 10-76 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Brush
NiBe Plate Combination at 1000°F
- 10-77 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Brush
NiBe Plate Combination at 1200°F
- 10-78 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Clevite
300 Plate Combination at 500°F
- 10-79 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Clevite
300 Plate Combination at 1000°F
- 10-80 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Haynes
No. 25 Plate Combination at 1000°F
- 10-81 Air Pressure Vs. Wear Rate of a CDJ-83 Carbon-Haynes
No. 25 Plate Combination at 500°F

LIST OF ILLUSTRATIONS (Cont'd)

Item 11

Figure
No.

- 11-1 Identification of Domestic Jet A and Jet A-1 Fuel Samples Used in Low Oxygen-Fuel Coker Tests
- 11-2 Inspection Data on Test Fuels
- 11-3 Summary of Tests to Determine Threshold Failure Temperature of Jet A and Jet A-1 Fuel Samples in ASTM-CFR Fuel Coker
- 11-4 Summary of Tests to Determine Threshold Failure Temperature of Test Fuels in CRC Research Fuel Coker at Controlled Fuel Oxygen Conditions with 175°F Reservoir Fuel Temperature
- 11-5 Fuel Oxygen Content During Research Coker Tests with 1.0 Psia Reservoir Pressure
- 11-6 Fuel Oxygen Content During Research Coker Tests with 1.5 Psia Reservoir Pressure
- 11-7 Summary of Threshold Failure Temperatures for Test Fuels at Air Saturation and Low Fuel Oxygen Content

Item 12

- 12-1 Estimated Fuel and Oil Temperatures Versus Mission
- 12-2 Gear Pitting Fatigue Life in Hydrotreated Machine with PWA 724 Gear Material
- 12-3 Test Data Summary for Lubricant L-919 Gear Pitting Fatigue Life in Hydrotreated Machine with PWA 724 Gear Material
- 12-4 General Properties of Lubricants Tested
- 12-5 Effect of Temperature and Pressure on Evaporation Loss of L-1535 Lubricant
- 12-6 Effect of Temperature and Pressure on Evaporation Loss of L-1537 Lubricant
- 12-7 Lubricant L-1530 Oxidation-Corrosion Data at Constant Airflow

CONFIDENTIAL

CONFIDENTIAL

CONFIDENTIAL

LIST OF ILLUSTRATIONS (Cont'd)

Item 13

Figure
No.

- 13-1 Typical Boeing SST Inlet Model Distortion Plots
- 13-2 Radial Total Pressure Profiles for Boeing SST Inlet
- 13-3 Distortion Versus Percent Bypass Door Opening and Percent Supercritical Operation for the Boeing SST Inlet

Item 14

- 14-1 Grain Structure of IMI 679 Titanium Alloy
- 14-2 Time to Creep 0.1% for IMI 679 Titanium Base Alloy Compared to PWA 1202 and PWA 1203
- 14-3 Tensile Properties of IMI 679 Titanium Base Alloy Compared to PWA 1202 and PWA 1203
- 14-4 Stress Required to Produce NaCL Salt Stress-Corrosion Cracking in 100 Hours for IMI 679 Titanium Base Alloy
- 14-5 Stress Required to Produce NaCL Salt Stress-Corrosion Cracking in 300 Hours for IMI 679 Titanium Base Alloy
- 14-6 Goodman Diagram of Fatigue Life for IMI 679 Titanium Base Alloy Notched Specimens. Test Temperature 70°F
- 14-7 Fracture Toughness of IMI 679 Titanium Base Alloy - Precracked Notched Charpy Impact Specimens
- 14-8 Tensile Properties of PWA 1007 Nickel Base Alloy. Disk Specimens Machined from Trepanned Rings
- 14-9 Structure of XGCF-6 and XGCF-7 Wyman-Gordon Test Rings from PWA 1007 Waspaloy Disks
- 14-10 Structure of CGNK-119 and CJDA-38 Ladish Test Rings from PWA 1007 Waspaloy Disks
- 14-11 Structure of Heat XGCF-6 and Heat XGCF-7 Wyman-Gordon Test Rings from PWA 1007 Waspaloy Disks
- 14-12 Structure of Heat CGNK-119 and Heat CJDA-38 Ladish Test Rings from PWA 1007 Waspaloy Disks
- 14-13 Test Setup for SST Ballistic Tests with Panels Rigidly Mounted
- 14-14 Overall Test Setup for SST Ballistic Tests
- 14-15 Typical Test Setup with Containment Ring Sandwiched between Two Plates

LIST OF ILLUSTRATIONS (Cont'd)

Item 14 (Cont'd)

Figure
No.

- 14-16 Ballistic Containment Test Results
- 14-17 Design and Weight Comparison Between the Conventional
Solid Disk and the Diffusion-Bonded Disk
- 14-18 Vacuum Retort Before Furnace Cycle
- 14-19 Vacuum Retort After Furnace Cycle
- 14-20 Overall View of Vacuum Pump, Retort, and Furnace
- 14-21 Opened Retort After Furnace Cycle Showing Bonded Disk
- 14-22 A 53-Inch Diameter Retort After Furnace Cycle
- 14-23 Test Specimen Bonding Parameters
- 14-24 Diffusion-Bonded Titanium (AMS-4928) Tensile Test Pieces.
Bonded at 1650°F and at 235, 425, and 650 Psi for 1/2
Hour in Vacuum Furnace
- 14-25 Metallographic Photograph of Specimen No. 12 Showing
Grain Growth Across Bond Joint

Item 15

- 15-1 Boeing Installation - Weight Summary
- 15-2 Lockheed Installation - Weight Summary
- 15-3 Weight Breakdown for STJ227 Engine
- 15-4 Weight of Turbojet Engine Configurations
- 15-5 High-Pressure Turbine Region

PAGE 10 OF 11

CONFIDENTIAL

CONFIDENTIAL

PRATT & WHITNEY AIRCRAFT

PWA-2600

INTRODUCTION

The overall objectives of the program conducted under contract FA-SS-65-18 were to continue: the design of the STF 219 duct-heating turbofan engine, liaison with aircraft manufacturers to ensure optimum engine and ejector-reverser installation, and verification of major component performance by approximately full-scale component testing. This program was a continuation of the contractor's design and test effort on supersonic transport powerplants and was aimed at achieving further advances in engine design and component state-of-the-art over those submitted in the Phase II-A proposal for the supersonic transport engine.

In accordance with the requirements of contract FA-SS-65-18, the program was divided into 15 major fields of effort corresponding to the tasks listed in Section B of the contract work statement. These fields of effort included engine design in addition to research and development on compressors, primary combustion, turbines, augmentors, inlet and exhaust systems, noise reduction, controls and accessories, bearings and seals, and fuels and lubricants. Also, investigations were conducted on installation optimization, materials and manufacturing techniques, and supporting design considerations such as maintainability, reliability, and value engineering. A discussion of the work accomplished in each of these fields is presented in separate sections of this report in an order corresponding to the work statement items of the contract.

PAGE NO. 1

CONFIDENTIAL

ITEM 1

ITEM 1
INSTALLATION COORDINATION



CONFIDENTIAL

PWA-2600

ITEM 1 - INSTALLATION COORDINATION

OBJECTIVE

The Contractor continued to work with the airframe contractors to ensure optimum installation arrangements for the engine and ejector reverser in the airframe. Engine cycle studies were also included in this work.

A. INSTALLATION COORDINATION MEETINGS
WITH AIRPLANE COMPANIES1. INSTALLATION COORDINATION WITH THE BOEING AIRPLANE
COMPANYa. Introduction

Six coordination meetings were held between the Contractor and the Boeing Airplane Company during Phase IIB. The most significant topics discussed in the course of these meetings are summarized below:

- Performance. IBM performance decks for both turbofan and turbojet were given to Boeing. Changes were incorporated in these programs, at Boeing's request, to enable more efficient use to be made of their computer time.
- Ejector concepts. Improved mechanical and aerodynamic ejector concepts were studied, the most recent of which is a sliding shroud ejector (a fixed shroud ejector was used during Phase IIA).
- Noise. Information was exchanged several times, and Boeing witnessed a full-scale noise test in which a J57 afterburning engine, using a "boiler plate" SST type ejector, was run.
- Engine-to-inlet compatibility. Various constants for the Contractor's analog computer program for studying engine-to-inlet compatibility were supplied to Boeing for both the turbofan and turbojet.

PAGE TWO - 1 - 1

CONFIDENTIAL

APPROVED BY THE BOARD OF
DIRECTORS OF THE AIR FORCE
ON 10/10/60

CONFIDENTIAL

CONFIDENTIAL

PRATT & WHITNEY AIRCRAFT

PWA-2600

- Turbine inlet temperature. The Contractor's turbine program was reviewed several times during Phase IIB. Data, hardware, and future planning were discussed.
- Weight. Weights were updated periodically. Weights for a specific installation were supplied to Boeing with each installation drawing.
- Engine-to-wing mating. A reduction in the base drag in the region between the engine and the wing at the wing trailing edge was made. Compatibility with Boeing's latest installation requirements was achieved.

The above items, with the exception of engine-to-wing mating, are covered in detail elsewhere in this report. This section of the report will, therefore, be devoted to a discussion of the engine-to-wing mating work done during Phase IIB.

b. Summary of Engine-to-Wing Mating Work

A reduction in base drag was accomplished by changing the shape of the ejector from round to octagonal, and by moving the ejector cant point aft relative to its Phase IIA position.

Figure 1-1 shows a comparison between the Phase IIA and Phase IIB configurations. Figure 1-2 shows the benefits obtained by moving the ejector cant point aft.

c. Nomenclature

The following nomenclature is used in describing the engine-to-wing mating work.

- Equivalent diameter. An octagonal ejector is used to eliminate base drag. As it is cumbersome to describe the size of an octagon, i.e., by dimensions across flats and corners, the terminology "equivalent diameter" or "equivalent round" has been adopted. This expression refers to the diameter of a round ejector which has the same geometric area as the particular octagonal ejector under discussion.
- Cant point. For all intents and purposes, the plane through which the ejector is canted intersects the engine center line at a point about which the ejector is a body of revolution. This point is called the ejector cant point. The location of

CONFIDENTIAL

PWA-2600

the cant point influences the following:

- Ejector cant angle
- Position of the ejector relative to the wing.

d. Background

(1) Selection of Cant Point Location

For a given size engine and ejector combination, and for a given exhaust gas target point, it is advantageous to move the cant point aft. As the cant point moves aft, the ejector cant angle decreases while the ejector stays close to the trailing edge of the wing with very little interruption of internal wing structure. This is illustrated in Figure 1-2. It is desirable to have a decreased cant angle because of ejector internal aerodynamic and mechanical reasons. It is also desirable to have the ejector close to the wing because of the consequent reduction in base drag, and simpler wing mating.

In the course of studying how to obtain these desirable features, the ejector was canted in three different places. In Phase IIA the ejector was canted at the rear mount plane. Early in Phase IIB, the cant point was moved aft to the rear face of the turbofan primary nozzle to determine the effect on base drag and ejector cant angle. Later in Phase IIB, the cant point was moved slightly forward to a plane which passes through the throat of the fan nozzle (or the afterburner nozzle in a turbojet). This latter position assured symmetry downstream of the nozzle choke point. Boeing selected this cant point as the one which best suited their installation. In each of the foregoing cases, however, the canting results in some non-symmetry upstream of the cant point which will impose additional mechanical and aerodynamic complexity.

(2) Mounts, Tailflaps, and Inlet Extensions

As the installation progressed, the engine (and mounts) were moved rearward to reduce base drag. Finally, in order to get the variable exhaust nozzle (tailflaps) entirely out from under the wing where it could adversely effect base drag, the ejector was positioned relative to the wing such that the hinge point of the tailflaps was either in line with or aft of the wing trailing edge. Since the Boeing inlet position was fixed relative to the wing, the engine had to fill the space between the ejector (as positioned by the tailflap hinge point) and the inlet. The shorter turbofan required an extension or spacer between the engine inlet and the Boeing inlet, see Figure 1-1. The longer turbojets with full afterburners did not require an inlet extension and, in some cases, they were long enough to necessitate moving the tailflap hinge point aft of the wing trailing edge.

CONFIDENTIAL

CONFIDENTIAL

PWA-2600

(3) Boeing's Installation Requirements

The differences between the installation drawings presented later in this section reflect the changes in Boeing's installation requirements as well as changes to the engine. As detailed design studies progressed, various aspects of the powerplant, including length and diameter, were revised. Close coordination with Boeing was essential to ensure that these revisions were compatible with the frequently changing airframe requirements.

The installation drawings were based on the latest Boeing requirements in the following areas:

- Wing contour. Wing cross section at the outbound nacelle location was provided by Boeing.
- Inlet position relative to the wing. This was changed as a function of the inlet flow field.
- Exhaust gas target point. This was fixed to be compatible with good cruise performance and by the location of the horizontal stabilizer.
- Permissible limits of mount locations. These limits were supplied by Boeing from time-to-time based on the latest wing configuration.
- Thrust reverser targeting requirements. Reverse thrust requirements and the possibility of re-ingestion were taken into consideration.
- Ejector position relative to the wing at the wing trailing edge. This position was varied from time to time depending on its influence on the favorable interference effect between the engine and the wing.

As a rule, the inlet, the exhaust gas target point, and the position of the ejector variable tailflap hinge point were all fixed relative to the wing. The objective was to fit the engine to these points using extensions, canting, etc., in such a manner as to result in the minimum amount of interruption of wing internal structure.

(4) Ejector Concepts

Improved aerodynamic and mechanical concepts were studied during

CONFIDENTIAL

REPRODUCED AT 1 YEAR AFTER AID
PERM. COPY 100-10

THIS DOCUMENT CONTAINS INFORMATION OF A CONFIDENTIAL NATURE AND IS NOT TO BE RELEASED TO THE PUBLIC OR TO OTHERS WITHOUT THE WRITTEN AUTHORIZATION OF THE PRATT & WHITNEY COMPANY

CONFIDENTIAL

PWA-2600

Phase IIB. These studies resulted in the sliding shroud ejector (a fixed shroud ejector was used in Phase IIA). The sliding shroud ejector uses more efficiently the energy available from the engine and secondary air streams, and reduces the minimum wrap of the ejector around the engine. The sliding shroud ejector also has a higher L/D than the fixed shroud ejector. Its longer length for a given diameter, together with the translating shroud capability, has the potential for more flexibility in accommodating thrust reverser targeting. This ejector appears on the latest turbojet and turbofan installations.

(5) Afterburner Concepts

The afterburner for the turbojets progressed from a short, partial (acceleration) type unit with a Mach 2.0 limitation to a longer, full afterburner with Mach 2.7 capability (Boeing indicated that they required a Mach 2.7 or full afterburning capability). This primarily accounts for the increased length of the turbojet engines toward the latter part of Phase IIB.

(6) Flow Schedules and Turbine Inlet Temperature

Above Mach 2.0 on both the turbofan and the turbojet, the corrected engine airflow as a function of Mach number may be selected over a range of values. This flexibility gives Boeing an opportunity to choose the airflow schedule which results in the best match between engine and inlet. A "high", "base" and "low" flow schedule were offered. In Phase IIA, Boeing selected a base flow turbofan. In order to provide an opportunity for comparison of turbojets, all three flow schedules were offered in the latter part of this Phase. For each flow schedule, configurations for both the 2000°F and 2300°F turbine inlet temperatures were presented. Boeing could, therefore, examine the trades involved in starting initial service at the lower temperature with subsequent growth to the higher temperature within the same external envelope.

e. Discussion of STF 219B Turbofan Installations

The study of new ejector and installation concepts began with the Phase IIA configuration, (see Figure 1-3), which was a 600 lb/sec engine with a cylindrical ejector of 76.00 inch diameter. This arrangement produced a base drag area between the ejector and the wing trailing edge of approximately 425 sq. in.

The object of the new studies was to reduce this base drag area, thereby improving on the Phase IIA installation. Wing contour, mount location,

CONFIDENTIAL

UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 10 YEARS
GROUP 1 EXCLUDED
THIS DOCUMENT CONTAINS INFORMATION RELATIVE TO THE
DEFENSE OF THE UNITED STATES OF AMERICA
IT IS HEREBY DECLARED TO BE UNCLASSIFIED
DATE 01-10-2001 BY 60322 UCBAW/SJS

- Figure 1-8 shows the best arrangement from a base drag reduction standpoint. The ejector tailflap hinge line was moved aft to the wing trailing edge and set tangent to it. The ejector was canted 9° at the primary nozzle plane. This configuration required a 51.50 inch extension, but reduced the base drag area to zero.

At this point in the program, Boeing specified a larger ejector diameter. Their area ruling dictated an ejector area/inlet area ratio = 1.7. Based on this, the ejector equivalent diameter was increased to 78.00 inches. Figure 1-9 shows an arrangement with this larger ejector canted $11^{\circ} 30'$ at the rear mount plane. The ejector tailflap hinge line was aligned with and set tangent to the wing trailing edge. This configuration required a 44.00 inch extension. The base drag remained at zero.

The ejector cant point was moved 10.00 inches forward of the primary nozzle plane to the throat of the fan nozzle. This preserved symmetry downstream of the fan nozzle choke point. Figure 1-10 shows an arrangement using this new cant point on the new 78.00 inch diameter equivalent ejector. The ejector tailflap hinge line was aligned and set tangent to the wing trailing edge. The ejector cant angle was $8^{\circ} 45'$. This required a 49.00 inch extension. Here again the base drag area was zero.

Figure 1-11 shows an arrangement for a 650 lb/sec engine with an 81.00 inch equivalent diameter ejector. The ejector is positioned so that the tailflap hinge line is aligned with the wing trailing edge. The upper surface of the octagonal ejector is set 3.00 inches above the wing upper surface at the wing trailing edge as requested by Boeing. The nacelle inlet capture area center point location of 29.60 inches down from the horizontal reference line was changed to 28.30 inches by Boeing. The ejector cant angle became $8^{\circ} 41.7'$ and a 17.4 inch extension was required.

Figure 1-12 shows a further Boeing revision. The inlet angle was reduced to $2^{\circ} 15'$ from 5° . Figure 1-12 is identical to Figure 1-11 except for the aforementioned inlet angle and the ejector cant angle which was increased to $11^{\circ} 2.1'$. A 30.63 inch extension was required.

Figure 1-13 shows an arrangement for a 600 lb/sec engine/ejector system and is similar to Figure 1-10 except that it was laid out to the latest Boeing geometry. The latest Boeing inlet centerline location of 27.30 inches down from the horizontal reference datum was used. The ejector was positioned as before, 3.00 inches above the wing with the tailflap hinge line aligned with the wing trailing edge. The octagonal

CONFIDENTIAL

PWA-2600

ejector has a 78.00 inch equivalent diameter and was canted 10.00 inches forward of the primary nozzle plane. The ejector cant angle is $10^{\circ} 55'$ and the configuration required a 36.98 inch extension.

The previous installation studies all involved fixed shroud ejector designs. A sliding shroud ejector was evolved in Phase IIB. This shroud makes more effective use of the energy in the engine and secondary air streams, reduces the minimum wrap of the ejector around the engine, and provides greater reverser targeting flexibility. On the turbofan, the shroud translates to three positions. The cruise position (blow-in doors closed) is the forward-most position. For take-off and up through the blow-in-door operating range, the shroud translates somewhat rearward relative to the cruise position. The shroud translates further rearward for reverse. The sliding shroud ejector is described in detail elsewhere in this report. Figure 1-14 shows a turbofan configuration which incorporates a sliding shroud ejector. Figure 1-14 also shows the most recent STF 219B accessory arrangement.

f. Discussion of STJ 227B Turbojet Installations

A 500 lb/sec afterburning turbojet engine installation was presented to Boeing for their initial Phase IIB studies. A number of new arrangements of the STJ 227B engine/ejector relative to the wing were investigated. The arrangements are summarized below. Two maximum turbine inlet temperatures were considered: 2000°F and 2300°F .

- Figure 1-15 shows an arrangement using an octagonal ejector of 75.00 inch equivalent diameter on a 2300°F turbine inlet temperature engine with partial augmentation. The inlet cowl was located 27.00 inches down from the horizontal reference line. Wing contour, reverser targeting, and the 5° inlet angle were supplied by Boeing and were used as the basis for these studies. The ejector was positioned so that the tailflap hinge line was aligned to and set tangent with the wing trailing edge. The ejector was canted at the rear engine mount plane with a $10^{\circ} 15'$ angle required to hit Boeing's exhaust gas target point. This configuration required a 13.40 inch extension at the engine inlet.
- Figure 1-16 shows an arrangement identical to Figure 1-15 except that the ejector cant point was moved aft to the throat of the afterburner nozzle, which reduced the cant angle to $7^{\circ} 40'$. This required a 13.80 inch extension.

PAGE NO. 1-8

CONFIDENTIAL

DECLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
GPO: 1984-000-000-000

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C. SECTION 793 AND 794, AND IS SUBJECT TO THE PENALTIES THEREOF. IT IS LOANED TO YOU BY THE NATIONAL ARCHIVES AND IS NOT TO BE REPRODUCED OR TRANSMITTED IN ANY FORM OR BY ANY MEANS, ELECTRONIC OR MECHANICAL, INCLUDING PHOTOCOPYING, RECORDING, OR BY ANY INFORMATION STORAGE AND RETRIEVAL SYSTEM.

CONFIDENTIAL

PRATT & WHITNEY AIRCRAFT

PWA-2600

The 2000°F maximum turbine inlet temperature engine incorporated a full afterburner from its inception and was thus longer than the 2300°F engine with partial augmentation. An 81.00 inch equivalent diameter ejector was used. Figure 1-17 shows an arrangement using this size octagonal ejector on a 2000°F turbine inlet temperature engine. The ejector, canted at the rear engine mount plane, required a cant angle of 8° 25'. The overall length of the engine and ejector eliminated the need for an inlet extension.

Figure 1-18 shows an arrangement identical to Figure 1-17 except that the ejector was canted at the throat of the afterburner nozzle, which reduced the required cant angle to 6° 30'.

At this point in the program the engine size was increased to 525 lb/sec. Figure 1-19 shows an arrangement of this engine with a partial afterburner and its required 76.8 inch equivalent diameter ejector for turbine inlet temperature of 2300°F. This configuration was positioned relative to the wing according to the latest Boeing data. A capture diameter of 56.18 inches located 27.75 inches down from the horizontal reference line with an inlet angle of 2° 15' was used. The ejector tail-flap hinge line was aligned with the wing trailing edge and with the upper surface of the octagonal ejector 3.00 inches above the wing upper surface at the wing trailing edge. The ejector was canted slightly forward of the afterburner nozzle throat at 10° 50'. A 21.00 inch inlet extension was required.

Figure 1-20 shows the corresponding arrangement for a fully augmented 2000°F engine with its required 83.00 inch equivalent diameter ejector. The engine was positioned relative to the wing similar to that shown in Figure 1-19 except that the length of the engine moved the ejector tail-flap hinge line 35.00 inches aft of the wing trailing edge. The ejector was canted at a point 2.40 inches forward of the afterburner nozzle throat at 8°. No extension was required.

The previous installations all had fixed shroud ejectors. Later installations used the sliding shroud ejector. On the turbojet, the shroud translates such that it has one position for forward flight, and another for reverse. This ejector is described in detail elsewhere in this report. Six full afterburning turbojet configurations incorporating the sliding shroud ejector were presented to Boeing. These six configurations comprised a high, base, and low flow version of the 2000°F and the 2300°F engine. Each configuration was adjusted to its particular flow schedule and temperature for comparative purposes. As in the case of the turbofan, once the comparison has been completed and a selection made, one configuration may then evolve which permits growth from

PAGE NO. 1-9

CONFIDENTIAL

DOWNLOADED AT 3 HOUR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DDC ON 8800-0

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C. SECTION 793 AND 794. IT IS TO BE KEPT SECRET AND NOT TO BE DISCLOSED TO ANY PERSON OR ENTITY IN ANY MANNER OR BY ANY MEANS.

CONFIDENTIAL

PWA-2600

2000 °F to 2300 °F within the same external envelope. The six configurations are summarized below:

- Figure 1-21 shows an installation study for a 2000 °F turbine inlet temperature, high flow engine. The ejector size used for this configuration was 78.00 inch diameter equivalent and was canted 1.50 inches forward of the afterburner nozzle throat at 7° 55'.
- Figure 1-22 shows an installation study for a 2000 °F turbine inlet temperature, base flow engine. The ejector size used for this configuration was 75.00 inch diameter equivalent and was canted 1.80 inches forward of the afterburner nozzle throat at 8° 35'.
- Figure 1-23 shows an installation study for a 2000 °F turbine inlet temperature, low flow engine. The ejector size used for this configuration was 72.00 inch equivalent diameter and was canted 1.75 inches forward of the afterburner nozzle throat at 8° 50'.
- Figure 1-24 shows an installation study for a 2300 °F turbine inlet temperature, high flow engine. The ejector size used for this configuration was 78.00 inch equivalent diameter and was canted 1.65 inches forward of the afterburner nozzle throat at 8° 9'.
- Figure 1-25 shows an installation study for a 2300 °F turbine inlet temperature, base flow engine. The ejector size used for this configuration was 75.00 inch equivalent diameter and was canted 1.80 inches forward of the afterburner nozzle throat at 8° 35'.
- Figure 1-26 shows an installation study for a 2300 °F turbine inlet temperature, low flow engine. The ejector size used for this configuration was 70.00 inch equivalent diameter and was canted 2.00 inches forward of the afterburner nozzle throat at 9° 45'.

It should be noted that in the above six installation studies the overall length of the engine/ejector eliminated the need for an inlet extension.

Figure 1-27 shows the most recent SIJ 227 accessory arrangement.

CONFIDENTIAL

CLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 5 YEARS
UNLESS INDICATED OTHERWISE

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE SPY ACT, 18 U.S.C. 793 AND 794.
IT IS THE POLICY OF THE UNITED STATES GOVERNMENT
TO MAINTAIN THIS INFORMATION IN SECRET UNLESS
REMOVAL FROM SECRET STATUS IS AUTHORIZED BY THE
COMMITTEE TO REVIEW INFORMATION IN SECRET STATUS

CONFIDENTIAL

PWA-2600

2. INSTALLATION COORDINATION WITH THE LOCKHEED CALIFORNIA CORPORATION

a. Introduction

Coordination meetings were held between the Contractor and the Lockheed California Corporation during Phase IIB. The most significant topics discussed in the course of these meetings are summarized below:

- Engine accessories. The Phase IIA arrangement was revised in order to relocate the accessories away from the bottom of the engine.
- Ejector. An octagonal blow-in-door ejector was studied as an alternative to the 12-sided design presented in Phase IIA.
- Engine cant and wing relationship. A study was made to determine the most desirable location for canting the engine as the result of Lockheed's request that the engine be canted 4° in a downward direction.
- Sonic boom. The size of the engine was changed in the light of the FAA's reduction in the sonic boom overpressure requirements.
- Turbojet installations. A number of installation sketches was prepared for various versions of the turbojet, which was reintroduced during Phase IIB.

b. Accessory Study

Pratt & Whitney Aircraft restudied the accessory arrangement proposed in Phase IIA. This arrangement consisted of engine accessories driven by a gearbox located on the bottom of the engine and a power take-off gearbox located on top of the engine which supplied power to drive the airframe accessories. Lockheed requested that all engine accessories containing combustible fluids be moved from the bottom of the engine to reduce the danger of fire in the event of a collapsed landing gear or belly landing.

Figure 1-28 shows a revised arrangement with the engine accessories split into two groups. Those accessories requiring a power drive were mounted on the gearbox located on the left side of the engine at the engine horizontal center line. The remaining accessories were grouped at the same elevation on the opposite side of the engine. All accessories

CONFIDENTIAL

UNCLASSIFIED AT 10 YEAR INTERVALS
DATE 08/01/2010 BY 60322 UCBAW/BJS/STP

ALL INFORMATION CONTAINED HEREIN IS UNCLASSIFIED EXCEPT WHERE SHOWN OTHERWISE

CONFIDENTIAL

PWA-2600

were removed from the bottom. This arrangement was rejected by Lockheed because the accessories in the upper quadrants interfered with their front engine mount structure. The arrangement also prevented ready access to the power take-off gearbox.

Continued study rejected the use of two separate towershafts and gearboxes for engine accessories because of internal gearing difficulties. A new arrangement (Figure 1-29) was devised with the engine-driven accessories located on the left side of the engine approximately 45° below the engine horizontal center line. The remaining accessories were located at the same level on the opposite side of the engine. Although this arrangement involved greater difficulty in removing the accessories from the bottom of the engine, the arrangement was generally acceptable for the 700 lb/sec engine size. When the size was reduced to 650 lb/sec this accessory arrangement became less attractive, as the units were crowded together in the reduced circumferential space. An alternate arrangement, Figure 1-30 evolved locating the engine accessory drive shaft on the right horizontal centerline, with the accessories grouped differently from Figure 1-28.

c. Octagonal Ejector

Early in Phase IIB Pratt & Whitney Aircraft studied the use of an octagonal blow-in-door ejector as an alternate to the circular door 12-sided design presented in Phase IIA. Figure 1-31 is a schematic drawing of the octagonal ejector sent to Lockheed. The octagonal ejector dominated studies during Phase IIB. All subsequent installation sketches sent to Lockheed represented some modification to this design. A detailed description of ejector studies is covered in the Ejector Reversers section of this report.

d. Engine Cant and Wing Relationship

During Phase IIB Lockheed moved both the inboard and outboard engines rearward on the wing approximately 5 to 6 feet. Concurrently, they requested that the engine be canted 4° downward. A study was made to determine the most desirable location for canting the engine, and a point just forward of the ejector was chosen. Figure 1-32 (outboard engine) and Figure 1-33 (inboard engine) show the STF219-L-700 engine installed in the Lockheed L-2000-4 wing with a 4° engine cant. This sketch indicated that the rearward relocation of the engine allowed the installation of the engine-ejector to significantly reduce wing blockage of the blow-in doors. During Phase IIA, the forward location of the engine placed the ejector in the thicker part of the wing and resulted in blockage of two blow-in doors.

PAGE NO 1-12

CONFIDENTIAL

DECLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 5 YEARS
GDD LTR 8306 10

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
DEFENSE OF THE UNITED STATES OF AMERICA. IT IS THE
POLICY OF THE UNITED STATES GOVERNMENT TO MAKE
AVAILABLE TO THE PUBLIC INFORMATION THAT IS NOT
EXCLUDED BY THE PROVISIONS OF THE NATIONAL
DEFENSE INFORMATION SECURITY ACT (50 USC 3605) OR
ANY OTHER LAWS, REGULATIONS, OR EXECUTIVE ORDERS.

CONFIDENTIAL

FWA-2600

e. Reduction of Sonic Boom Overpressure Requirements

The FAA's reduction of the sonic boom overpressure reduced the maximum thrust requirement for the engine. The engine is now sized for cruise thrust levels instead of the greater thrust levels previously required for transonic acceleration. In addition, refinement of the aircraft design with accompanying increases in lift/drag values lowered thrust requirements. As a result the size of the engine was reduced to 650 lbs/sec airflow size.

Figures 1-34 and 1-35 are installation drawings for the new size of engine. Critical areas (i.e. accessory arrangement and ejector O.D.) were reviewed to assure proper nacelle clearance. These drawings reflect the changes in basic engine length derived from the continuing detailed design of the engine. The most significant change was the addition of a diffuser section between the third stage turbine blades and the rear struts. This results in the reduction of the exhaust gas velocity from the turbine to the desired value. This feature is further explained in the section of this report on Turbine Design. A revised weight for the 650 lbs/sec STF219 engine was developed.

f. Miscellaneous Studies

Lockheed expressed concern about the area available for passing secondary airflow between the front mount ring O.D. and the nacelle wall. Information was then supplied to Lockheed describing the weight changes incurred in reducing the front mount ring O.D. by 0.5 and 1.0 inches to increase the secondary air passage. The design of the ring had been optimized in Phase IIA.

While the detail design of the STF219 turbofan continued, a study was made to position the P_{t2} and T_{t2} probes, which are necessary for biasing the engine fuel control. The probes were located in the engine front mount ring as shown in Figure 1-36.

At the end of Phase IIA, Lockheed requested a study of flame arrestors in the secondary airstream. Although spring-loaded flapper valves in the secondary air stream to prevent a reverse or forward flow of secondary air were included in the Phase IIA design, it was considered possible that during conditions of very low rearward secondary airflow, a combustible mixture from a leak in the engine compartment might ignite when it reached the hot exhaust section. This could propagate forward through the flapper valve openings into the engine compartment.

PAGE NO. 1-13

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DDO DW 830010

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C. SECTION 793 AND 794. THE TRANSMISSION OF THE REVEALATION OF ITS CONTENTS TO ANY PERSON OR AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

CONFIDENTIAL

PWA-2600

Lockheed provided a sketch of a suggested flame arrestor showing the general construction, frontal area, and desired pressure drop. The resulting study is shown on Figure 1-37. The flame arrestors were located immediately forward of the flapper valves. Their installation requires a mounting structure, attachment flanges, and accessibility for replacement of damaged units. The original study based on the 700-lbs/sec engine showed that the flame arrestors could be installed without difficulty and still provide the required area to maintain a minimum pressure drop.

Continued detail design of the engine and the reduction in engine size from 700 lbs/sec to 650 lbs/sec resulted in slightly less room for the flame arrestors. Similarly, continued ejector studies resulted in a tighter wrap between the engine and the ejector reducing still further the area available for the flame arrestors. A study was made of the entire secondary airflow passage from the engine inlet to the ejector throat because of these changes and the possible increase in secondary airflow to nearly 18 percent of the primary airflow.

Figure 1-38 shows that the minimum area available for secondary airflow occurs at the flame arrestors, flapper valves, and rear mount rings. Figure 1-39 shows more detail of these areas. This minimum area is now smaller than earlier studies showed. With an increase in secondary airflow and a corresponding increase in pressure drop, continued studies will be required in this area to provide a proper installation.

g. Turbojet Installations

The reintroduction of the turbojet engine during Phase IIB required updating of the Phase IIA installation. As a result, two preliminary installation drawings for a partial afterburning and a full afterburning turbojet were developed for the initial study (Figures 1-40 and 1-41). Subsequently, as the turbojet design evolved, a series of six installation sketches were prepared each showing the effect of turbine inlet temperature and airflow schedule selection on the size of the engine and ejector. Figure 1-42 presents a typical turbojet engine. The major dimensions and differences between each of the six engines are summarized on page 1-16. All versions of the turbojet are fully augmented and weights for each of the six engines were quoted.

Figure 1-43 shows external dimensions and a proposed accessory arrangement for the high temperature, base flow engine.

CONFIDENTIAL

DOWN-MADE AT 2 YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS
DATE OF REVIEW

THIS DOCUMENT CONTAINS NEITHER RECOMMENDATIONS NOR
CONCLUSIONS OF THE NATIONAL AERONAUTICS AND SPACE
ADMINISTRATION. IT IS THE PROPERTY OF THE NATIONAL AERONAUTICS
AND SPACE ADMINISTRATION. IT IS LOANED TO YOUR ORGANIZATION
AND IT AND ITS CONTENTS ARE NOT TO BE DISTRIBUTED OUTSIDE
YOUR ORGANIZATION.

CONFIDENTIAL

PWA-2600

The 4° downward cant of the engine is required for the turbojet engine as well as the turbofan. Various combinations of engine positions relative to the L-2000-4 wing shape were tried. The current design has placed the cant in the turbojet engine slightly aft of the rear mount structure. All subsequent installation studies showed the cant at this location with an octagonal sliding shroud ejector.

The additional length of the turbojet engine plus the rearward movement of the engines relative to the wing makes it possible to install the engine in the L-2000-4 wing without blockage of any of the ejector blow-in doors. Reaching this optimum position requires setting the ejector higher into the wing and increasing the cant angle to approximately 6° (Figure 1-44). As seen in the figure, the engine rear mount ring appears to be in a favorable position for direct mounting to the wing.

An alternate scheme placed the top of the ejector flush with the top of the wing. This allowed full utilization of the blow-in doors and introduced the added feature of reduced cant angle. Unfortunately this arrangement lowered the engine relative to the wing sufficiently to eliminate a practical mounting installation.

Continued detail design of the engine will create further areas requiring coordination and revised installation techniques and procedures.

CONFIDENTIAL

UNCLASSIFIED AT 1 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
(C) 1964 PWA

THIS DOCUMENT CONTAINS INFORMATION RELATIVE TO THE
DEFENSE OF THE UNITED STATES OF AMERICA. IT IS
HEREBY DECLARED TO BE UNCLASSIFIED EXCEPT WHERE
SHOWN OTHERWISE. IT IS THE POLICY OF THE UNITED STATES
GOVERNMENT TO MAKE SUCH INFORMATION AVAILABLE TO THE
PUBLIC AS SOON AS POSSIBLE AFTER REVIEW BY THE
APPROPRIATE AGENCIES.

CONFIDENTIAL

PWA-2600

TABLE 1-1
PRINCIPAL INSTALLATION DIMENSIONS
FOR LOCKHEED CALIFORNIA CORPORATION
STJ227 TURBOJET ENGINE, 525 LBS./SEC. SIZE
TRANSLATING SHROUD EJECTOR, 4° CANT ANGLE

Turbine Inlet Temp. (°F)	Airflow Schedule	Inlet Diameter (in.)	Overall Length (in.)	Ejector Equivalent Diameter (in.)	Distance Between Mount Planes (in.)
2000	High Flow	56.30	307.32	78.00	119.15
2000	Base Flow	56.30	305.54	75.00	119.15
2000	Low Flow	56.30	303.56	72.00	119.15
2300	High Flow	56.30	299.54	78.00	121.15
2300	Base Flow	56.30	296.24	75.00	121.15
2300	Low Flow	56.30	296.26	70.00	121.15

CONFIDENTIAL

REPRODUCED AT THE REQUEST OF THE
NAVY DEPARTMENT
1950

CONFIDENTIAL

PRATT & WHITNEY AIRCRAFT

PWA-2600

B. ENGINE CYCLE STUDIES

1. INTRODUCTION

Two engines were studied during Phase II B, the STF219, a non-mixed flow duct-heating turbofan, and the STJ227, an afterburning turbojet. The design parameters for these engines are presented in Table 1-2.

TABLE 1-2

Design Parameters of Afterburning Turbojet (STJ227)
and Duct-Heating Turbofan (STF219) Engines

	<u>STJ227</u>		<u>STF219</u>	
Turbine Inlet Temperature (°F)				
Take-Off and Acceleration	2000	2300	2000	2300
Cruise	1900	2200	1900	2200
Corrected Airflow (lb/sec)				
Nominal	525	525	650	650
Cruise (Mn = 2.7)	372	372	308	308
Over-All Pressure Ratio	9.3	9.3	11.9	11.9
Fan Pressure Ratio			2.7	2.7
Bypass Ratio			1.3	1.3
Maximum Engine Diameter (inches)	75	75	81	81
Engine Weight Including Ejector (lb)	10470	10455	9560	9560

During Phase II A, the airframe manufacturers were supplied with sufficient information to permit them to make preliminary engine-aircraft performance estimates. At this time, both airframe manufacturers selected the STF219 engine. Boeing initially expressed an interest in the STJ227 engine with 25 percent augmentation, but this engine exceeded the limits for stable afterburner combustion above Mach 2, being specifically designed for augmentation during transonic operation from Mach 1.2 to Mach 1.8. Above Mach 2, the combustion chamber Mach number increased rapidly above the combustion stability limit and the combustion chamber inlet temperature dropped into the marginal auto ignition range. However, during Phase II B, the sonic overpressure

CONFIDENTIAL

PRATT & WHITNEY AIRCRAFT

PWA-2700

Limits were increased from 2.0 to 2.5 paf for transonic acceleration and from 1.5 to 1.7 paf for supersonic cruise. This shift resulted in the engine being sized for supersonic cruise rather than for transonic acceleration and made the STJ227 turbojet engine with full augmentation an attractive powerplant. Consequently, both engines were restudied.

The performance of the uninstalled engines at several critical flight conditions is shown in Table 1-3. The thrust specific fuel consumption over a wide range of power settings is shown in Figures 1-45 through 1-47. The effect of reoptimizing the STF219 engine is shown in Table 1-4 and Figures 1-48 through 1-50. Similar data for the STJ227 engine is presented in Table 1-5 and Figures 1-51 through 1-53. Detailed design information is presented in Section 2.

TABLE 1-3

Performance of Uninstalled
STJ227 and STF219 Engines

	Afterburning Turbojet STJ227		Duct Heating Turbofan STF219	
Take-Off Maximum Thrust (lb)	57000	59800	52500	57000
Nominal Airflow Size (lb/sec)	525	525	650	650
Cruise Turbine Inlet Temperature (°F)	1900	2200	1900	2200
<u>Transonic Acceleration, Mach 1.2, at 45000 Ft</u>				
Thrust (lb)	20600	21400	18400	19800
Specific Fuel Consumption (lb/hr/lb)	1.91	1.83	1.88	1.84
<u>Supersonic Cruise Mach 2.7 at 65000 Ft, Thrust = 9800 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.44	1.45	1.54	1.48
<u>Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.05	1.06	0.88	0.91
<u>Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.25	1.25	0.95	0.96

Source: 1-10

CONFIDENTIAL

CONFIDENTIAL

PWA-2600

TABLE 1-4

Performance of STEAR Engine at Completion of Phase IIA
and at Completion of Phase IIB

	Basic Engine		Initial Engine	
	Phase IIA	Phase IIB	Phase IIA	Phase IIB
Supersonic Cruise Turbine inlet Temperature (°F)	2200	2200	1900	1900
Take-Off Maximum Thrust (lb)	56800	57000	52300	52500
Airflow Size (lb/sec)	650	650	650	650
Maximum Diameter (in)	79.7	81	79.7	81
Engine Weight Including Ejector (lb)	9150	9560	9200	9560
<u>Transonic Acceleration, Mach 1.2 at 45000 Ft</u>				
Thrust (lb)	19700	19800	18300	18400
Specific Fuel Consumption (lb/hr/lb)	1.84	1.84	1.88	1.88
<u>Supersonic Cruise, Mach 2.7 at 65000 Ft, Thrust = 9800 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.49	1.48	1.55	1.54
<u>Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	0.92	0.91	0.89	0.88
<u>Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	0.97	0.96	0.96	0.95

CONFIDENTIAL

CONFIDENTIAL

PWA-2600

TABLE 1-5

Performance of STJ227 Engine at Completion of Phase IIA
and at Completion of Phase IIB

	Basic Engine Phase IIA	Phase IIB	Initial Engine Phase IIB
Supersonic Cruise Turbine Inlet Temperature ($^{\circ}\text{F}$)	2200	2200	1900
Take-Off Maximum Thrust (lb)	60600	59800	57000
Airflow Size (lb/sec)	525	525	525
Maximum Diameter (in)	78	75	75
Engine Weight Including Ejector (lb)	9850	10455	10470
<u>Transonic Acceleration, Mach 1.2 at 45000 Ft</u>			
Thrust (lb)	21400	21400	20600
Specific Fuel Consumption (lb/hr/lb)	1.86	1.83	1.91
<u>Supersonic Cruise, Mach 2.7 at 65000 Ft, Thrust 9800 lb</u>			
Specific Fuel Consumption (lb/hr/lb)	1.50	1.45	1.44
<u>Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb</u>			
Specific Fuel Consumption (lb/hr/lb)	1.10	1.06	1.05
<u>Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb</u>			
Specific Fuel Consumption (lb/hr/lb)	1.33	1.25	1.25

CONFIDENTIAL

CONFIDENTIAL

PWA-2600

Particular emphasis during Phase IIB was placed on the augmentation systems, and IBM performance decks were provided to the airframe manufacturers to permit airframe-powerplant optimization with the fully augmented turbojet engine. In addition, a study of inflight thrust measurement techniques based on the STF219 engine was conducted.

2. AUGMENTATION SYSTEM STUDIES

The relaxation of the sonic boom overpressure limit permitted a re-optimization of the engine cycles, and, in particular, the augmentation systems. For the STF219 engine, the effect of reducing the amount of augmentation and the magnitude of the thrust discontinuity between un-augmented and minimum augmented operation were studied. For the STF227 engine, the effects of the turbine inlet temperature and engine air flow on augmentor performance were studied.

a. STF219

Since the engines are now sized for supersonic cruise rather than for transonic acceleration, the fuel consumption during climb can be reduced by reducing the duct heater augmentation thrust and temperature. A reduction in the maximum augmentation thrust of 1 to 5 percent corresponds to an augmentation temperature reduction of 100 to 300°F and increases the aircraft range by 20 to 35 miles.

The Phase IIA evaluation indicated that the evaluating team considered the thrust discontinuity between non-augmented and minimum augmented thrust to be too great. Consequently, a study was conducted to determine the g loading applied to a passenger when the duct heaters on all four engines are lit simultaneously with several augmentation ratios. The results are plotted in Figure 1-54. The upper curve shows the g loading for the performance data presented during Phase IIA for which the fuel-air ratio for lighting was 0.01. For these conditions, the g loading would be only 0.05. However, studies conducted during Phase IIB have indicated that the minimum duct heating fuel-air ratio could be reduced to 0.006, thereby reducing the g loading to 0.04. Stable combustion has been demonstrated at much lower fuel-air ratios, and, with additional development, it would be possible to decrease the minimum fuel-air ratio to 0.003, corresponding to a maximum g loading of 0.01. The problem has been discussed with airframe manufacturers, however, and it was learned that passengers on present commercial jet aircraft experience a g loading of about 0.1, and, in some cases, they experience g loadings as high as 0.33. It appears, therefore, that the g loading produced during the SST

CONFIDENTIAL

CONFIDENTIAL

PWA-2600

flight will be significantly lower than that of current jet aircraft and, therefore, that there is little reason to develop a duct heater capable of ignition with a fuel-air ratio below 0.008.

b. STJ227

(1) Effect of Turbine Inlet Temperature

The engines were designed for operation with a turbine inlet temperature of 2200°F. However, initially the engines will probably be operated with a turbine inlet temperature of 1900°F, and, therefore, the effect of operating at the lower turbine inlet temperature must be considered. If the augmentation ratio or thrust increase is to be maintained at the lower turbine inlet temperature, the weight and maximum engine diameter are significantly increased. If the cycle pressure ratio is to remain unchanged, the turbine expansion ratio must be increased to provide the required compressor power. However, increasing the expansion ratio increases the turbine exit Mach number and the afterburner combustion chamber Mach number. Consequently, the engine diameter must be increased to provide acceptable afterburner inlet conditions. The effect of turbine inlet temperature on the maximum engine diameter is shown in Figure 1-55.

With a turbine inlet temperature of 1900°F, unaugmented thrust at cruise is marginal and the capability for augmentation during cruise at nonstandard temperatures or at off-design altitudes is desirable. The IBM decks supplied to the airframe manufacturers, therefore, provided performance, weight, and installation data for engines with augmentation limited to take-off and transonic operation; limited to take-off, acceleration, and supersonic cruise; or unlimited and used throughout the mission. Optimization studies conducted by the airframe manufacturers and based on these data resulted in the selection of the fully augmented version of the STJ227 engine.

(2) Effect of Supersonic Cruise Airflow

The engine corrected airflow at cruise affects the augmentor design in a manner similar to that of the turbine inlet temperature. In order to increase the airflow, the power output of the turbine must be increased, and, therefore, the expansion ratio must be increased. Consequently, the maximum engine diameter must be increased to satisfy the augmentor design criteria.

Continued reduction of the supersonic cruise airflow eventually results in the turbine power requirement being established at acceleration

CONFIDENTIAL

Classified by 61500 INTER-ALL
DECLASSIFIED BY 61500 INTER-ALL
DATE 08-01-2001

THIS DOCUMENT CONTAINS NEITHER
RECOMMENDATIONS NOR CONCLUSIONS
OF THE NATIONAL BUREAU OF AERONAUTICS
UNLESS INDICATED OTHERWISE

CONFIDENTIAL

PWA-2600

conditions rather than at cruise conditions. This effect is shown in Figure 1-56.

3. INFLIGHT PERFORMANCE MEASUREMENT SYSTEM

Inflight thrust measurement in supersonic vehicles is considerably more complicated than in subsonic vehicles. In subsonic vehicles, engine pressure ratio, speed, and exhaust temperature have been used in conjunction with a series of charts to indicate the performance and condition of the engine. These parameters provide an accurate method of determining the performance of engines with fixed nozzle geometry. For the more sophisticated engines required for the supersonic transport, however, these parameters are not adequate since the inlet, the exhaust nozzle, and the interference effects between the propulsion system and the airframe significantly affect the actual force applied to the vehicle. Consequently, thrust measurement will require the use of additional parameters in conjunction with a compact, lightweight (about 30 pounds) computer such as those currently being used in missile guidance systems. The parameters required are listed in Table 1-6 together with the measurement accuracy.

Table 1-6

Inflight Performance Measurement Parameters

<u>Parameter</u>	<u>Accuracy (Percent)</u>
P_{am} , free stream static pressure	± 1.25
P_{t3} , fan discharge total pressure	± 1.25
P_{s3} , fan discharge static pressure	± 1.25
T_{t3} , fan discharge total temperature	± 1.0
$(P_{t3} - P_{s3}) / P_{s3}^*$	± 4.0
P_{t7} , gas generator total exhaust pressure	± 1.0
A_{jduct} , duct exhaust nozzle throat area **	± 3.0
W_f , engine fuel flow	± 0.5

Notes:

- * This parameter may be obtained from engine control where it is used for duct exhaust nozzle positioning.
- ** Direct measurement of A_{jduct} may not be required.

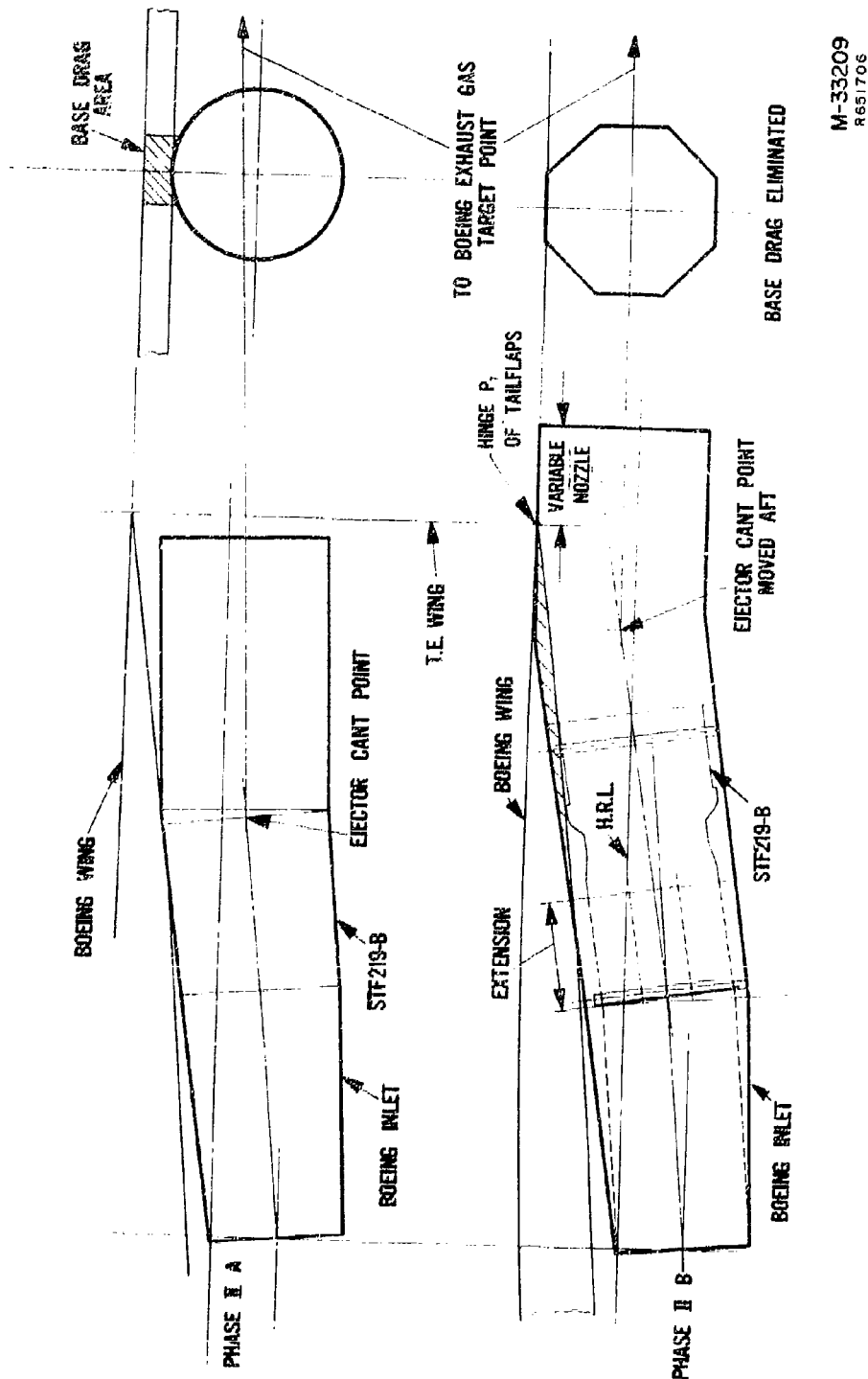
The thrust and thrust specific fuel consumption obtained from these measurements could be in error by as much as the arithmetical sum of the individual errors, but it is unlikely that all errors would be

CONFIDENTIAL

REPRODUCTION OF THIS REPORT IS PROHIBITED
EXCEPT BY SPECIFIC WRITTEN PERMISSION
OF THE PRATT & WHITNEY COMPANY

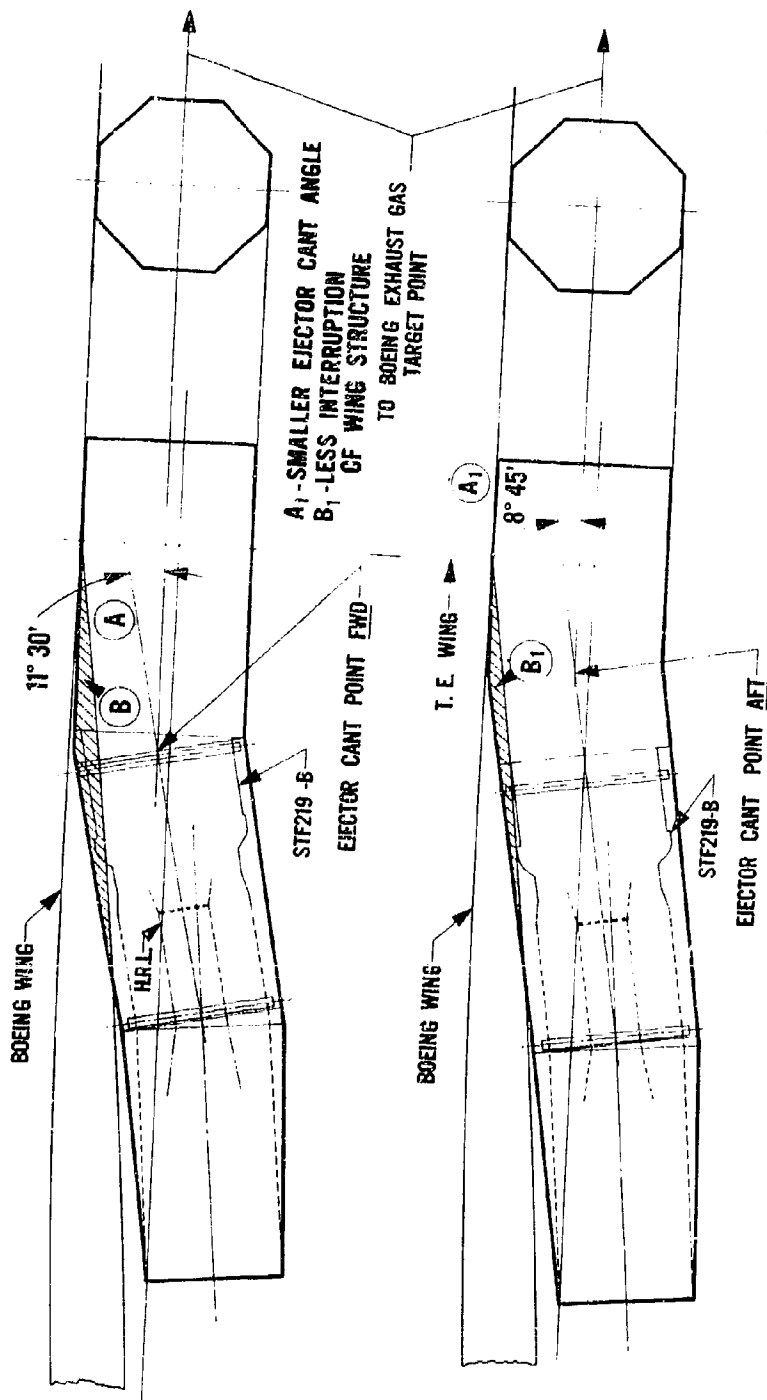
Probable Error in Inflight Performance Measurement

<u>Flight Condition</u>	<u>Power Setting</u>	<u>Probable Error (Percent)</u>	
		<u>Total Thrust</u>	<u>TSFC</u>
Sea-Level Take-Off	Maximum Duct Heating	3.0	3.1
Transonic Acceleration (Mach 1.2)	Maximum Duct Heating	4.2	4.3
Supersonic Cruise (Mach 2.7)	Partial Duct Heating		12.3
Subsonic Cruise (Mach 0.9)	Part Throttle		5.5



BOEING INSTALLATION - SST

Figure 1-1

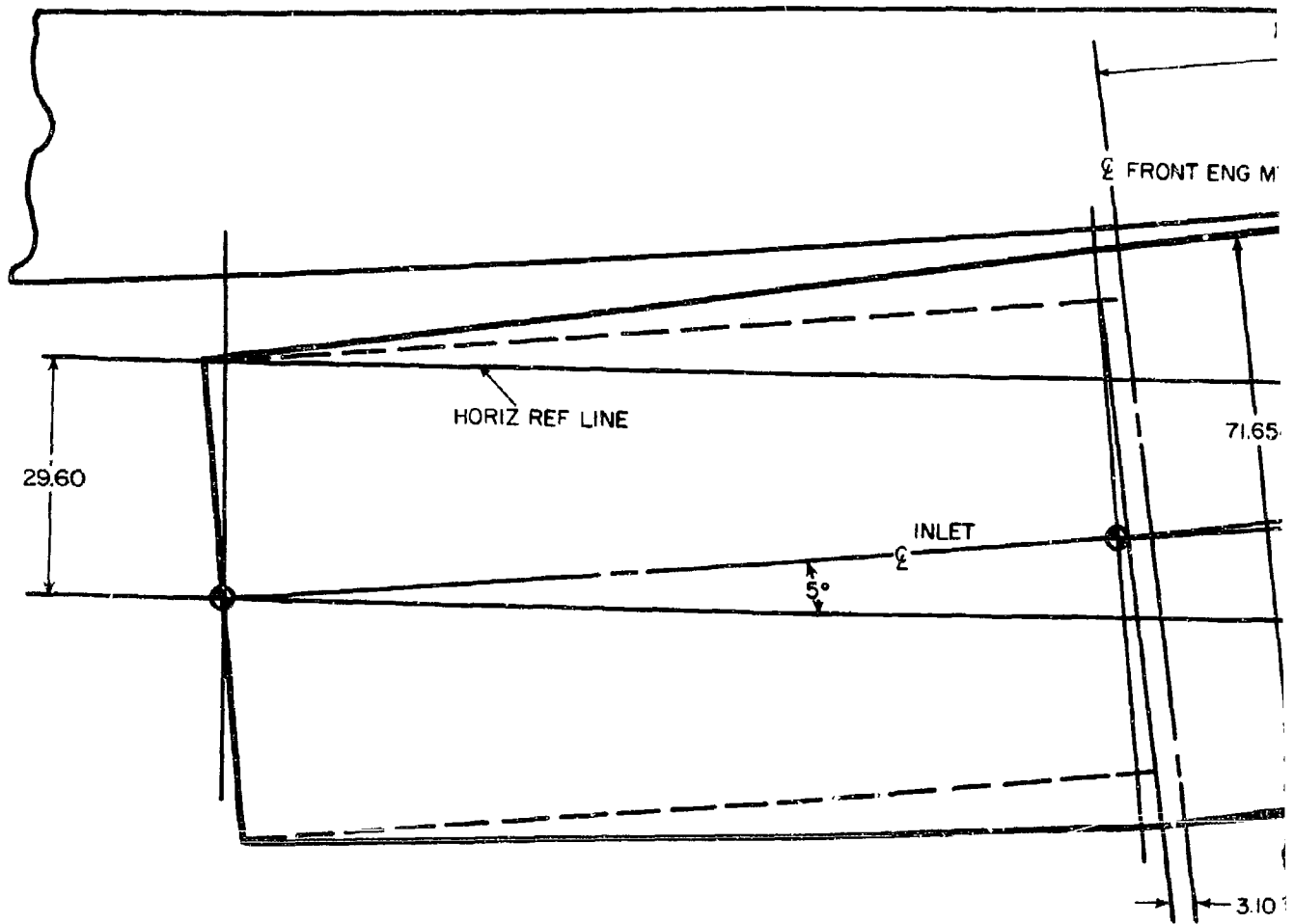


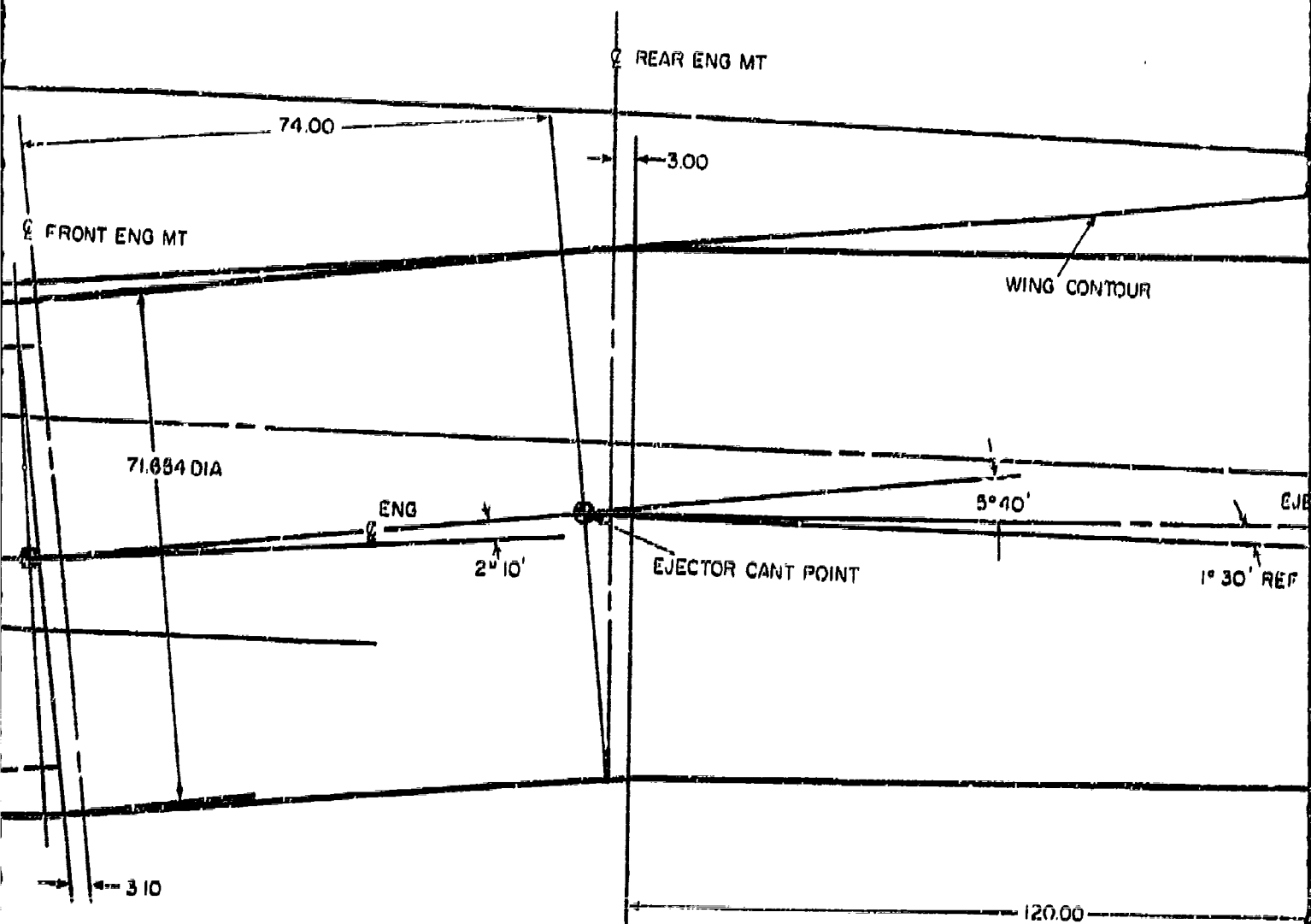
M-33208
R65106

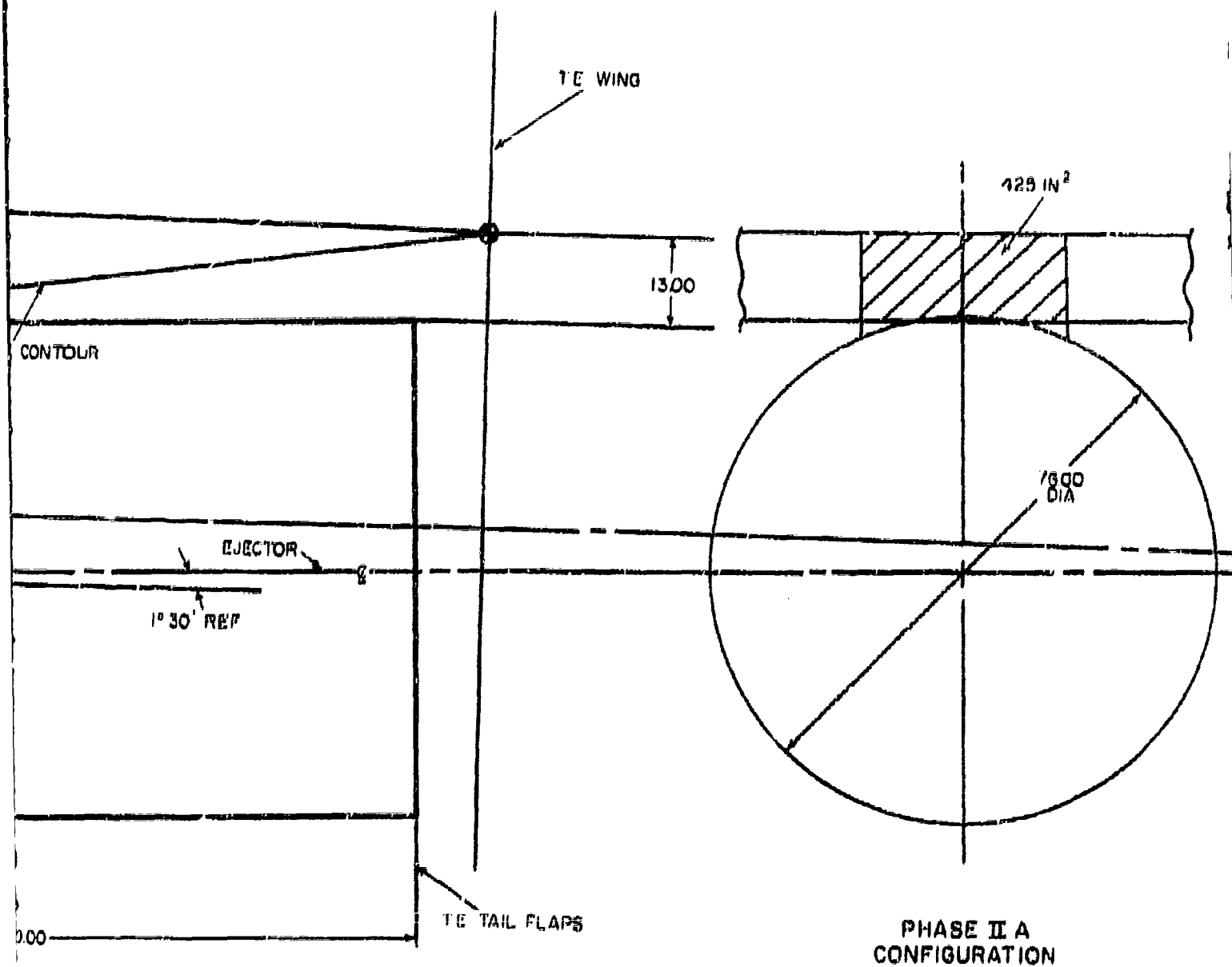
EFFECT OF MOVING THE CANT POINT AFT

Figure 1-2

PRATT & WHITNEY AIRCRAFT







PW A-2600

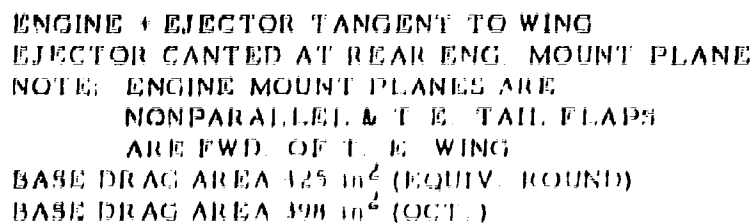


Figure 1-3

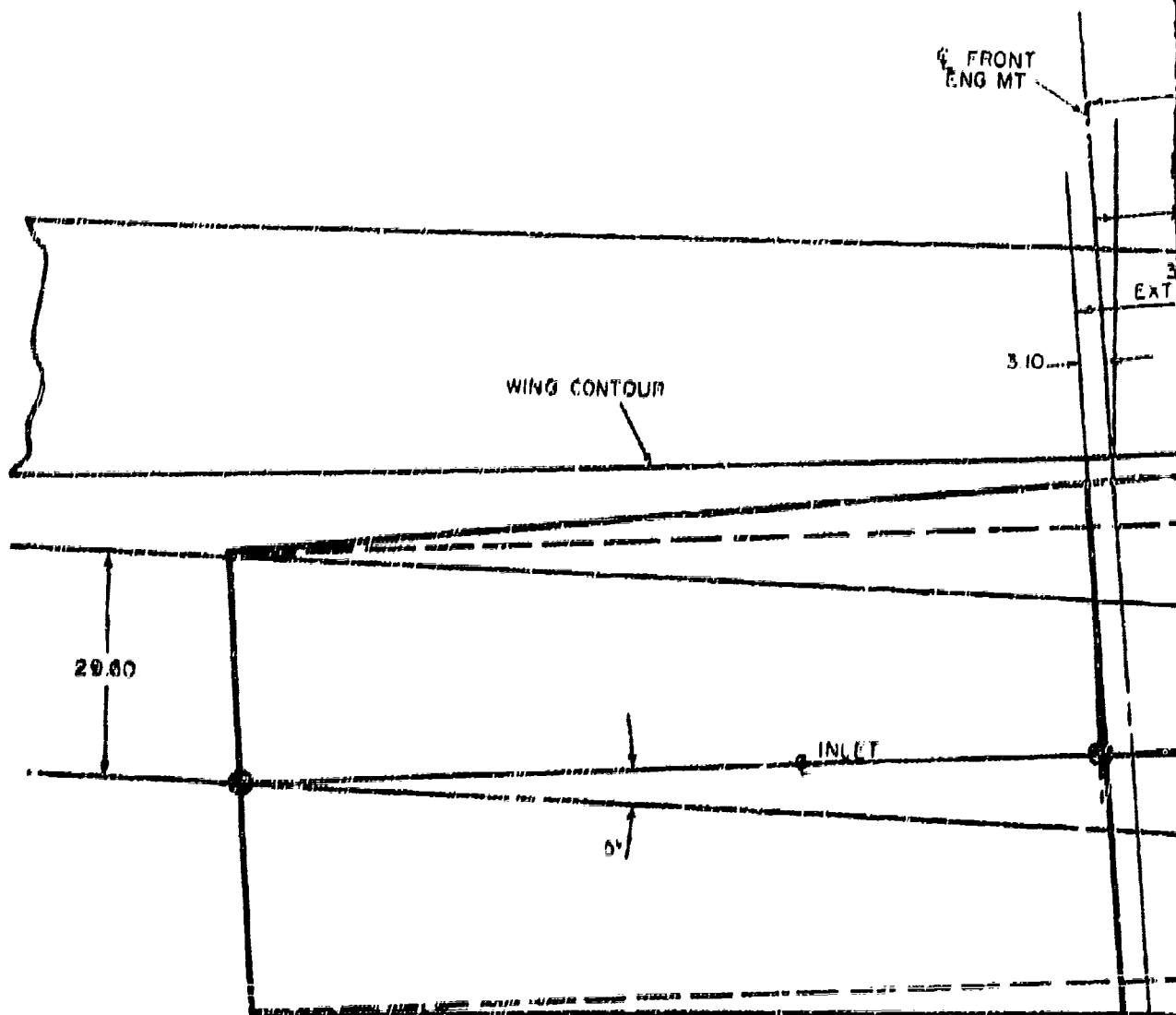
CONFIDENTIAL

1. 凡在本行開辦之各項業務，均應遵守本行所訂之各項規章，並應遵守國家及地方有關之法律法規。

[illegible]

4

PRATT & WHITNEY A. B. CHART



FRONT
ENG MT

114.00

E REAR MT
ORIG. LOC

74.00

E REAR
ENG MT

3.00

12

33.50
EXTENSION

3.10

HRL

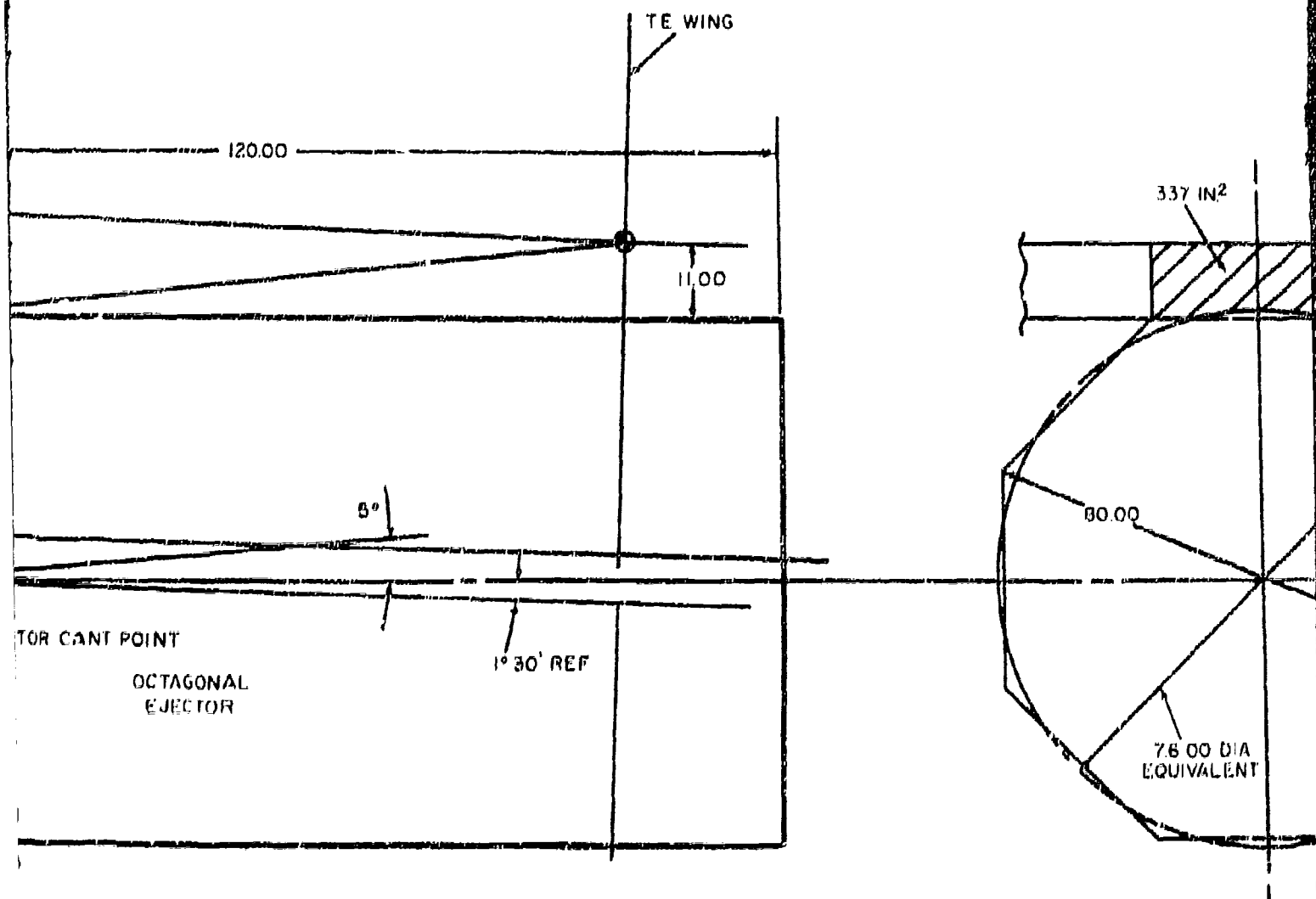
E ENG

1° 30'

EJECTOR CANT POINT

OCTAGONAL
EJECTOR

CONFIDENTIAL



ARRANGEMENT SHOWING RE
 MOVED AFT 40.00
 ENGINE & EJECTOR ARE TA
 EJECTOR CANTED AT REA
 PLANE
 ENGINE, MOUNT PLANES /
 BASE DRAG AREA 337 in²

STF219B 600 LBS. /SEC

Figure 1

CONFIDENTIAL
 3

PW A - 2600

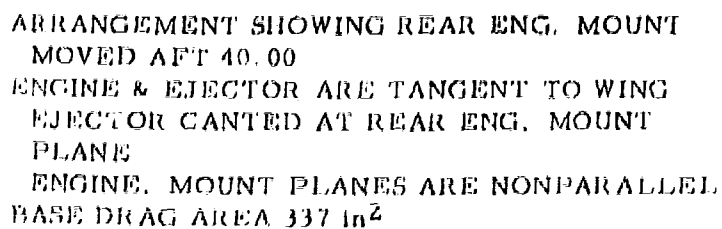
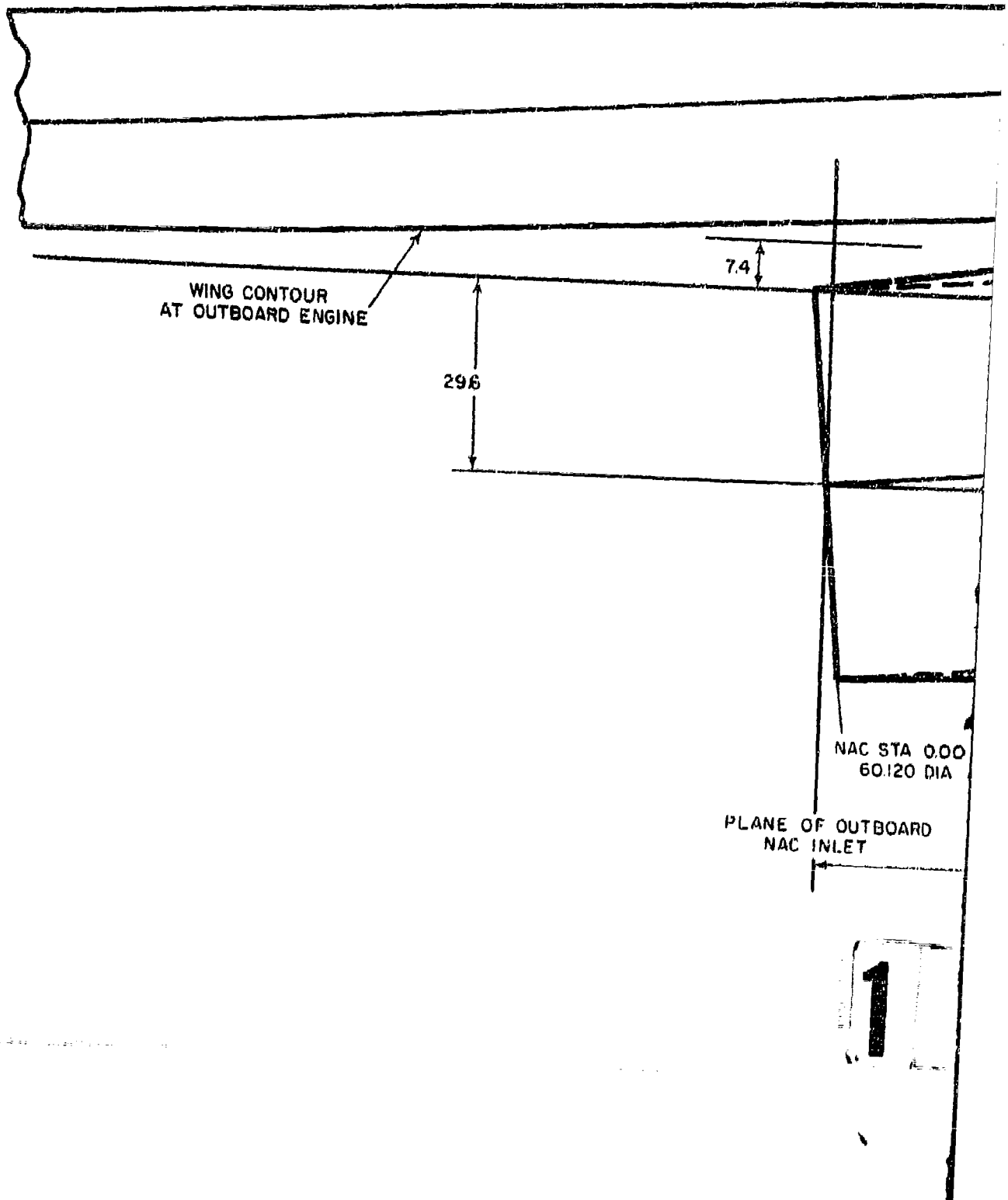


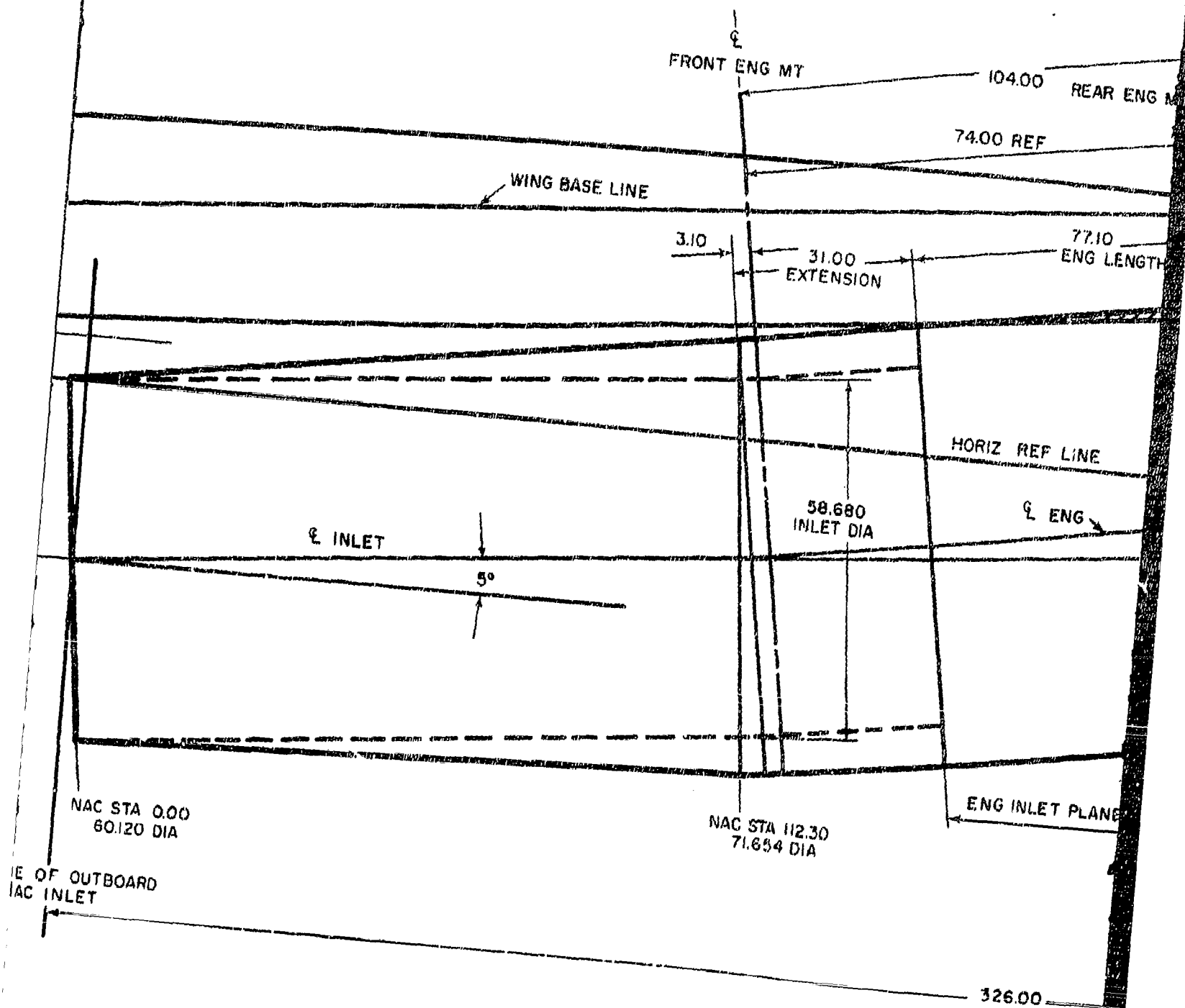
Figure 1-4

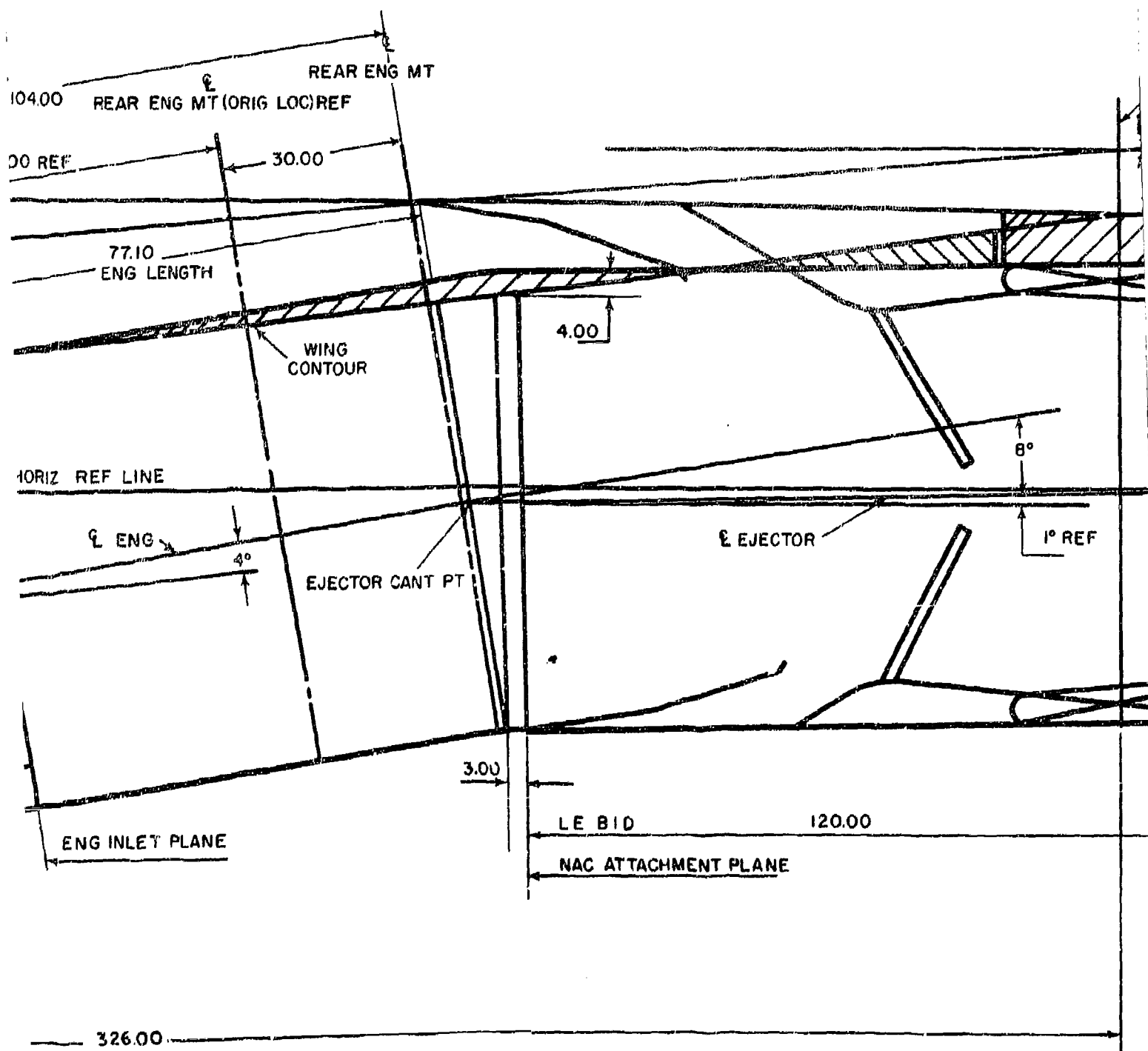
CONFIDENTIAL

[illegible]

PRATT & WHITNEY AIRCRAFT







CON

TE WING

4.3°

7.88

44.00

39.50

8°

1° REF

80.00

EJECTOR CANTED AT REAR MOUNT PLANE
ARRANGEMENT MOVING REAR ENG. MOUNT
30.00 AFT. & INSERTING ENG. 4.00 INTO
WING

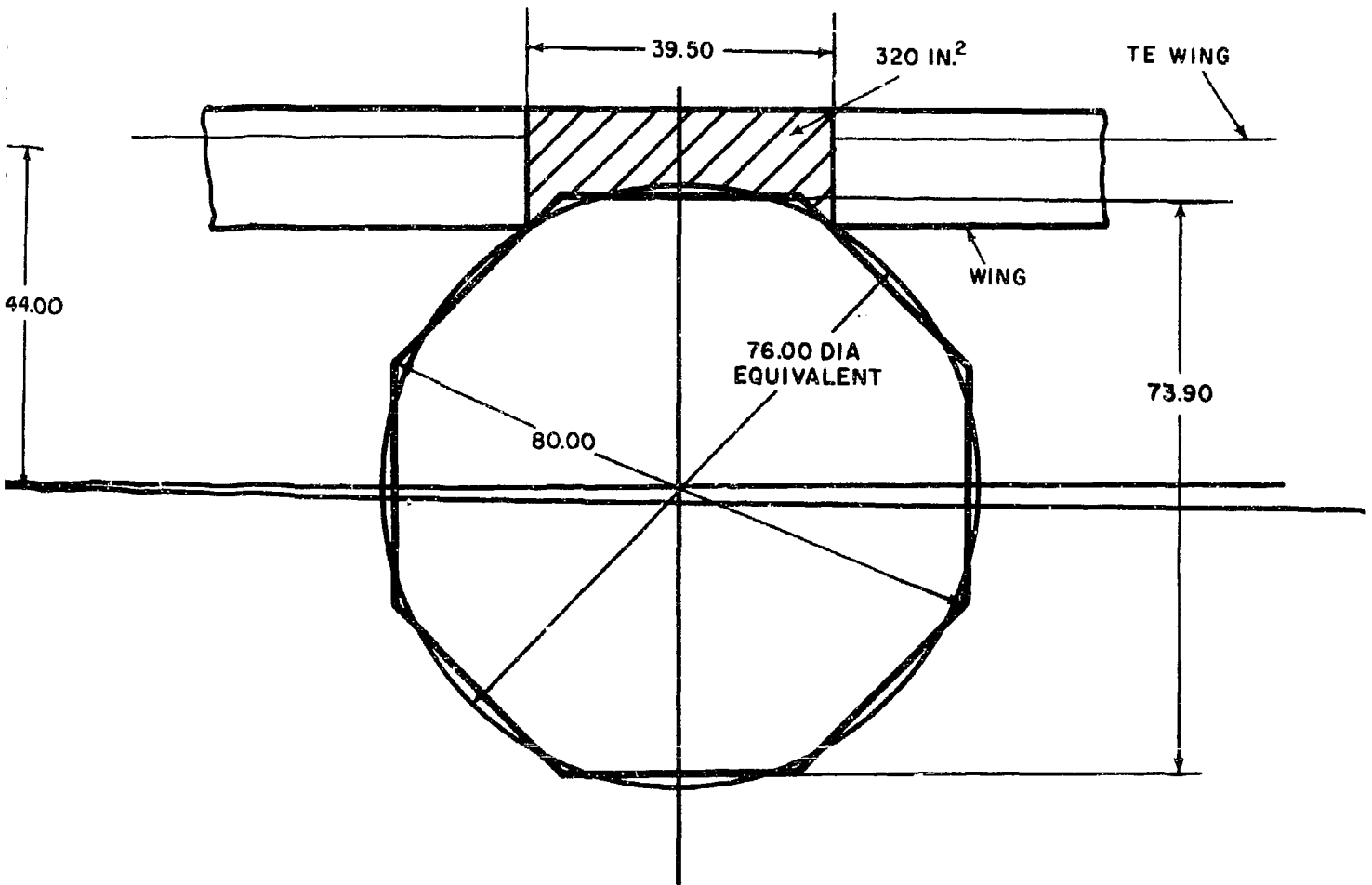
STF219B 600 1

CO

4

CONFIDENTIAL

PWA-2600



EJECTOR CANTED AT REAR MOUNT PLANE
ARRANGEMENT MOVING REAR ENG. MOUNT
30.00 AFT. & INSERTING ENG. 4.00 INTO
WING

STF219B 600 LBS./SEC. TURBOFAN

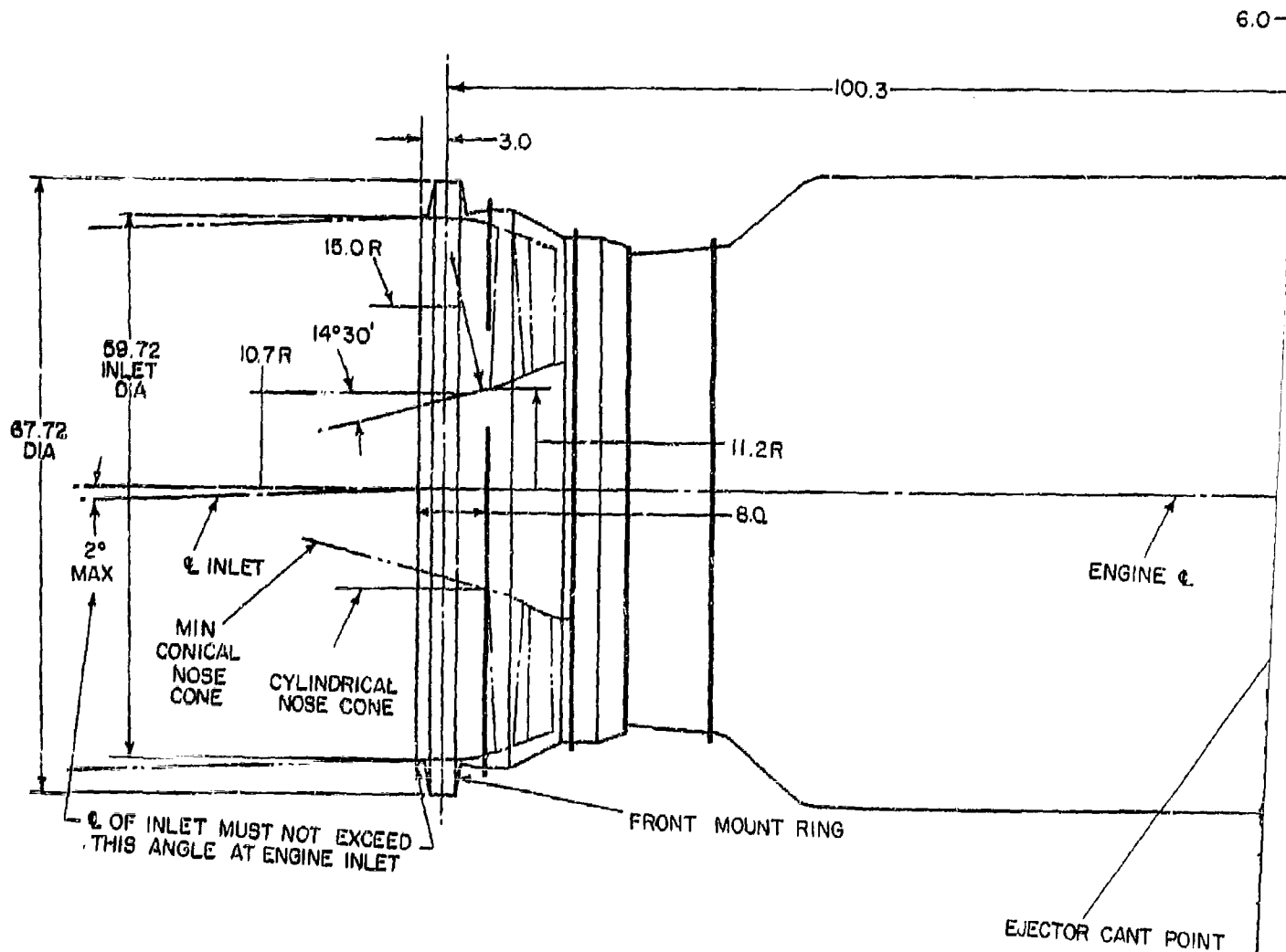
Figure 1-5

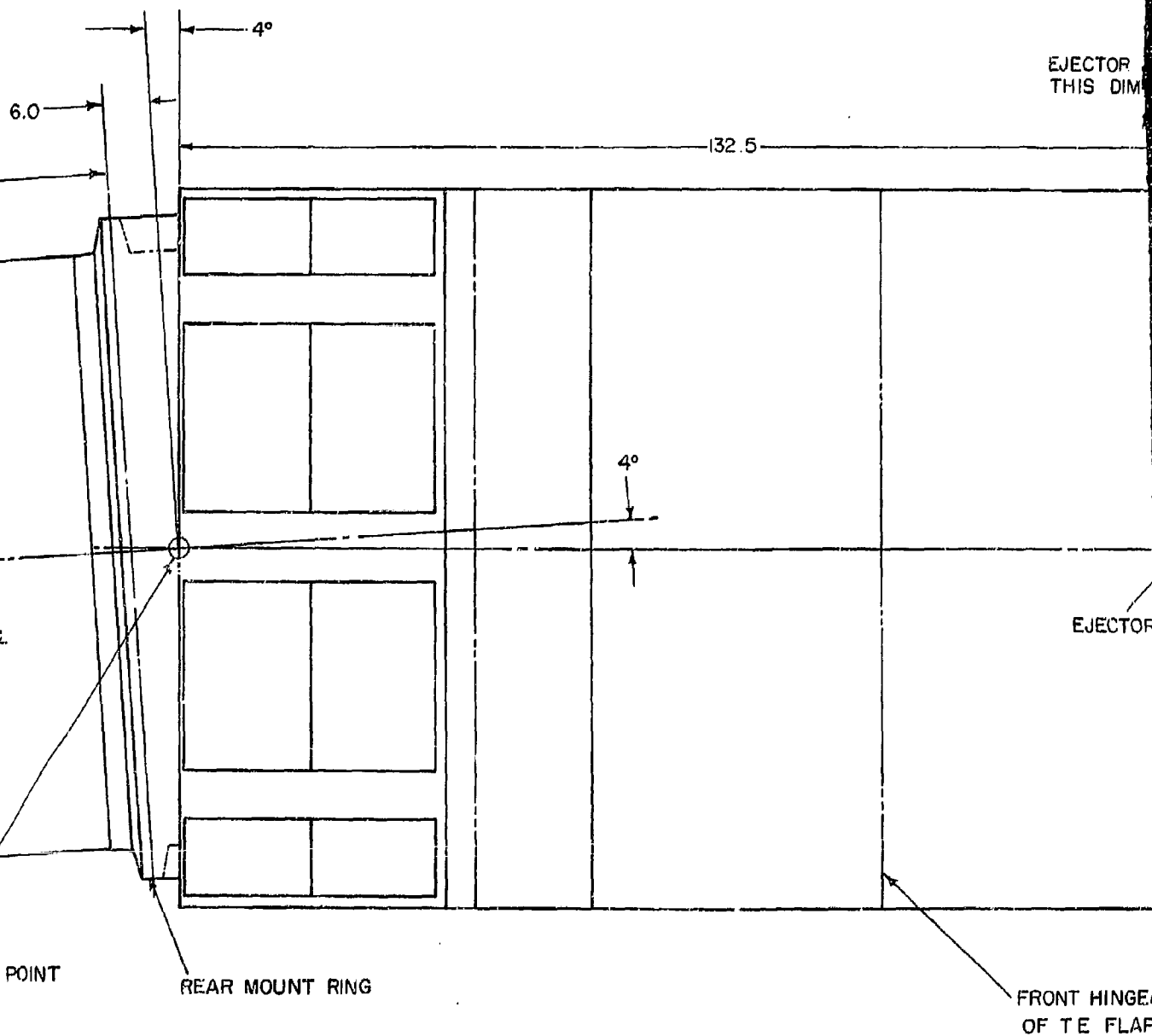
CONFIDENTIAL

DOWNLOADED AT 5 YEAR INTERVALS
UNCLASSIFIED AFTER 12 YEARS
DOW. DR. 83010

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.,
SECTION 793 AND 794, AND THE TRANSMISSION OF THE
REVELATION OF ITS CONTENTS TO ANY PERSON OR
AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

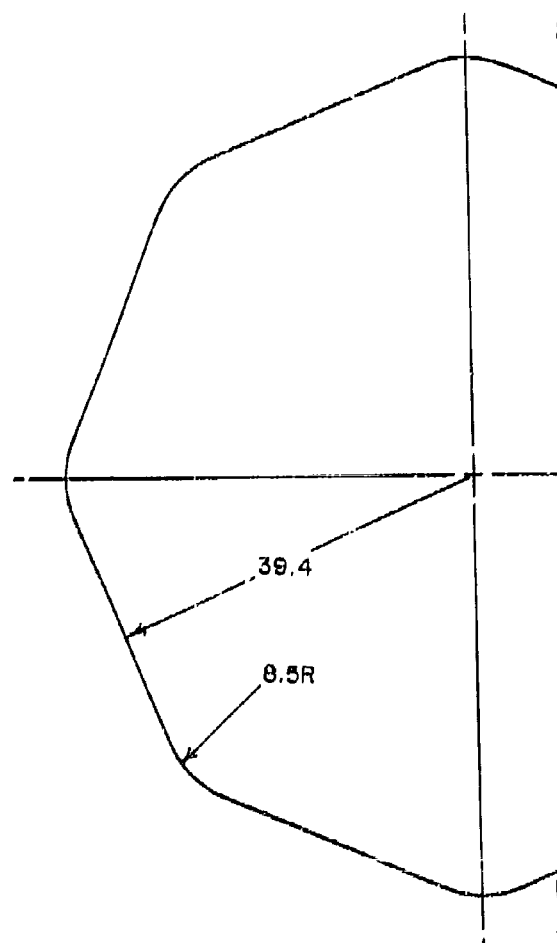
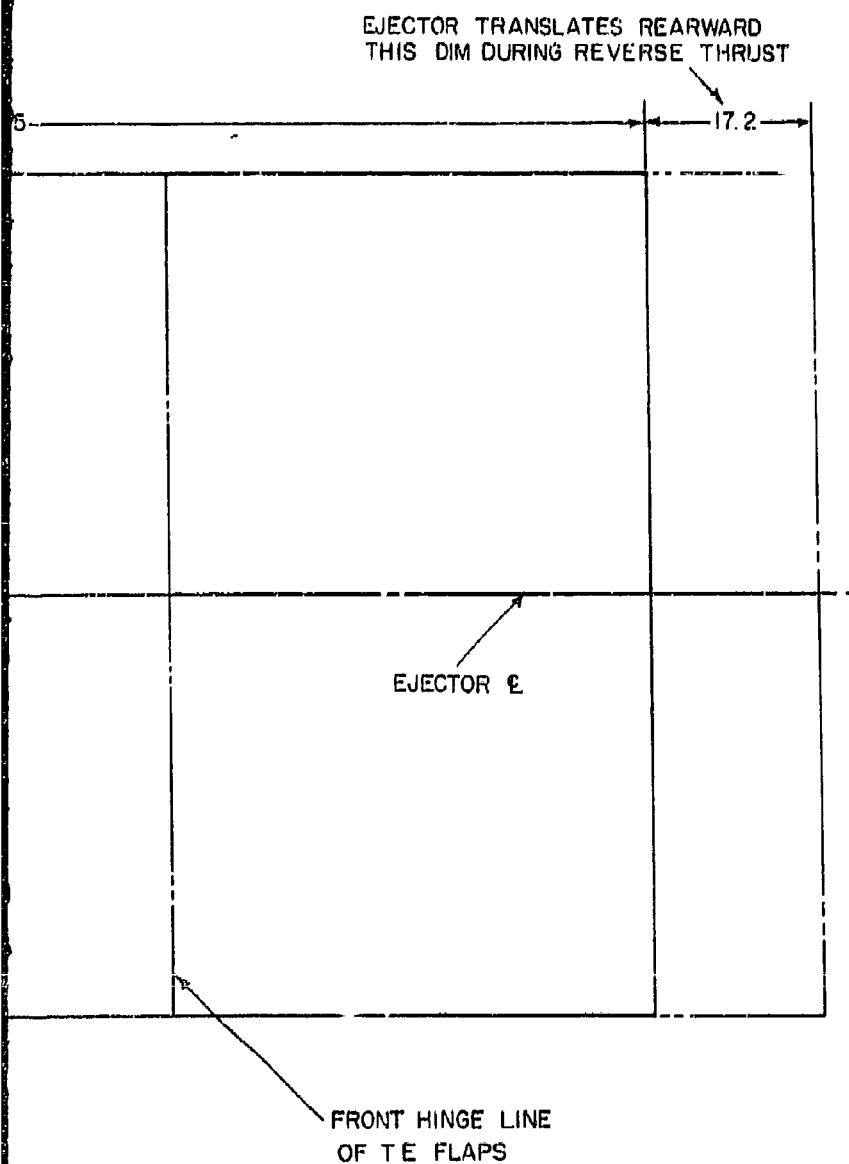
5





EJECTOR CANTED AT REAR
MOUNT PLANE

CONFIDENTIAL



REAR

STF219B 600 LBS. /SEC. TURBOFAN

Figure 1-6

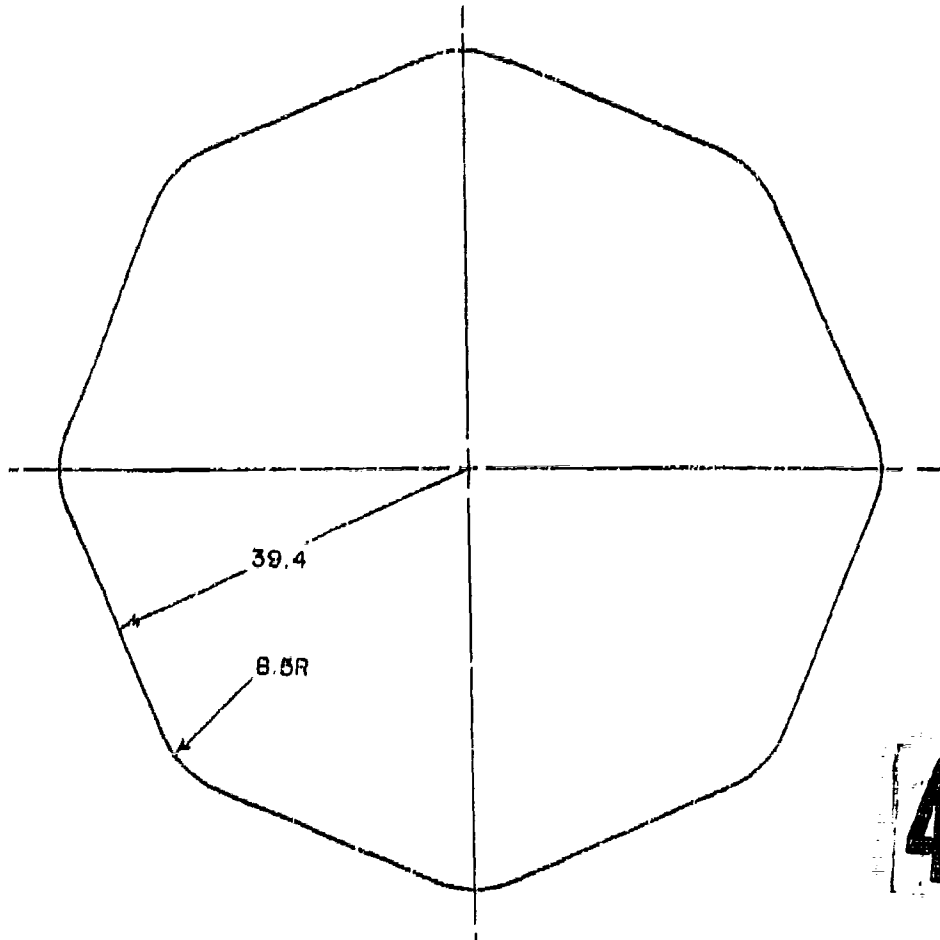
CONFIDENTIAL

CONFIDENTIAL

PWA-2600

ATES REARWARD
REVERSE THRUST

17.2



STF219B 600 LBS./SEC. TURBOFAN

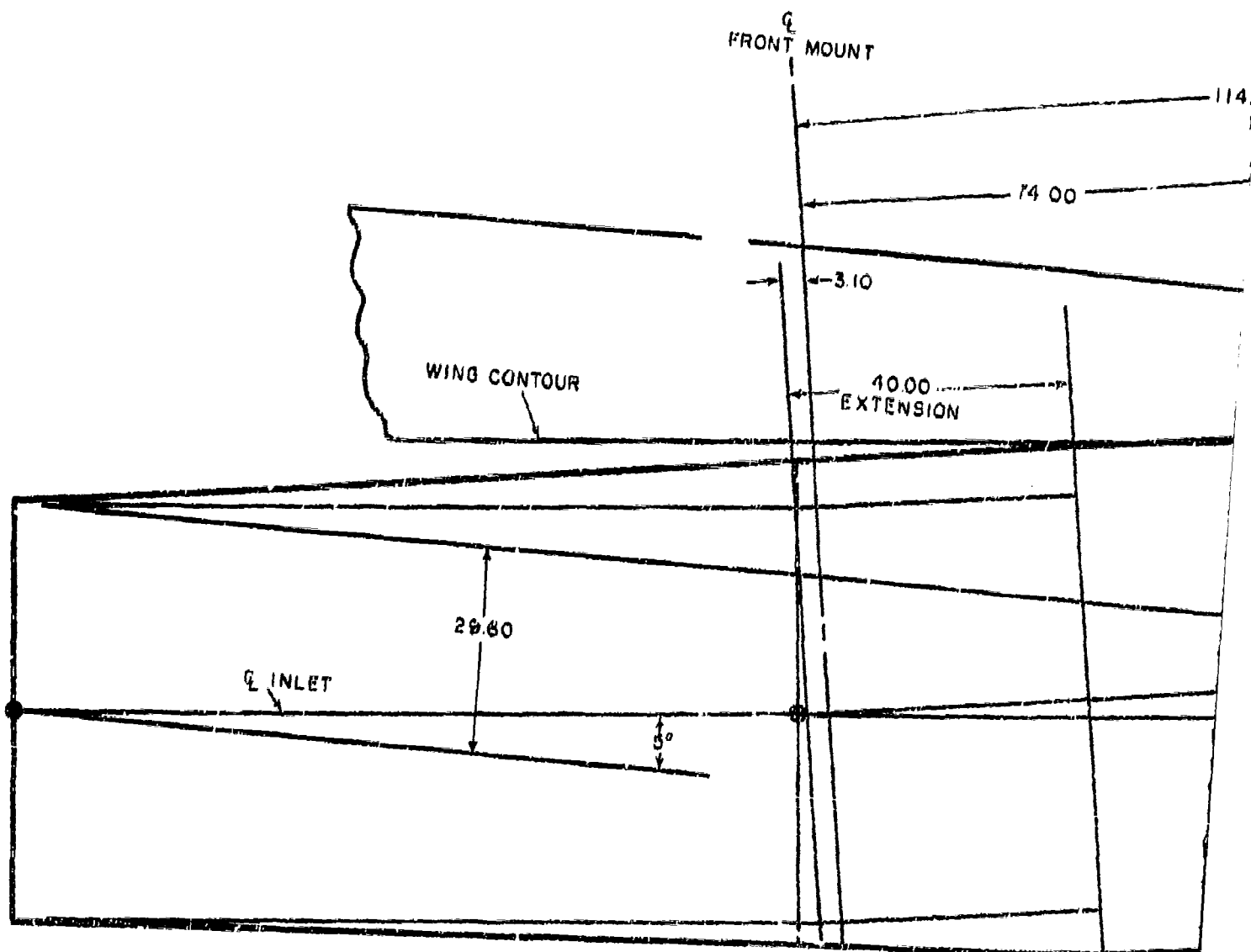
Figure 1-6

CONFIDENTIAL

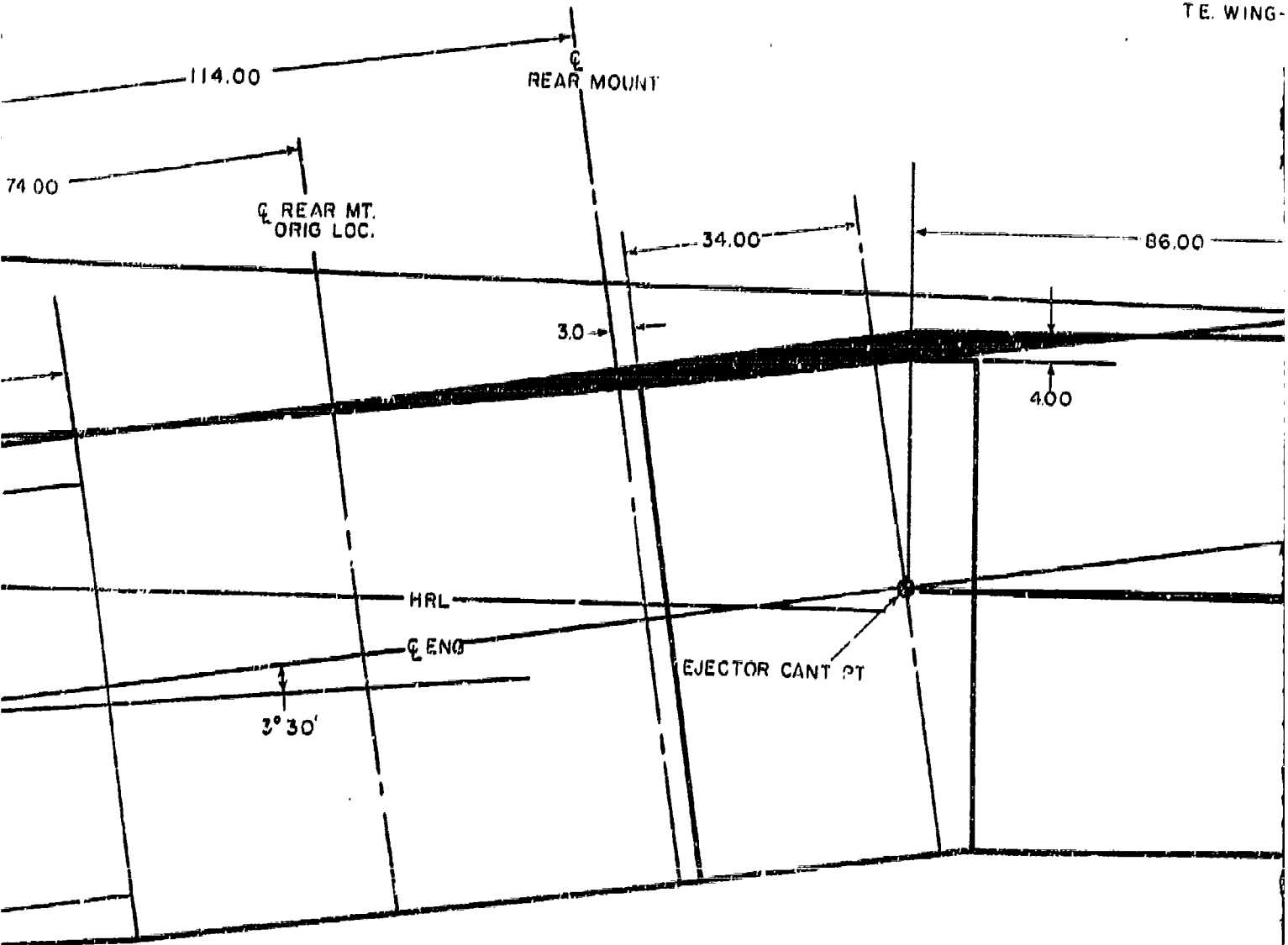
CONFIDENTIAL - SECURITY INFORMATION
EXCLUDED FROM AUTOMATIC DOWNGRADING AND
DECLASSIFICATION SCHEDULE

THIS DOCUMENT CONTAINS INFORMATION OF A
CONFIDENTIAL NATURE AND IS NOT TO BE
DISCLOSED TO THE PUBLIC OR TO OTHER
PERSONS WITHOUT AUTHORITY OF THE
SECRETARY OF THE AIR FORCE

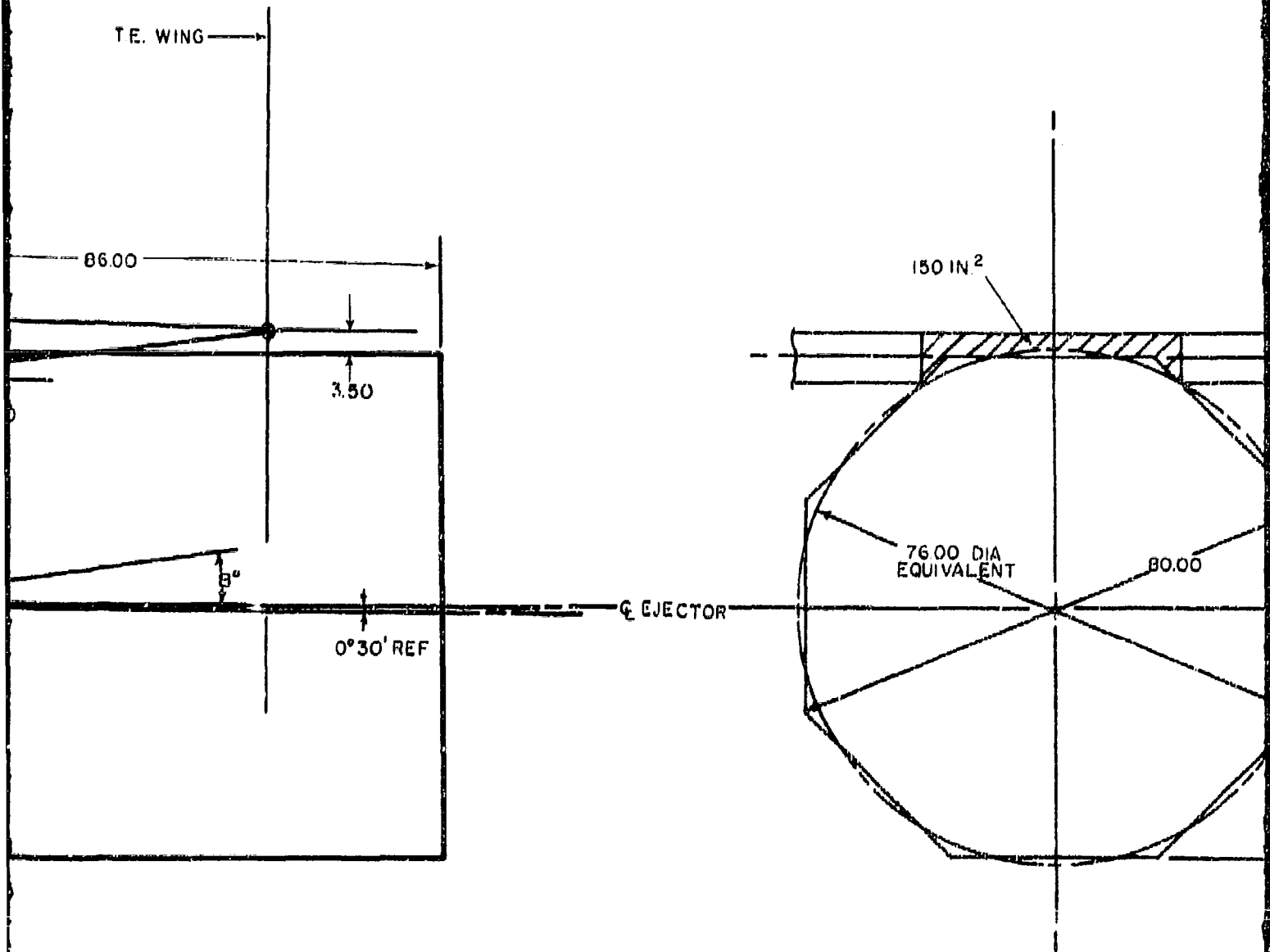
PRATT & WHITNEY AIRCRAFT



TE. WING-



CONFIDENTIAL



ARRANGEMENT MOVING REAR ENG. MT
40.00 & INSERTING ENG. 4.00 INTO WIN
EJECTOR CANTED AT NOZZLE PLANE
ENG. MOUNT PLANES ARE PARALLEL
BASE DRAG AREA = 150 in²

STF219B 600 LBS. /SEC. TURBOFAN

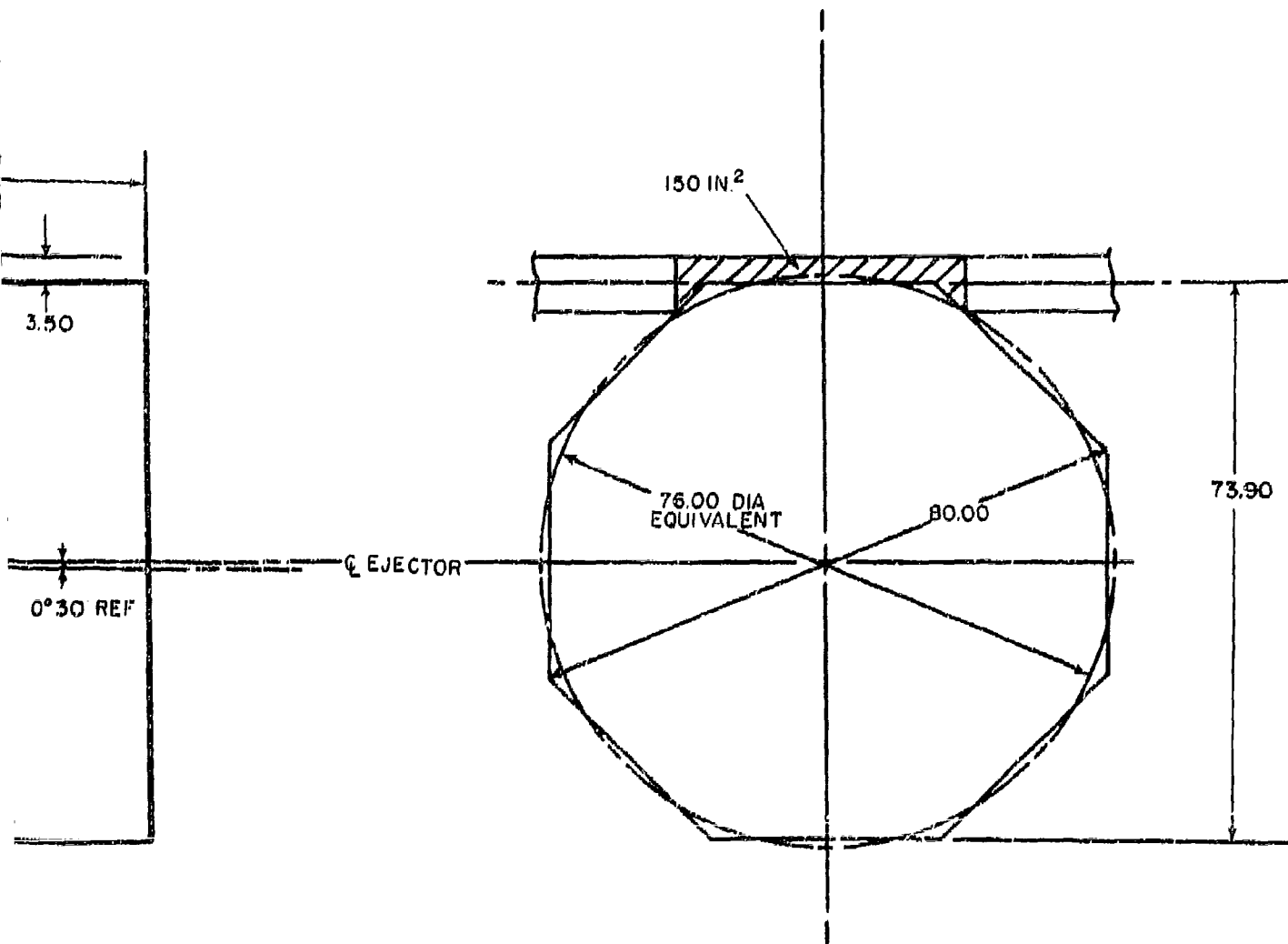
Figure 1-7

CONFIDENTIAL

UNCLASSIFIED AT 8-10
UNCLASSIFIED BY 1010
DATE 01-10-01
FOR MORE INFORMATION SEE 1010
UNCLASSIFIED BY 1010
DATE 01-10-01
FOR MORE INFORMATION SEE 1010

CONFIDENTIAL

PWA-2600



ARRANGEMENT MOVING REAR ENG. MT AFT
40.00 & INSERTING ENG. 4.00 INTO WING
EJECTOR CANTED AT NOZZLE PLANE
ENG. MOUNT PLANES ARE PARALLEL
BASE DRAG AREA = 150 in²

STF219B 600 LBS./SEC. TURBOFAN

Figure 1-7

CONFIDENTIAL

1. THIS DRAWING IS TO BE USED FOR THE PURPOSES OF THE PROJECT ONLY.
2. IT IS NOT TO BE USED FOR ANY OTHER PURPOSE.
3. IT IS NOT TO BE REPRODUCED OR TRANSMITTED IN ANY FORM OR BY ANY MEANS, ELECTRONIC OR MECHANICAL, INCLUDING PHOTOCOPYING, RECORDING, OR BY ANY INFORMATION STORAGE AND RETRIEVAL SYSTEM.
4. IT IS NOT TO BE DISTRIBUTED OUTSIDE THE PROJECT.

FRONT END

3.10

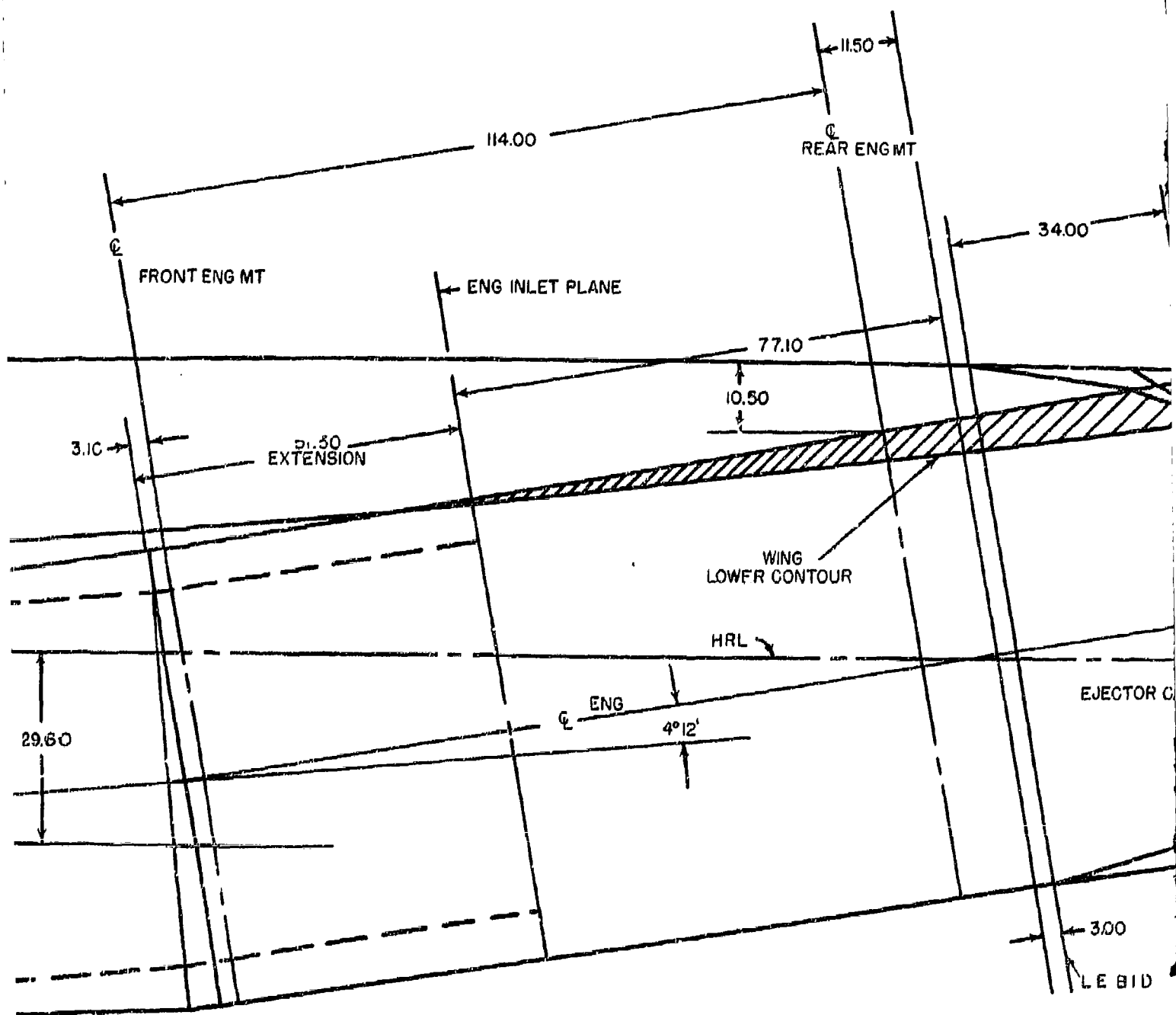
29.60

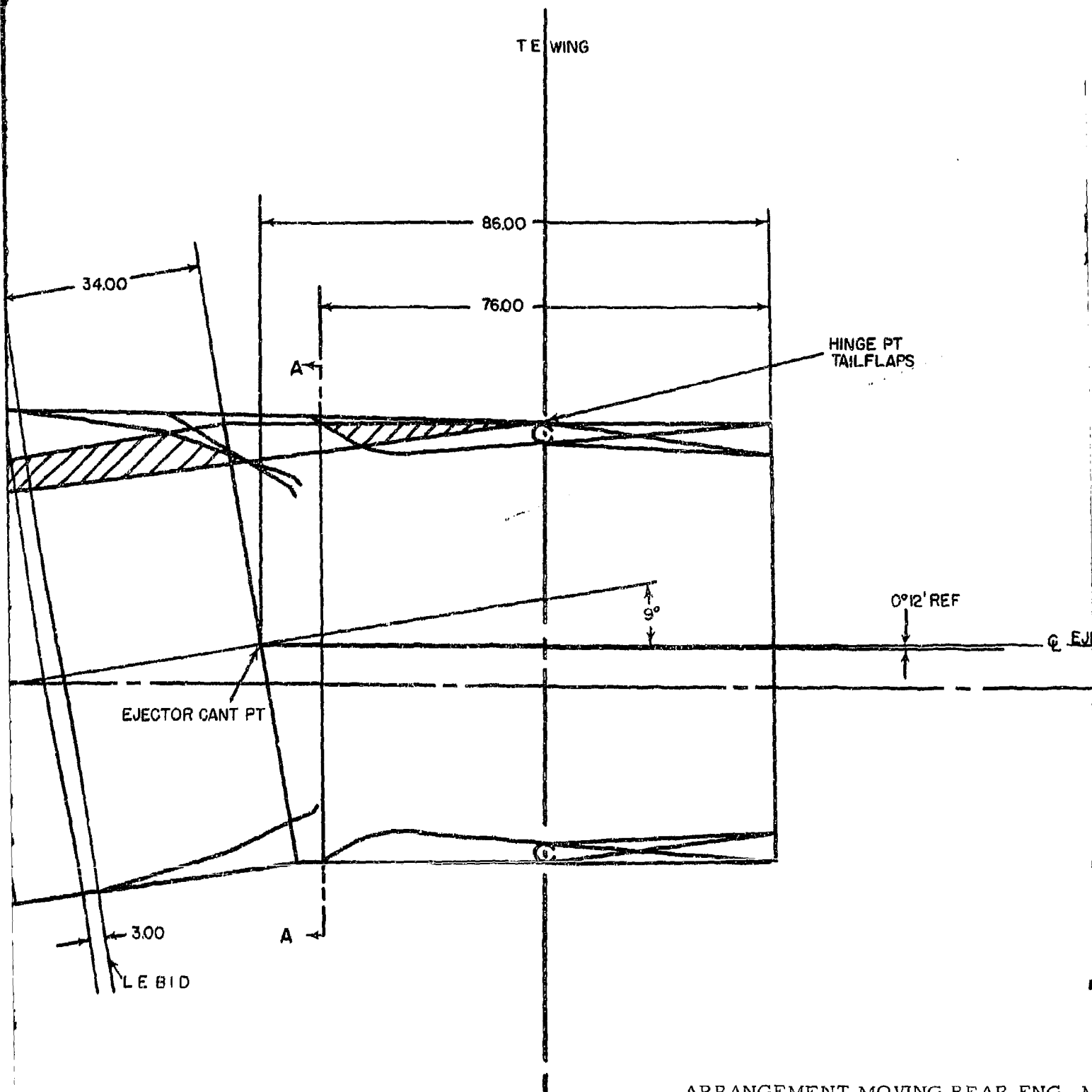
INLET

5"

WING PIVOT PLANE

1

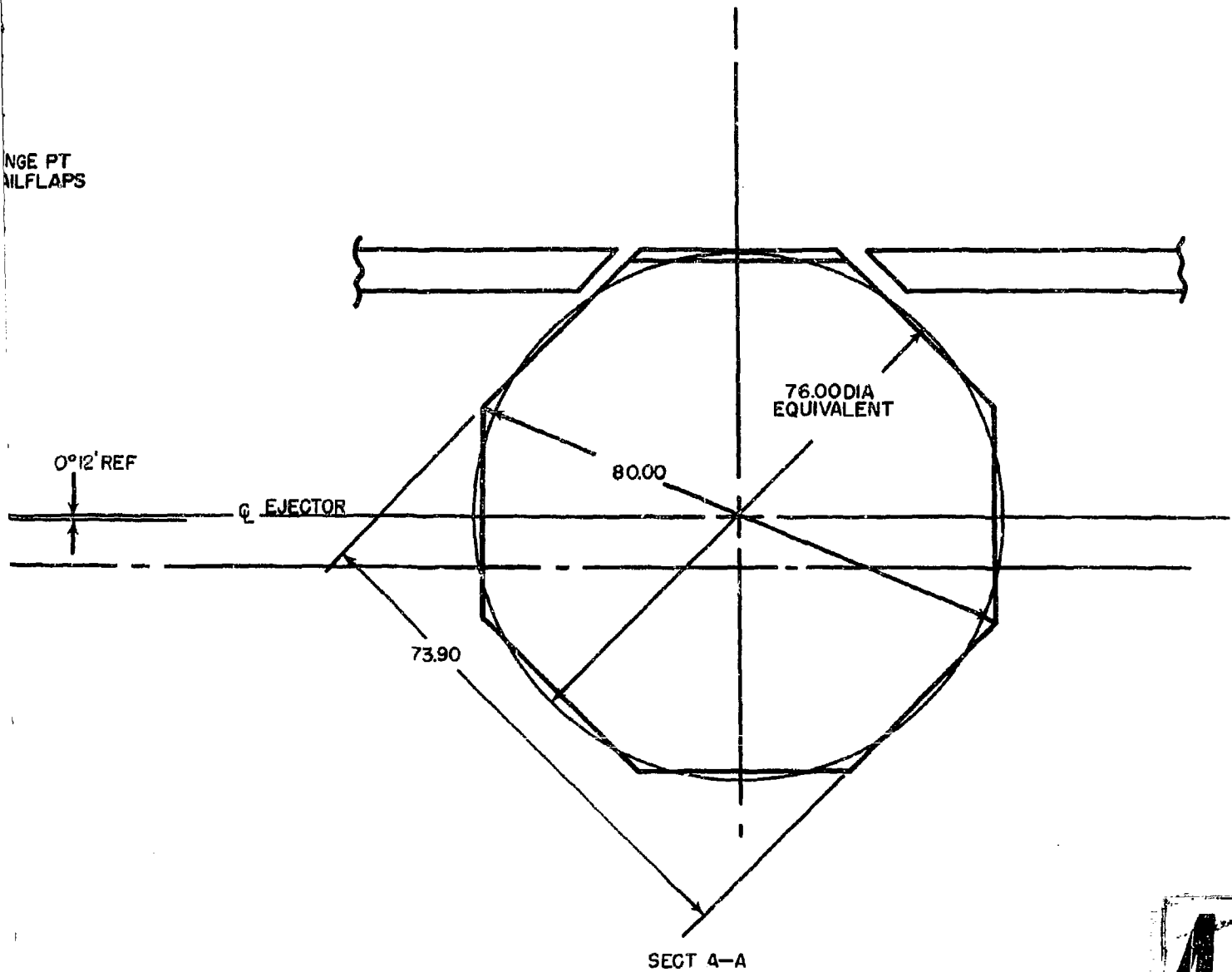




ARRANGEMENT MOVING REAR ENG. M
40.00 AFT & SETTING EJECTOR TAIL F
AT T. E. OF WING
EJECTOR CANTED AT NOZZLE PLANE

CONFIDENTIAL

PWA-2600



MOVING REAR ENG. MT.
EJECTOR TAIL FLAPS
NG
ED AT NOZZLE PLANE

STF219B 600 LBS./SEC. TURBOFAN

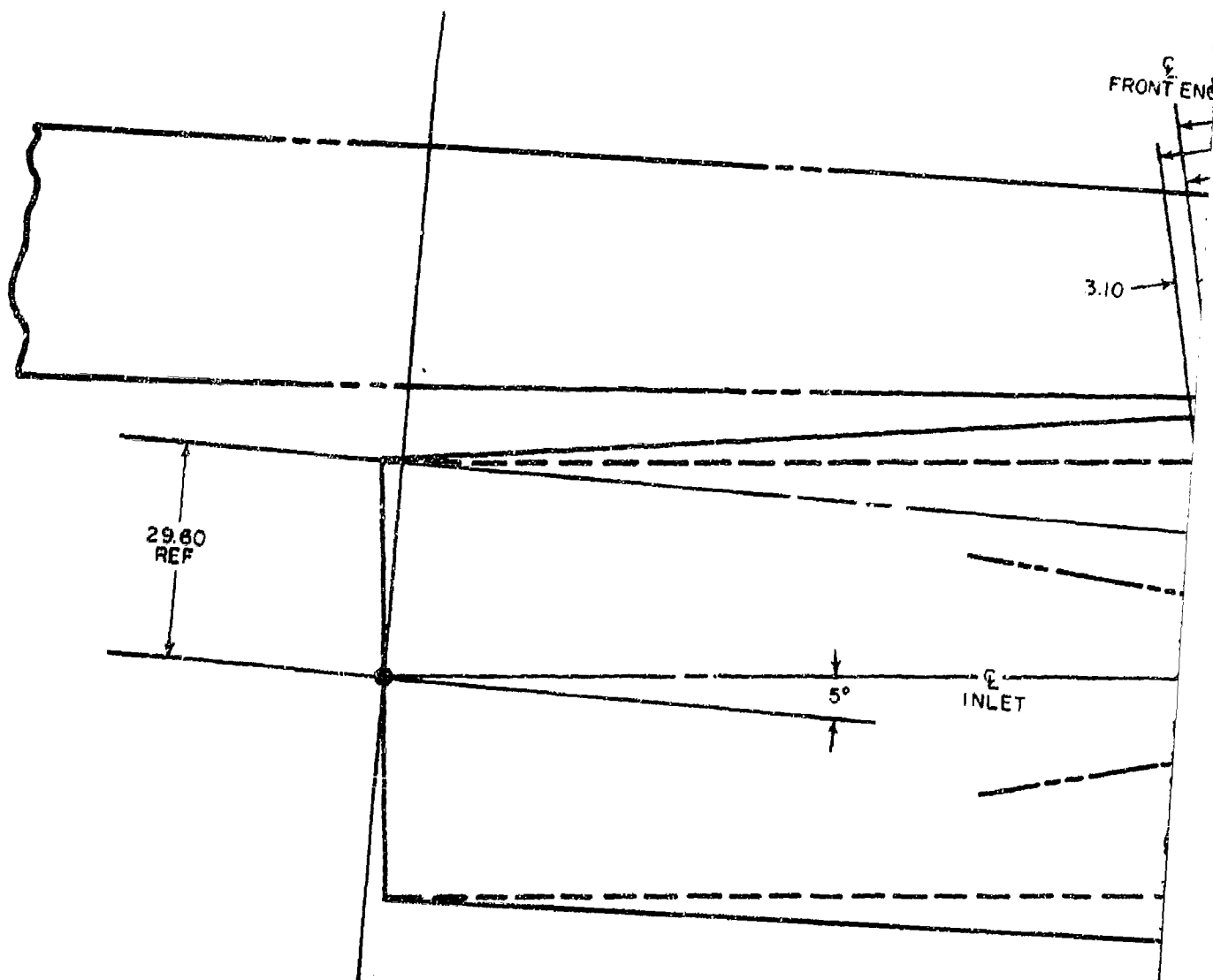
Figure 1-8

CONFIDENTIAL

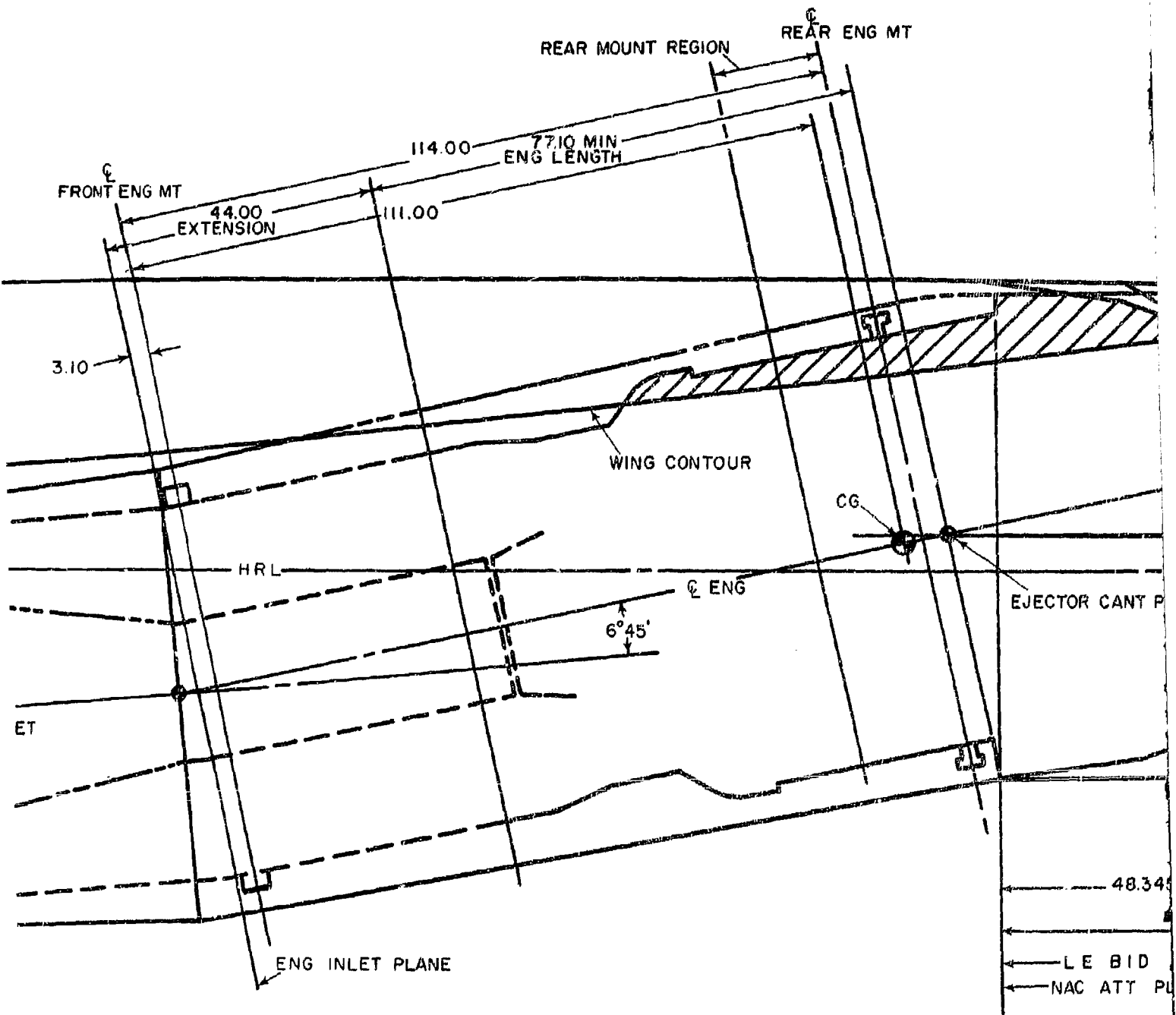
DECLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DDO GPO 820010

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE INTELLIGENCE INFORMATION ACT, U.S.C. 1-1
SECTIONS 1-101 AND 1-102. THE TRANSMISSION OR THE
REVELATION OF ITS CONTENTS IN ANY MANNER TO AN
UNAUTHORIZED PERSON IS PROHIBITED BY LAW

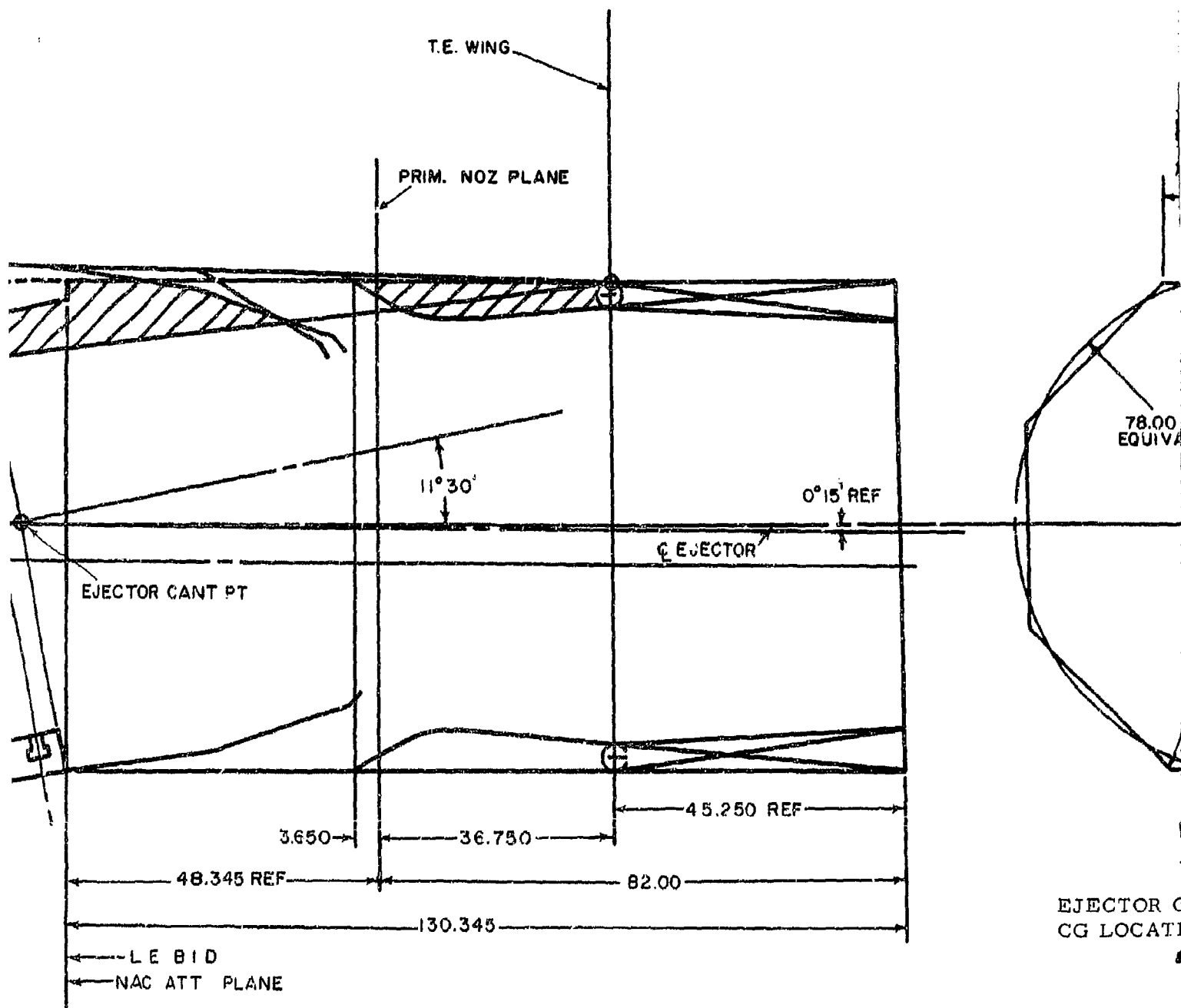
PRATT & WHITNEY AIRCRAFT



1



CONFIDENTIAL



STF219B 600 LBS. /S

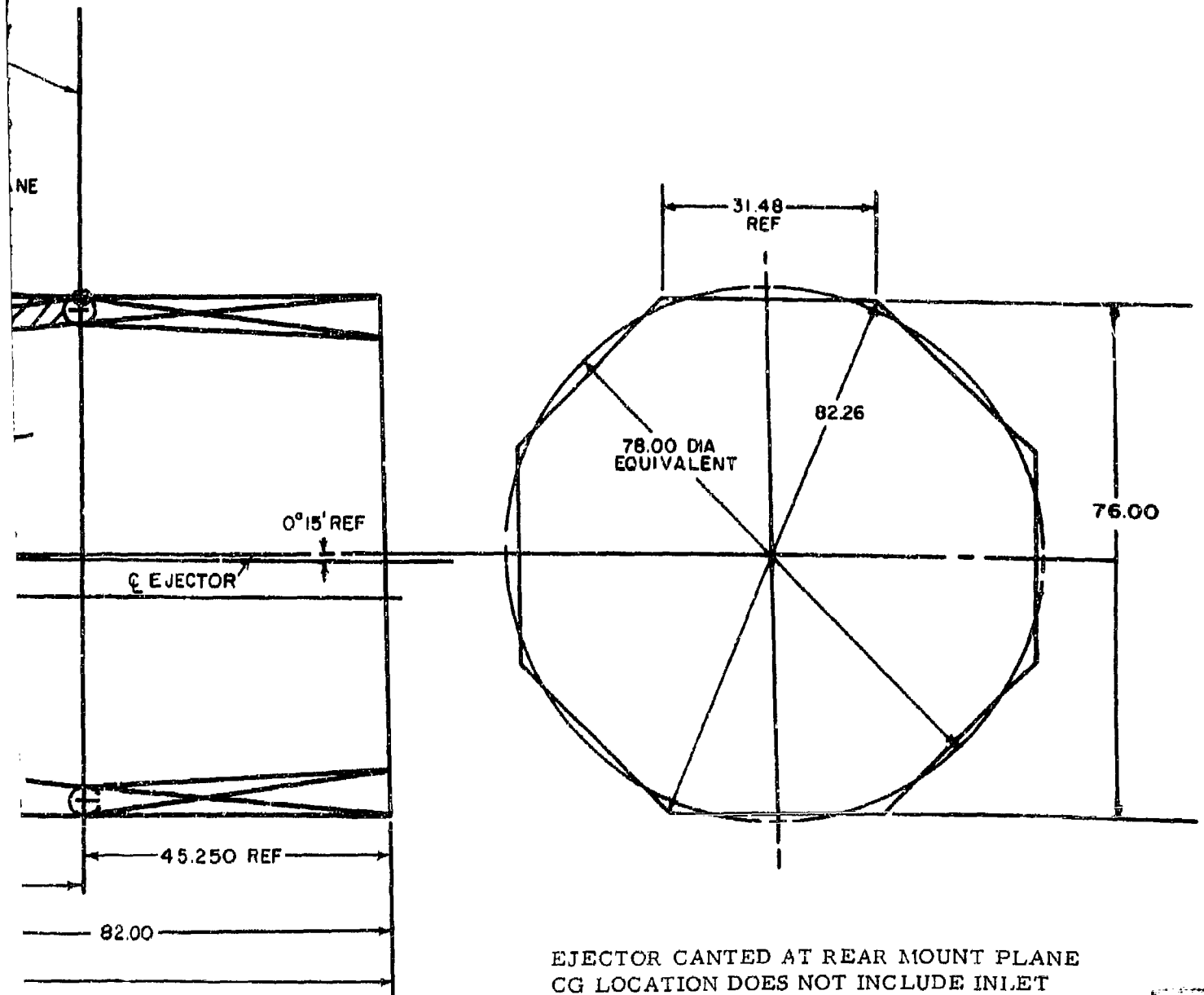
Figure

CONFIDENTIAL

3

CONFIDENTIAL

PWA-2600



STF219B 600 LBS./SEC. TURBOFAN

Figure 1-9

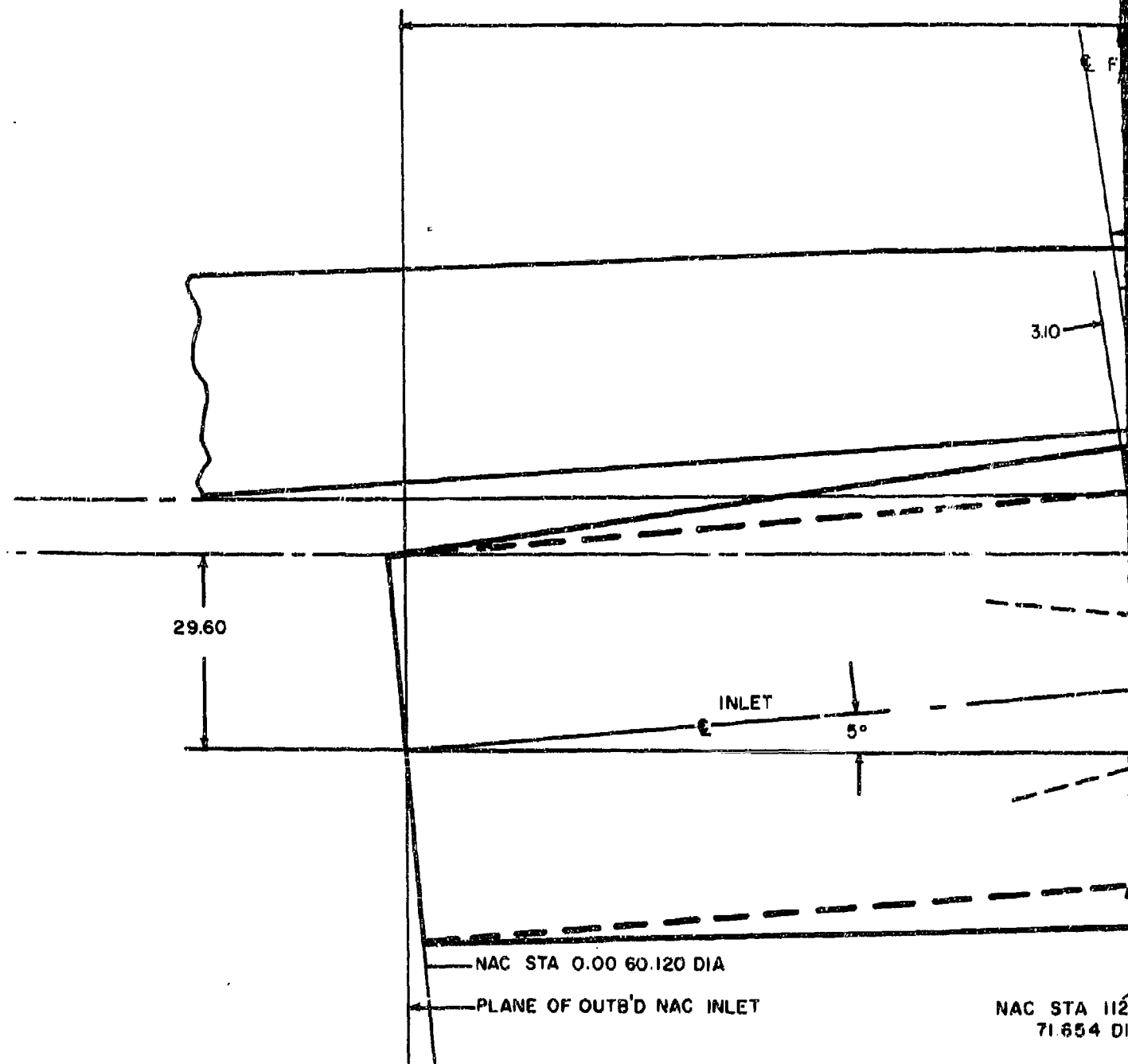
CONFIDENTIAL

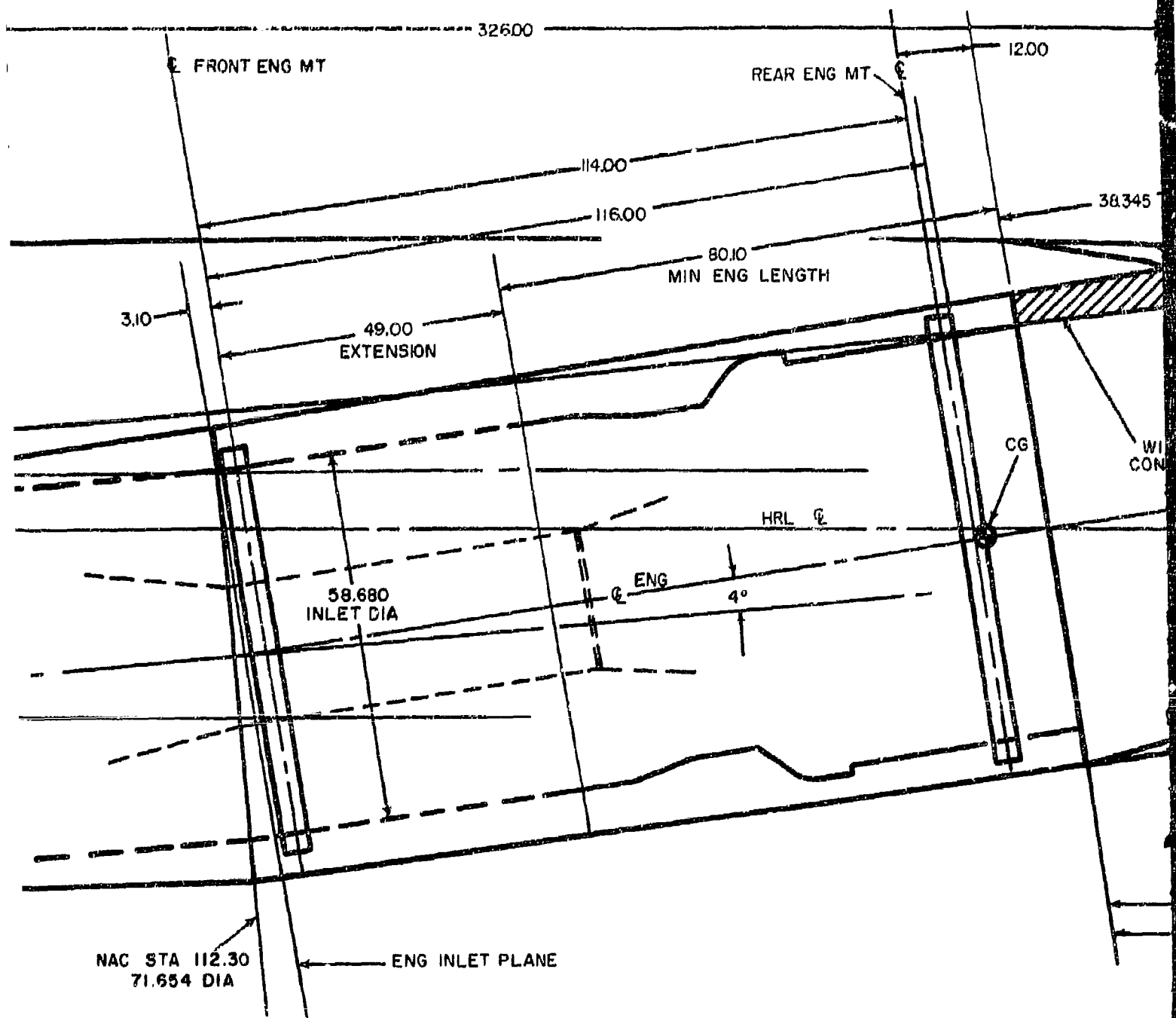
UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
GPO: 1981 O-800-10

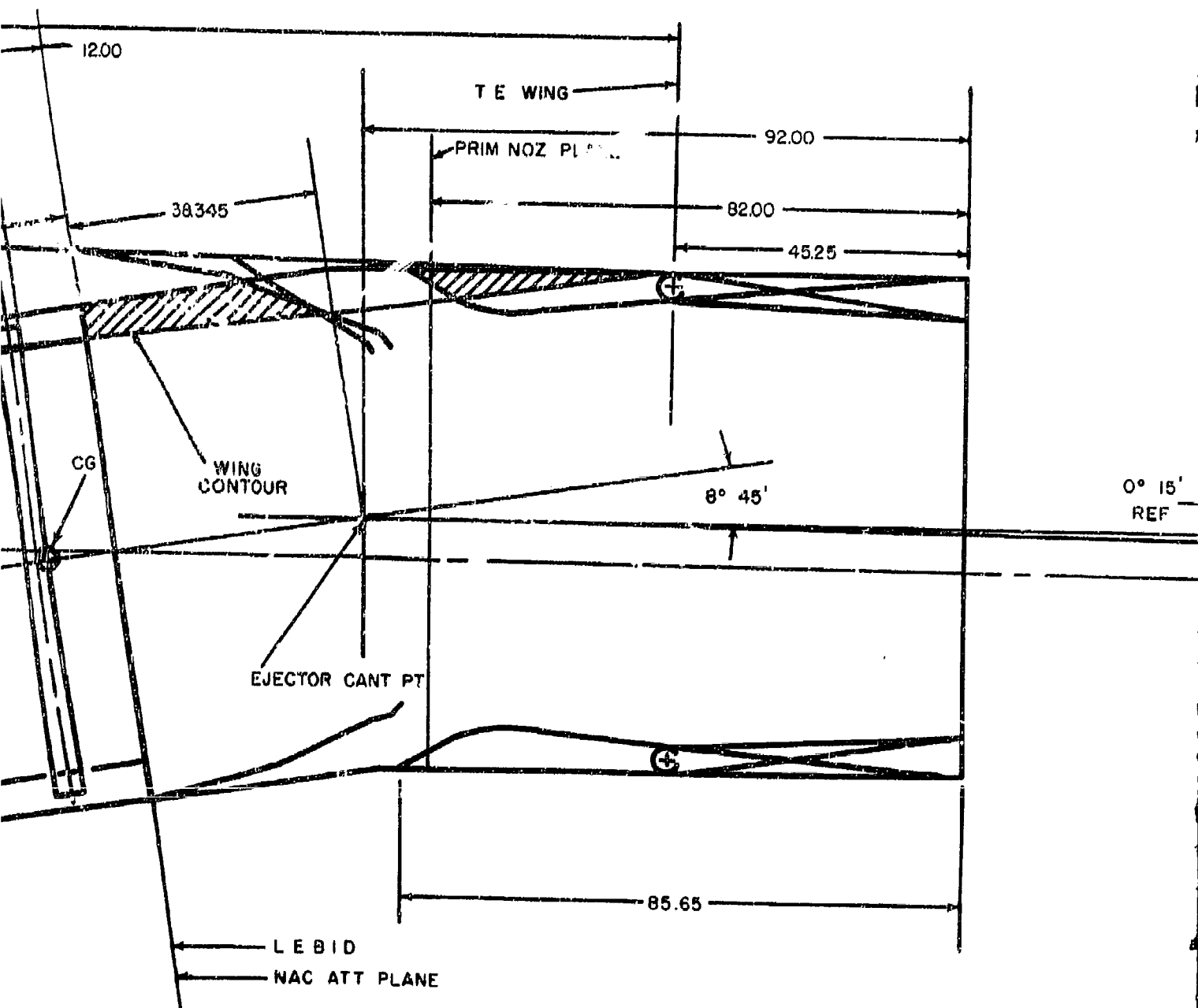
THIS DOCUMENT CONTAINS INFORMATION SPECIFIC TO THE
NATIONAL DEFENSE OF THE UNITED STATES. IT IS THE
POLICY OF THE DEPARTMENT OF DEFENSE TO
CLASSIFY AND CONTROL THE DISSEMINATION OF
INFORMATION TO THE MAXIMUM EXTENT POSSIBLE
AND TO DECLASSIFY IT WHEN IT IS NO LONGER
NECESSARY FOR THE NATIONAL DEFENSE.

4

PRATT & WHITNEY AIRCRAFT

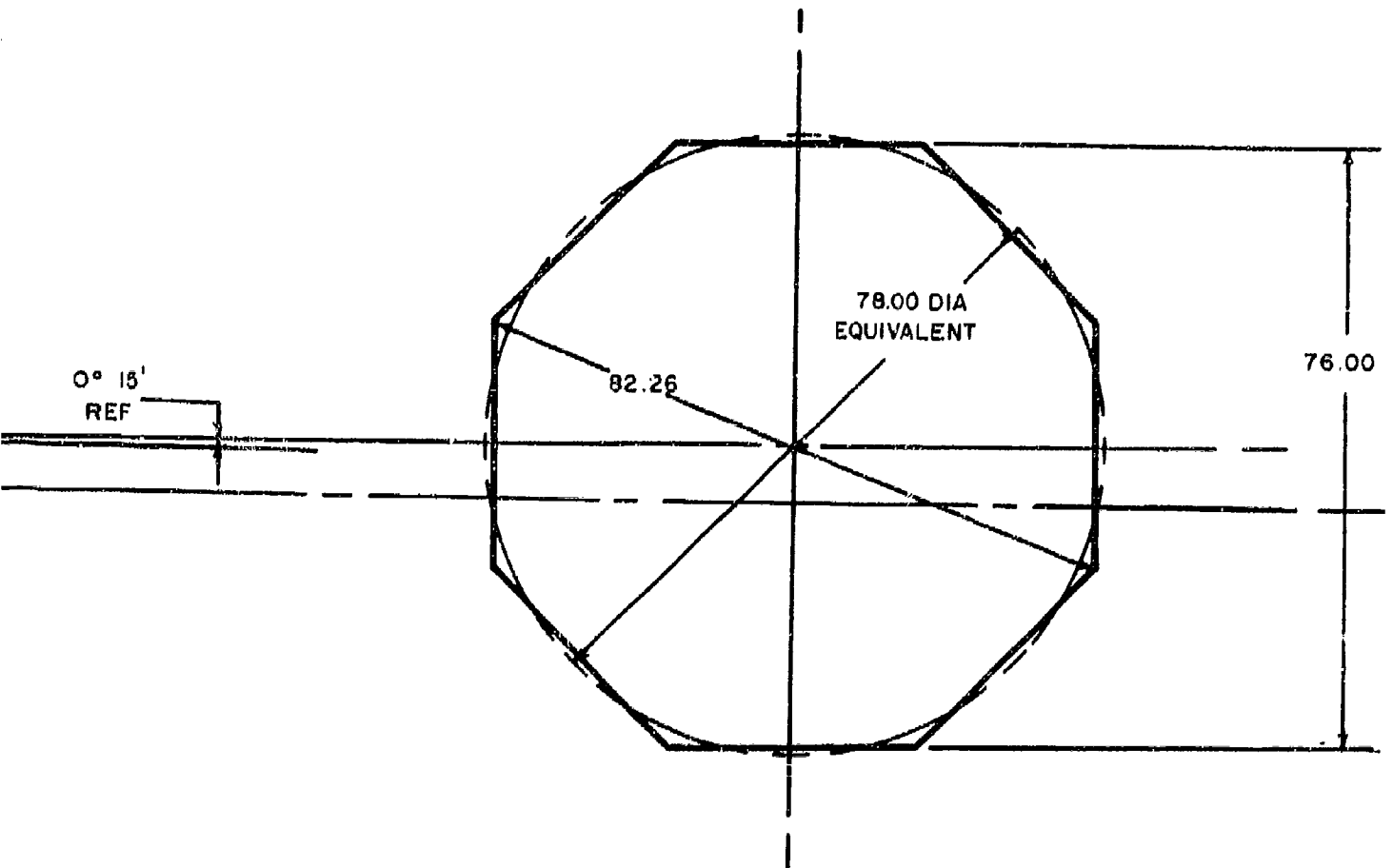






CONFIDENTIAL

PWA-2600



EJECTOR CANTED 10.00 FWD. OF PRIMARY
NOZZLE PLANE
CG LOCATION DOES NOT INCLUDE INLET

STF219B 600 LBS./SEC. TURBOFAN

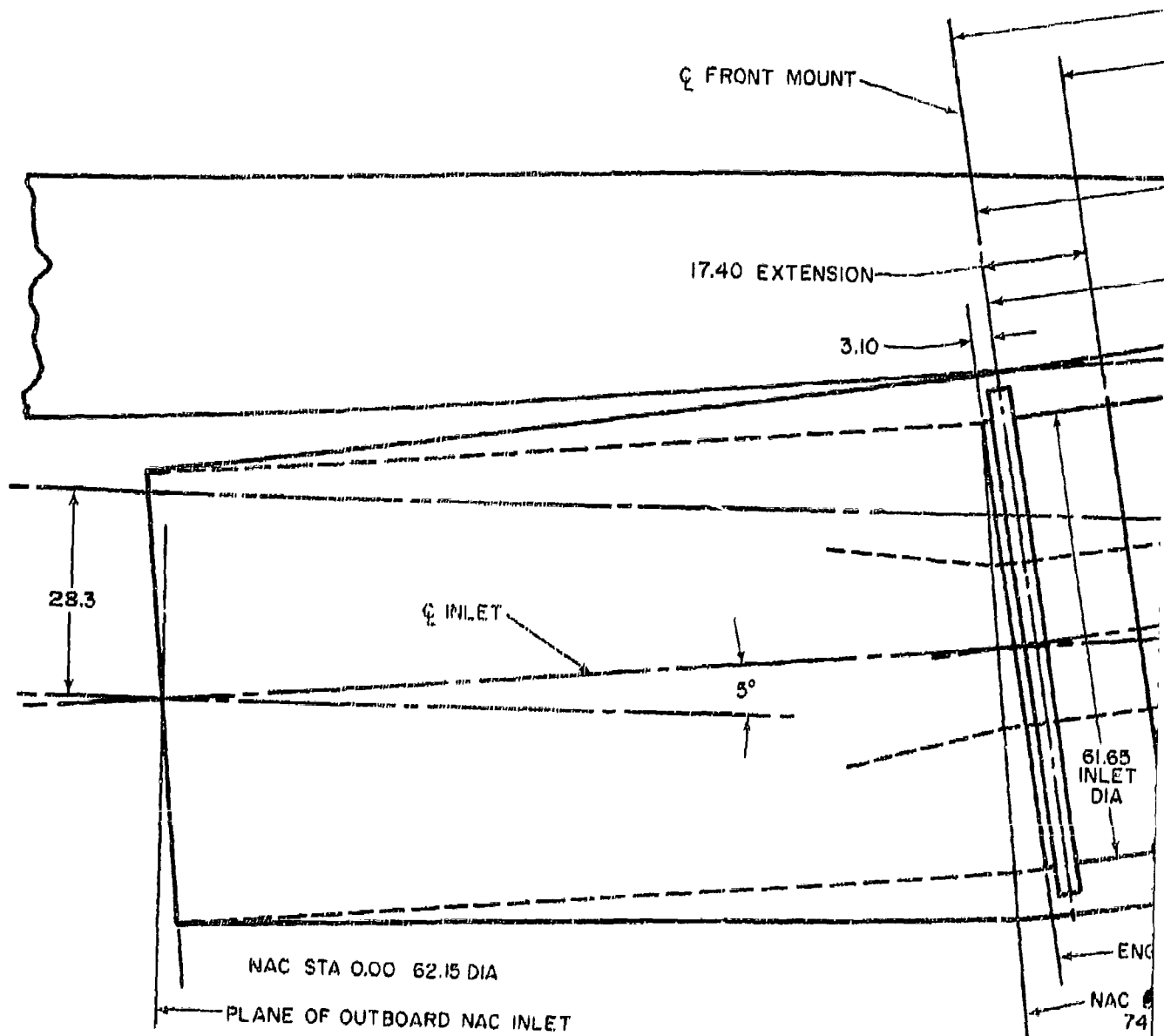
Figure 1-10

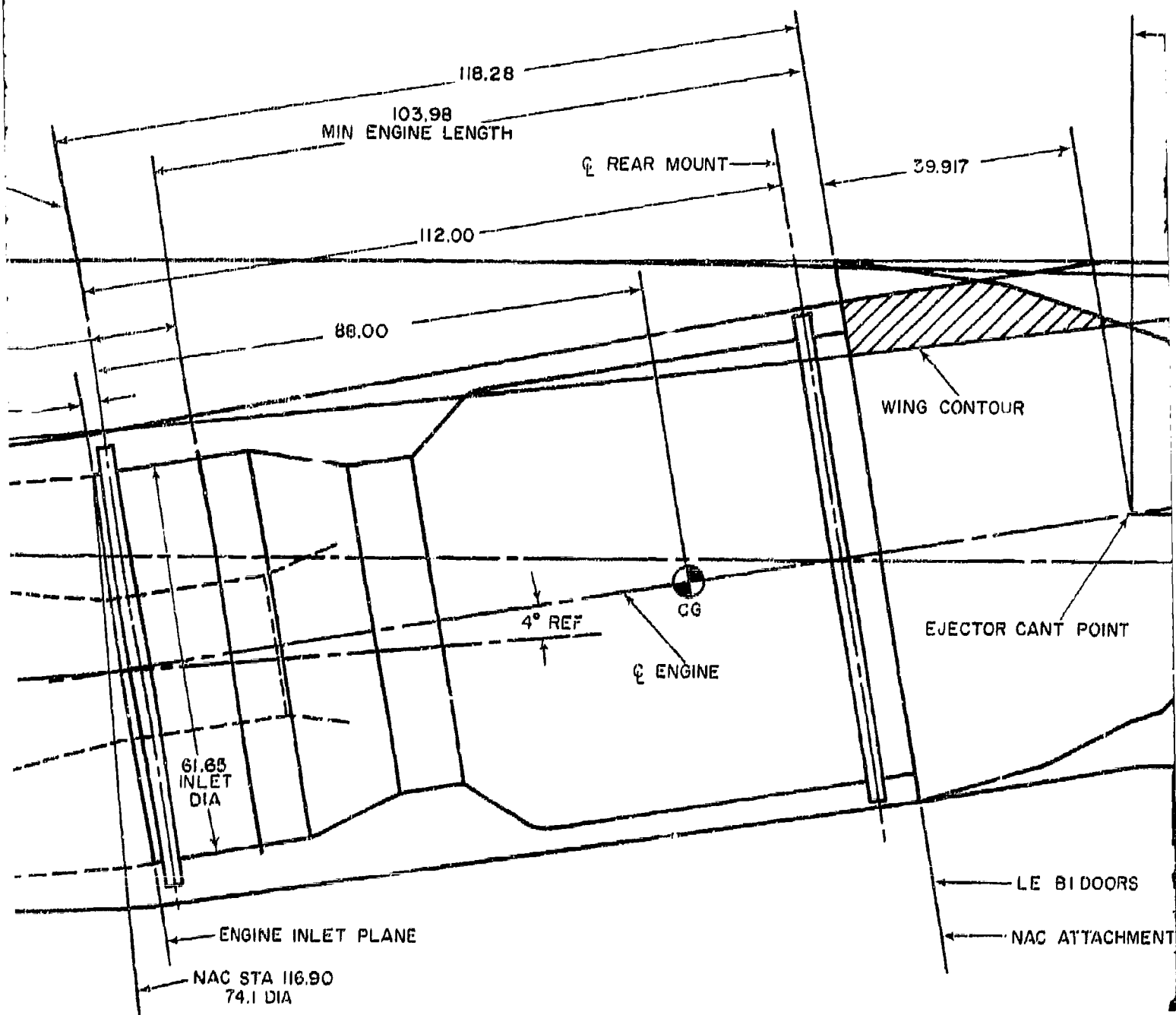
CONFIDENTIAL

UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 10 YEARS
GDS FOR EMD-0
THIS DOCUMENT CONTAINS INFORMATION RELATIVE TO THE
NATIONAL DEFENSE OF THE UNITED STATES WHICH IS
THE PROPERTY OF THE UNITED STATES GOVERNMENT. IT IS
EXTREMELY CONFIDENTIAL AND ITS DISCLOSURE TO THE
PUBLIC OR TO ANY OTHER PERSON IS PROHIBITED BY LAW.
UNCLASSIFIED VERSION IS PROHIBITED BY LAW

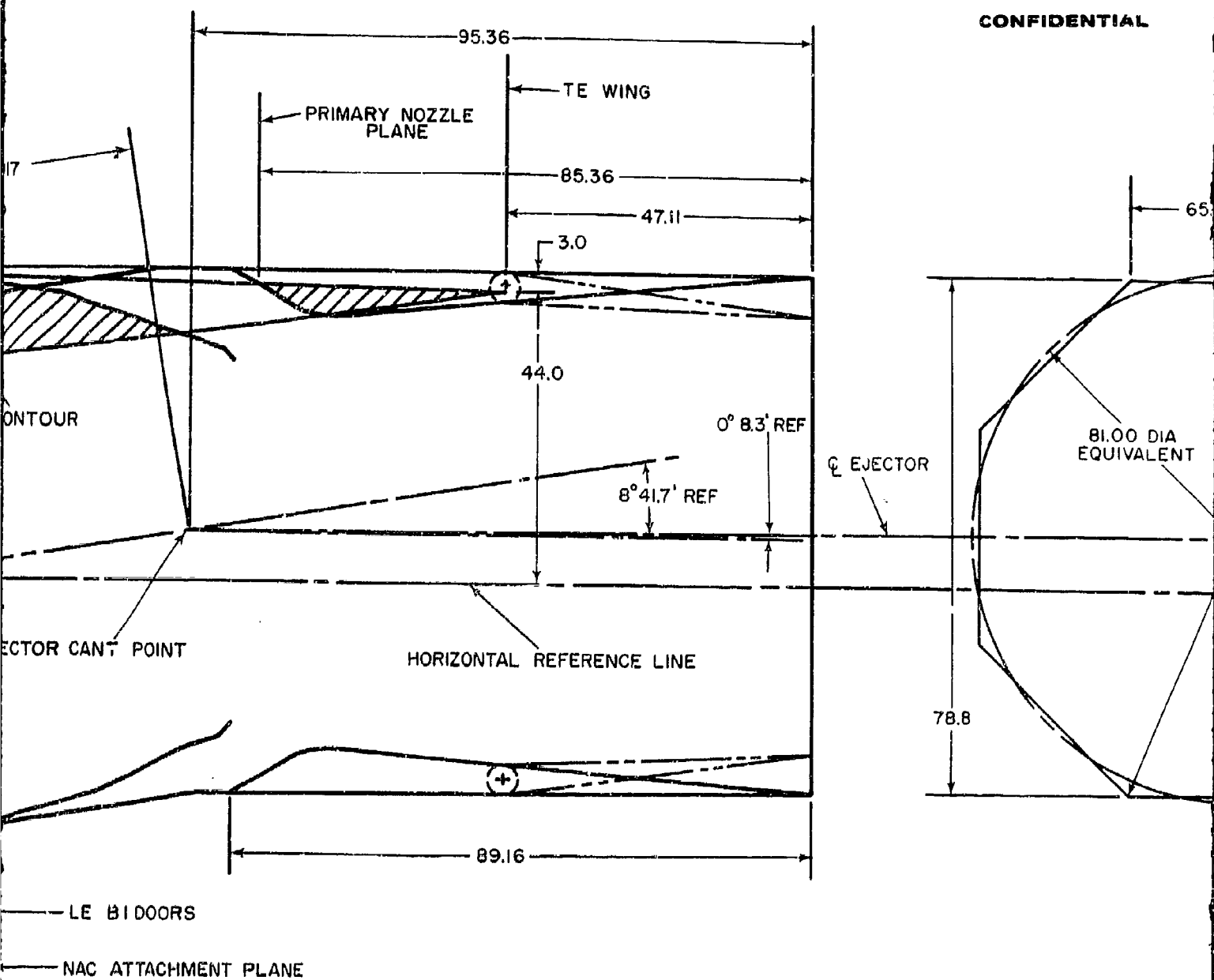
4

PRATT & WHITNEY AIRCRAFT





CONFIDENTIAL



CG LOCATION DOES NOT INCLUDE INLET

EJECTOR CAN
NOZZLE PL
CG LOCATION

STF219 650 LBS./SEC. TUR

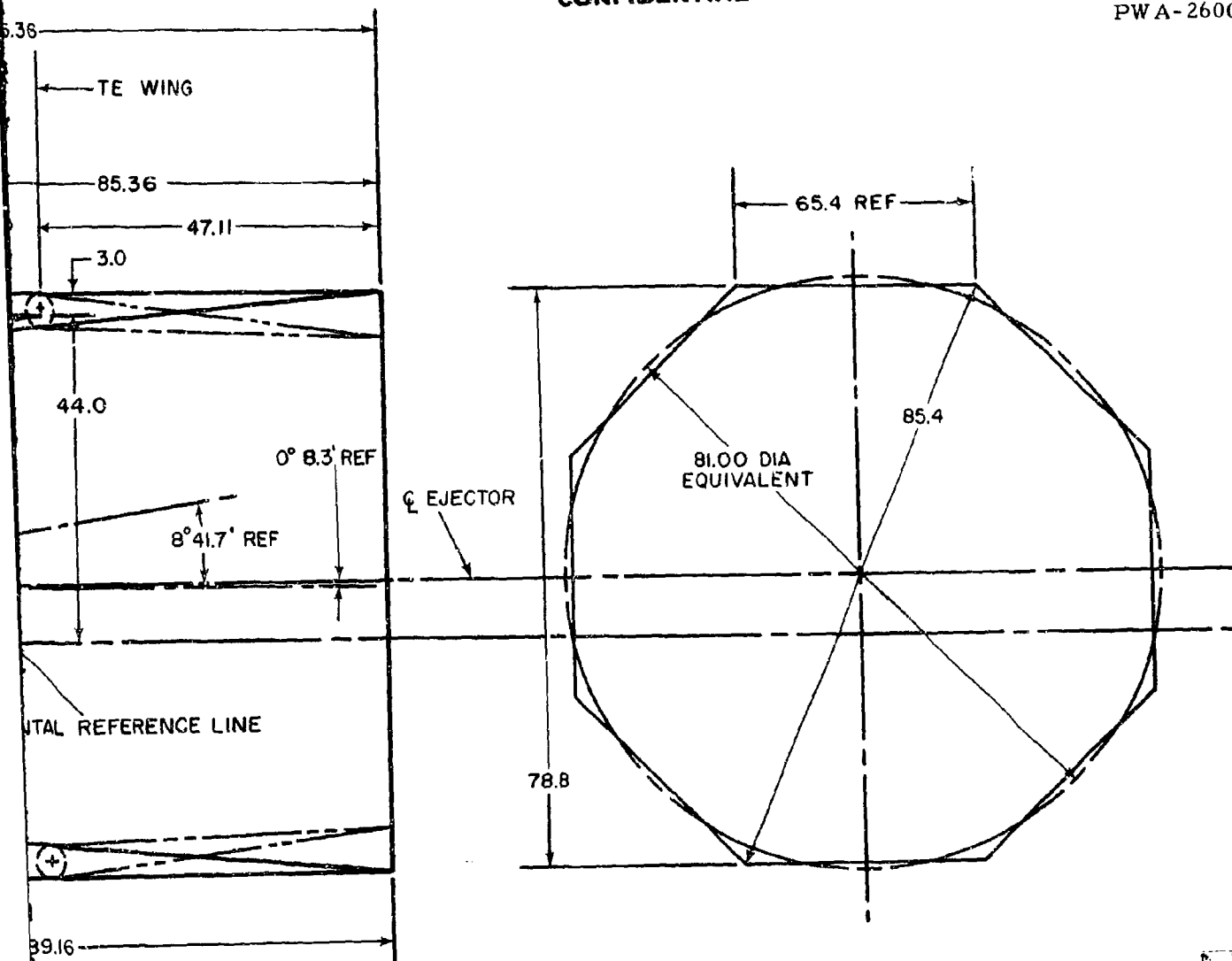
Figure 1-11

CONFIDENTIAL

3

CONFIDENTIAL

PWA-2600



CG LOCATION DOES NOT INCLUDE INLET

EJECTOR CANTED 10.00 FWD. OF PRIMARY
NOZZLE PLANE
CG LOCATION DOES NOT INCLUDE INLET

STF219 650 LBS./SEC. TURBOFAN

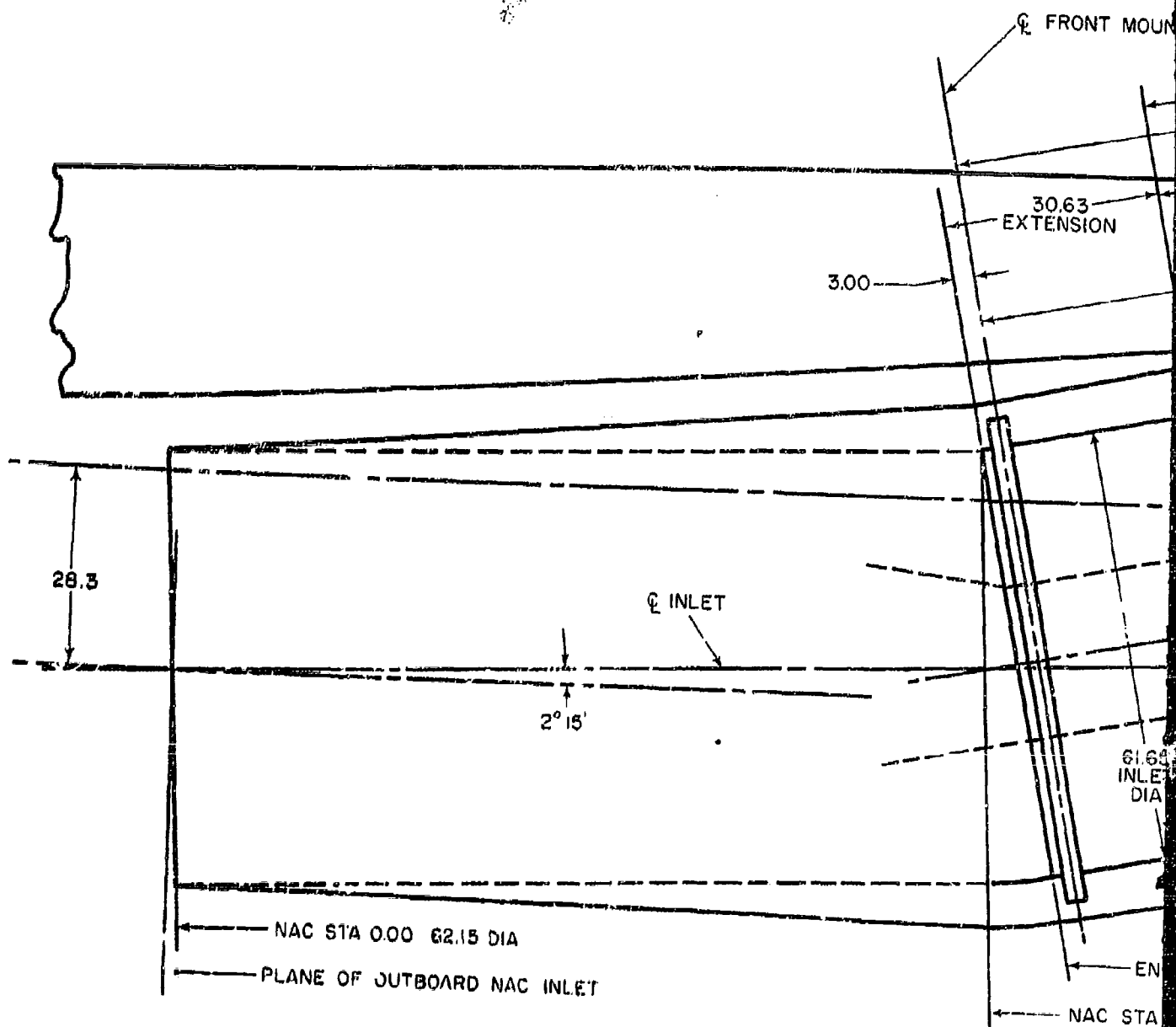
Figure 1-11

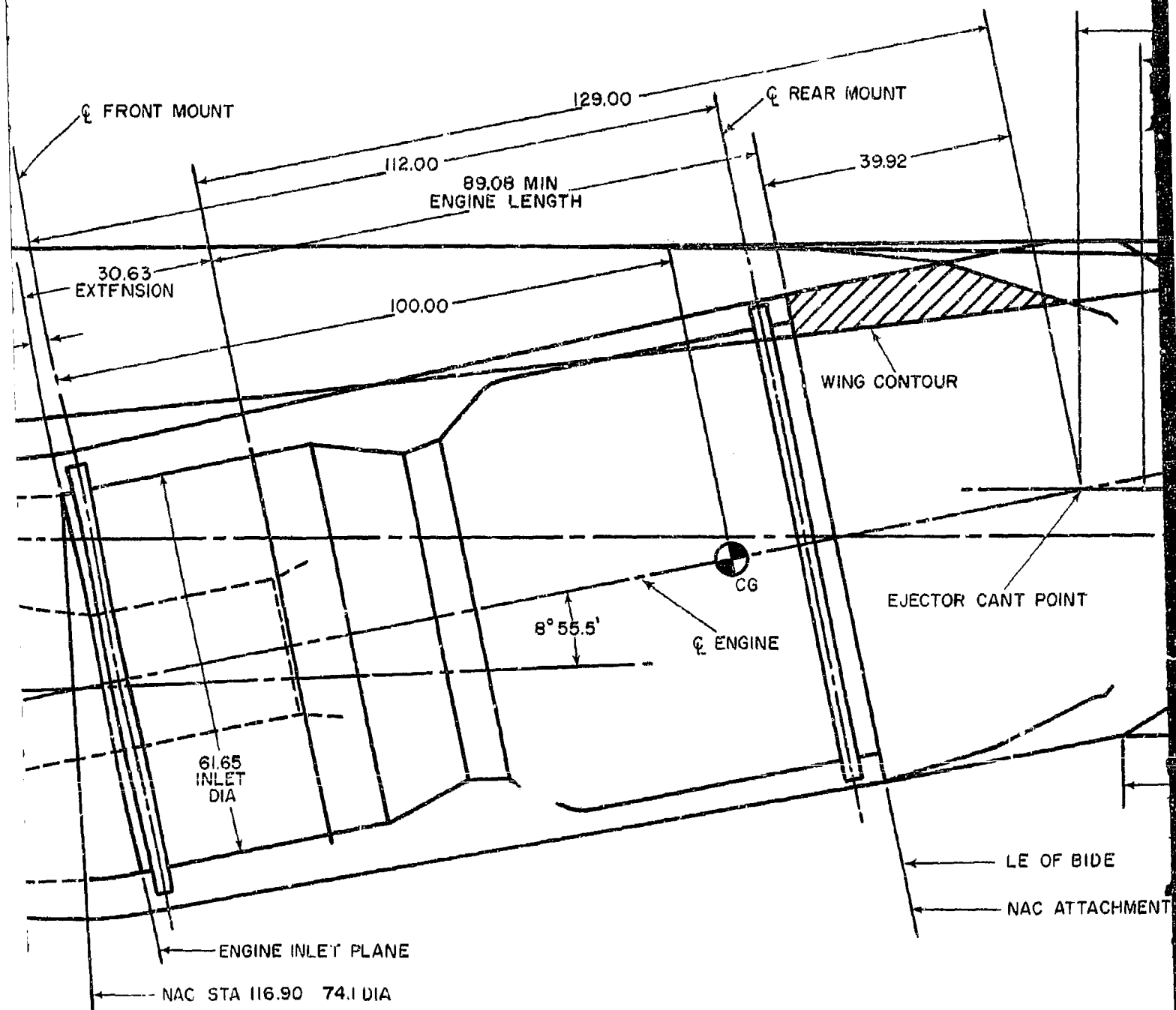
CONFIDENTIAL

DOWNGRADED AT 1 YEAR INTERVALS
DECLASSIFIED AFTER 10 YEARS
DDO OR 50010

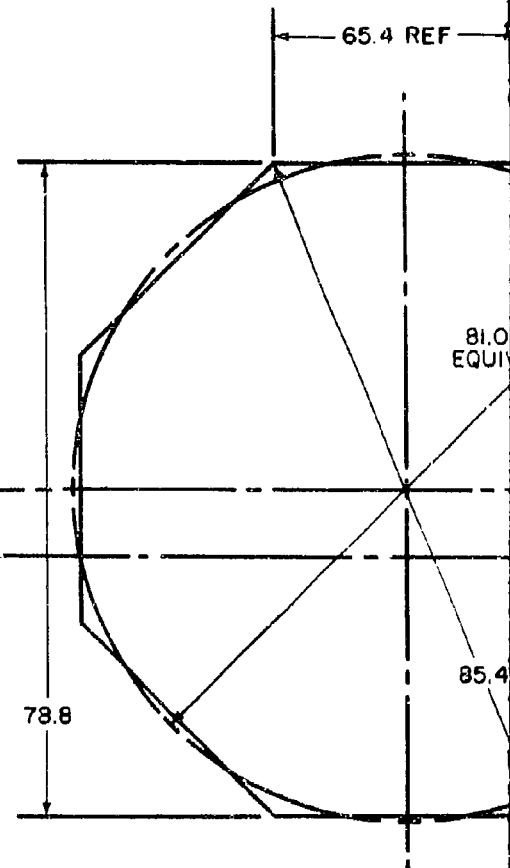
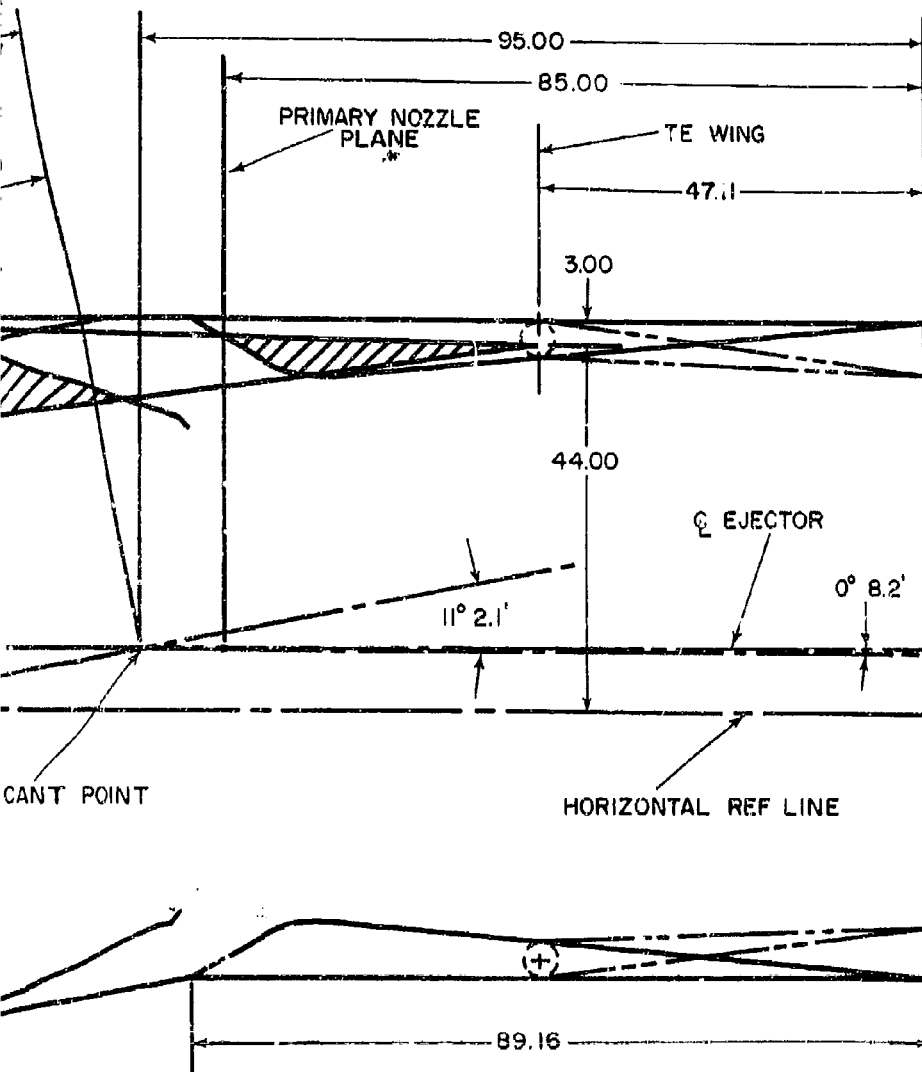
THIS DOCUMENT CONTAINS INFORMATION RELATIVE TO THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.
SECTION 793 AND 794, AND IS SUBJECT TO PENALTIES FOR
REVELATION OF ITS CONTENTS TO AN UNAUTHORIZED PERSON.

4





CONFIDENTIAL



— LE OF BIDE
— NAC ATTACHMENT PLANE

EJECTOR CANTED 10.00 FWD. NOZZLE PLANE
CG LOCATION DOES NOT INCLUDE INLET

CG LOCATION DOES NOT INCLUDE INLET

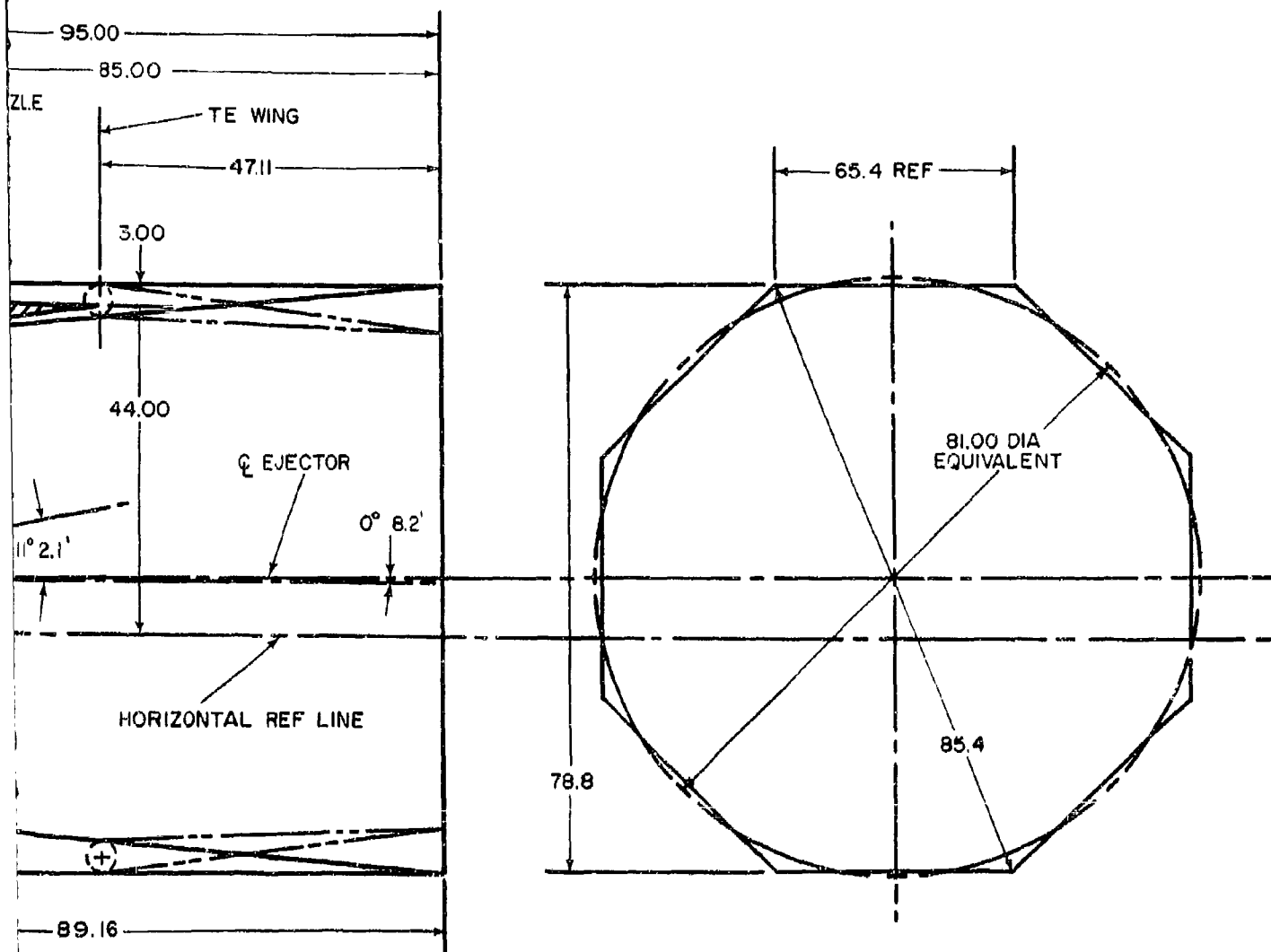
STF219 650 LBS./SEC. TURBOFAN

Figure 1-12

CONFIDENTIAL

CONFIDENTIAL

PWA-2600



EJECTOR CANTED 10.00 FWD. OF PRIMARY
NOZZLE PLANE
CG LOCATION DOES NOT INCLUDE INLET

CG LOCATION DOES NOT INCLUDE INLET

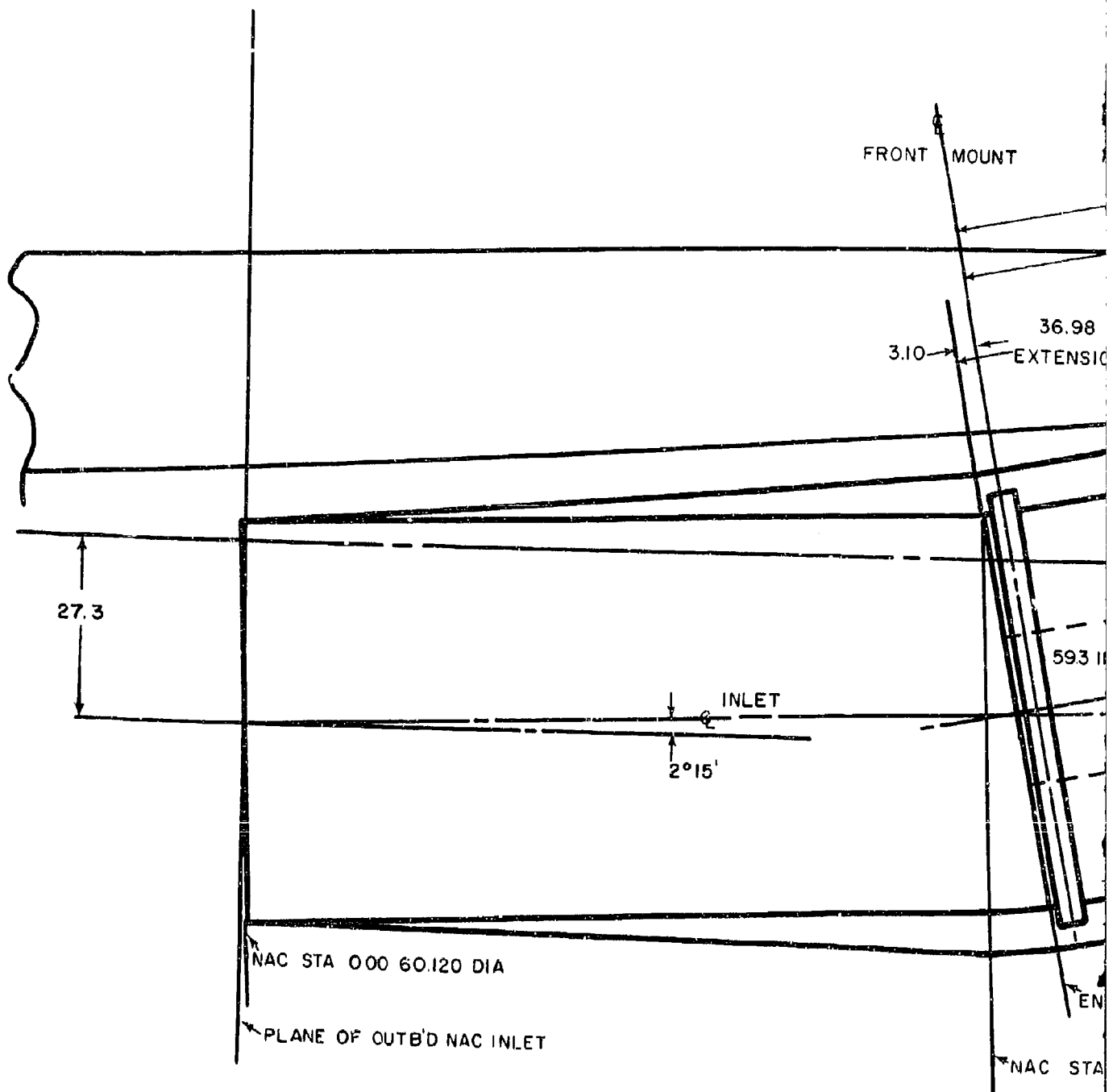
STF219 650 LBS./SEC. TURBOFAN

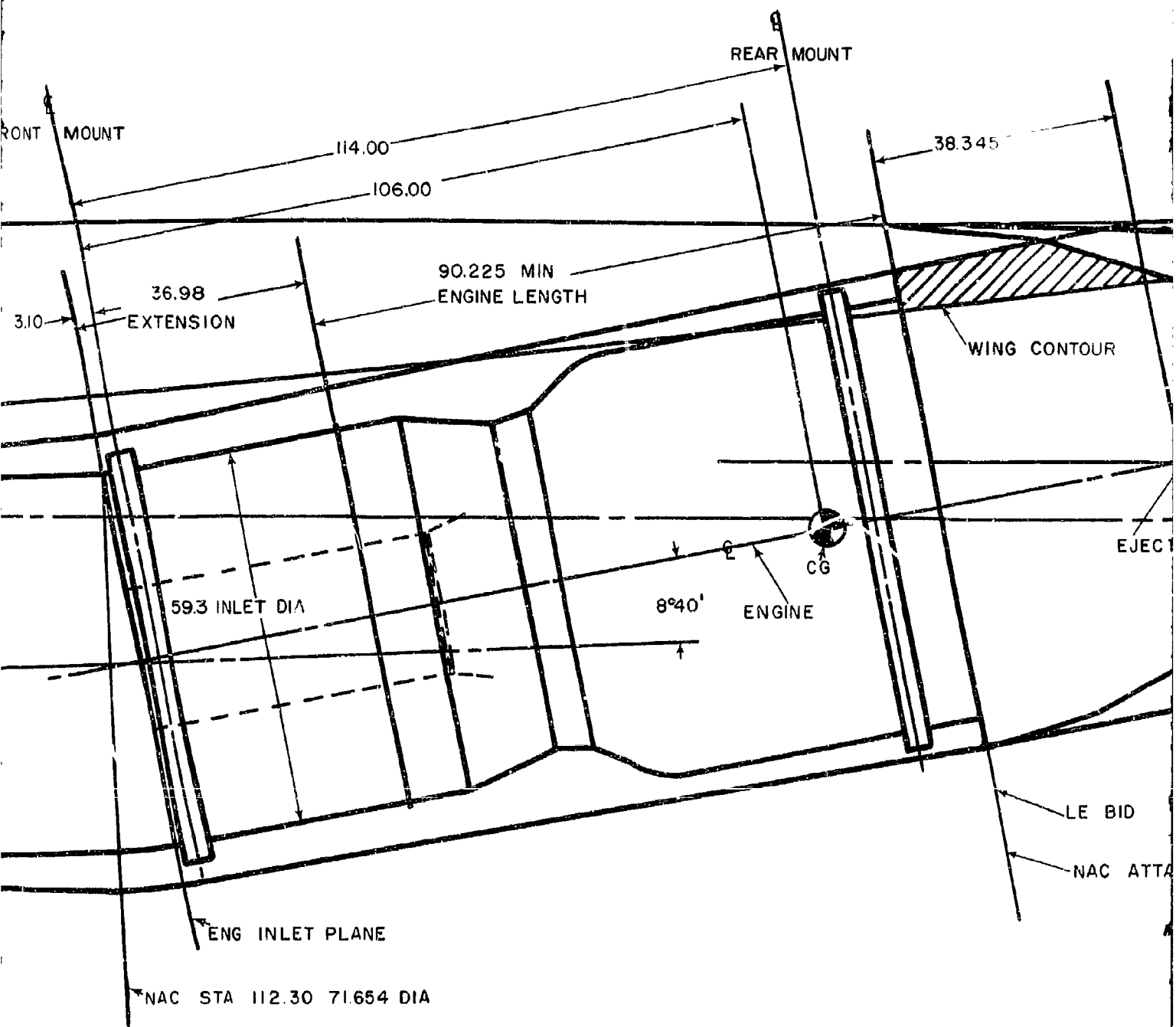
Figure 1-12

CONFIDENTIAL

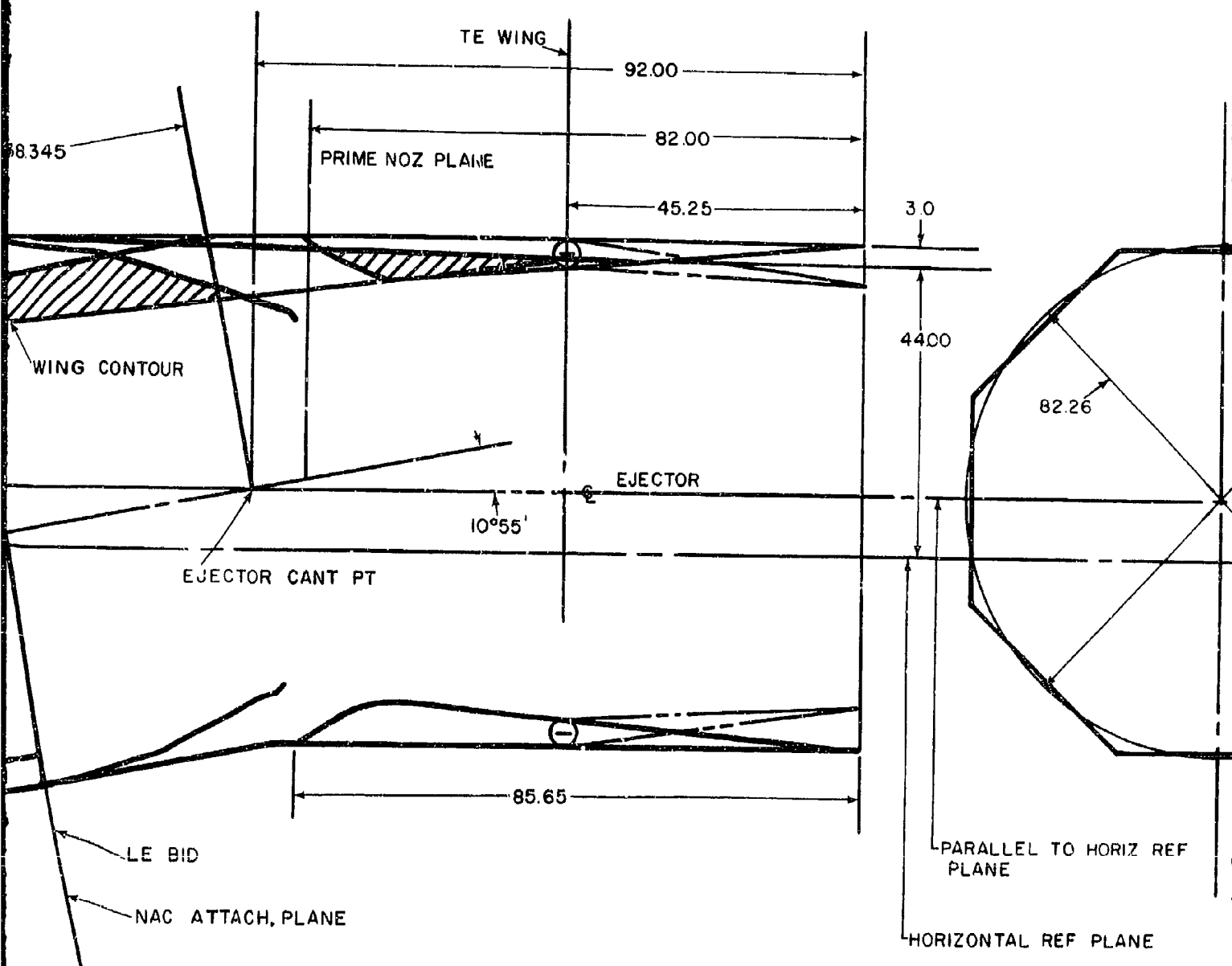
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL
REPRODUCED AT 1 YEAR INTERVAL

4





CONFIDENTIAL



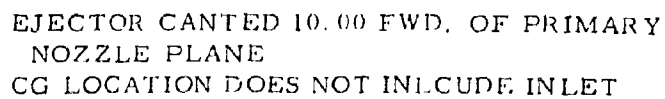
EJECTOR CANTED 10°
NOZZLE PLANE
CG LOCATION DOES NOT

STF219 600 LBS./SEC. TURB

Figure 1-13

CONFIDENTIAL

PW A-2600

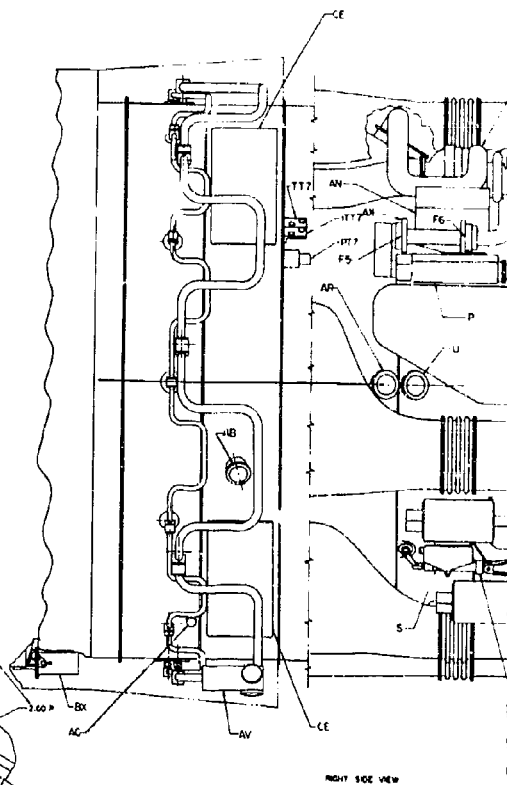
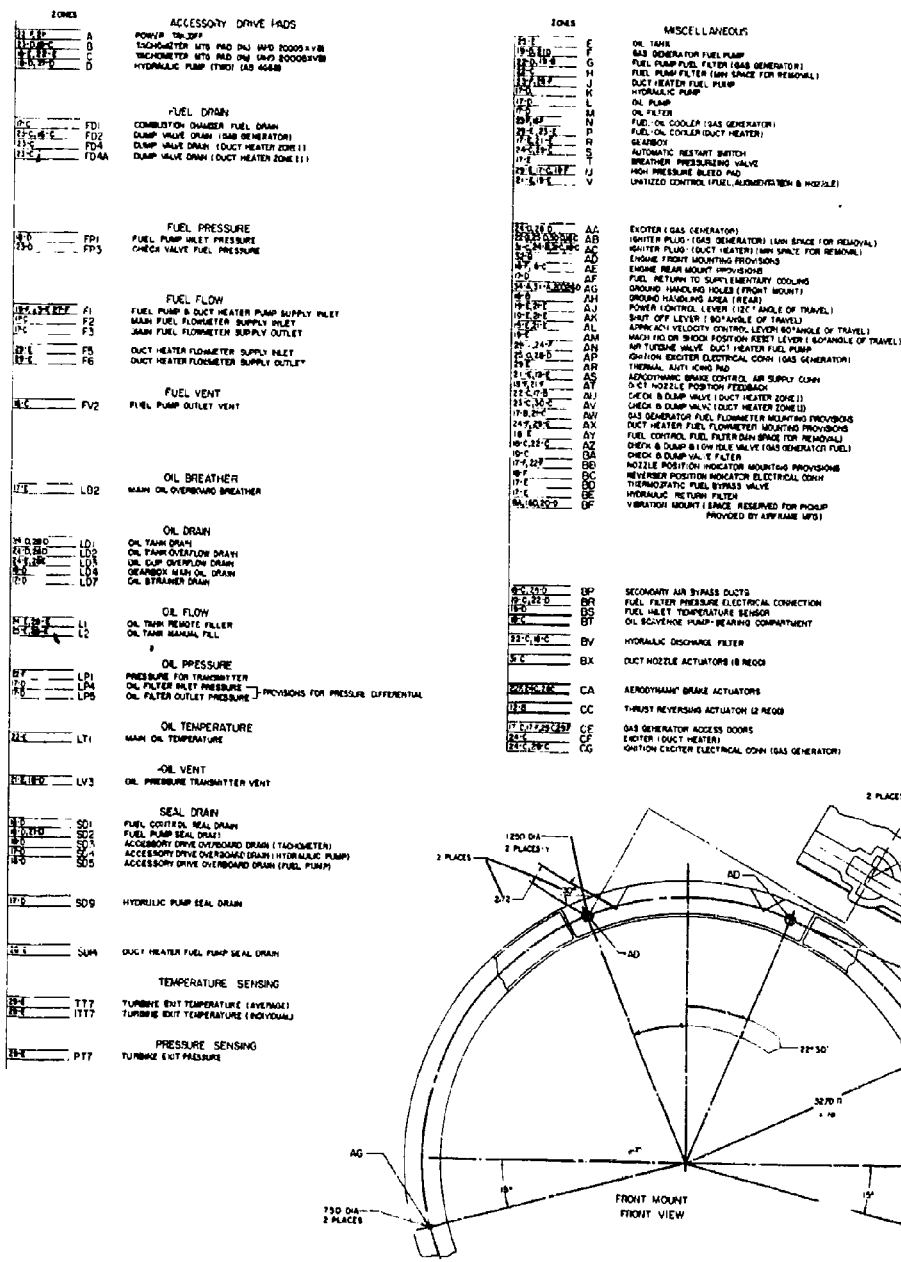


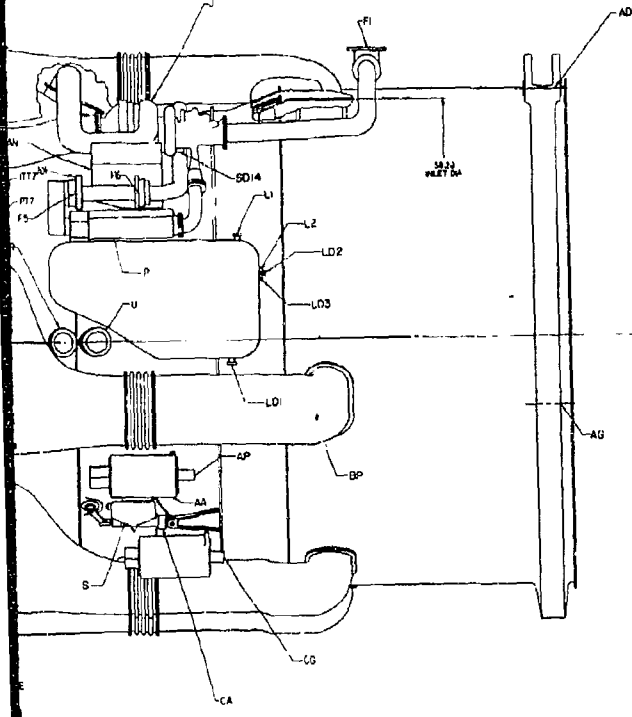
CONFIDENTIAL

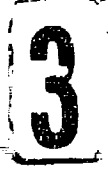
1997年6月10日 星期一 1997年6月10日 星期一

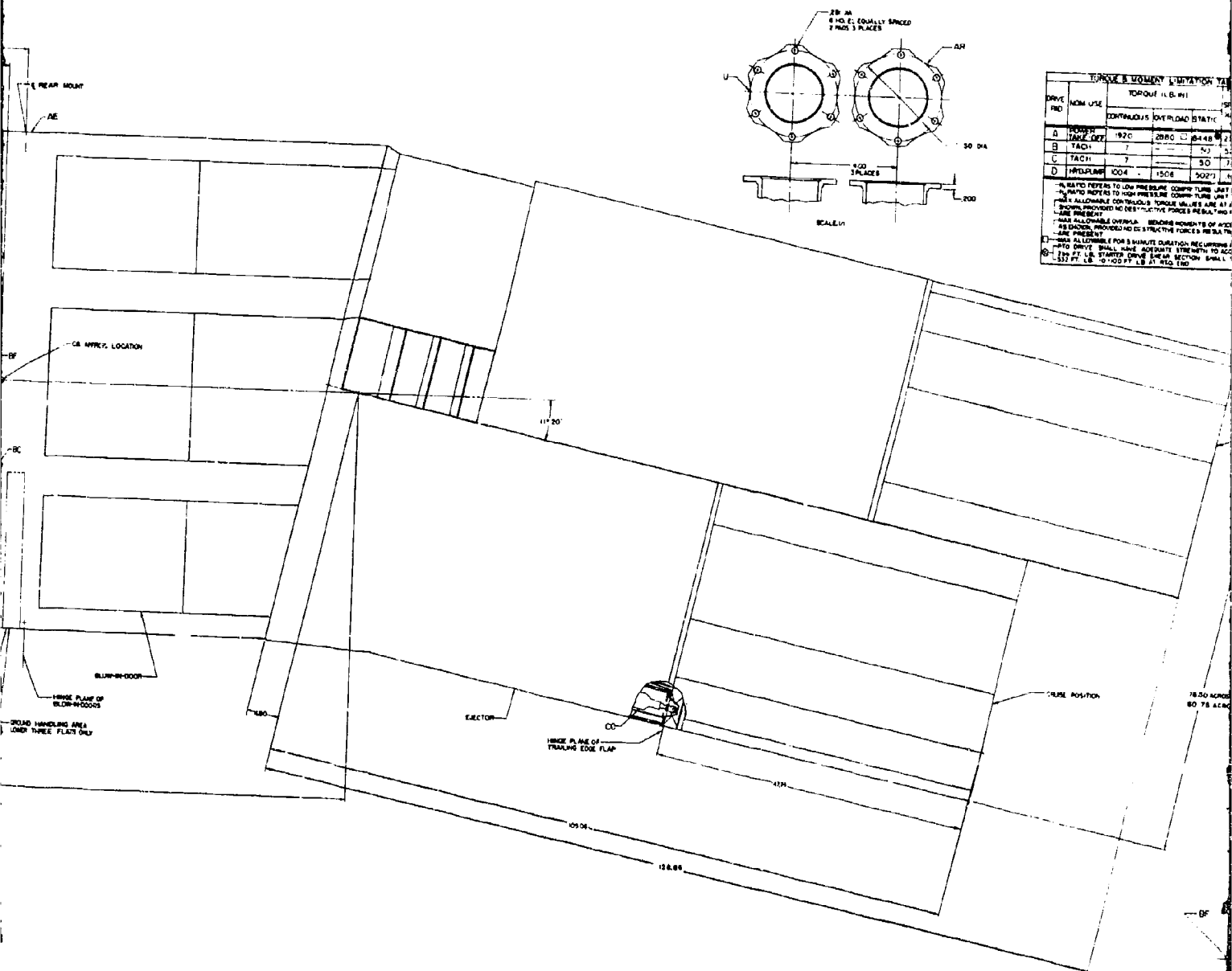
한국의 경우, 1990년대 초부터 시작된 민주화 운동은 사회 전반에 걸쳐서 민주주의의 가치를 확산시켰다. 이 시기에 민주주의는 단순히 정치 체제나 통치 형태를 넘어서, 시민들의 일상생활과 문화생활에까지 영향을 미쳤다. 예를 들어, 1987년 6월의 민주화 운동은 한국 사회에 민주주의의 가치를 심어주었고, 이후 1990년대에는 민주주의가 대중적인 문화 현상이 되었다. 이 시기에 민주주의는 대중적인 문화 현상이 되었다. 이 시기에 민주주의는 대중적인 문화 현상이 되었다.

PRATT & WHITNEY AIRCRAFT

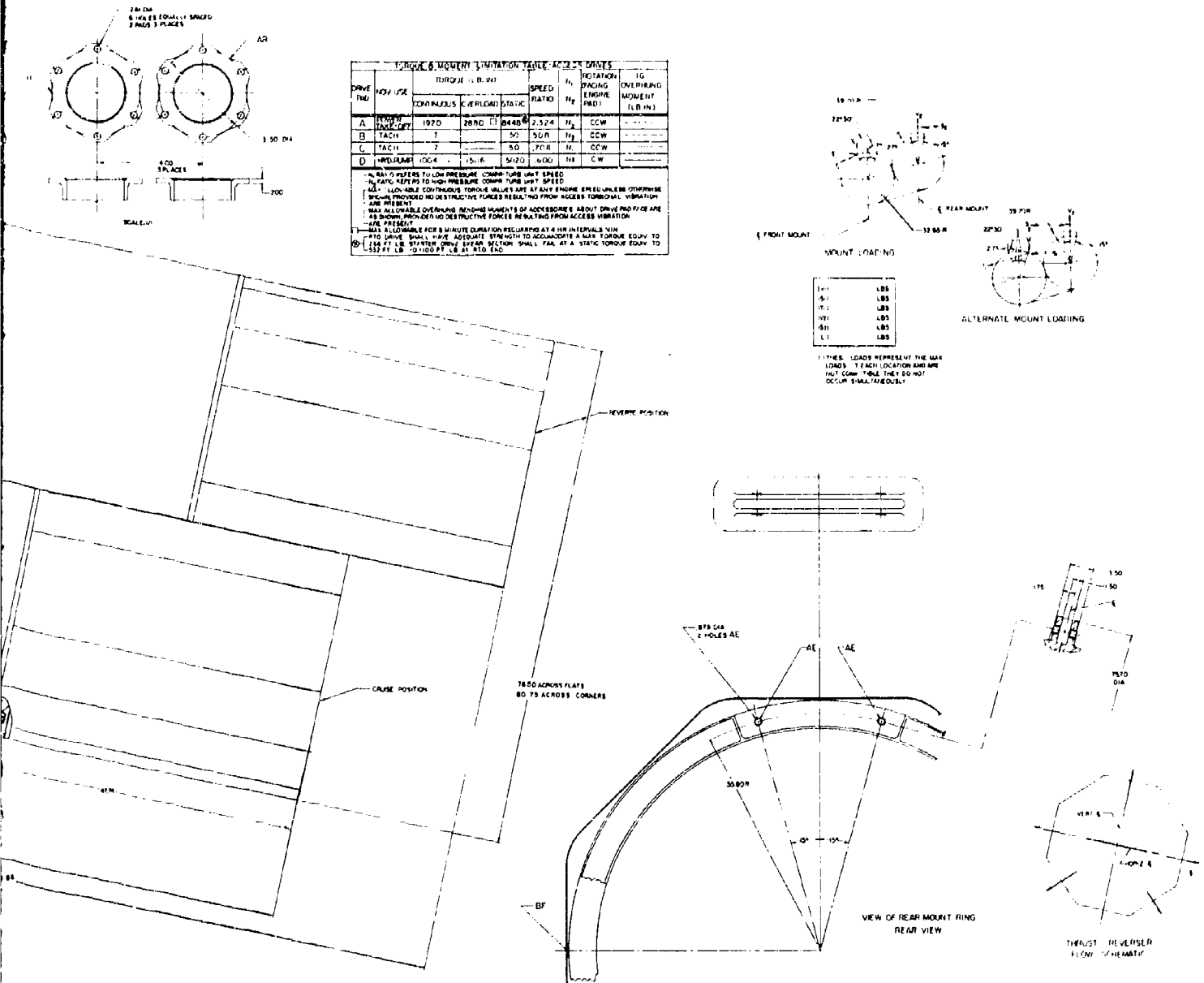






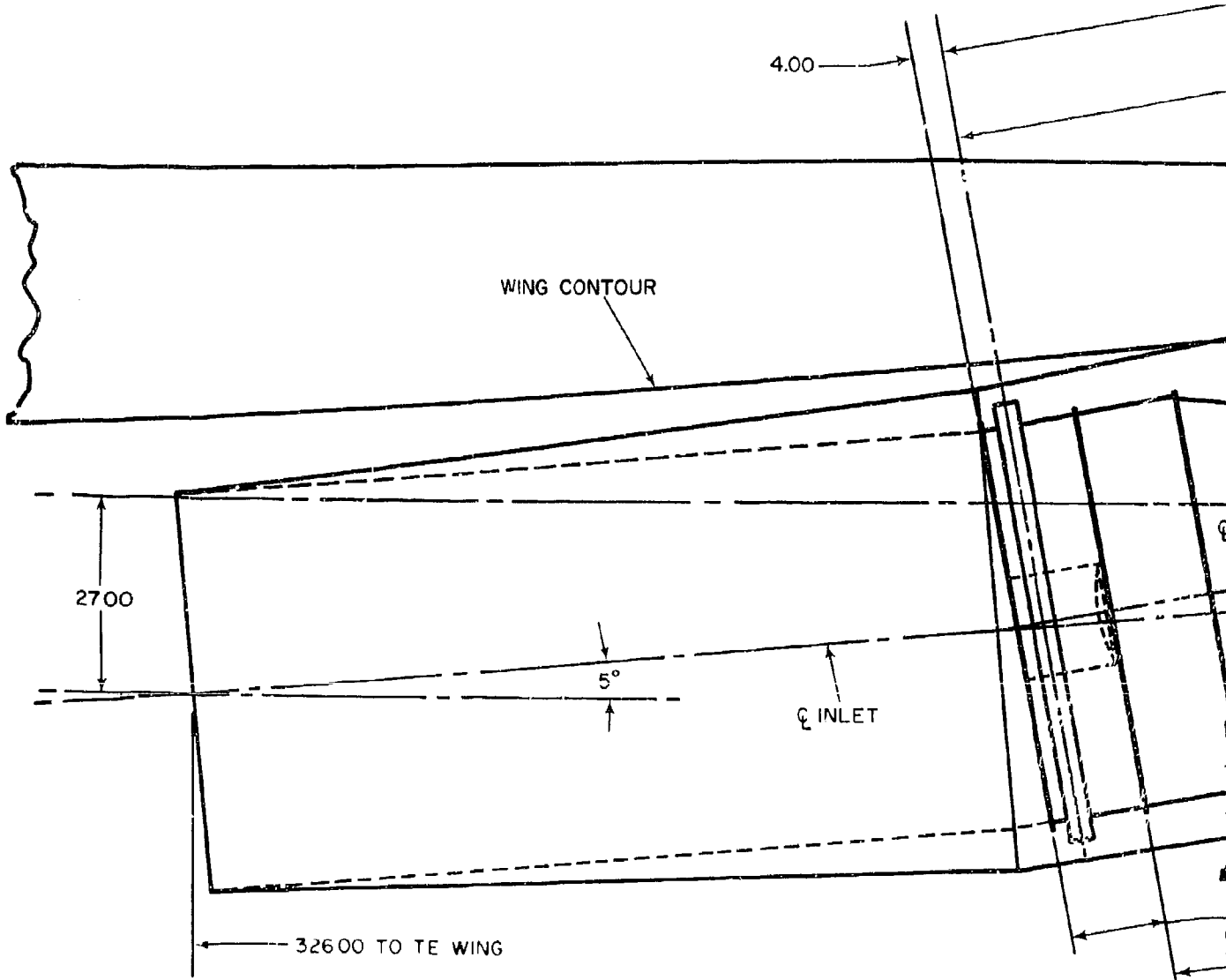


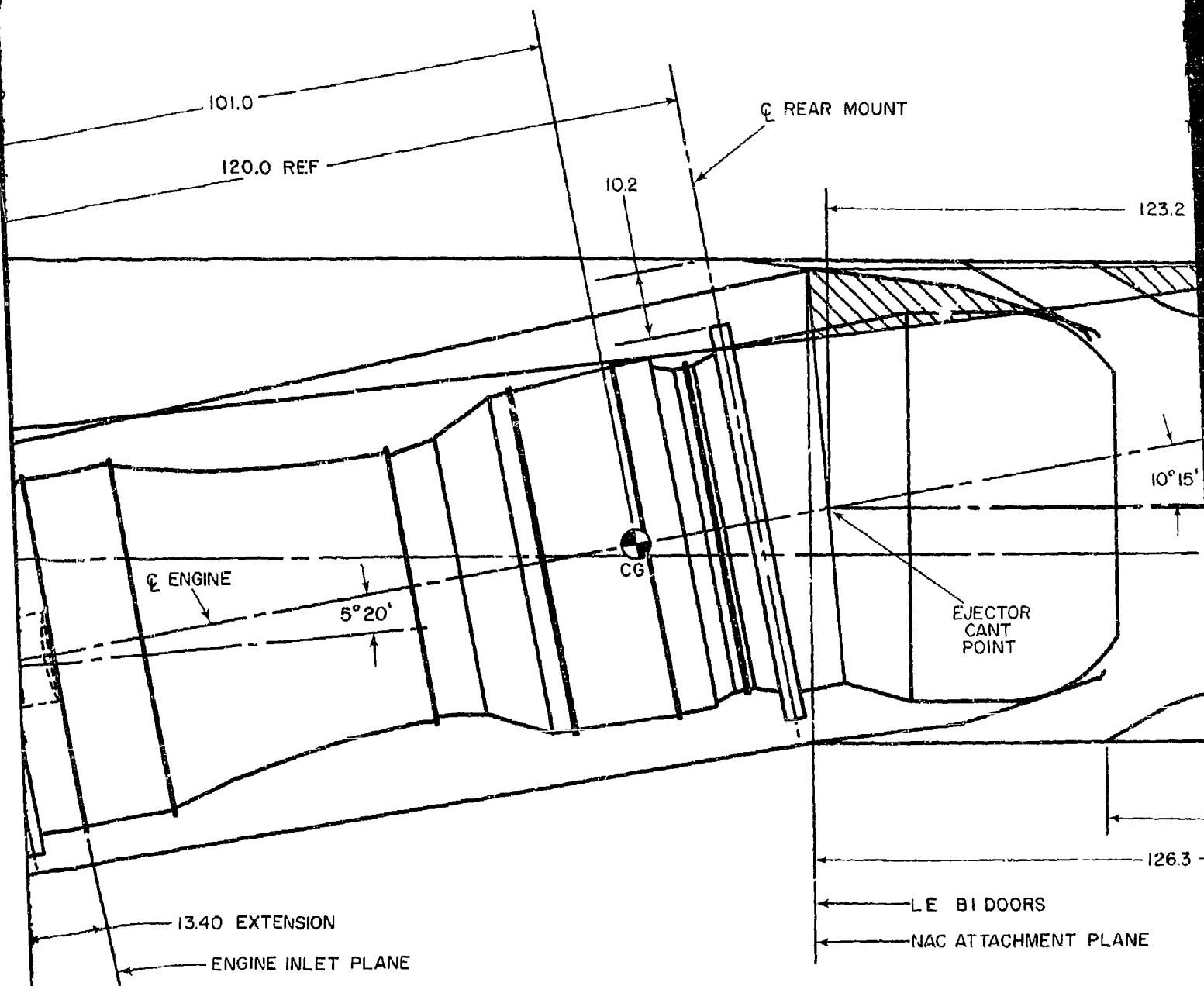
PROPOSED TURBINE



PROPOSED TURBOFAN ACCESSORY ARRANGEMENT

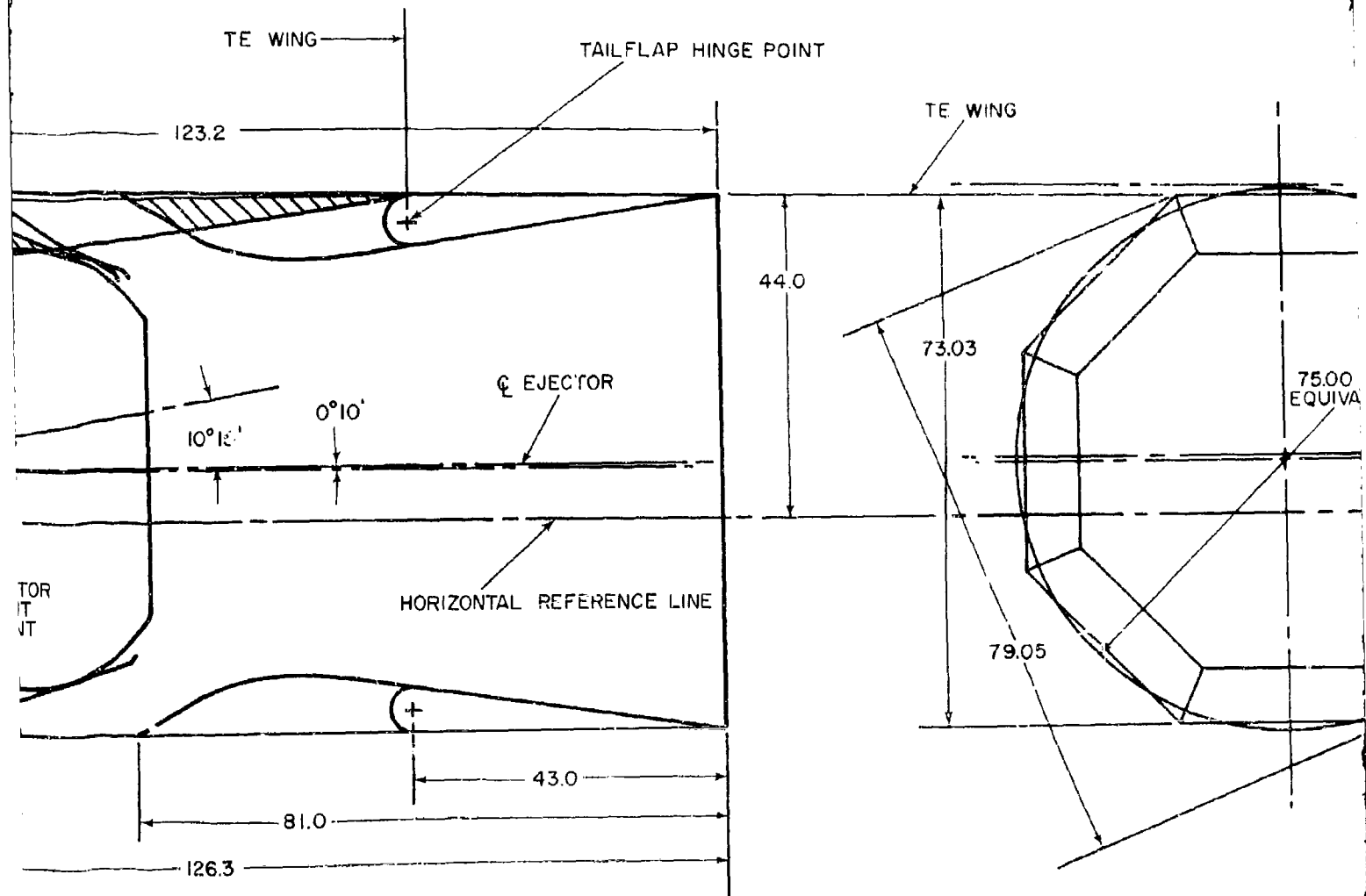
Figure 1-14





2300°F MAX. TURBINE I
EJECTOR CANTED AT R
CG LOCATION DOES NOT

CONFIDENTIAL



DOORS

CHMENT PLANE

CG LOCATION DOES NOT INCLUDE THE INLET

F MAX. TURBINE INLET TEMP.
 TOE CANTED AT REAR MOUNT PLANE
 LOCATION DOES NOT INCLUDE THE INLET

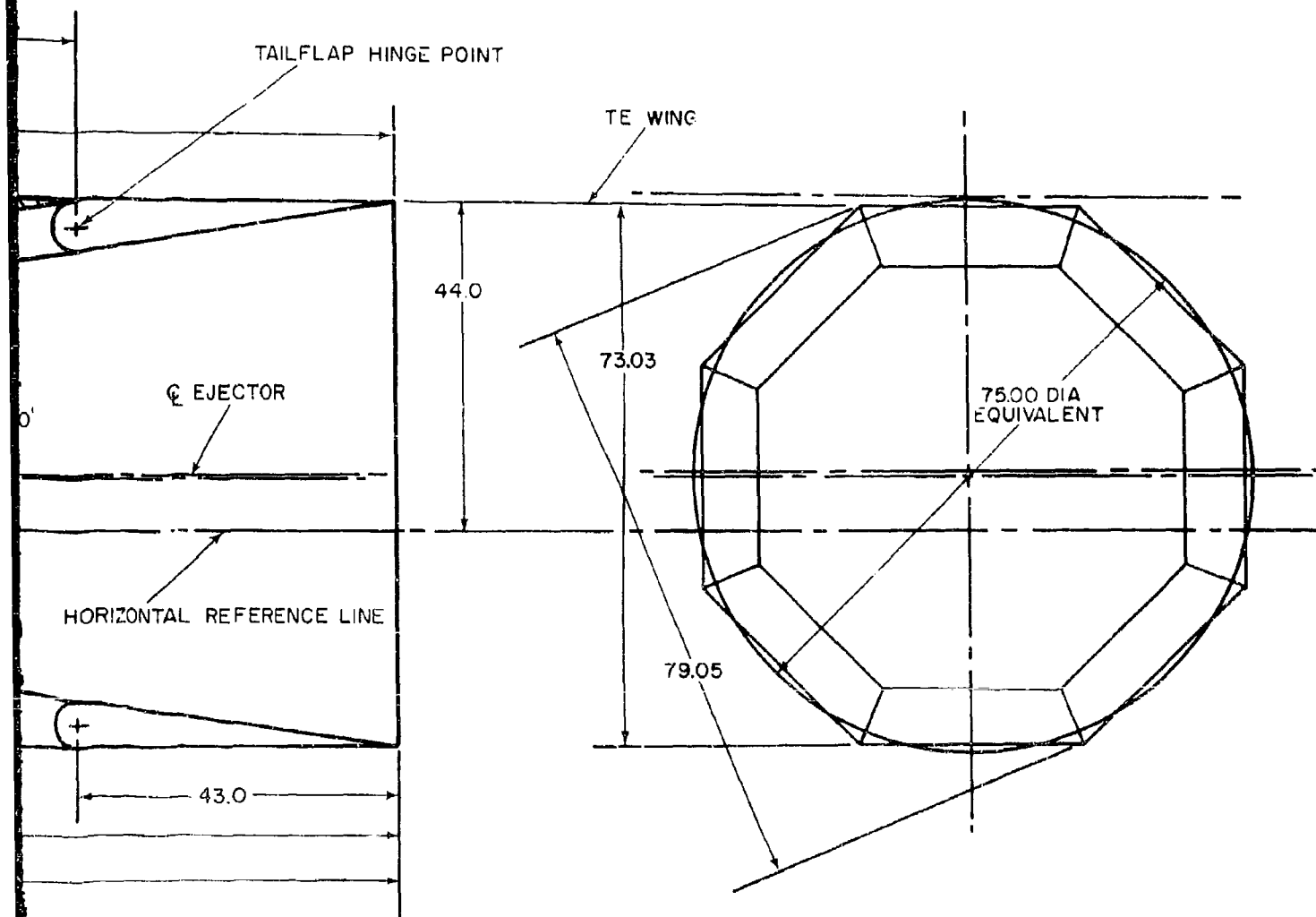
STJ227 500 LBS./SEC. TURBOJET

Figure 1-15

CONFIDENTIAL

CONFIDENTIAL

PWA-2600



CG LOCATION DOES NOT INCLUDE THE INLET

TEMP.
OUNT PLANE
IDE THE INLET

STJ227 500 LBS./SEC. TURBOJET

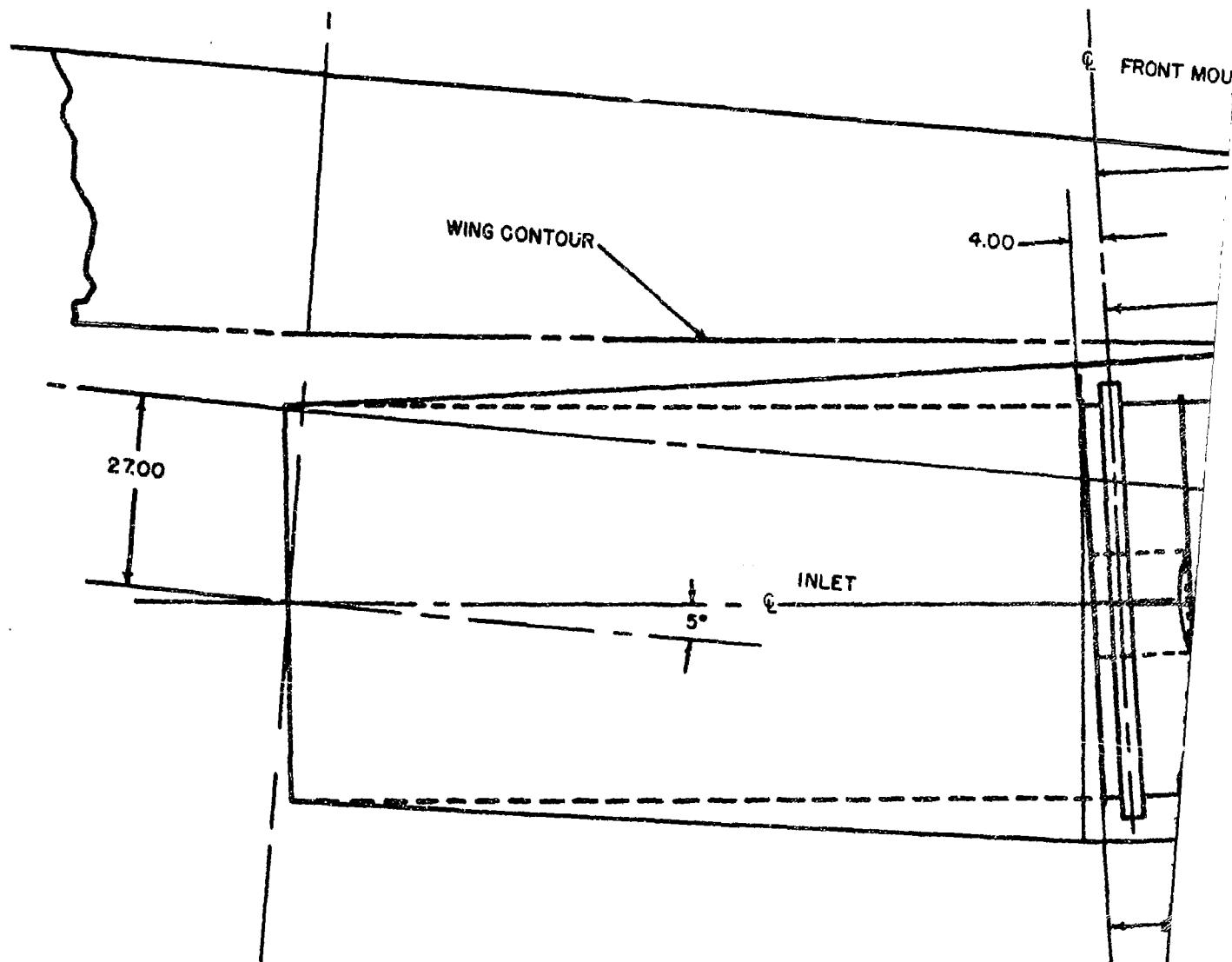
Figure 1-15

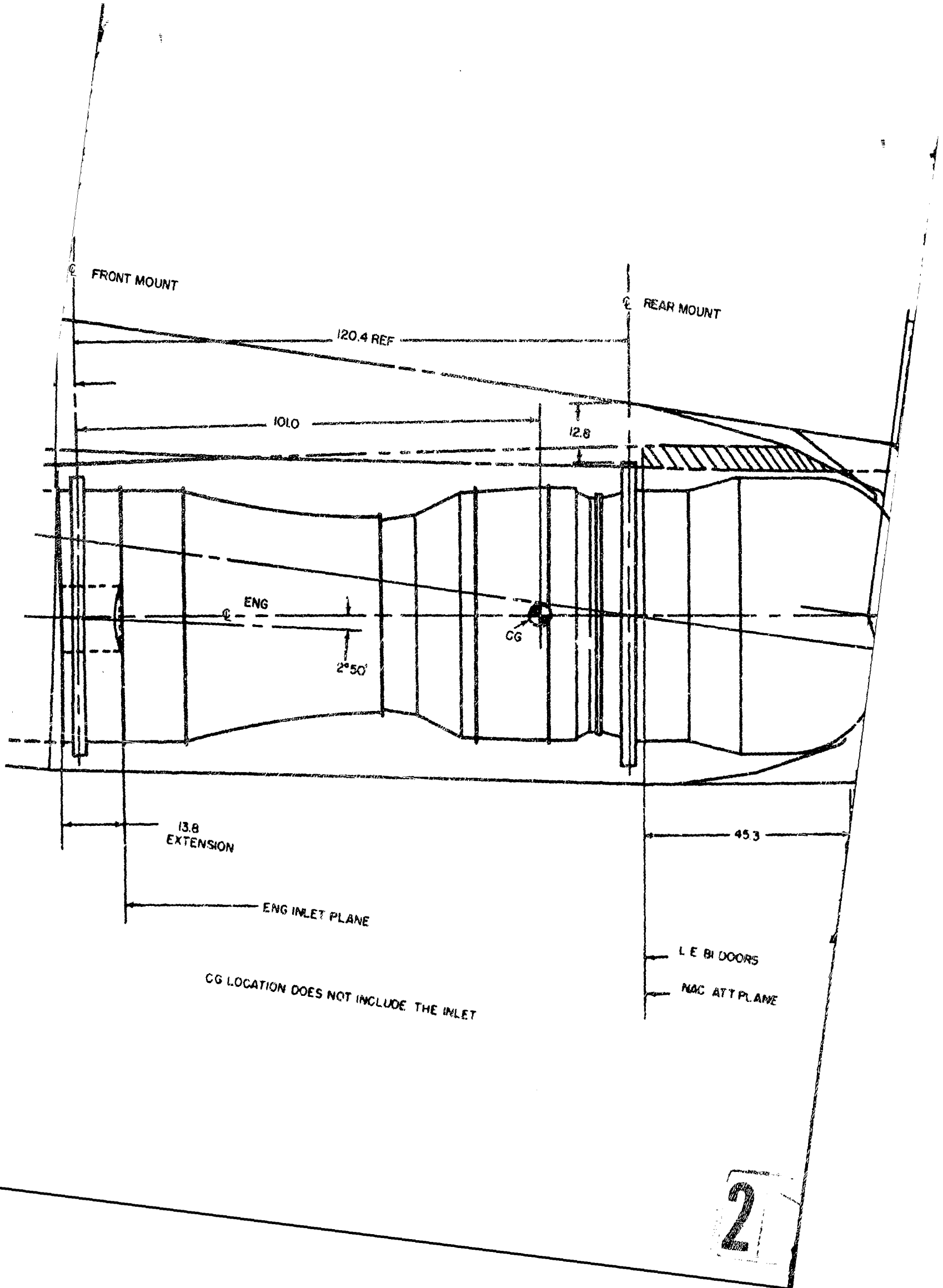
CONFIDENTIAL

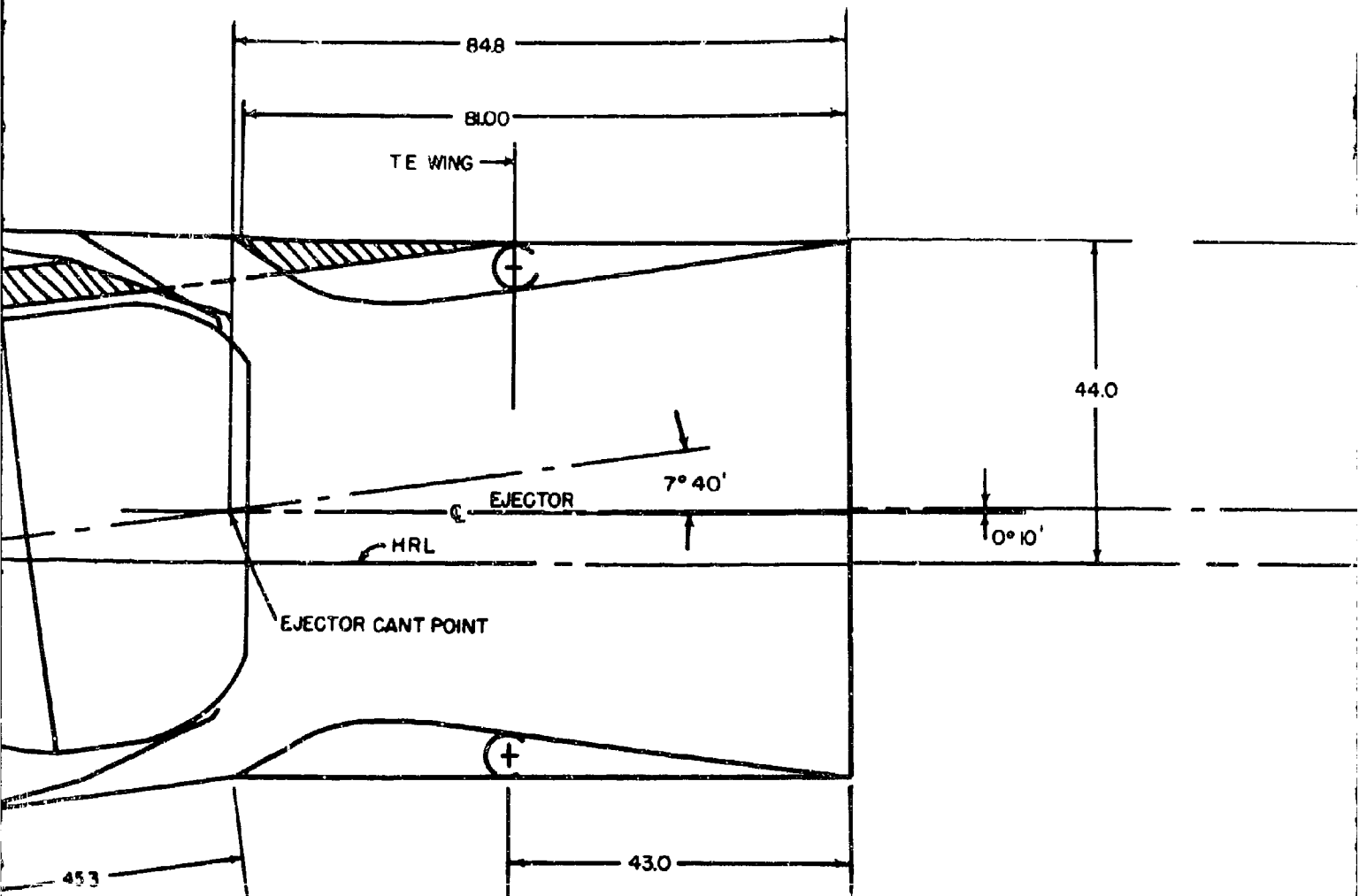
UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DATE 01/10/10
THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
DEFENSE OF THE UNITED STATES OF AMERICA. IT IS
THE POLICY OF THE UNITED STATES OF AMERICA TO
REPRODUCE AND DISTRIBUTE THIS INFORMATION IN THE
PUBLIC DOMAIN. IT IS THE POLICY OF THE UNITED STATES
OF AMERICA TO REPRODUCE AND DISTRIBUTE THIS
INFORMATION IN THE PUBLIC DOMAIN.

4

PRATT & WHITNEY AIRCRAFT





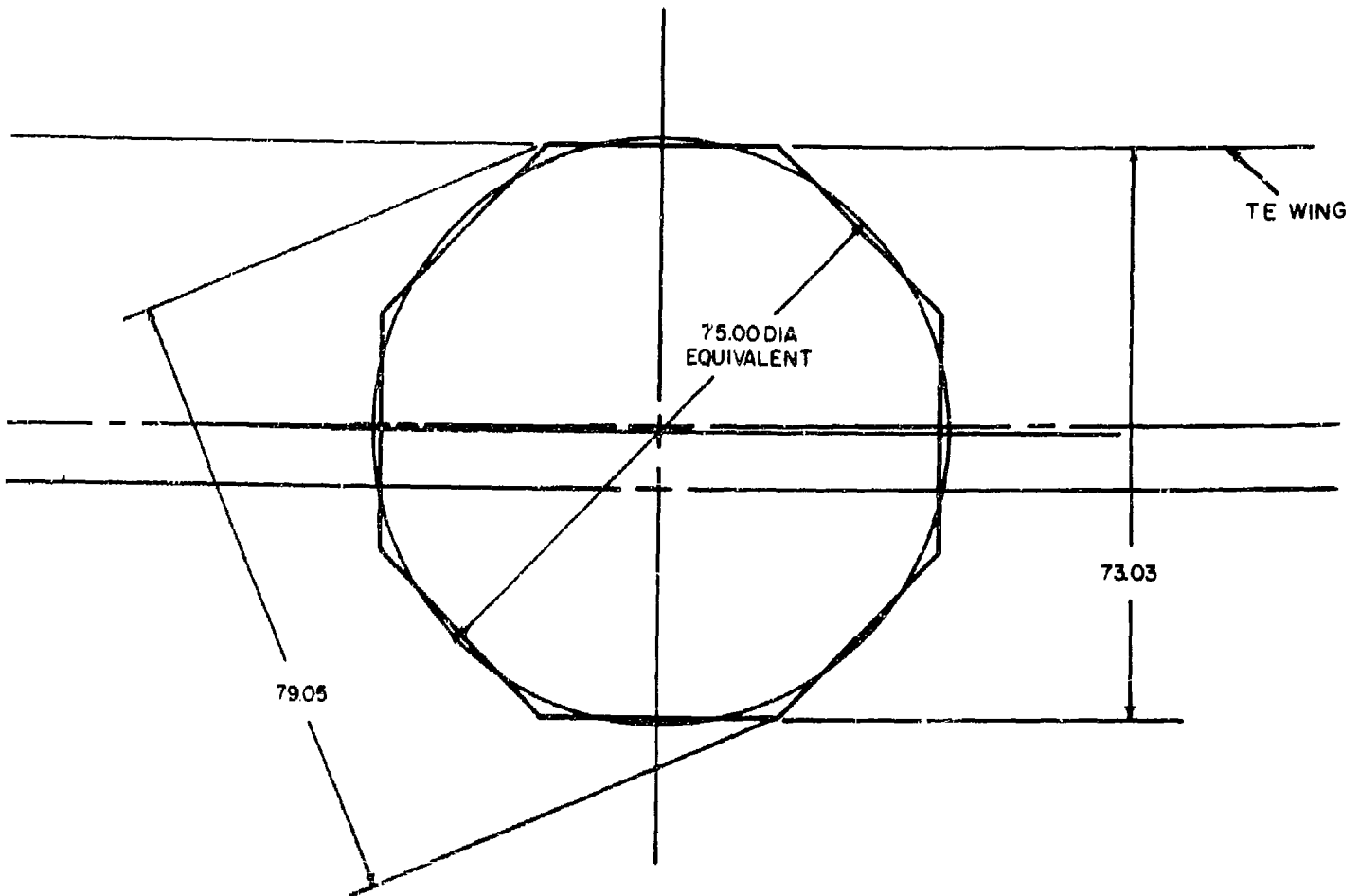


E IN DOORS

NAC ATT PLANE

CONFIDENTIAL

PWA-2600



2300°F MAX. TURBINE INLET
TEMPERATURE
EJECTOR CANTED AT PRIMARY
NOZZLE PLANE

STJ227 500 LBS./SEC. TURBOJET

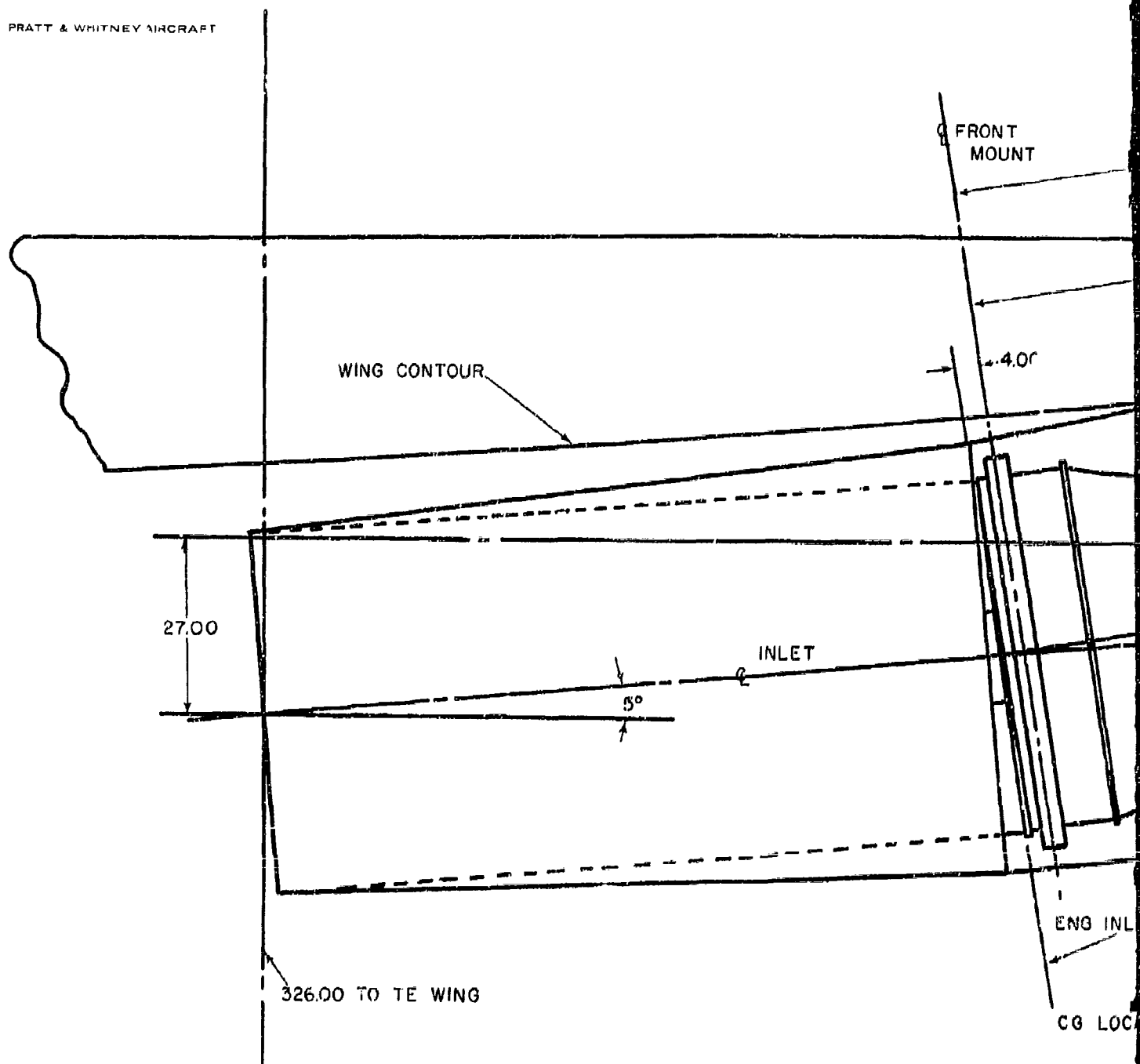
Figure 1-16

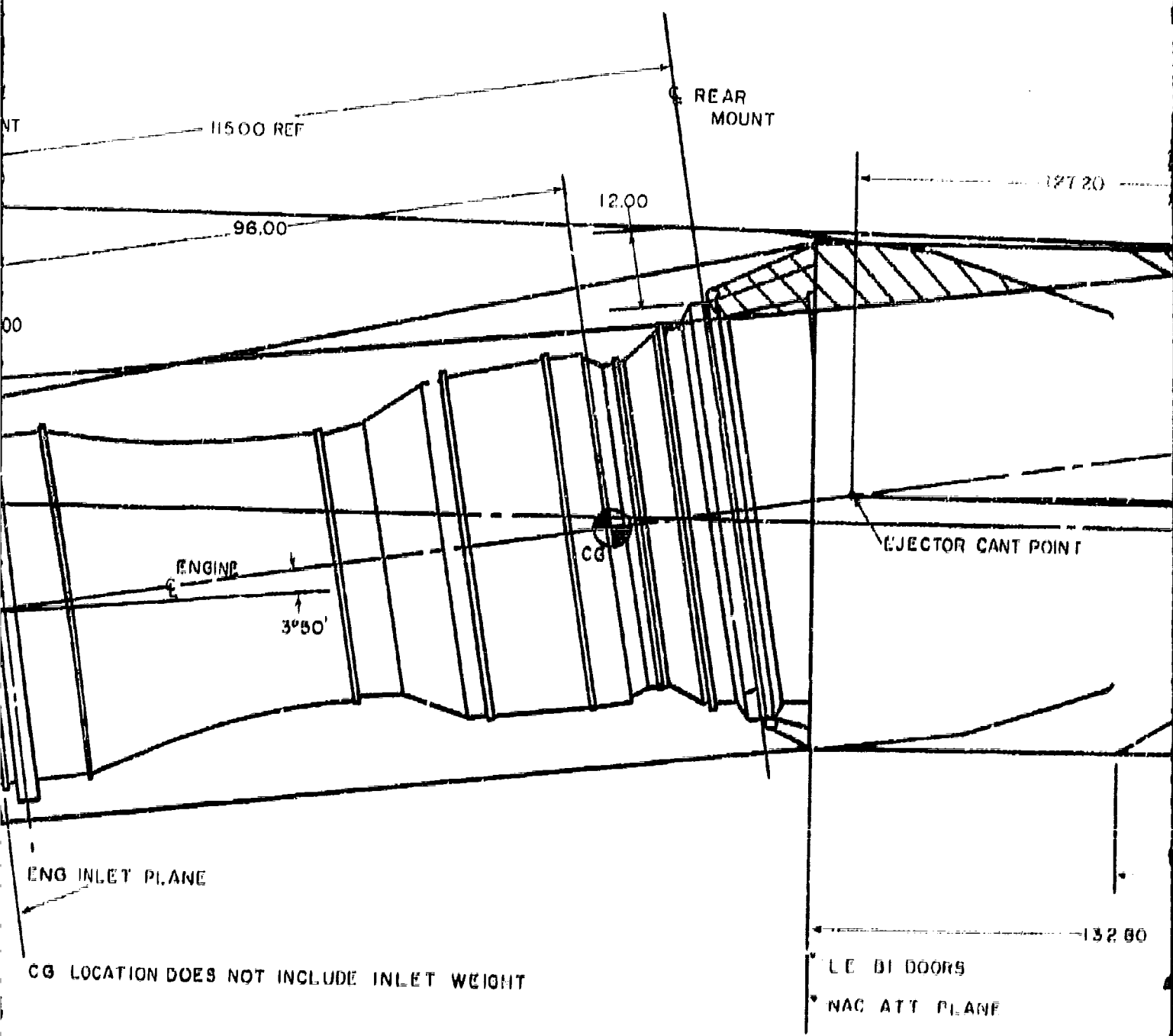
CONFIDENTIAL

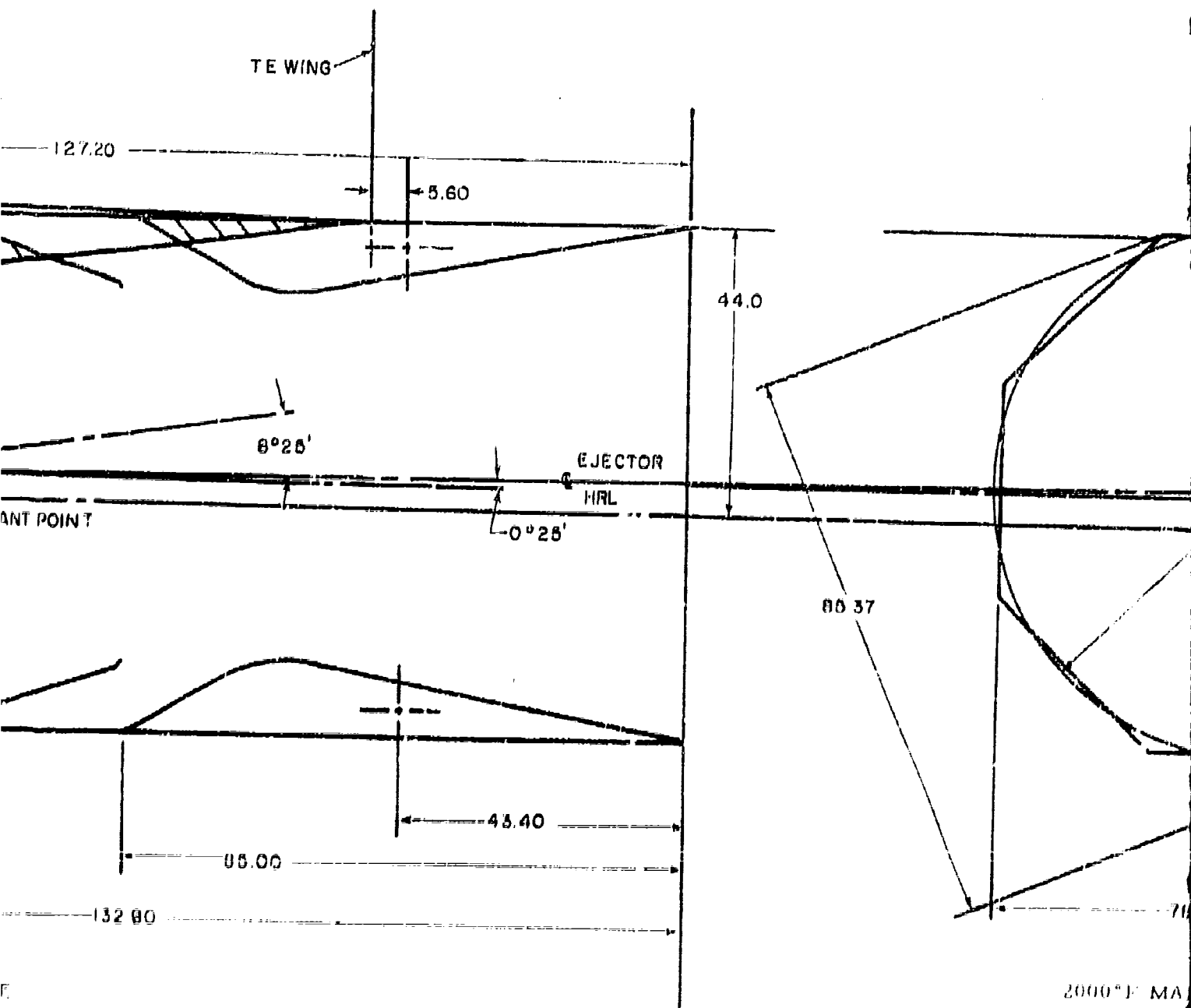
DECLASSIFIED AT 5 YEAR INTERVALS
EXCEPTED AFTER 15 YEARS
DOD OR 500000
THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.
SECTION 793 AND 794, THE TRANSMISSION OR THE
REVELATION OF ITS CONTENTS IN ANY MANNER TO
AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

4

PRATT & WHITNEY AIRCRAFT



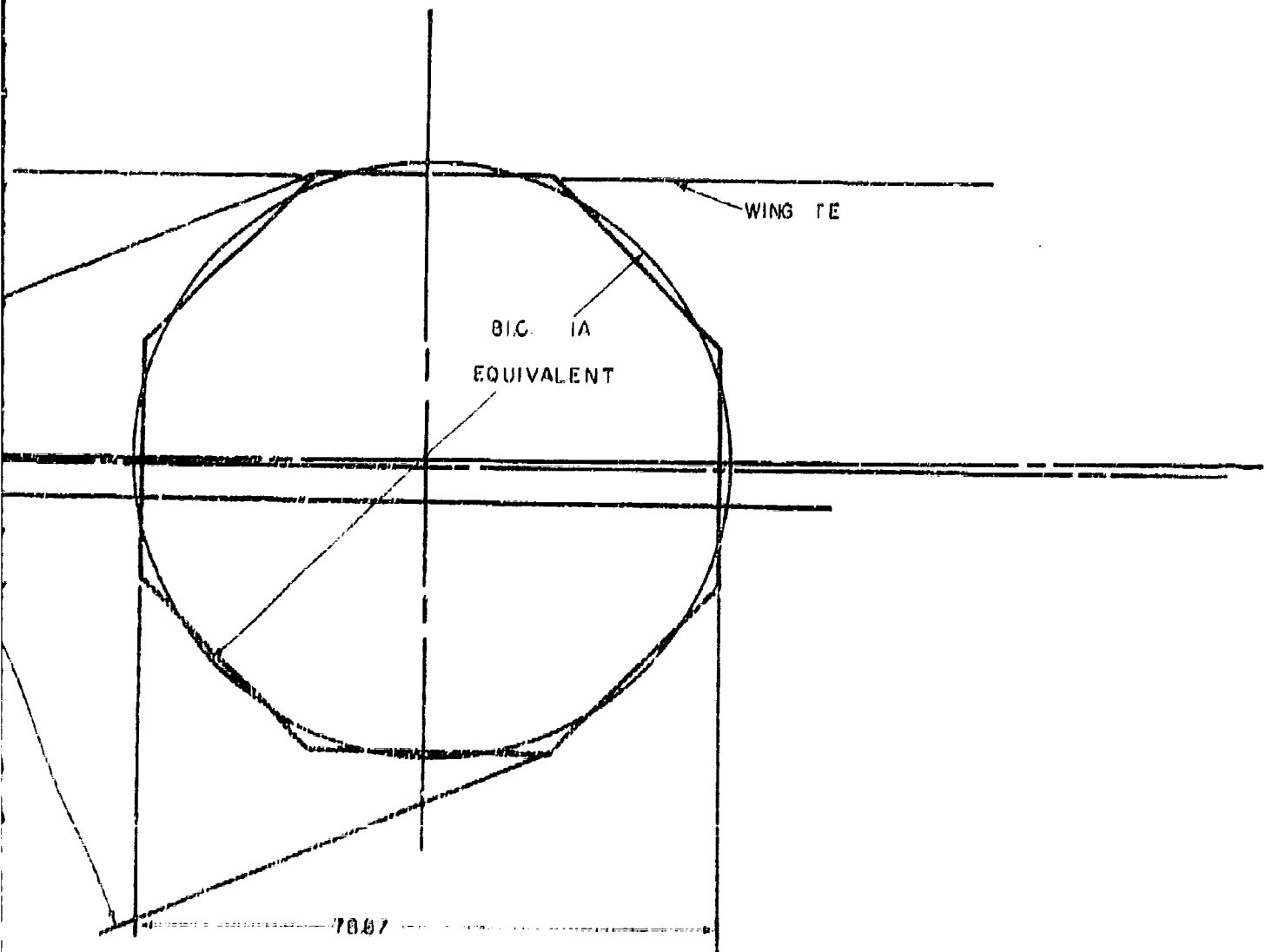




2000°F MAX
TEMPERATURE
EJECTOR
MOUNT 19

CONFIDENTIAL

PWA-2600



2000°F MAX. TURBINE INLET
TEMPERATURE
EJECTOR CANTED AT REAR
MOUNT PLANE

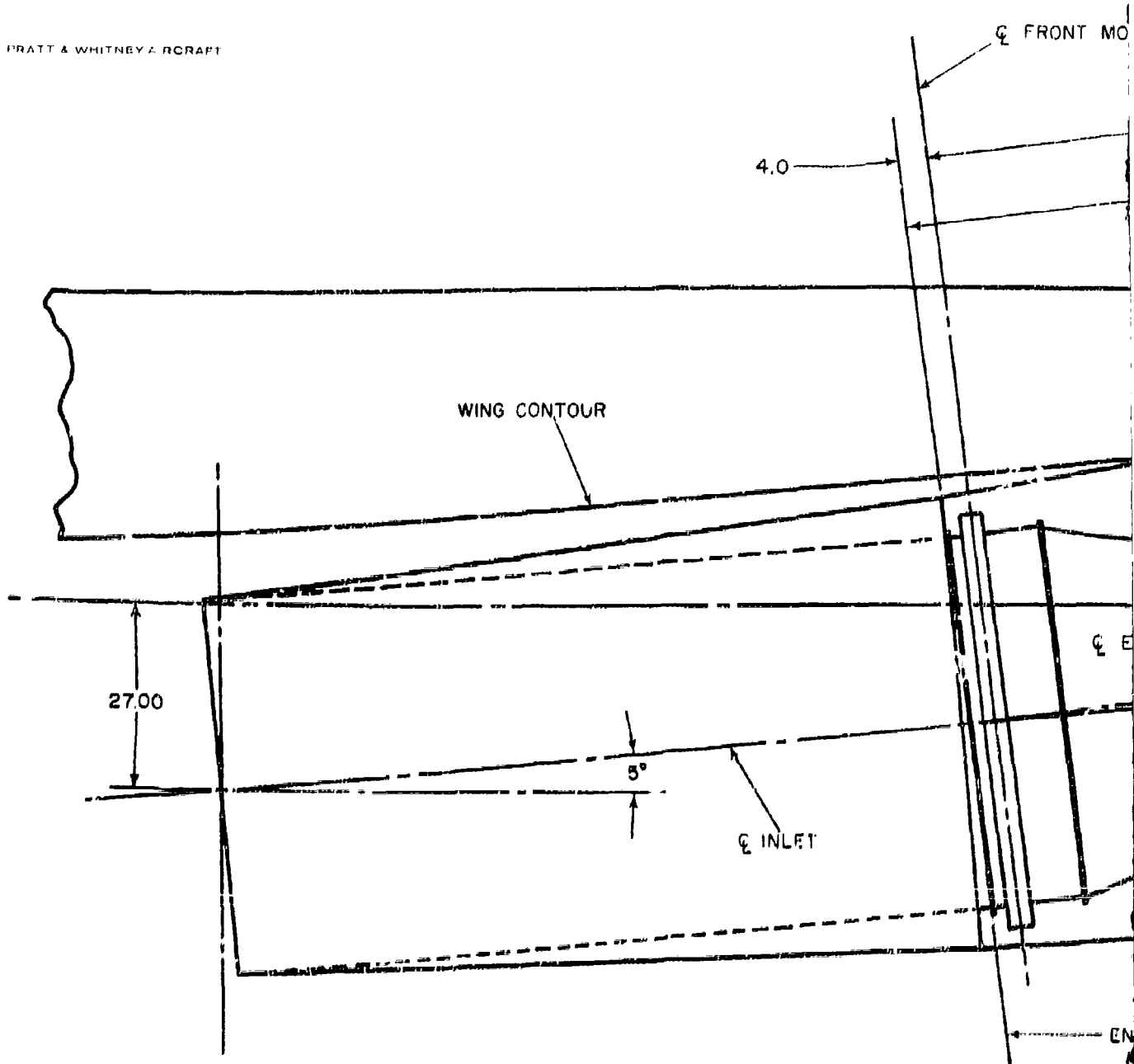
STJ227 500 LBS./SEC. TURBOJET

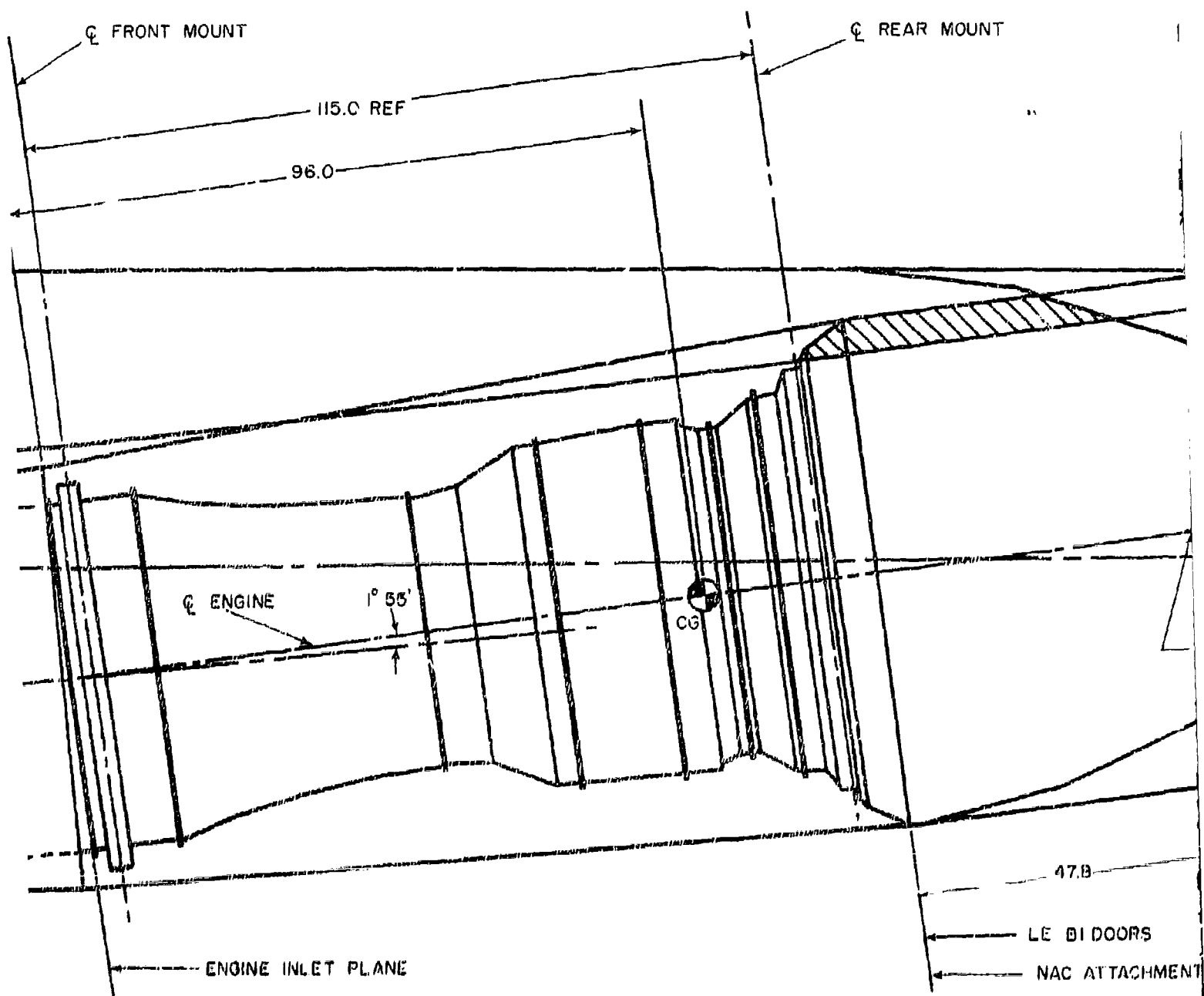
Figure 1-17

CONFIDENTIAL

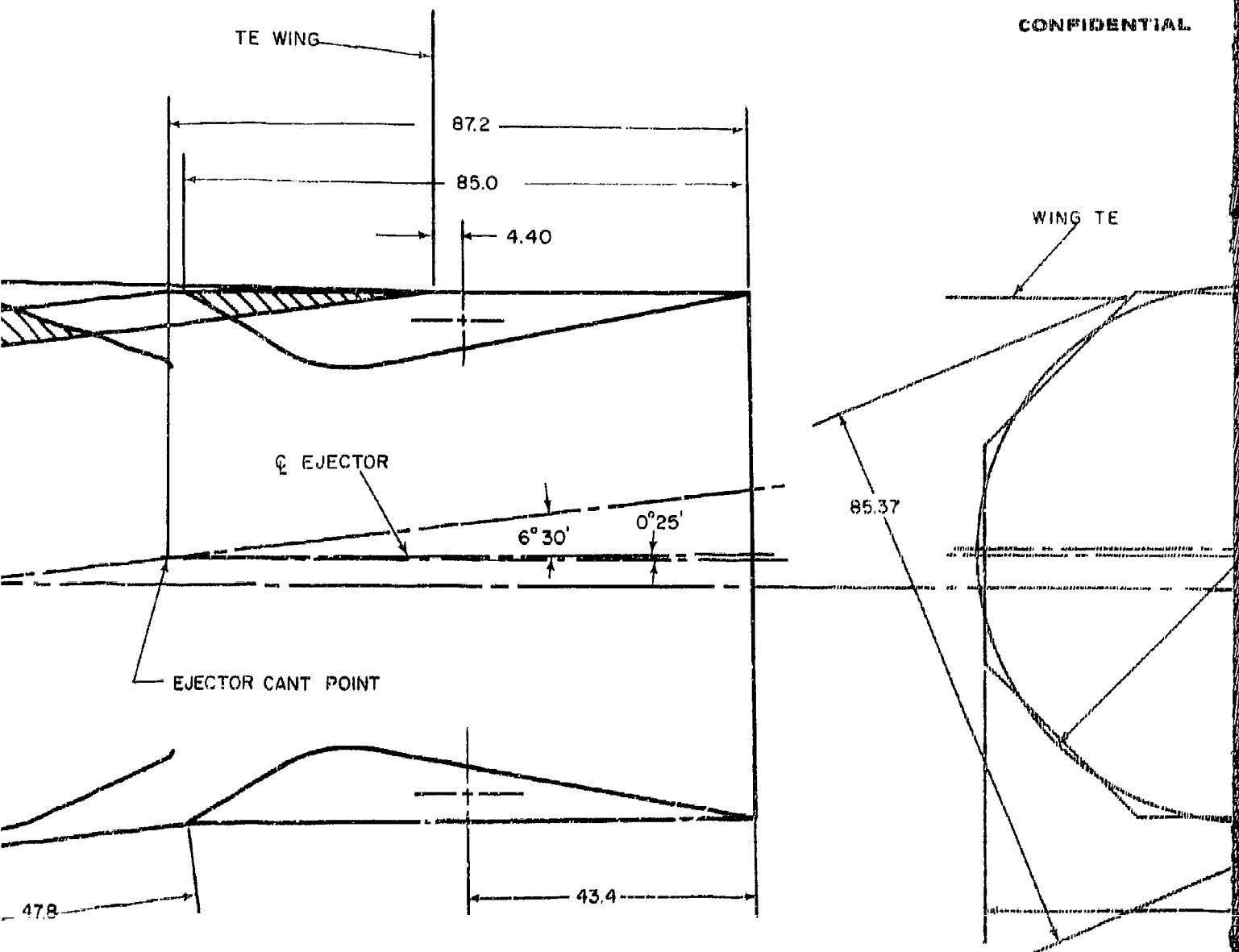
4

PRATT & WHITNEY AIRCRAFT





CONFIDENTIAL



E 81 DOORS

NAC ATTACHMENT PLANE

CG LOCATION DOES NOT INCLUDE INLET WEIGHT

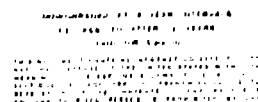
2000°F MAX TU
TEMPERATUR
EJECTOR CANT
NOZZLE PLAN

STJ227 500 LBS./SEC. TU

Figure 1-18

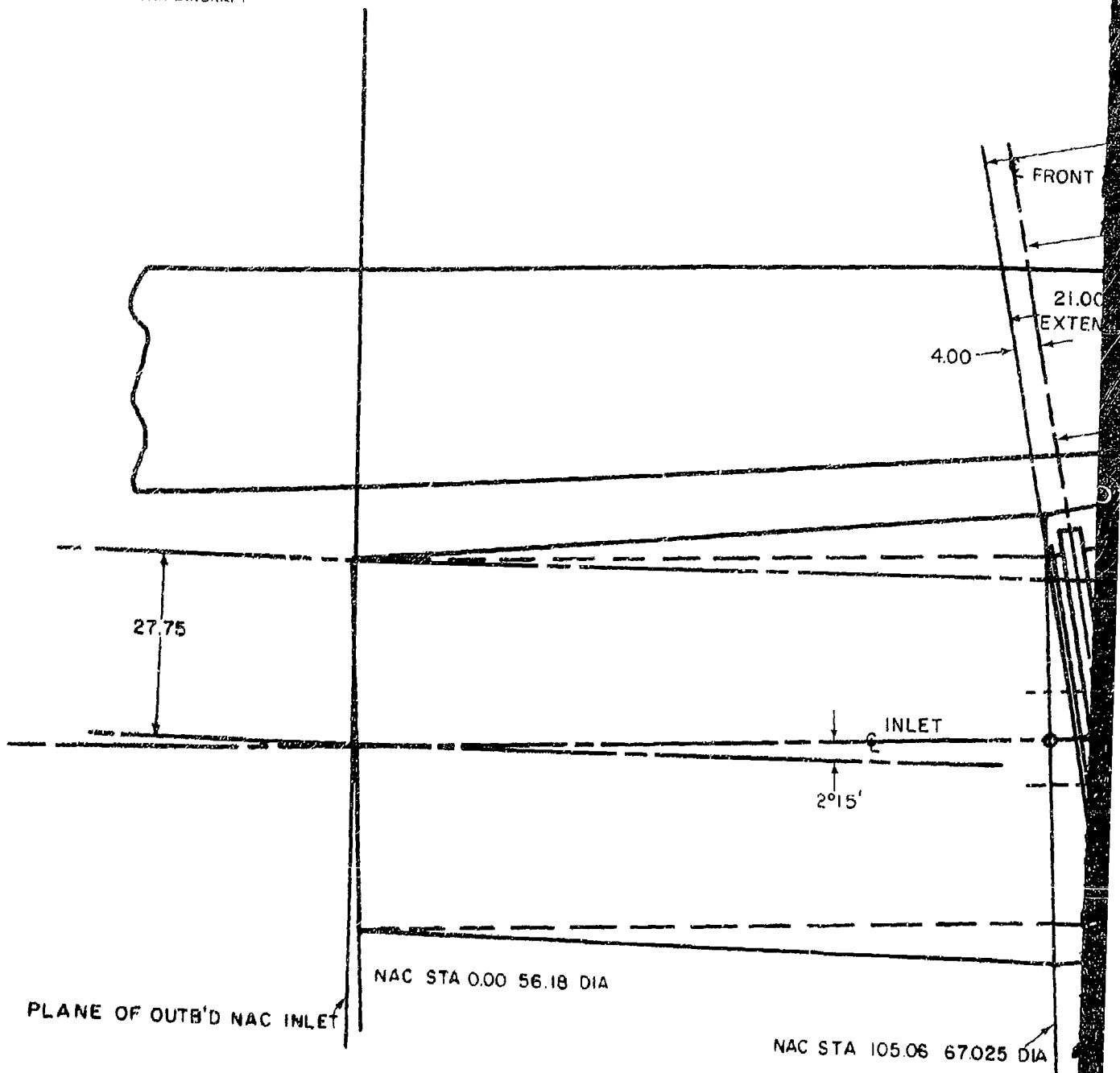
CONFIDENTIAL

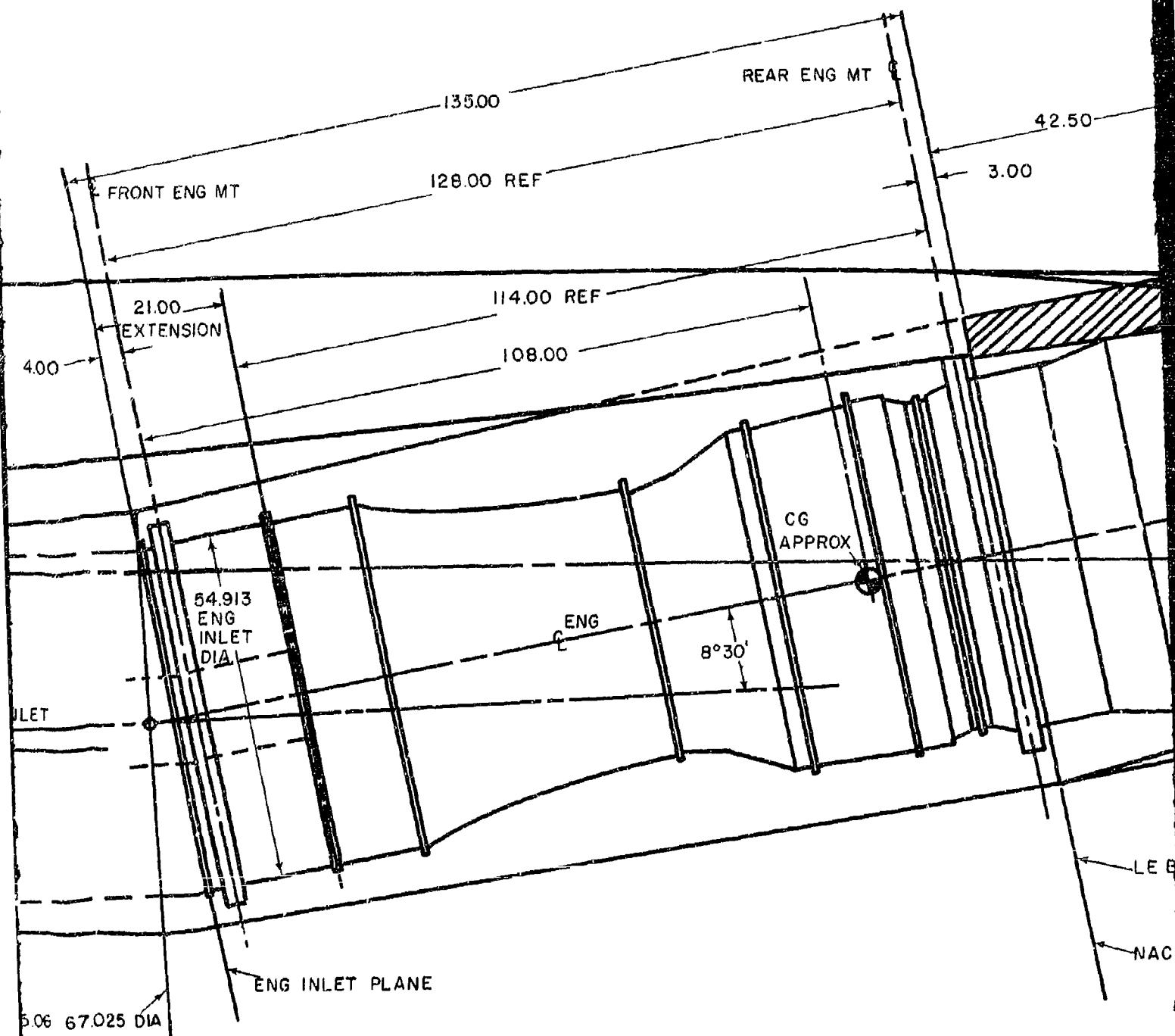
PWA-2600



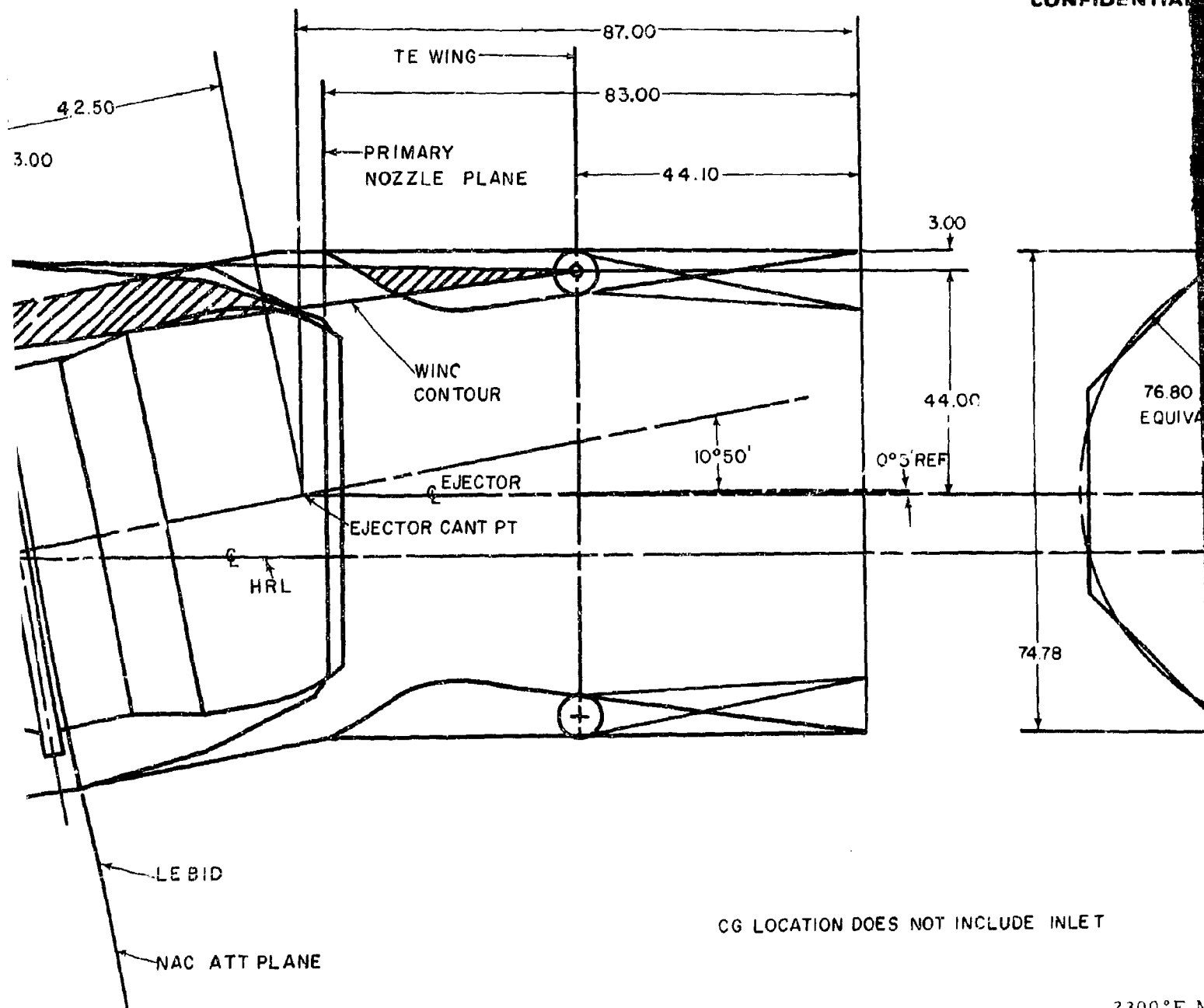
4

PRATT & WHITNEY AIRCRAFT





CONFIDENTIAL



CG LOCATION DOES NOT INCLUDE INLET

2300°F M
TEMPE
EJECTOR
PRIMAR
*NOTE:

STJ227 525 LBS./SEC.

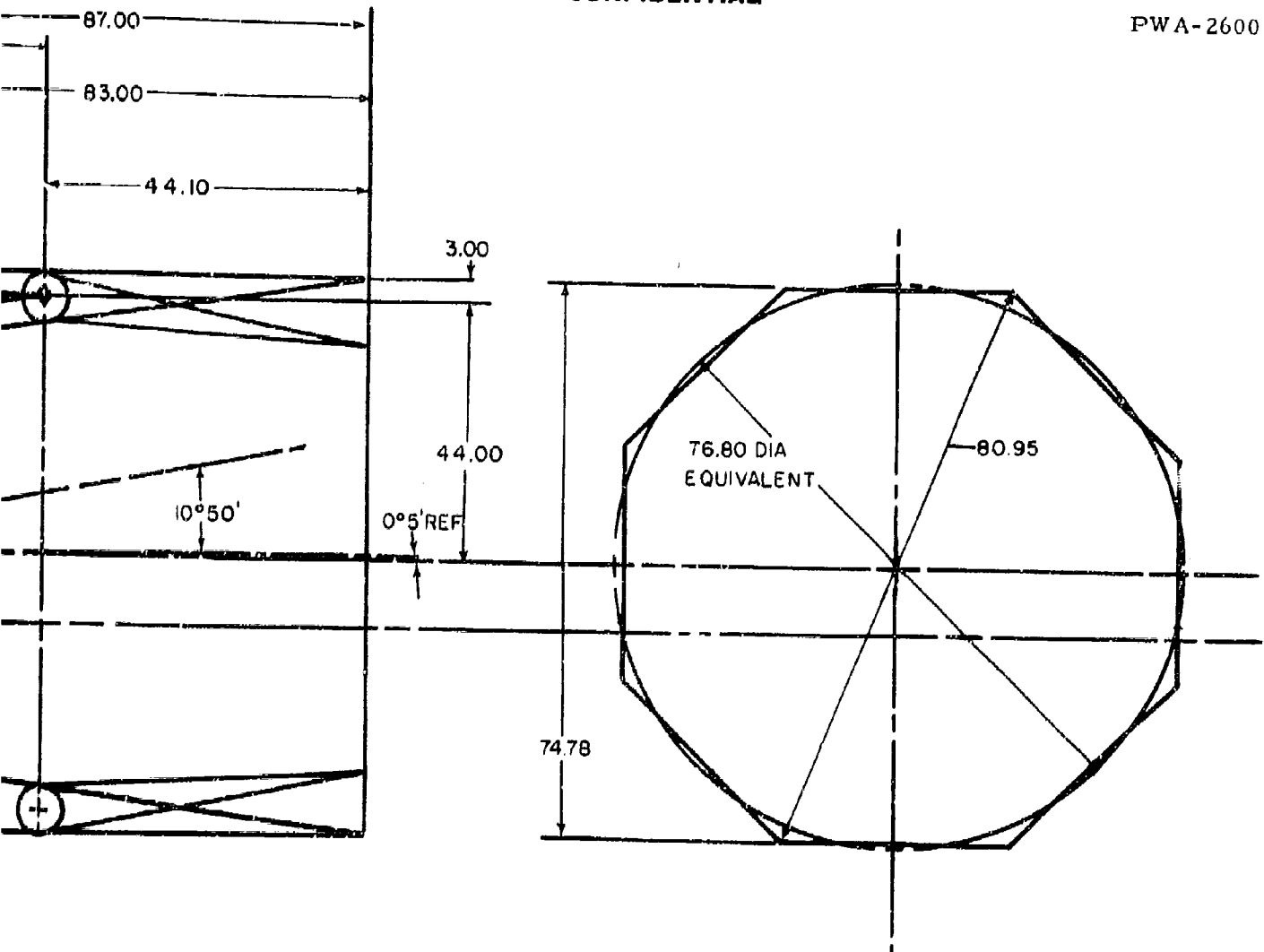
Figure 1-19

CONFIDENTIAL

3

CONFIDENTIAL

PWA-2600



CG LOCATION DOES NOT INCLUDE INLET

2300°F MAXIMUM TURBINE INLET
TEMPERATURE

EJECTOR CANTED 4.00 FWD. OF
PRIMARY NOZZLE PLANE

*NOTE: CG LOCATION DOES NOT
INCLUDE INLET

STJ227 525 LBS./SEC. TURBOJET

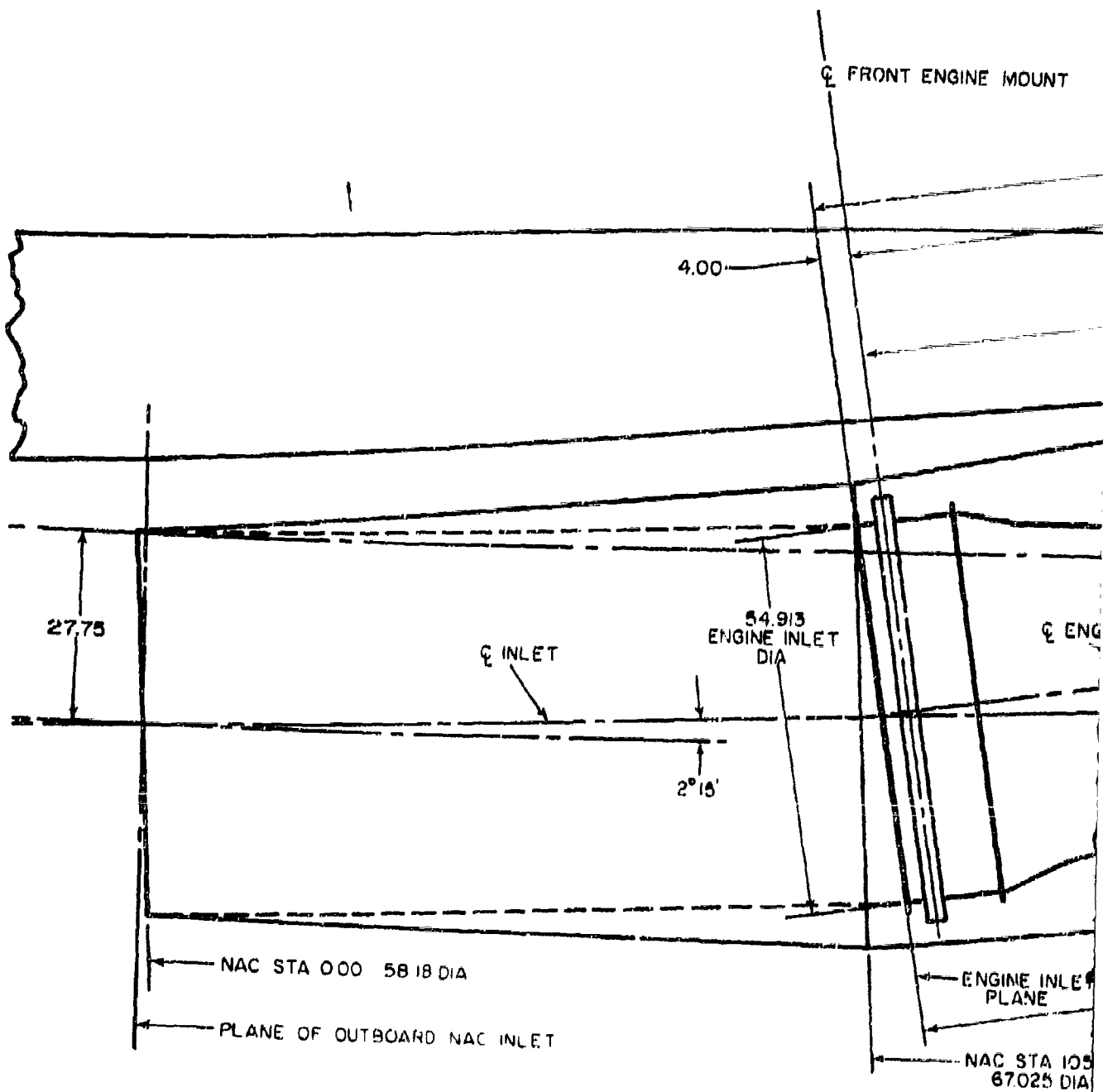
Figure 1-19

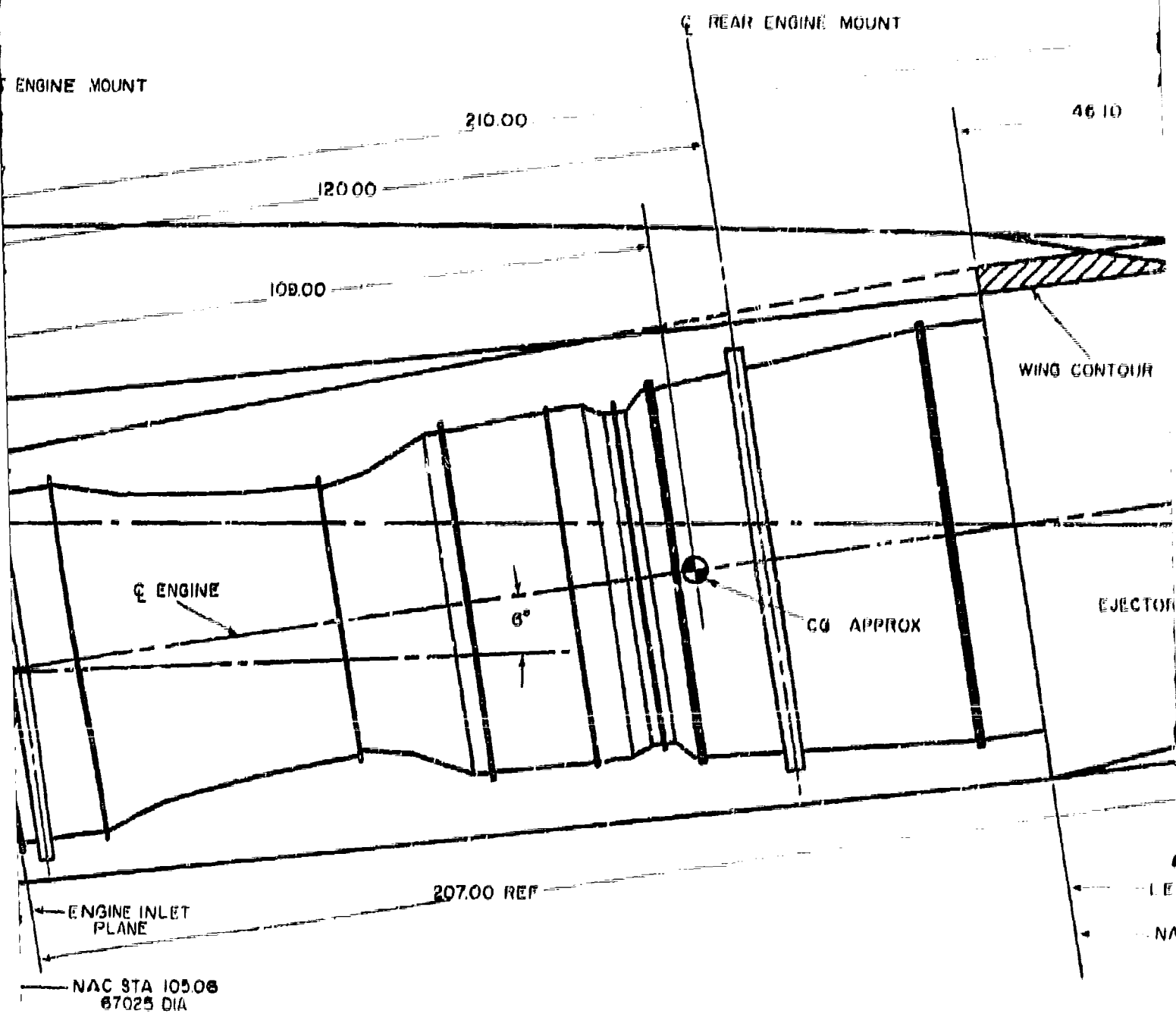
CONFIDENTIAL

CLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
UNLESS INDICATED OTHERWISE

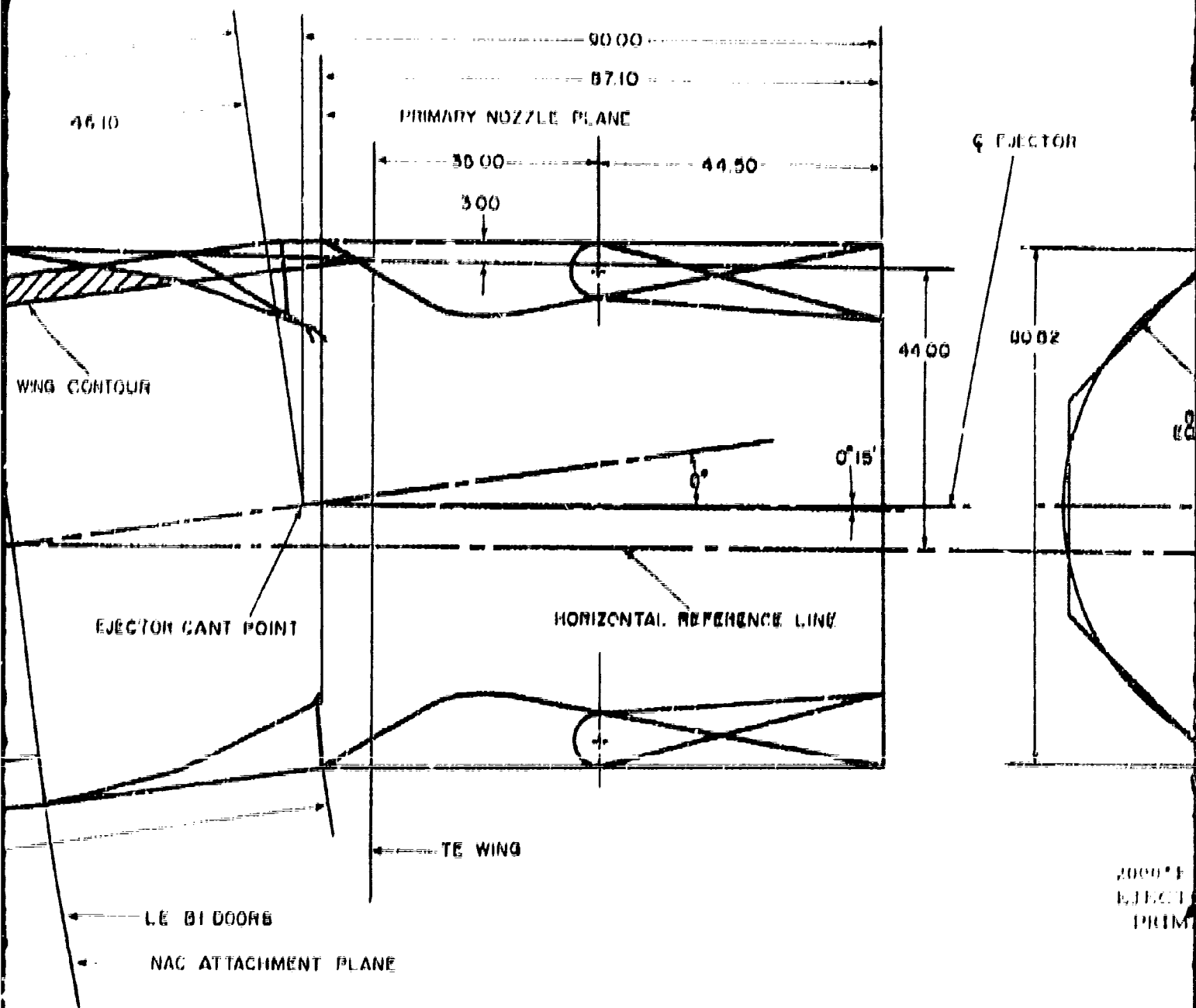
THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.
SECTION 793 AND 794, TRANSMISSION OF THE
SUBSTANCE OF ITS CONTENTS TO ANY PERSON OR
AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

4





CON

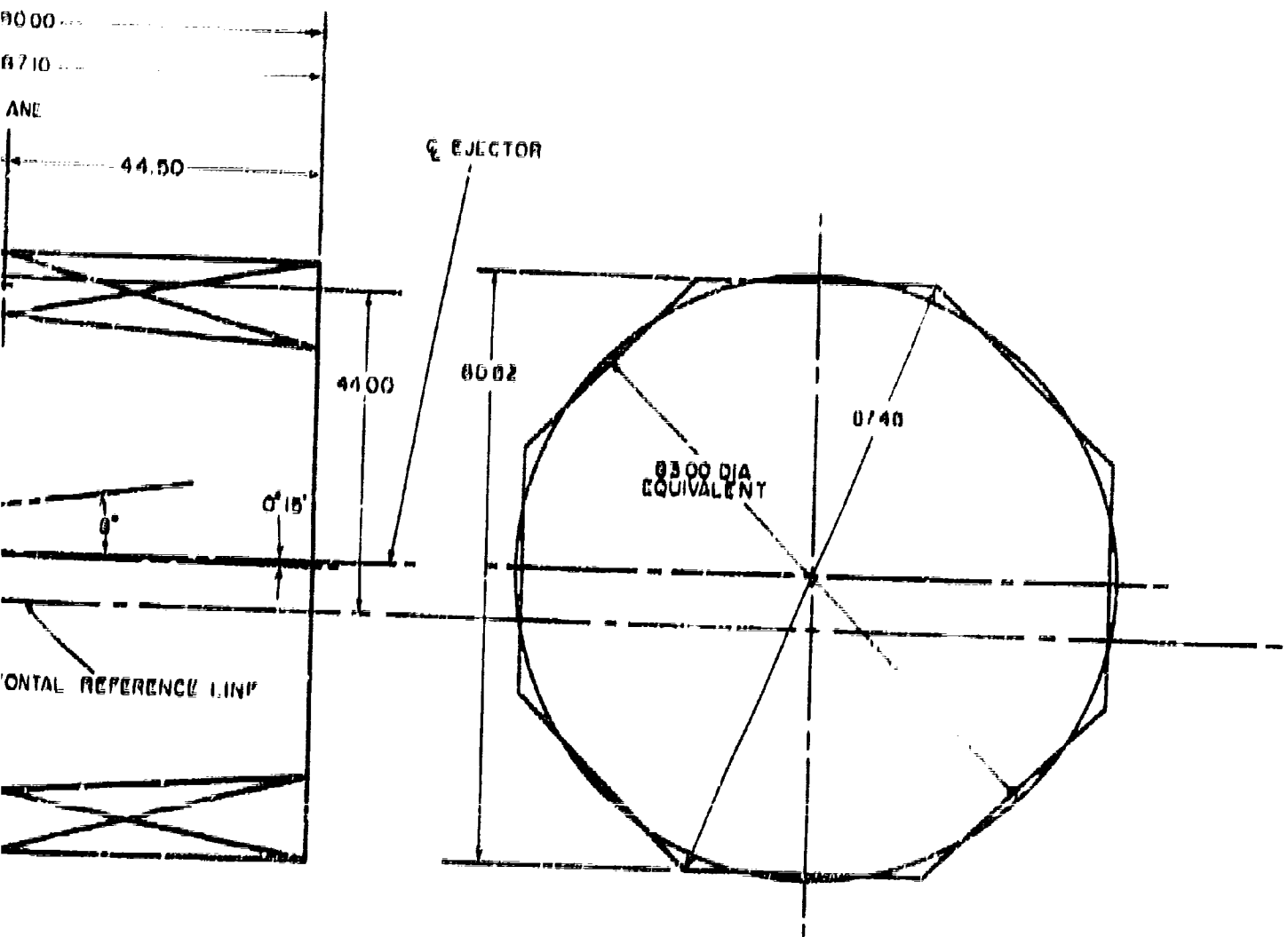


51.227 575 10

CON

CONFIDENTIAL

PWA-2600



2000°F MAXIMUM TURBINE INLET
EJECTOR CANTED 2° 30' FWD. OF
PRIMARY NOZZLE PLANE

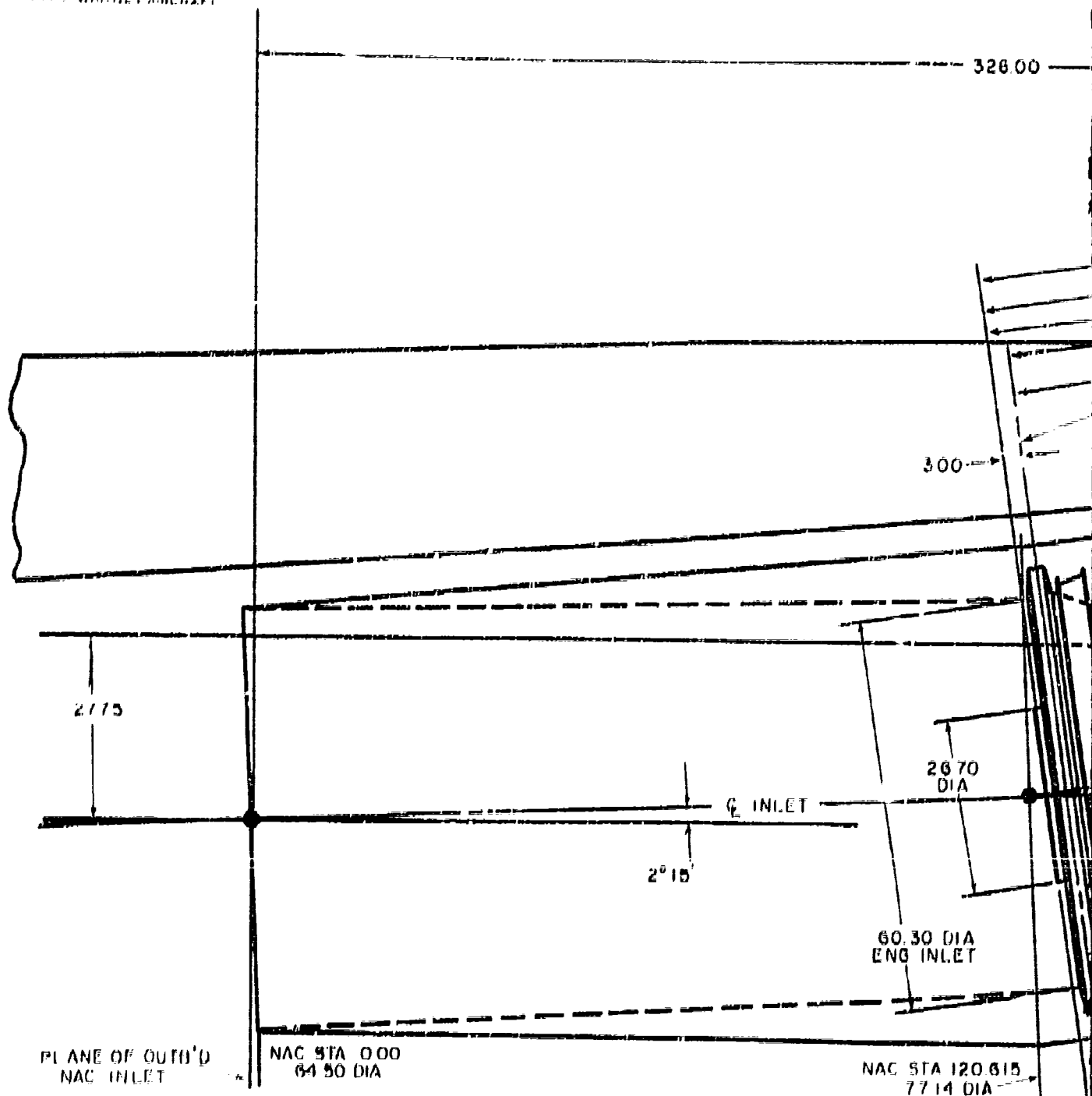
5TJ227 525 LBS./SEC. TURBOJET

Figure 1-20

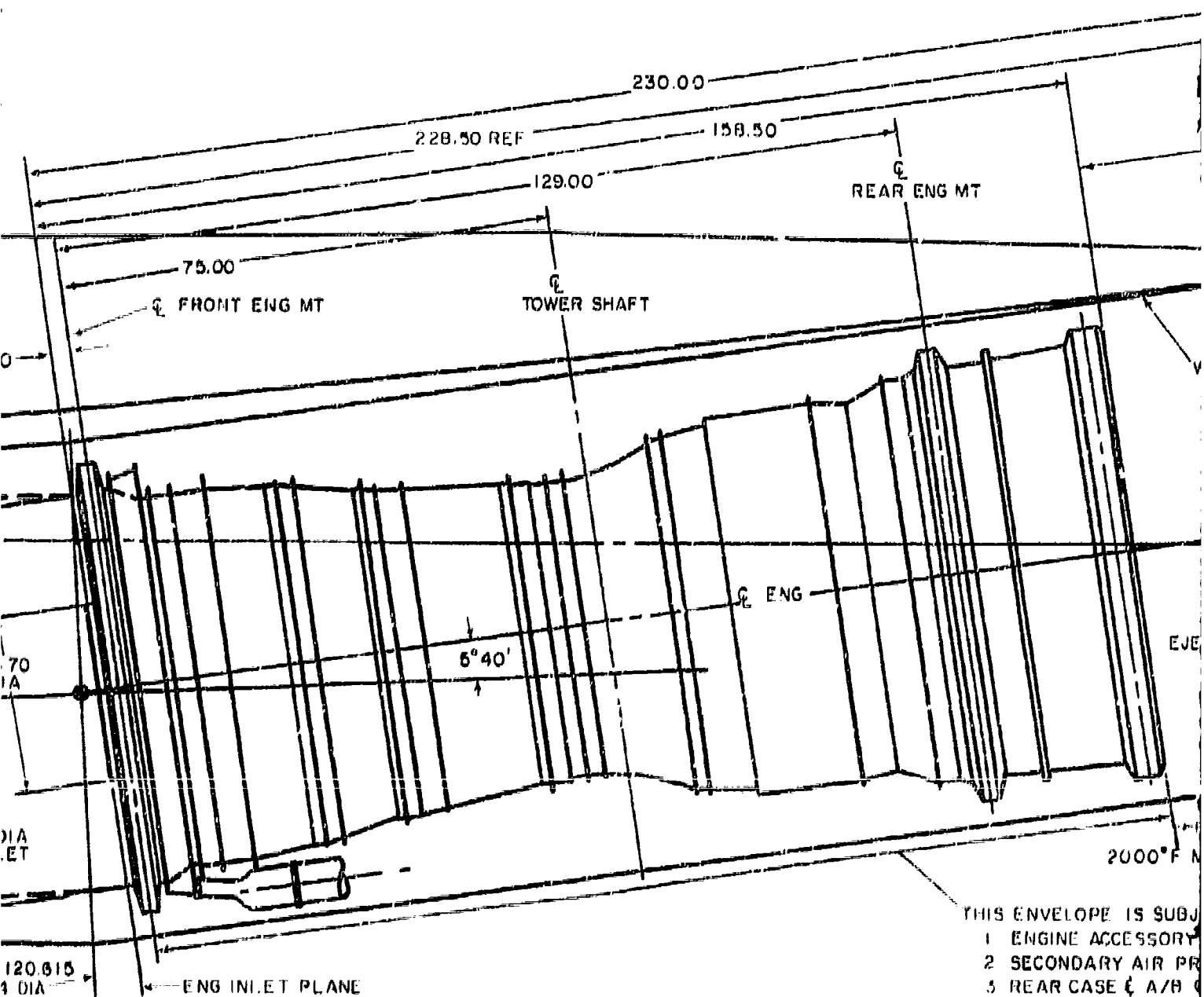
CONFIDENTIAL

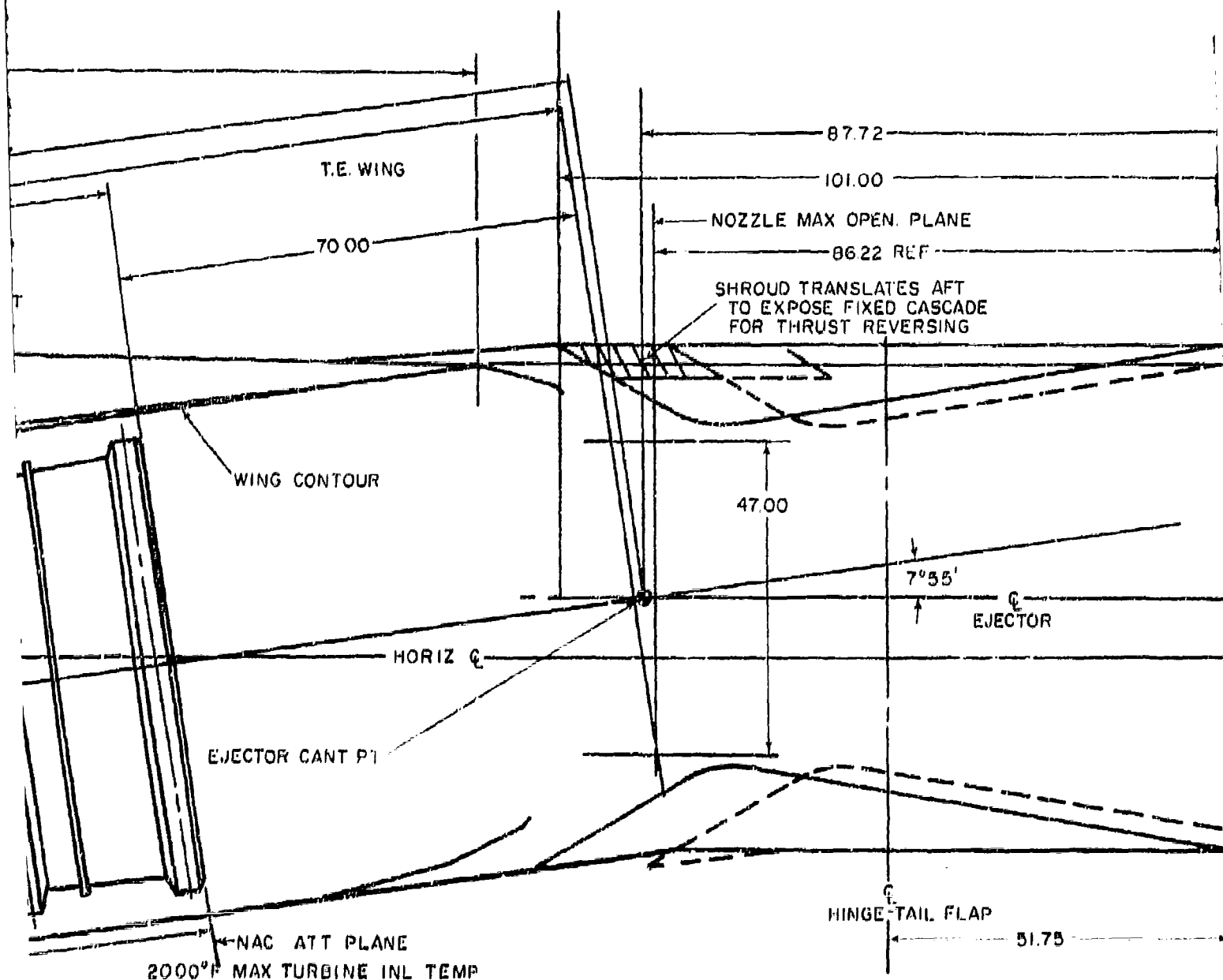
4

PRATT & WHITNEY AIRCRAFT



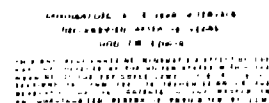
326.00





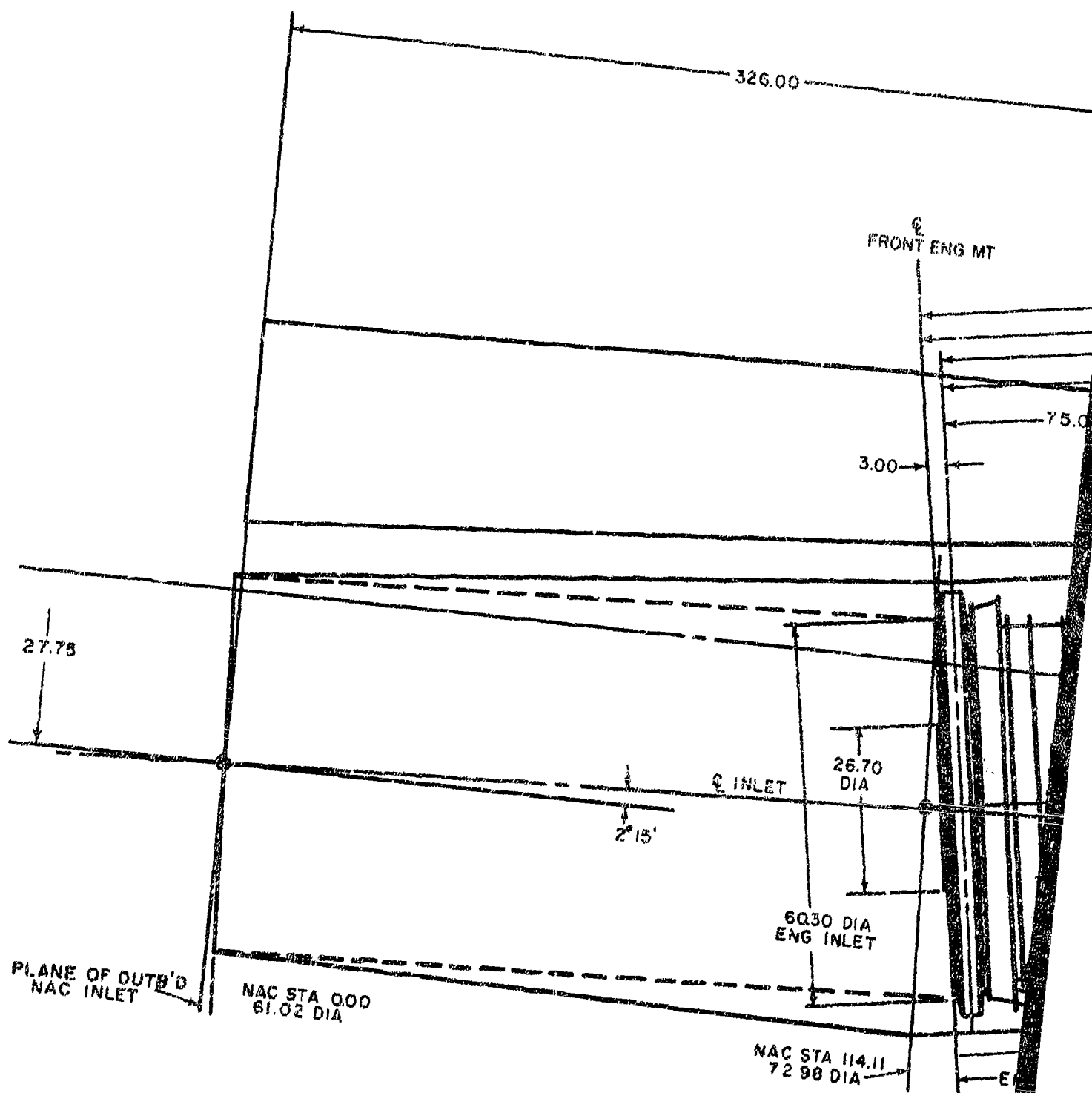
1. ENVELOPE IS SUBJECT TO CHANGE FOR:
 2. ENGINE ACCESSORY ARRANGEMENT
 3. SECONDARY AIR PROVISIONS FOR EJECTOR
 4. REAR CASE & A/O COOLING PROVISIONS

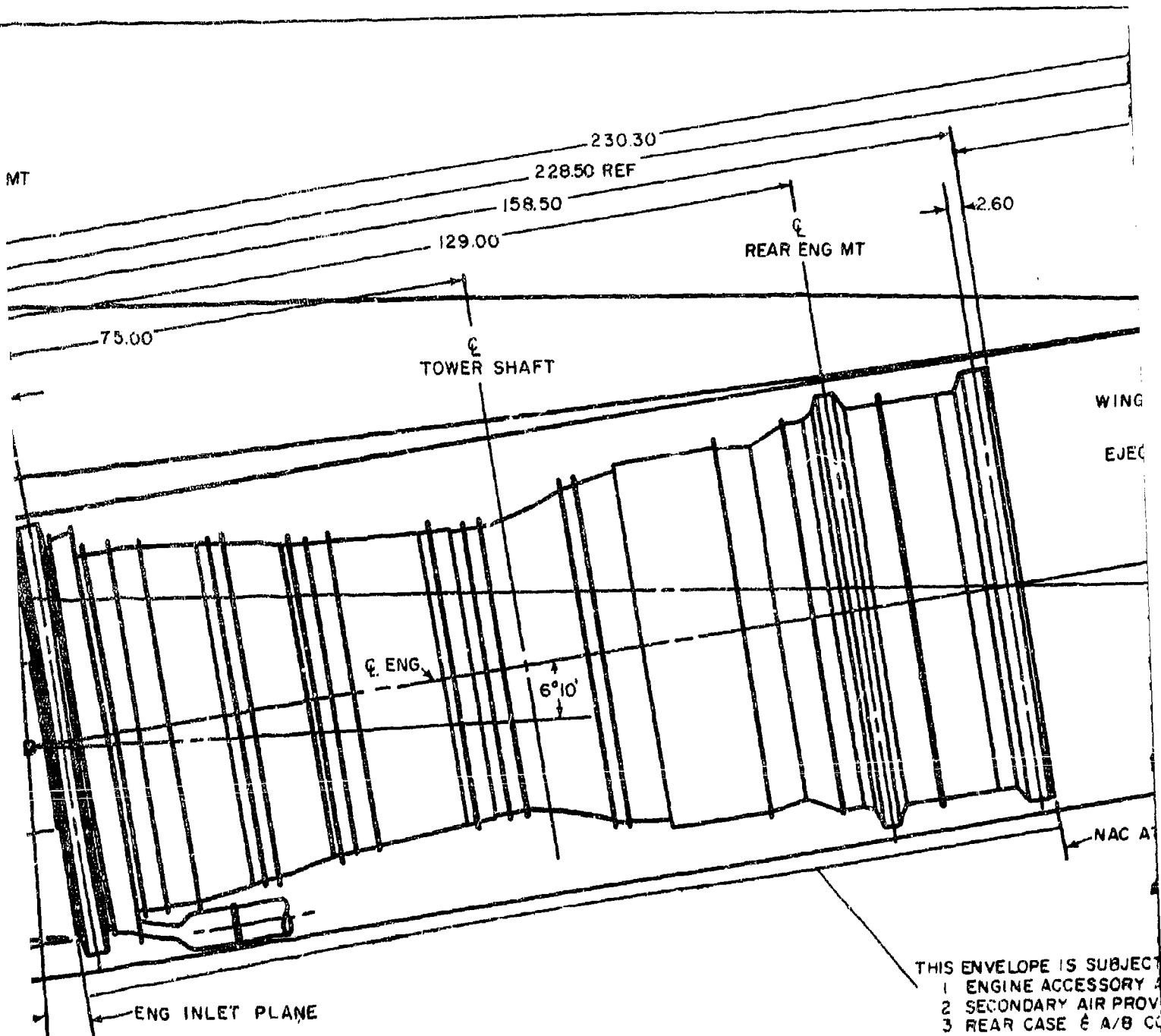
F3W A-2600

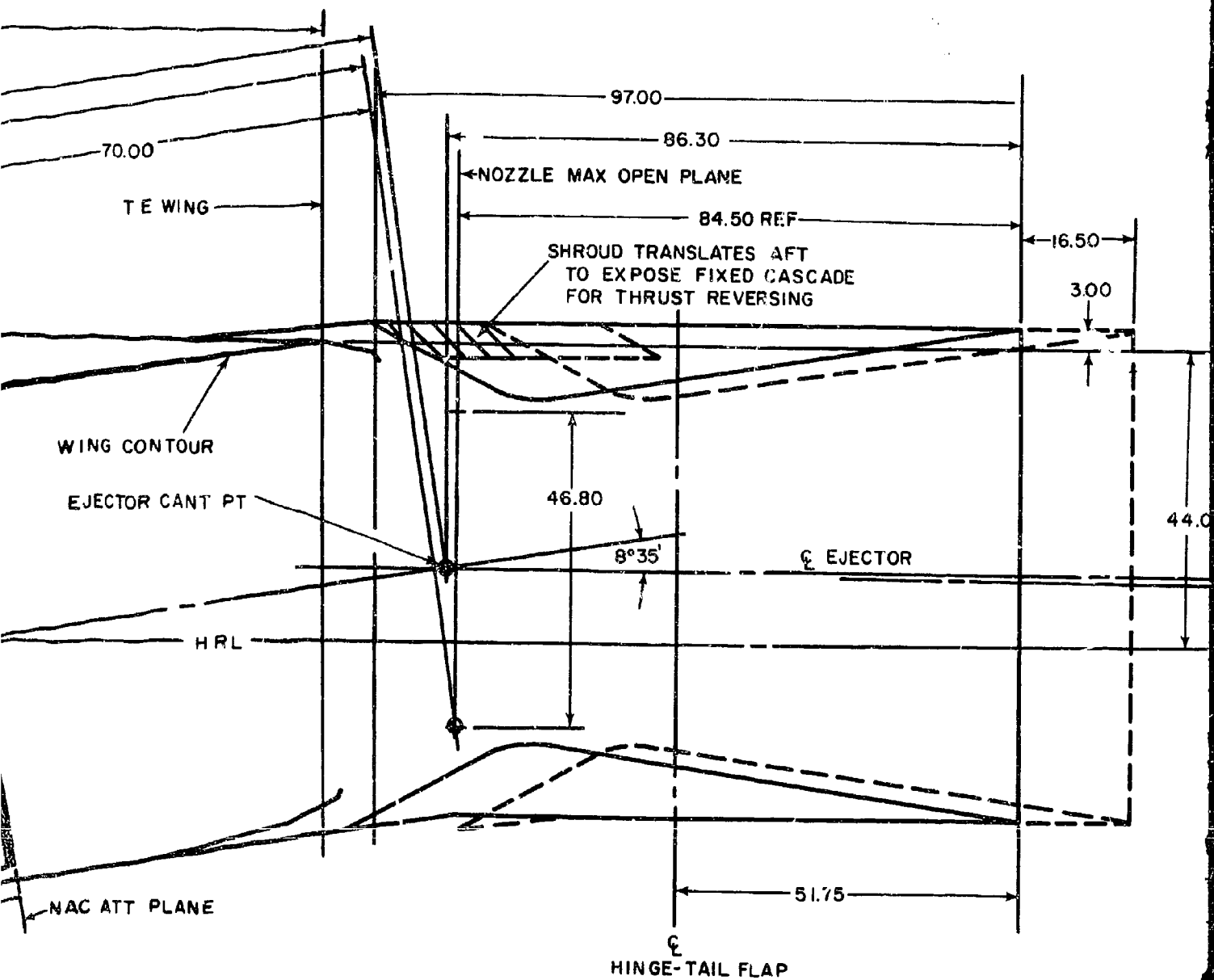


4

PRATT & WHITNEY AIRCRAFT



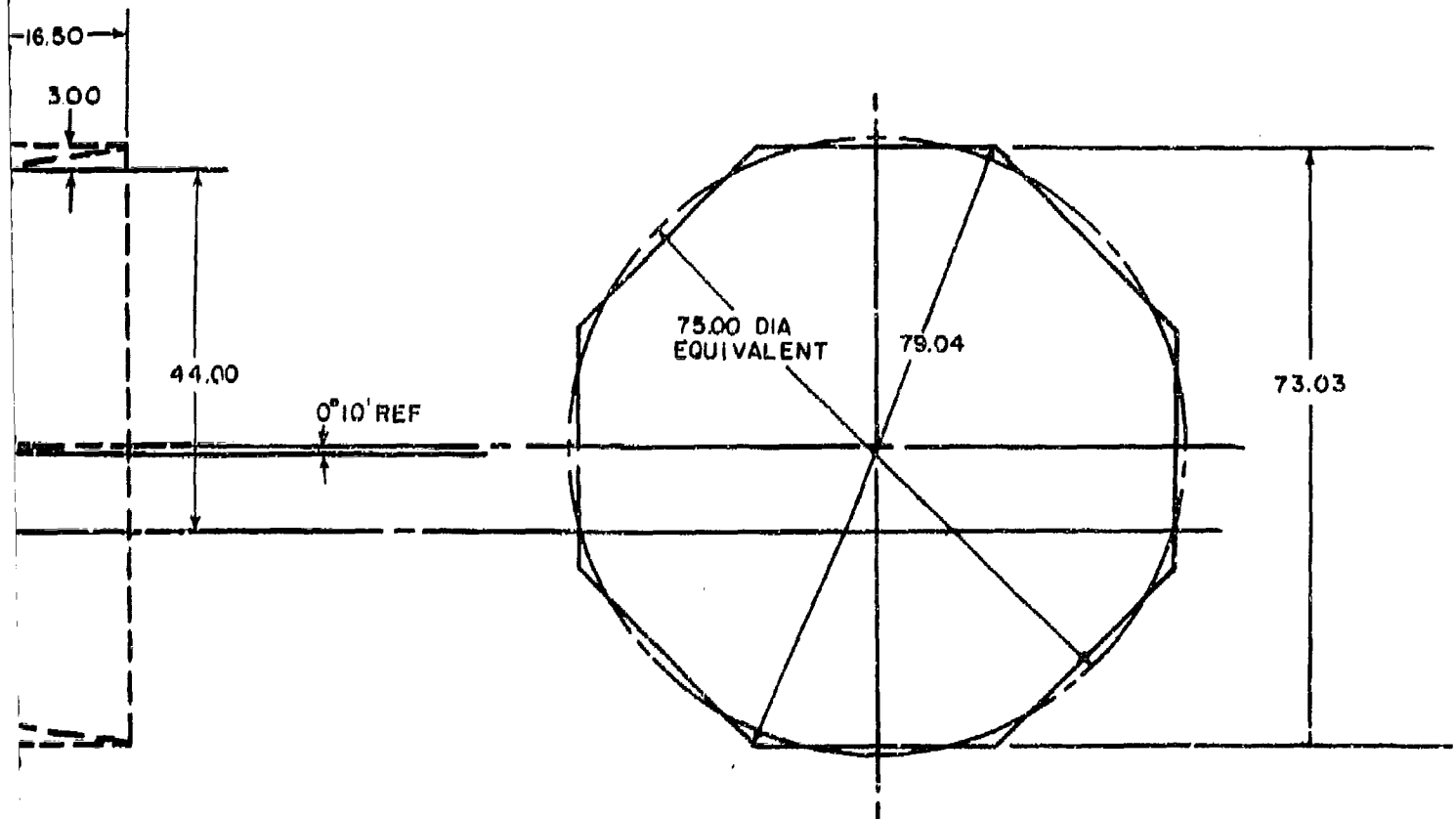




IS SUBJECT TO CHANGE FOR
 ACCESSORY ARRANGEMENT
 AIR PROVISION FOR EJECTOR
 SE & A/B COOLING PROVISION

CONFIDENTIAL

PWA-2600



2000°F TURBINE INLET TEMPERATURE
L. H. SIDE VIEW OUTBOARD POD
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISION FOR EJECTOR
3. REAR CASE AND A/B COOLING PROVISION

STJ227, 525 LBS./SEC. (BASE FLOW) TURBOJET

Figure 1-22

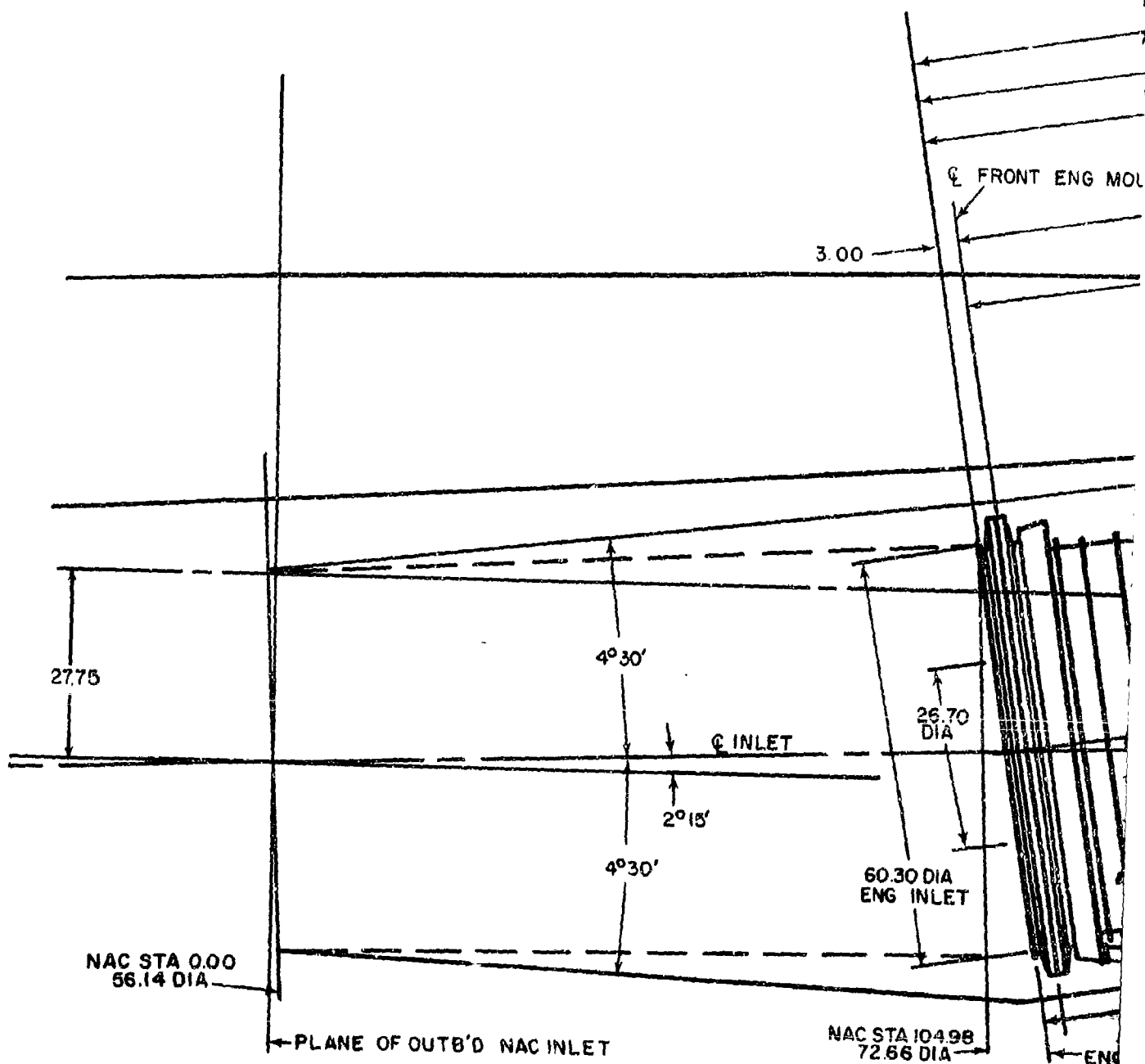
CONFIDENTIAL

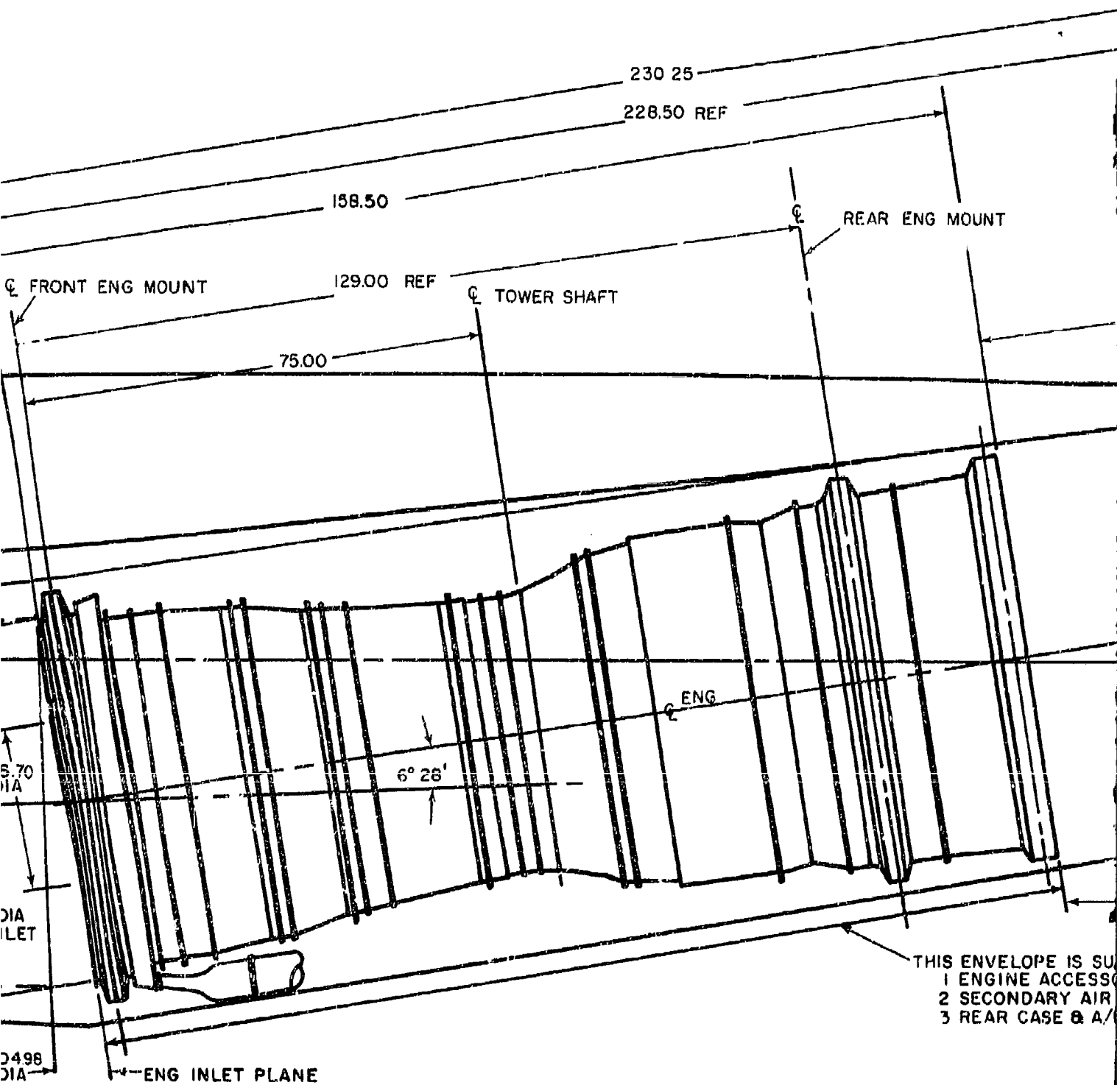
UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFY AFTER 10 YEARS
EXCEPT WHERE SHOWN OTHERWISE

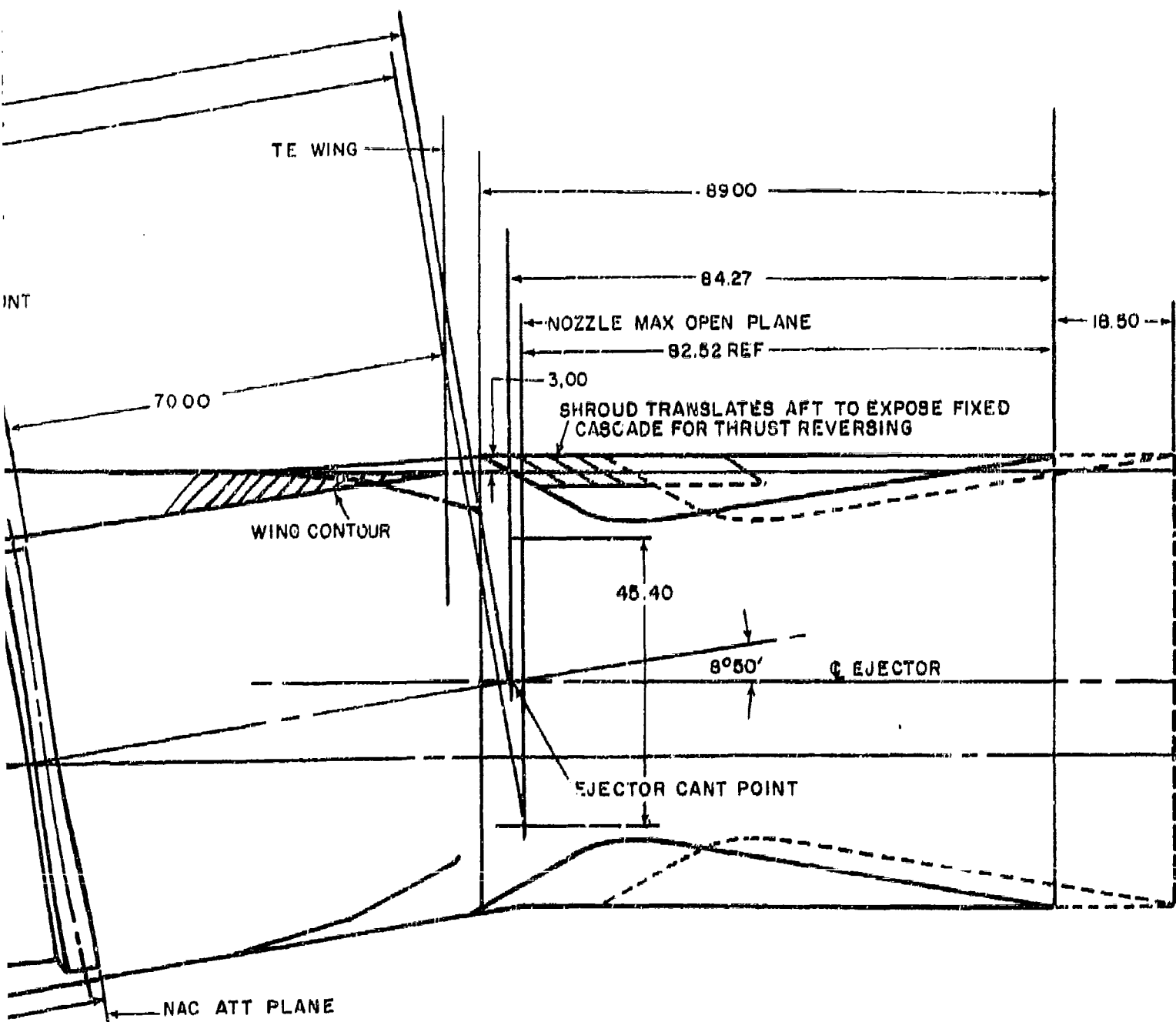
THIS DOCUMENT CONTAINS INFORMATION THAT MAY BE
REGULATED BY THE EXPORT ADMINISTRATION ACT OF 1969
AND THE INTERNATIONAL TRADE AND COMMERCE ACT OF 1980
AND IS SUBJECT TO THE CONTROLS OF THE EAR AND ITAR
AND IS NOT TO BE RELEASED OR DISCLOSED TO THE PUBLIC

4

PRATT & WHITNEY AIRCRAFT

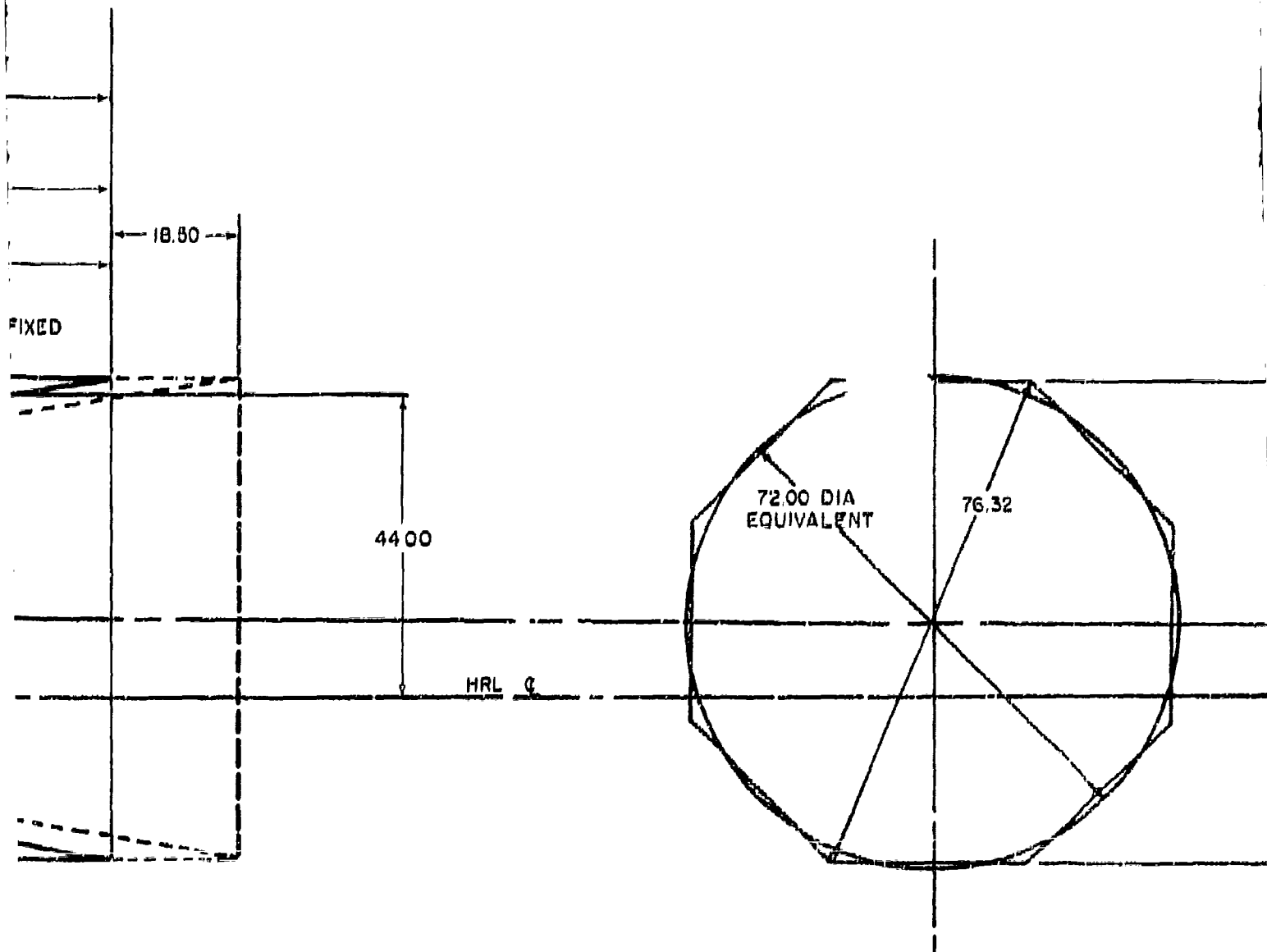






ENVELOPE IS SUBJECT TO CHANGE FOR
ENGINE ACCESSORY ARRANGEMENT
SECONDARY AIR PROVISIONS FOR EJECTOR
NEAR CASE & A/B COOLING PROVISIONS

CONFIDENTIAL



2000°F TURBINE INLET TEMPERATURE
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY A/R PROVISIONS FOR EJECTOR
3. REAR CASE & A/B COOLING PROVISIONS

STJ227, 525 LBS./SEC. (LOW FLOW) TURBOJET

Figure 1-23

CONFIDENTIAL

PW A-2600

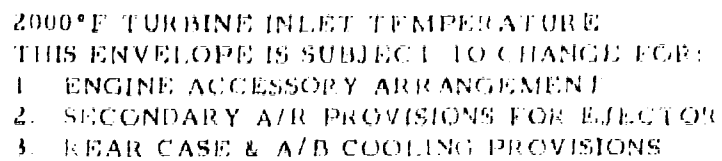
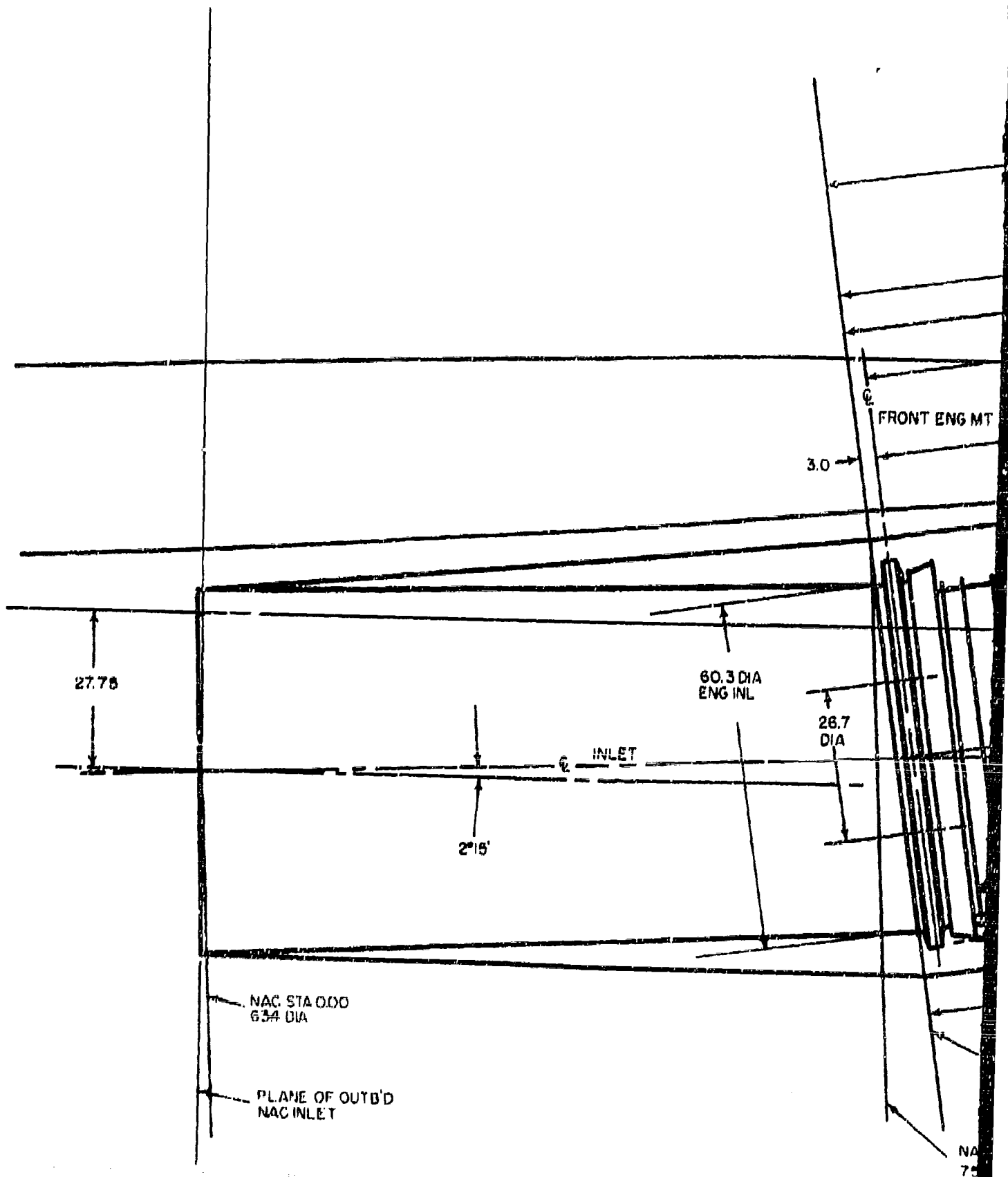


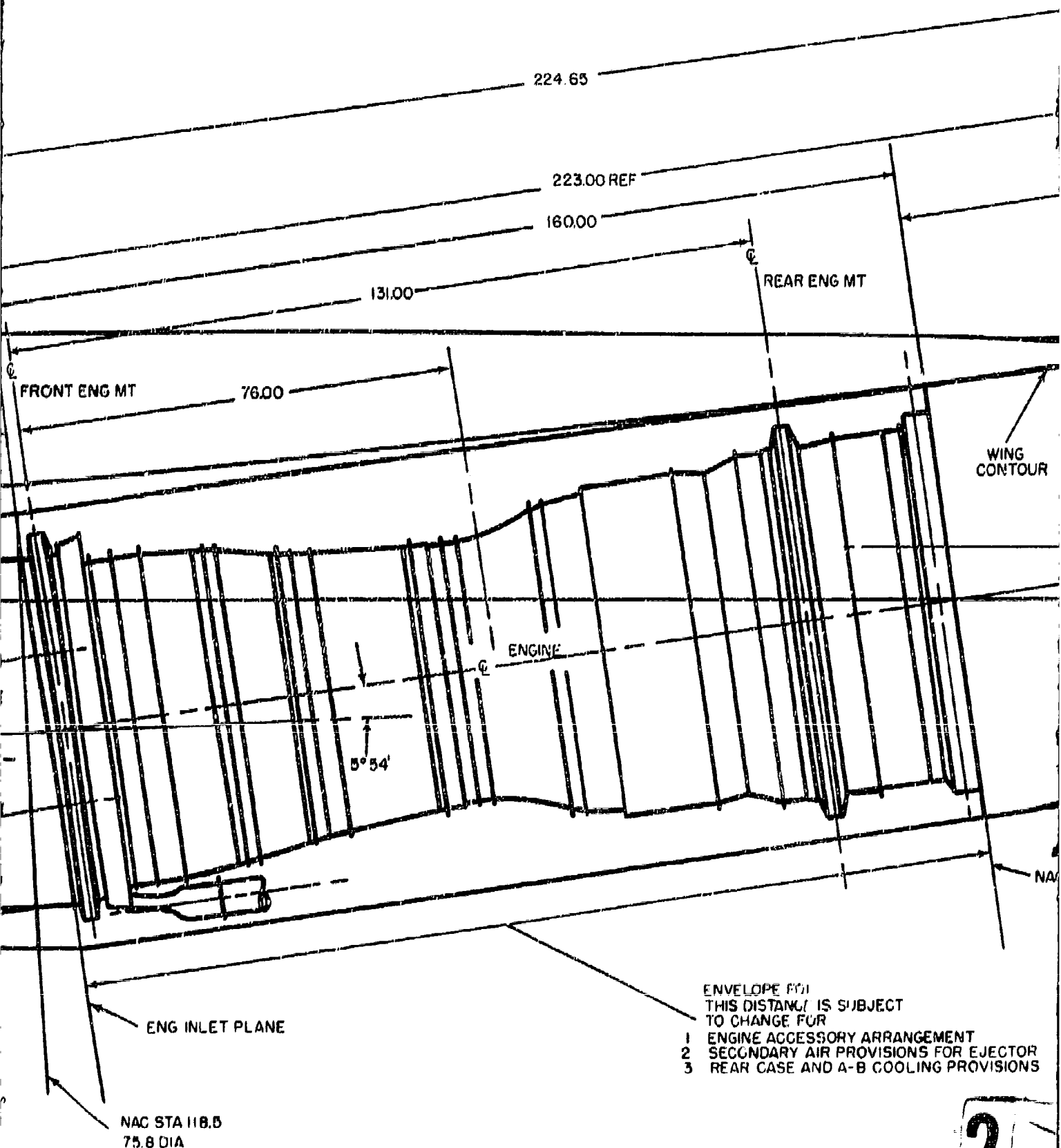
Figure 1-23

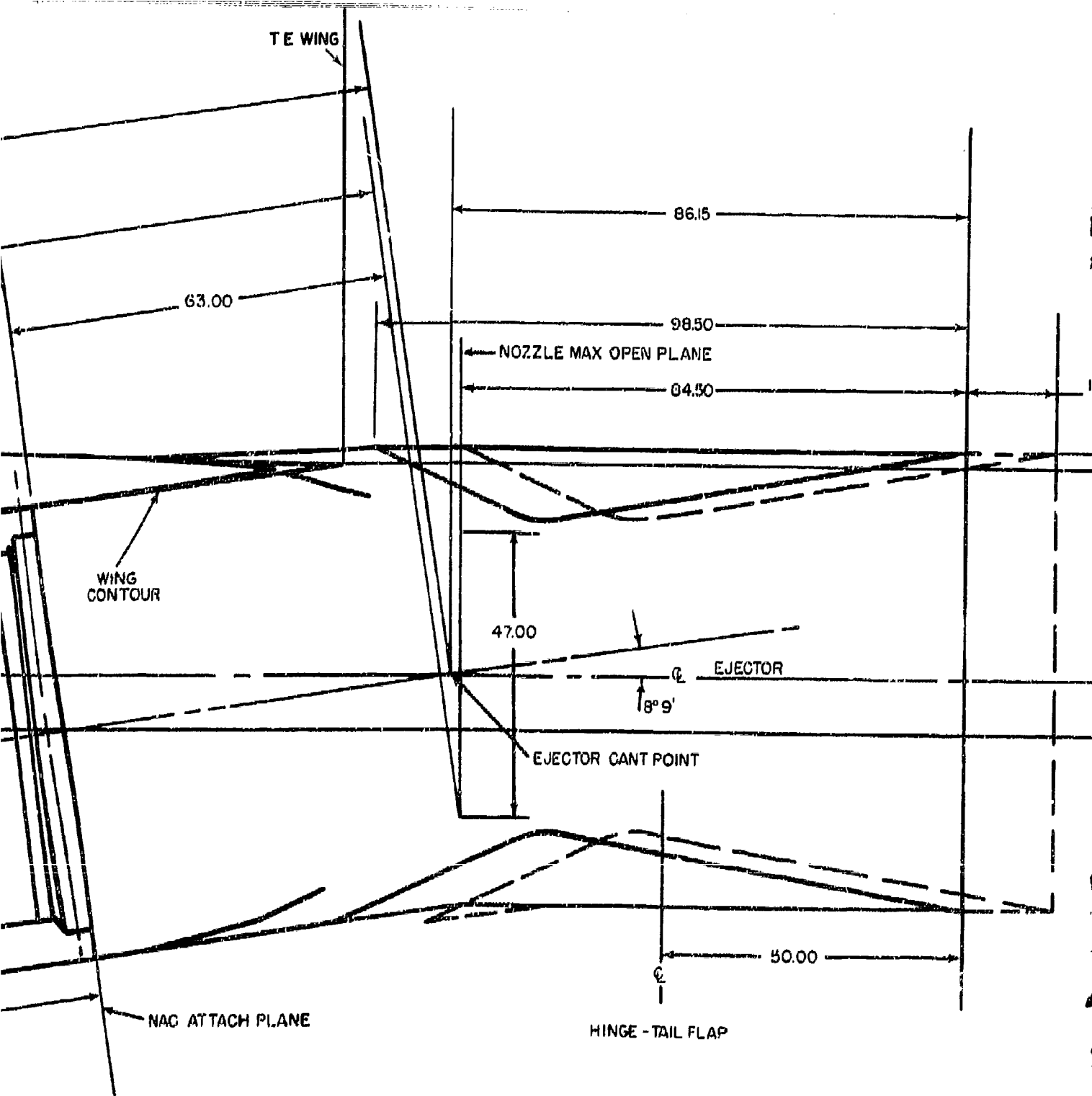
CONFIDENTIAL

《中国大百科全书·音乐舞蹈卷》(中国大百科全书出版社, 1989年)第145页

5

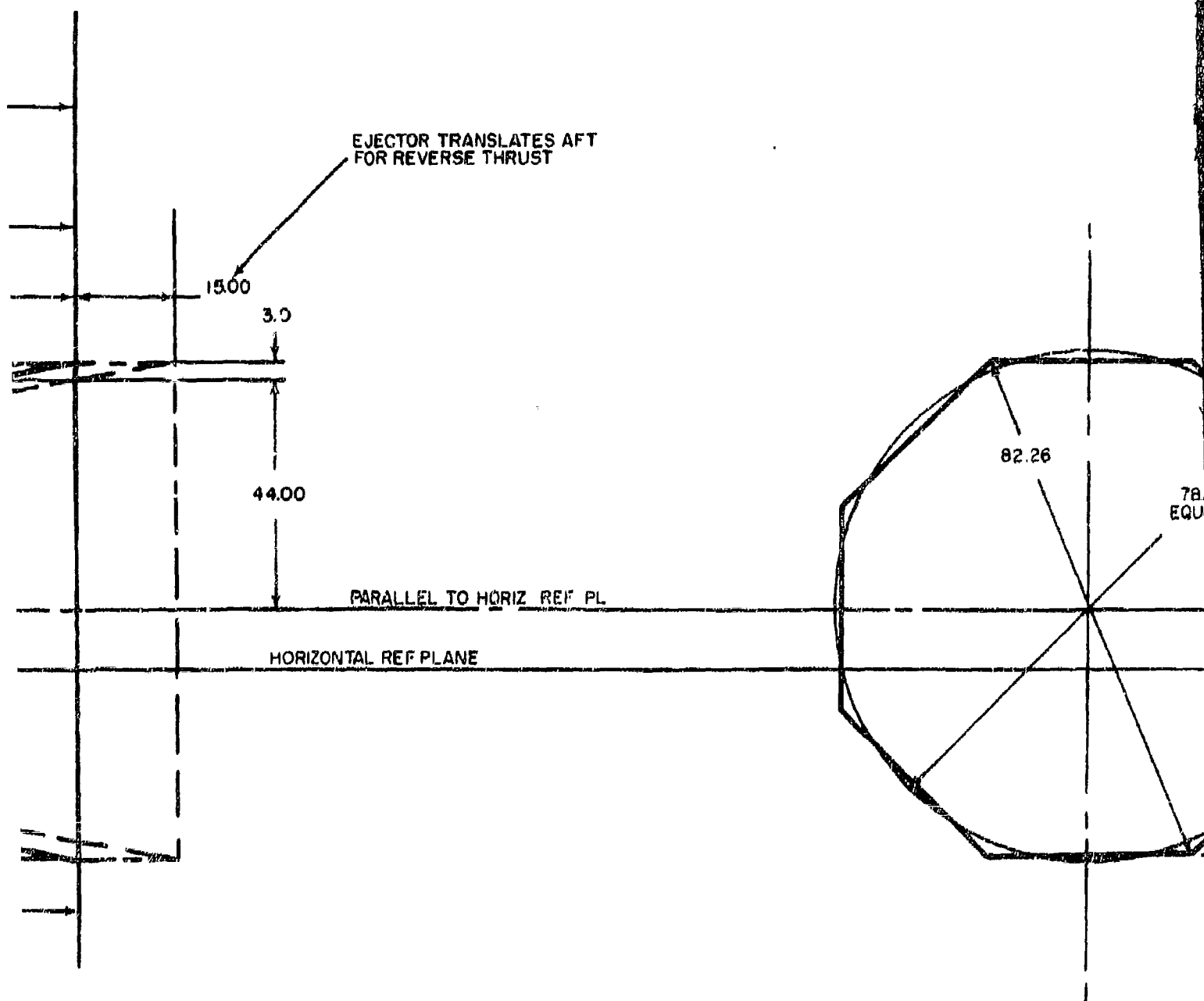






ST
 ANGEEMENT
 IONS FOR EJECTOR
 OLING PROVISIONS

CONFIDENTIAL



2300°F TURBINE INLET TEMPERATURE
STJ227, 525 LBS./SEC. (HIGH FLOW)

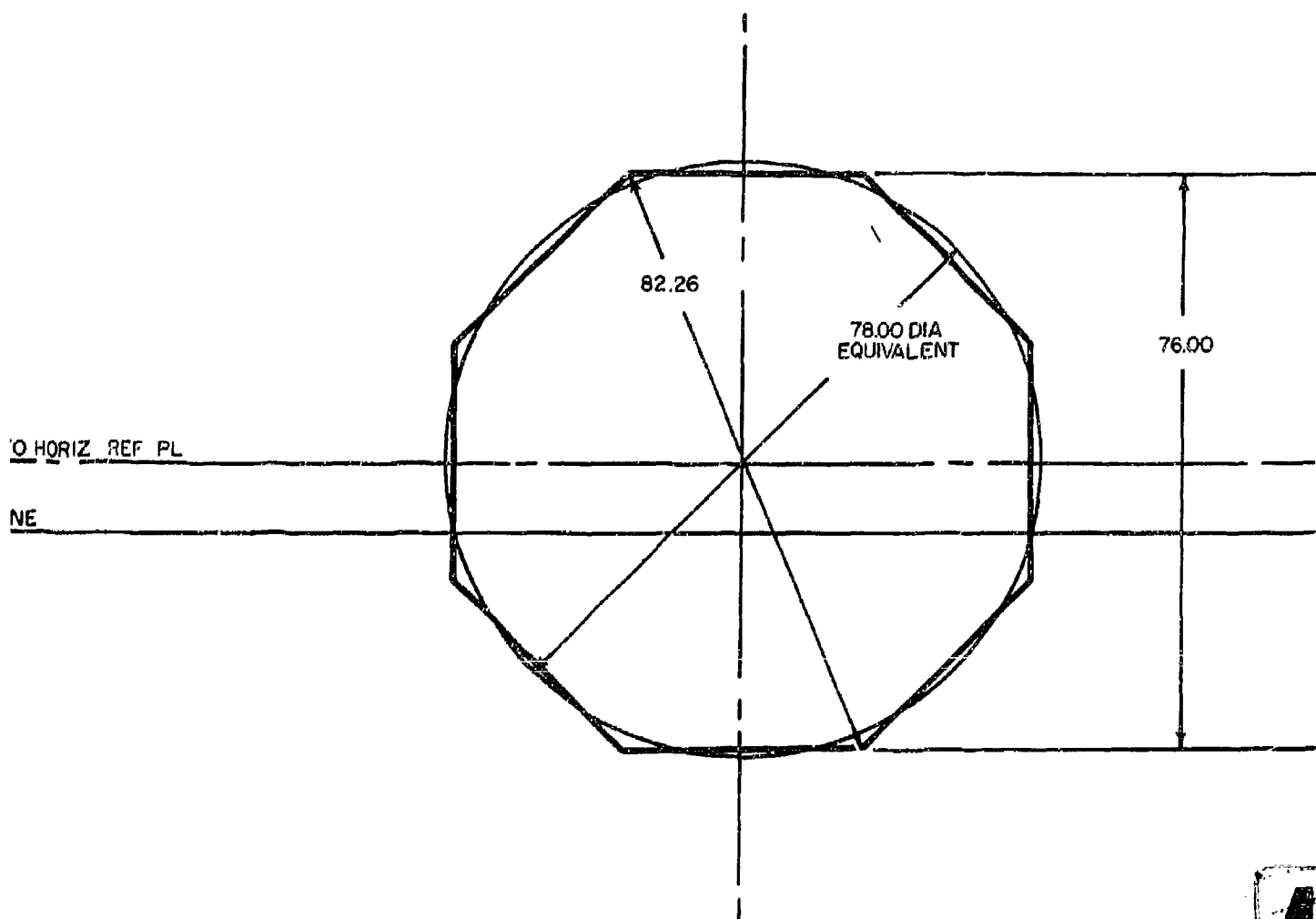
Figure 1-24

CONFIDENTIAL

CONFIDENTIAL

PWA-2600

TRANSLATES AFT
SE THRUST



2300°F TURBINE INLET TEMPERATURE
STJ227, 525 LBS./SEC. (HIGH FLOW) TURBOJET

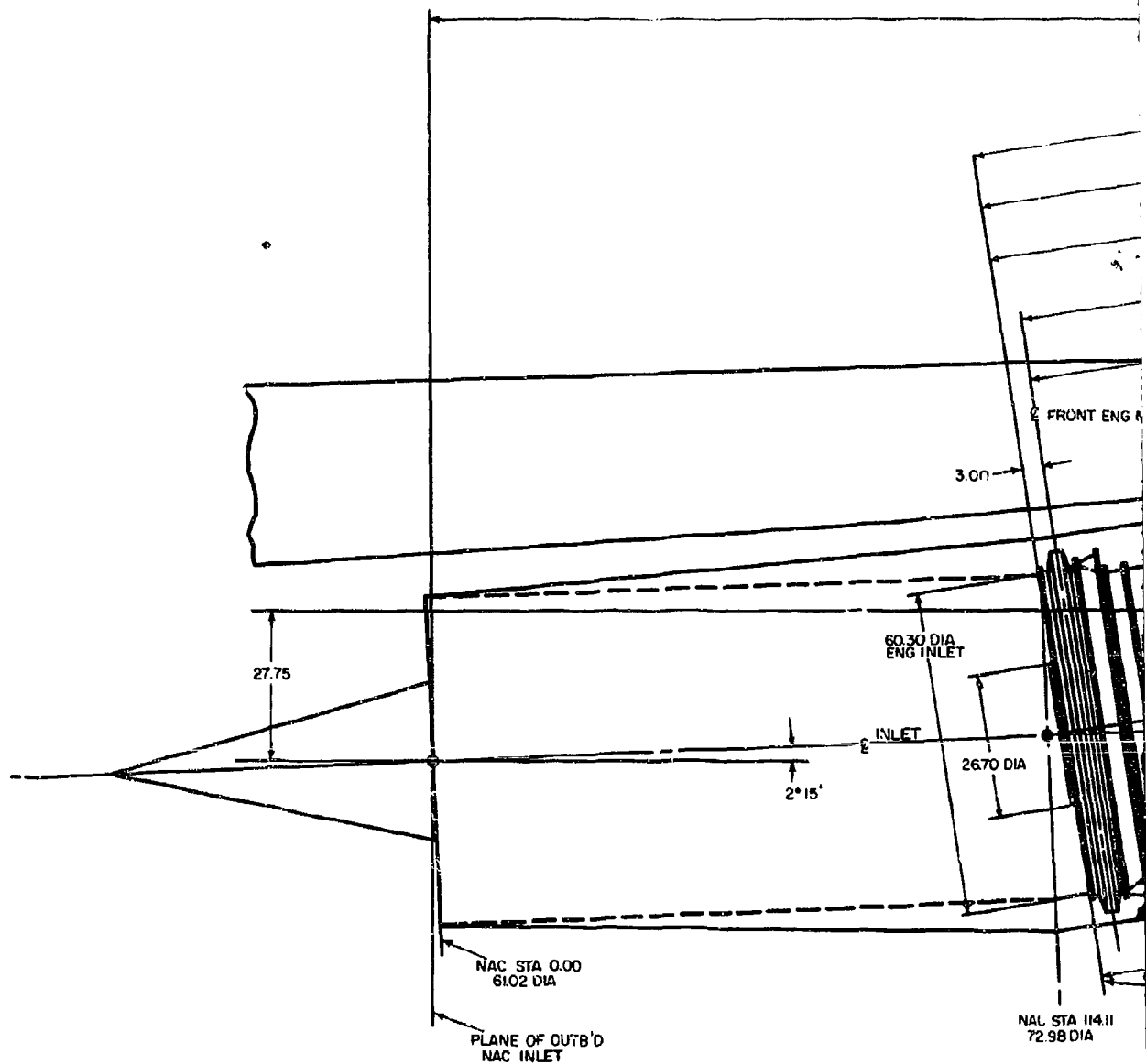
Figure 1-24

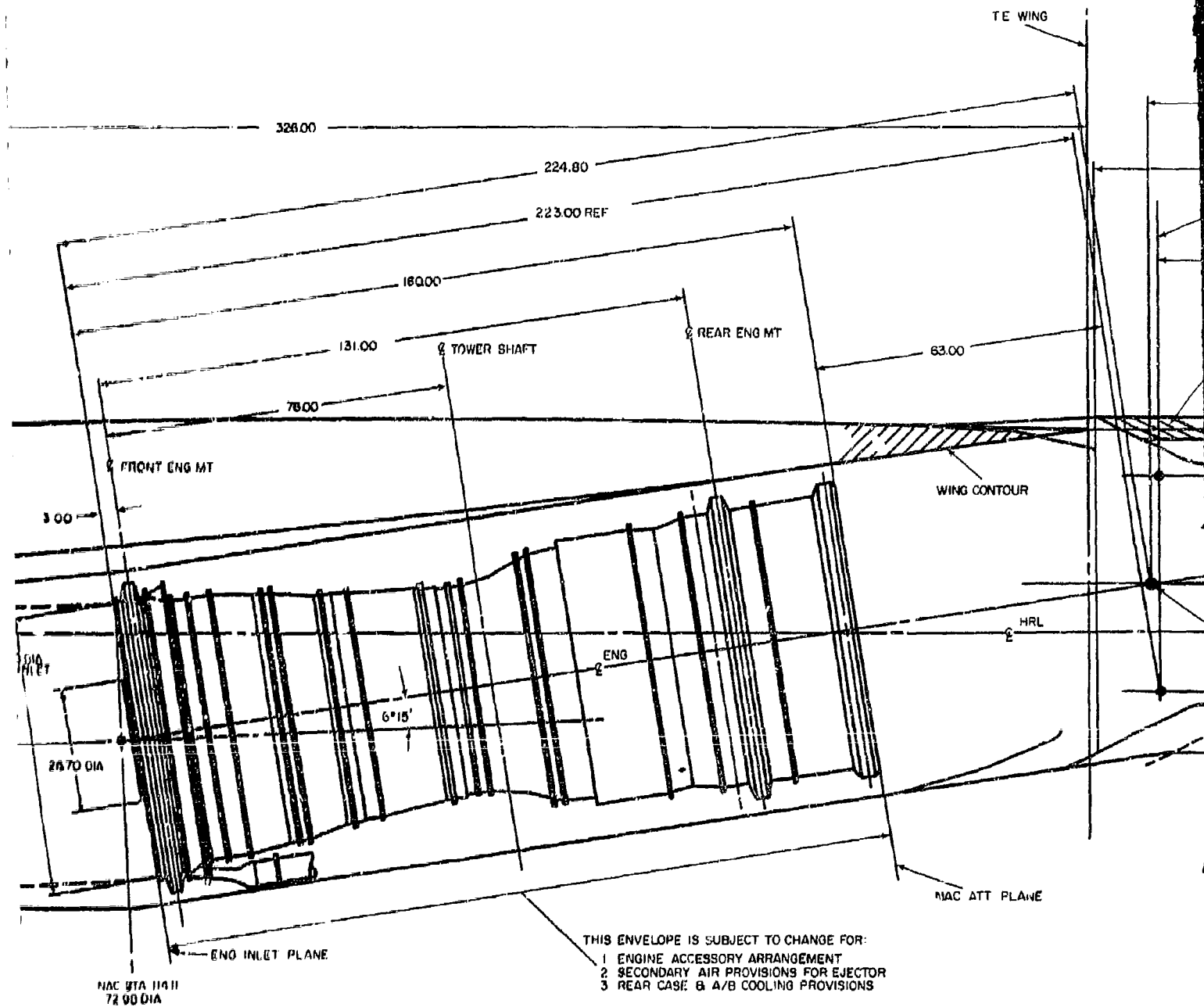
CONFIDENTIAL

UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS
DDC DOW 5300-10
THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.,
SECTION 793 AND 794, THE TRANSMISSION OF THE
REVEALATION OF ITS CONTENTS TO ANY PERSON OR
IN ANY MANNER IS PROHIBITED BY LAW

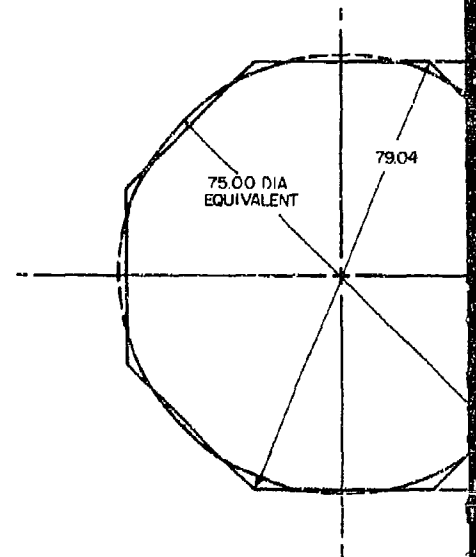
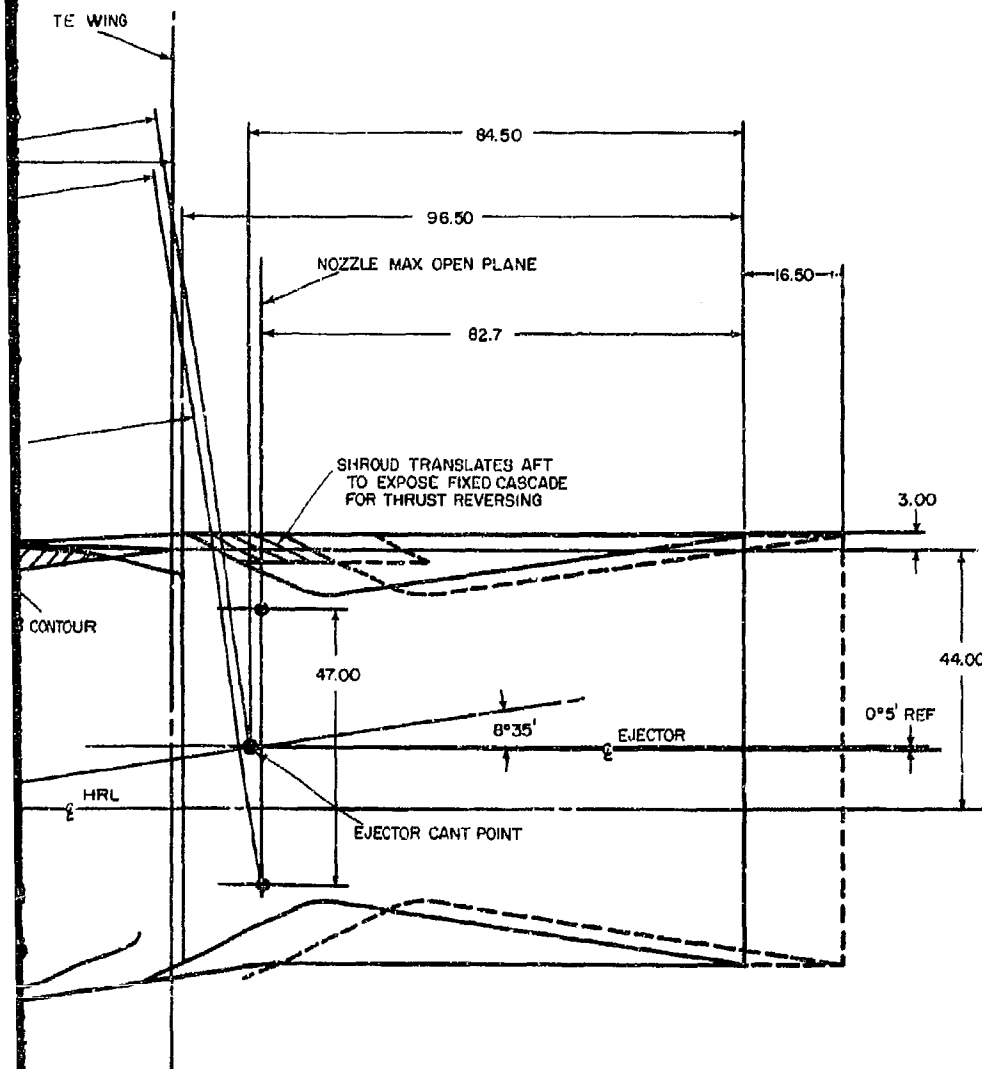
4

PRATT & WHITNEY AIRCRAFT





CONFIDENTIAL



2300°F TURBINE INLET TEMPERATURE
THIS ENVELOPE IS SUBJECT TO CHANGES
1. ENGINE ACCESSORY ARRANGEMENTS
2. SECONDARY AIR PROVISIONS FOR
3. REAR CASE & A/B COOLING PROVISIONS

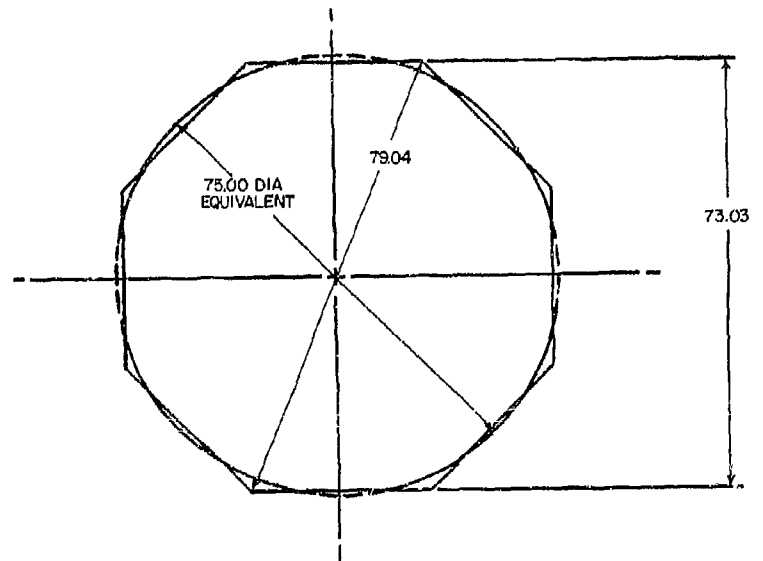
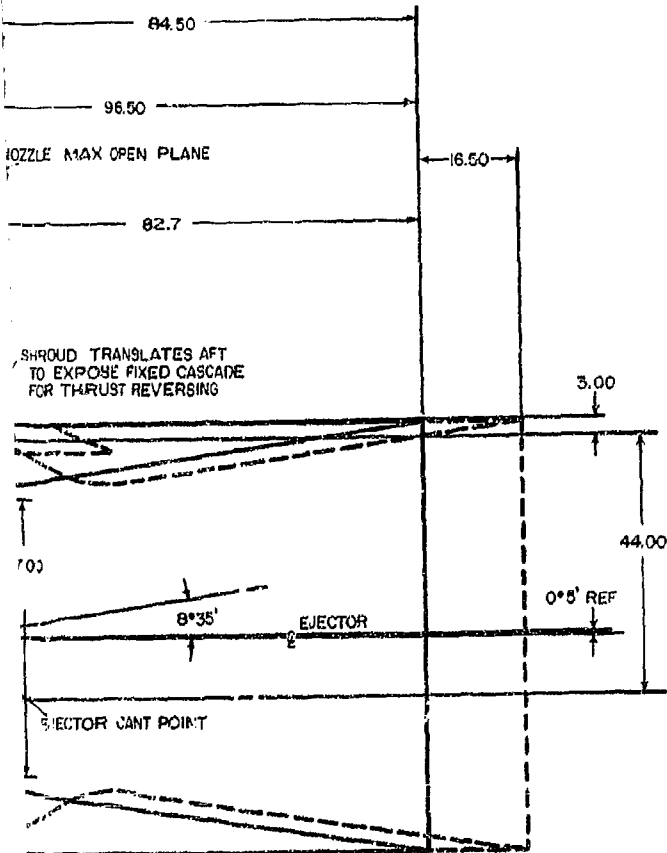
STJ227, 525 LBS./SEC. (BASE FLOW) TURBOJET

Figure 1-25

CONFIDENTIAL

CONFIDENTIAL

PWA-2600



2300°F TURBINE INLET TEMPERATURE
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR EJECTOR
3. REAR CASE & A/B COOLING PROVISIONS

STJ227, 525 LBS./SEC. (BASE FLOW) TURBOJET

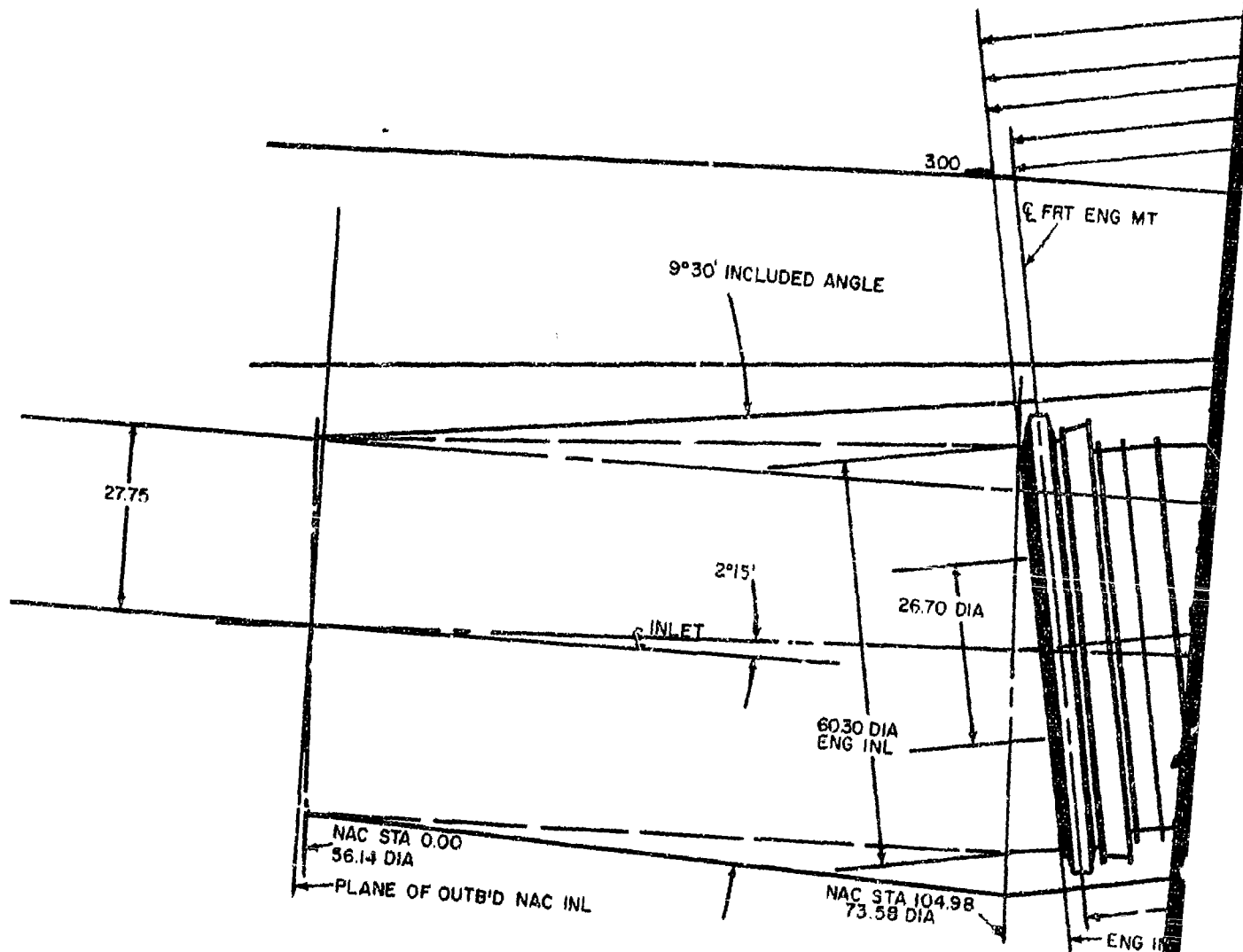
Figure 1-25

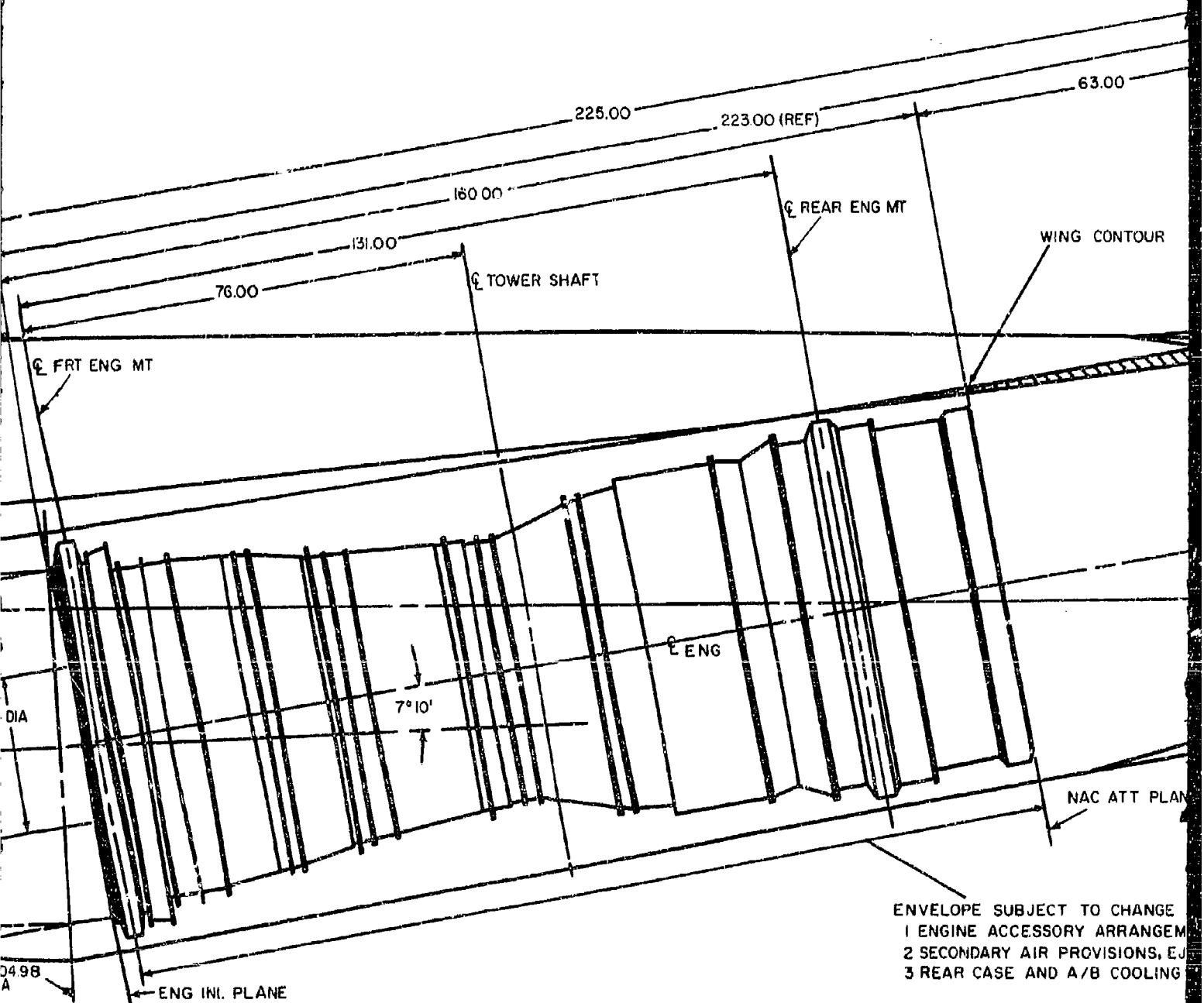
CONFIDENTIAL

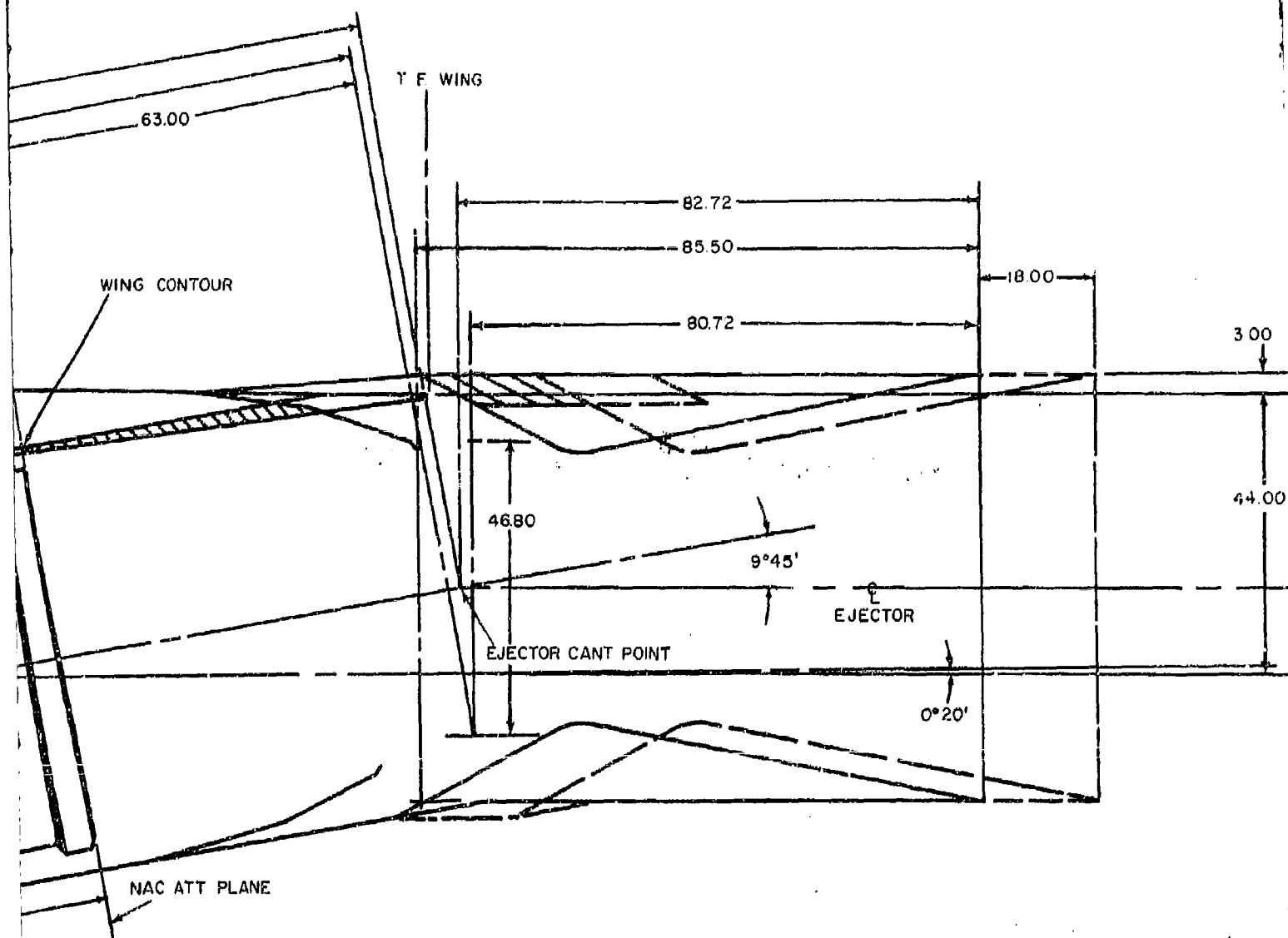
DECLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DDO ON 9900-10
THIS DOCUMENT CONTAINS INFORMATION THAT IS THE
PROPERTY OF THE UNITED STATES GOVERNMENT
AND IS NOT TO BE DISTRIBUTED OUTSIDE THE
DEFENSE DEPARTMENT WITHOUT THE
APPROPRIATE AUTHORITY

4

PRATT & WHITNEY AIRCRAFT



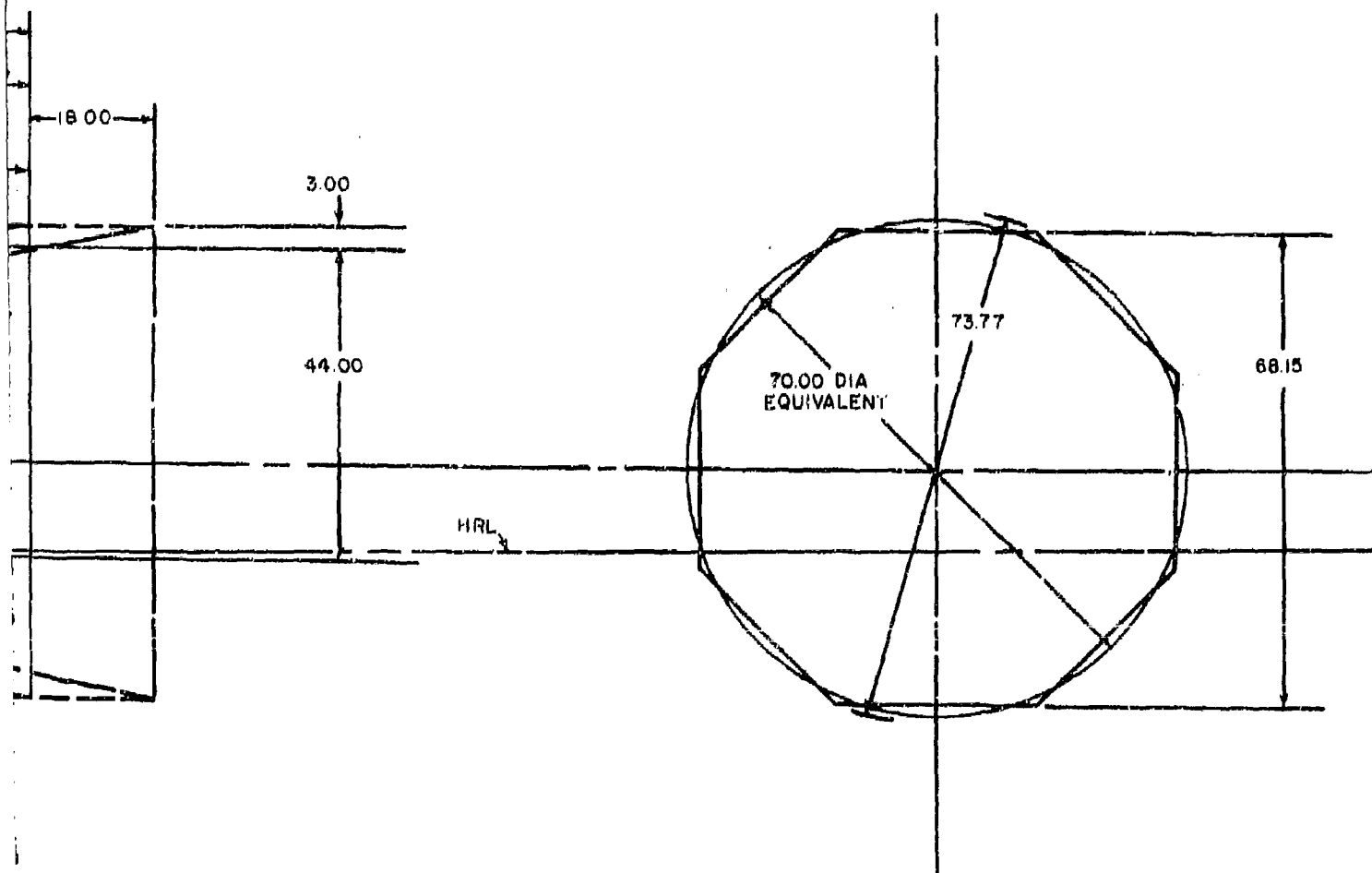




PE SUBJECT TO CHANGE
 IE ACCESSORY ARRANGEMENT
 IDARY AIR PROVISIONS, EJECTOR
 CASE AND A/B COOLING PROVISIONS

CONFIDENTIAL

PWA-2600



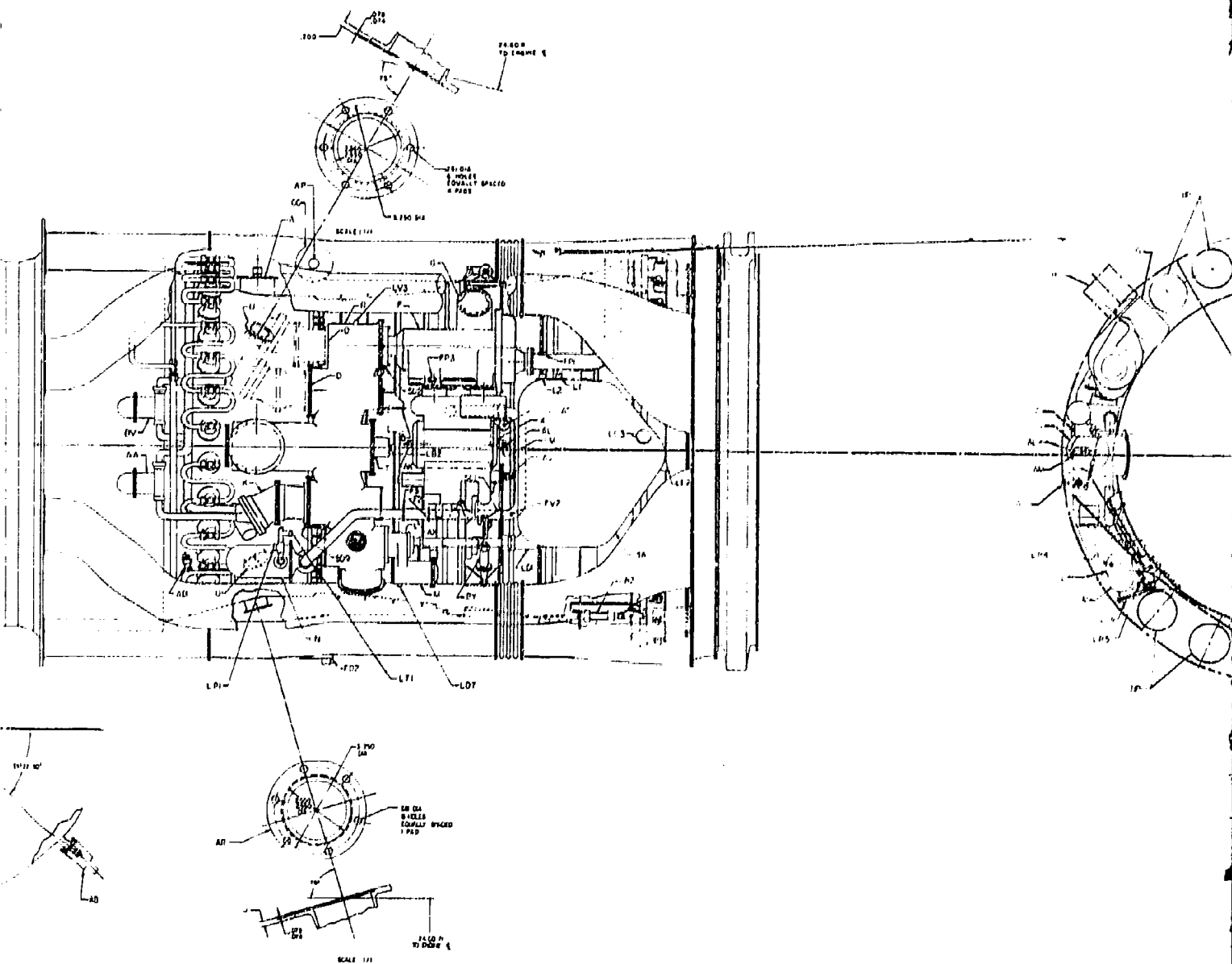
STJ227, 525 LBS./SEC. (LOW FLOW) TURBOJET

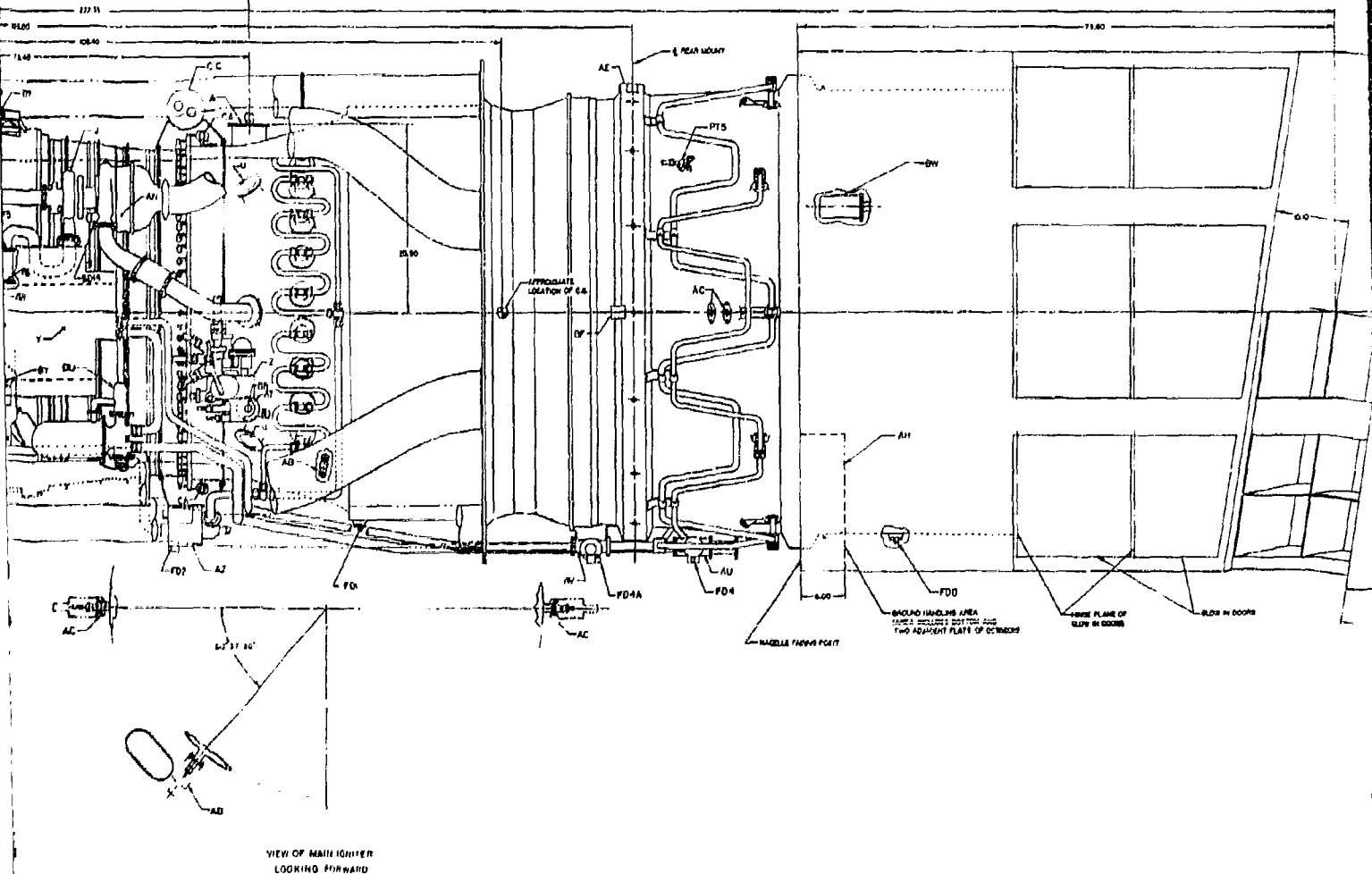
Figure 1-26

CONFIDENTIAL

4

REPRODUCED AT 1:1 SCALE
FROM PWA-2600
1960-01-01



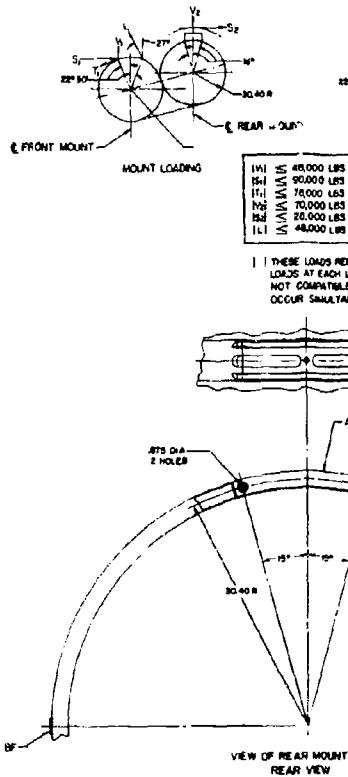


CONFIDENTIAL

TORQUE & MOMENT LIMITATION TABLE-ACCESS DRIVES							
DRIVE PNO	NOMINAL JSE	TORQUE (L.B.F.)			SPEED RPM	ROTATION (FACING DRIVE PNO)	+ G OVERHUNG (L.B.F.) + L.D.P.
		WORTHLOAD	OVERLOAD	STATIC			
A	2534	3500	2400	2.84	CCW		
B	7	50	762	6.64	CCW		
C	HYDRA-MAT	1470	4750	980	CCW		

* MAY ALLOWABLE COUNTER-TORQUE VALUES ARE AT MAXIMUM SPEED VALUES
OVERHUNG DRIVEN MEMBER IS NOT DETAILED FOR DEFLECTIVE FORCES RESULTING FROM LOADS IN TORSIONAL
MODE
MAY ALLOWABLE OVERHUNG MEMBER MOMENTS OF ACCORDANCE WITH LOADS IN TORSION
ARE LESSER PROVIDED NO DEFLECTIVE FORCES RESULTING FROM ACCIDENTAL WEARION LOADS

* MAY ALLOWABLE FOR 5 INCHES CLAMP ON WEARION AT 4 AND 10 INCHES
AT 4 INCHES CLAMP ON WEARION TO ACCORDANCE WITH LOADS IN TORSION
TO 1/2 INCH 1/2 INCH CLAMP ON WEARION SMALL PNO AT A STATIC TORQUE
TO 1/2 INCH 1/2 INCH CLAMP AT 4 INCHES CLAMP ON WEARION
TO 1/2 INCH 1/2 INCH CLAMP ON WEARION



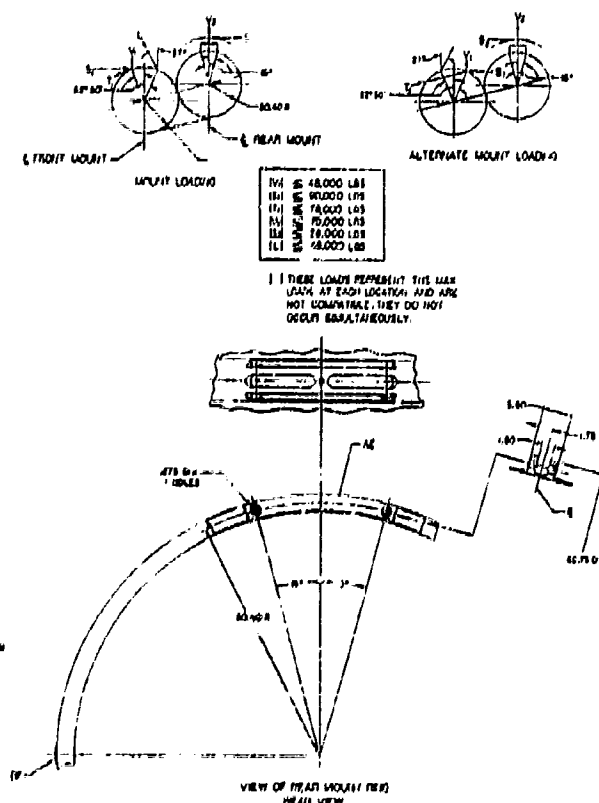
PROPOSED TURBOJET ACCESSORY DRIVE

Figure 1-27

CONFIDENTIAL

TURBOJET ACCESSORY MOUNTING DATA						
MOUNT	MOUNTING	TORQUE (LB IN)			SPEED (RPM)	ROTATION (TORQUE SHAPE NOT)
		CONTINUOUS	INTERMITTENT	STATIC		
A	FLANGE	1215	3600	4800	3000	CCW
B	FLANGE	1215	3600	4800	3000	CCW
C	FLANGE	1215	3600	4800	3000	CCW

1. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE TORQUES. 2. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE SPEEDS. 3. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE ROTATIONS. 4. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE VIBRATIONS. 5. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE SHOCKS. 6. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE TEMPERATURES. 7. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE HUMIDITIES. 8. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE SALINITIES. 9. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE POLLUTANTS. 10. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE WEATHER. 11. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE NOISE. 12. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE RADIATION. 13. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE ELECTROMAGNETIC INTERFERENCE. 14. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE ESD. 15. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE STATIC DISCHARGE. 16. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE SURFACE CONTAMINATION. 17. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE CORROSION. 18. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE FATIGUE. 19. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE CRACKS. 20. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE DEFORMATION. 21. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE DIMENSIONAL STABILITY. 22. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE WEIGHT. 23. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE BALANCE. 24. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE ALIGNMENT. 25. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE CLEARANCE. 26. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE PROTECTION. 27. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE MAINTENANCE. 28. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE REPAIR. 29. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE REPLACEMENT. 30. ALL MOUNTS SHALL BE DESIGNED TO WITHSTAND THE ABOVE DISPOSAL.



(A)	48,000 LBS
(B)	60,000 LBS
(C)	70,000 LBS
(D)	70,000 LBS
(E)	70,000 LBS
(F)	70,000 LBS
(G)	70,000 LBS
(H)	70,000 LBS
(I)	70,000 LBS

THESE LOADS REPRESENT THE MAX LOAD AT EACH LOCATION AND ARE NOT COMPARABLE. THEY DO NOT OCCUR SIMULTANEOUSLY.

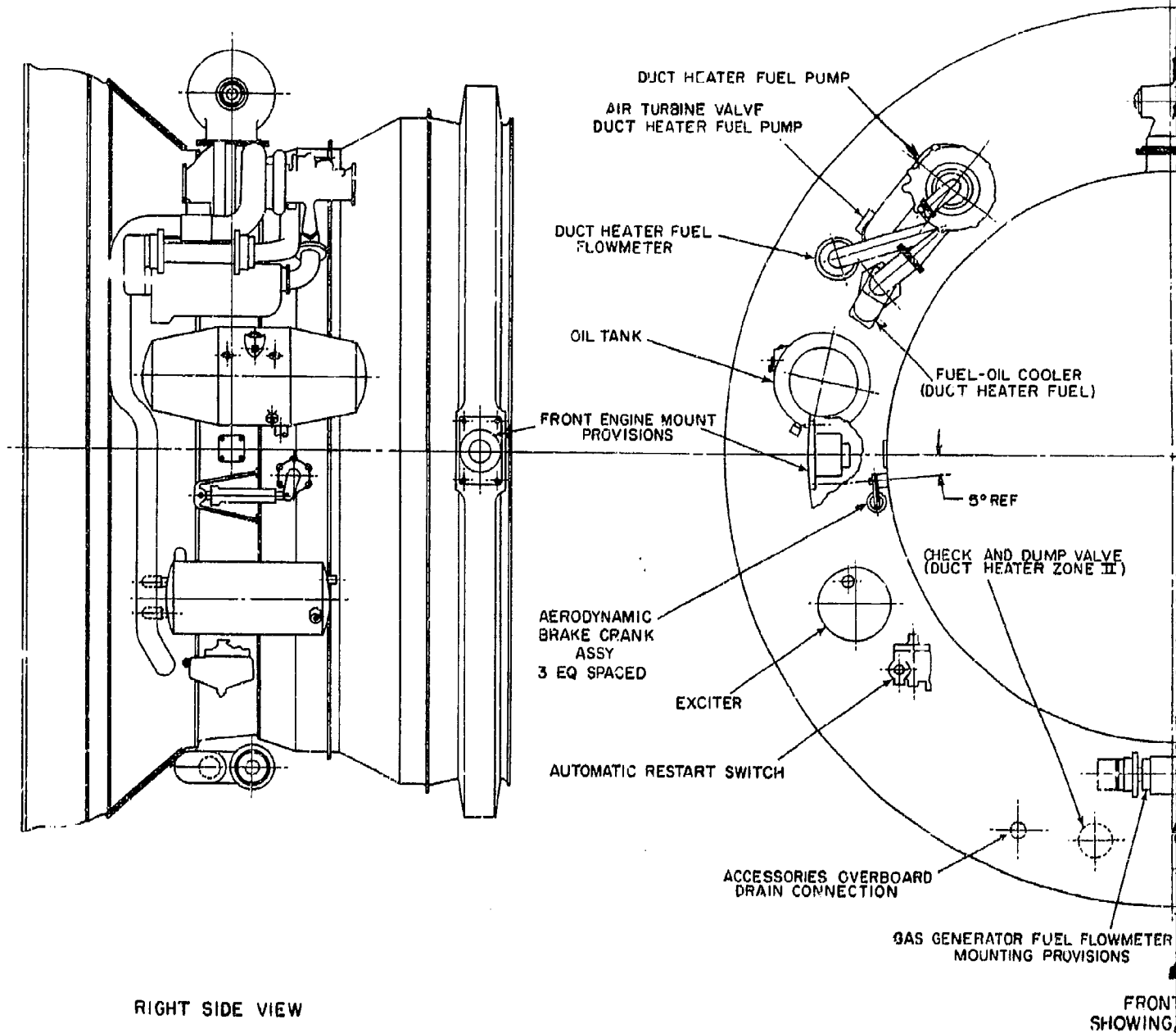
6

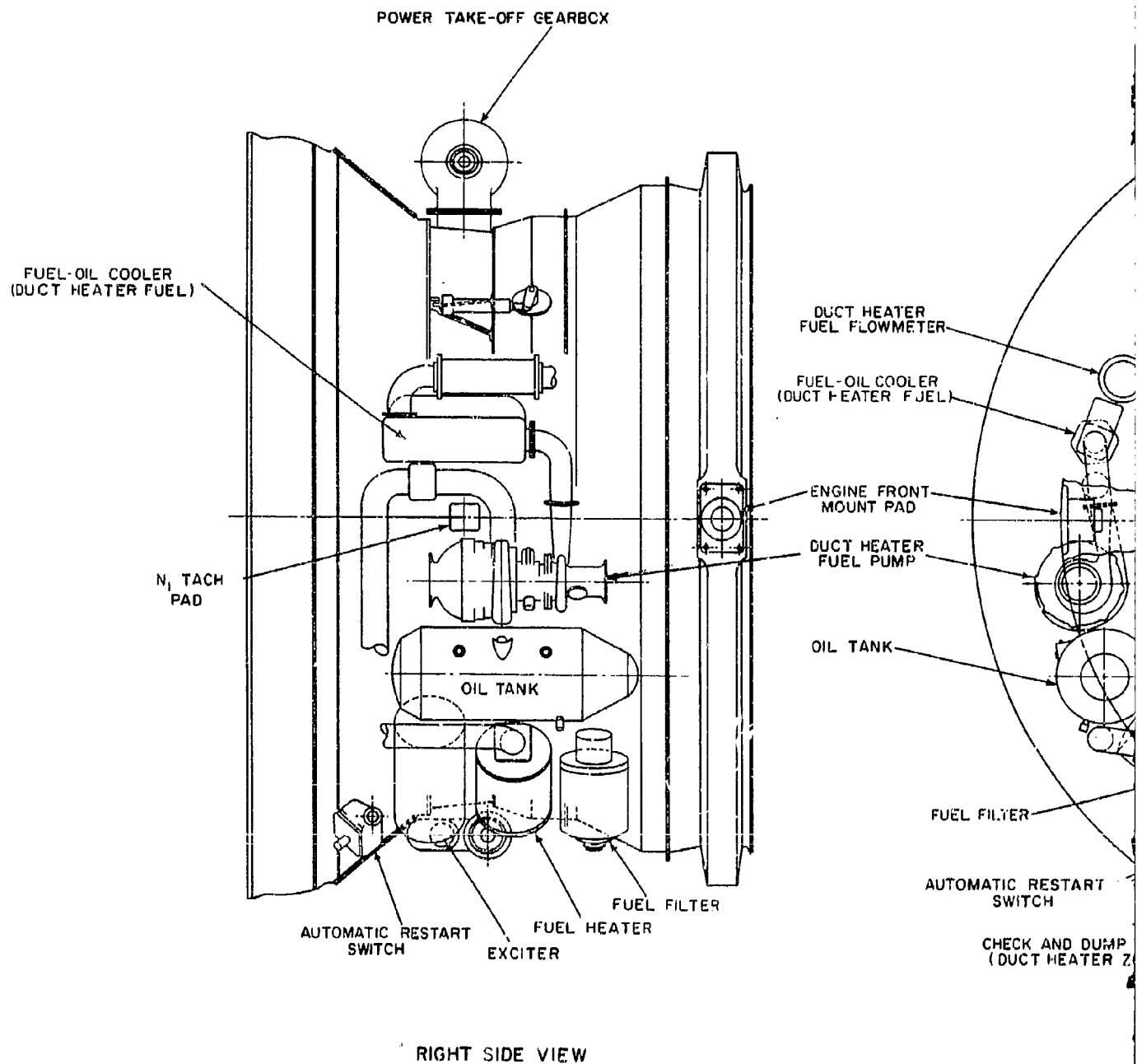
PRELIMINARY INSTALLATION AND PRINTED 6/8/60

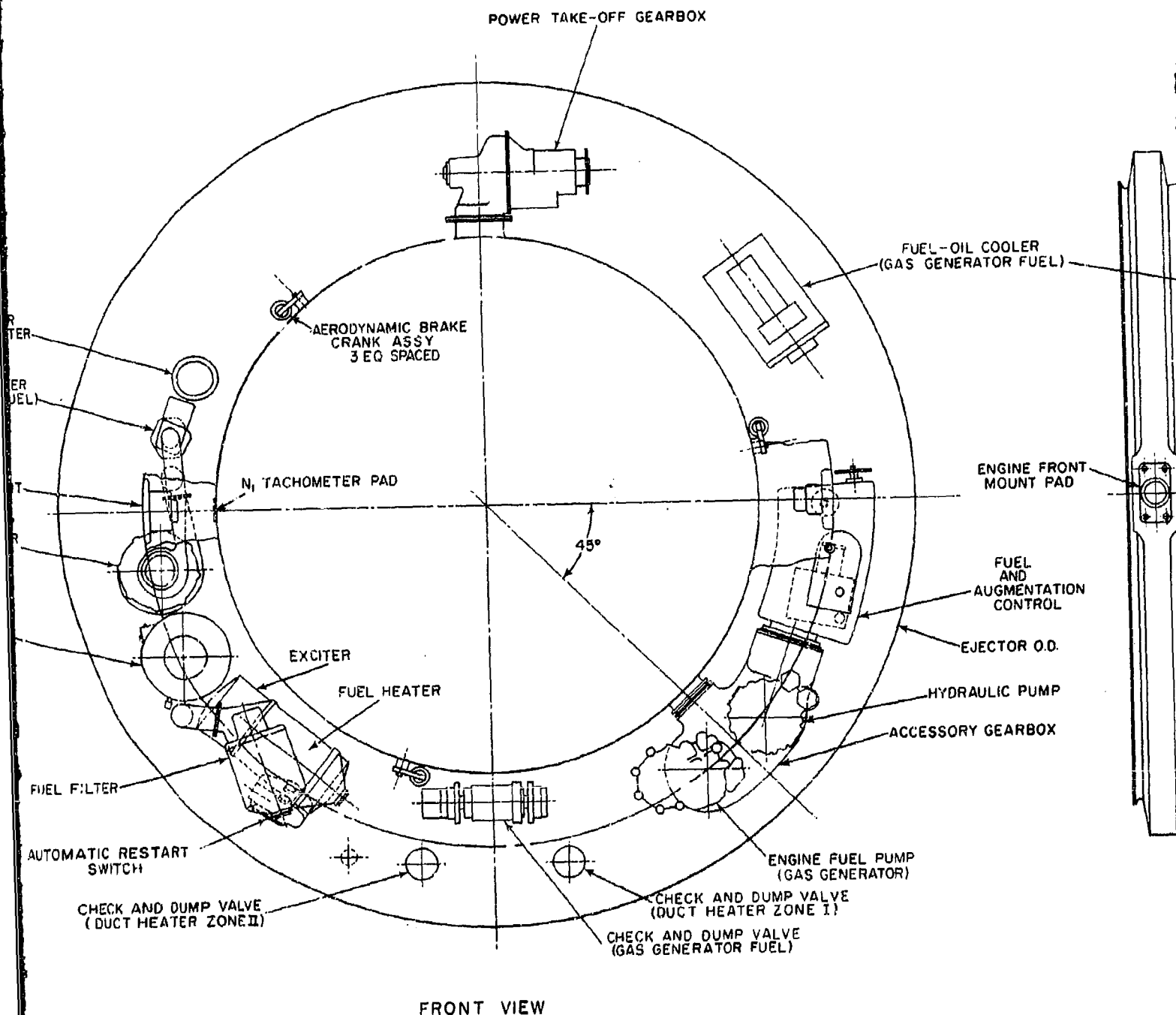
PROPOSED TURBOJET ACCESSORY ARRANGEMENT

Figure 1-27

INFORMATION IS A TECHNICAL REPORT
 PREPARED BY THE AIR FORCE
 UNDER CONTRACT NO. AF33(616)-2-6000
 WORK ELEMENT NO. 1-27
 THIS REPORT IS THE PROPERTY OF THE AIR FORCE
 AND IS LOANED TO YOUR ORGANIZATION.
 IT AND ITS CONTENTS ARE NOT TO BE
 DISTRIBUTED OUTSIDE YOUR ORGANIZATION.
 THE AIR FORCE WILL BE RESPONSIBLE FOR
 THE PROTECTION OF THIS INFORMATION.
 THE AIR FORCE WILL BE RESPONSIBLE FOR
 THE PROTECTION OF THIS INFORMATION.

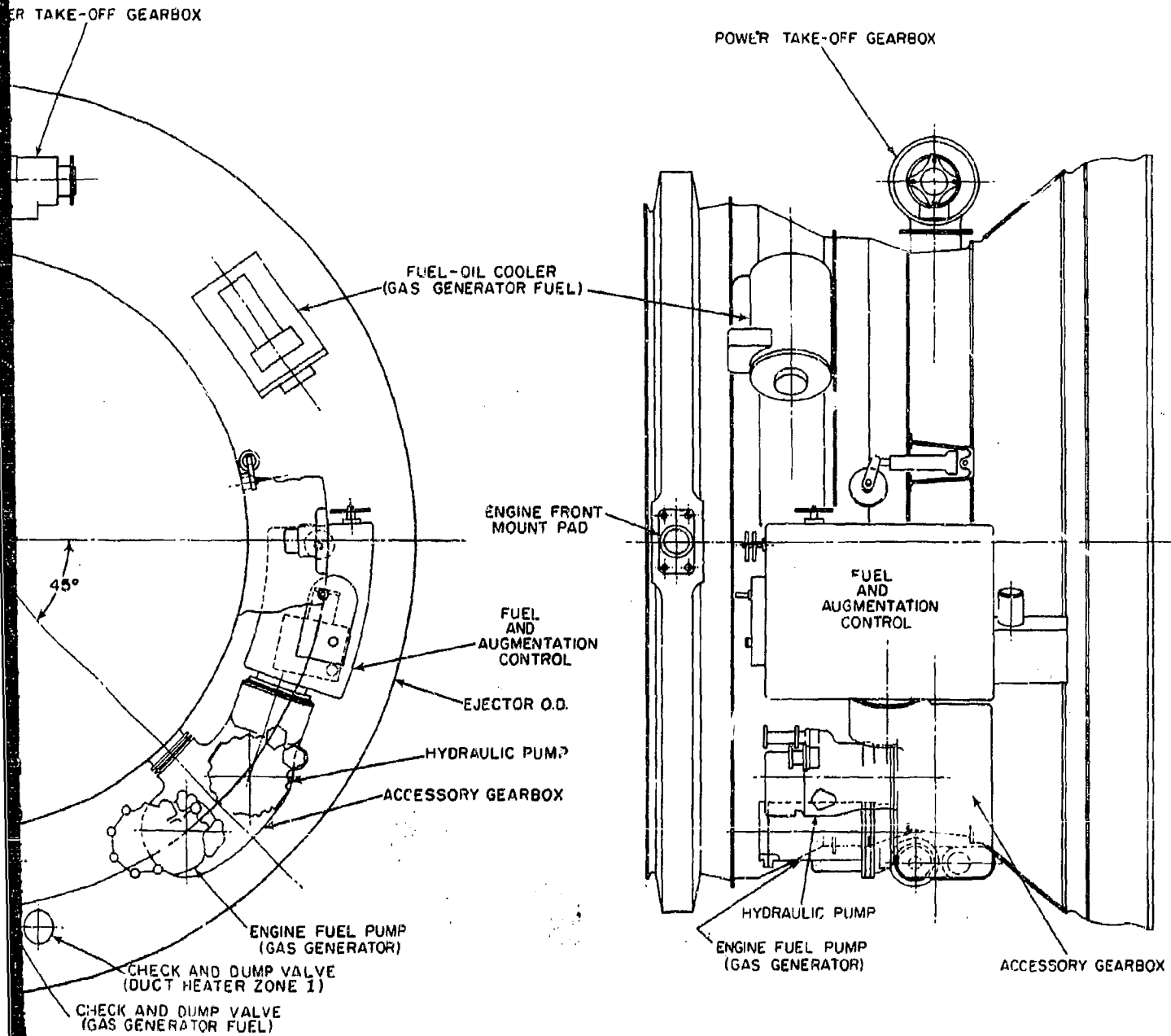






REVISED ACCESSORY ARRANGEMENT
MOUNT STRUCTURE AT

Figure 1-20



LEFT SIDE VIEW

REVISED ACCESSORY ARRANGEMENT MAKING ROOM FOR
MOUNT STRUCTURE AT TOP OF ENGINE

Figure 1-29

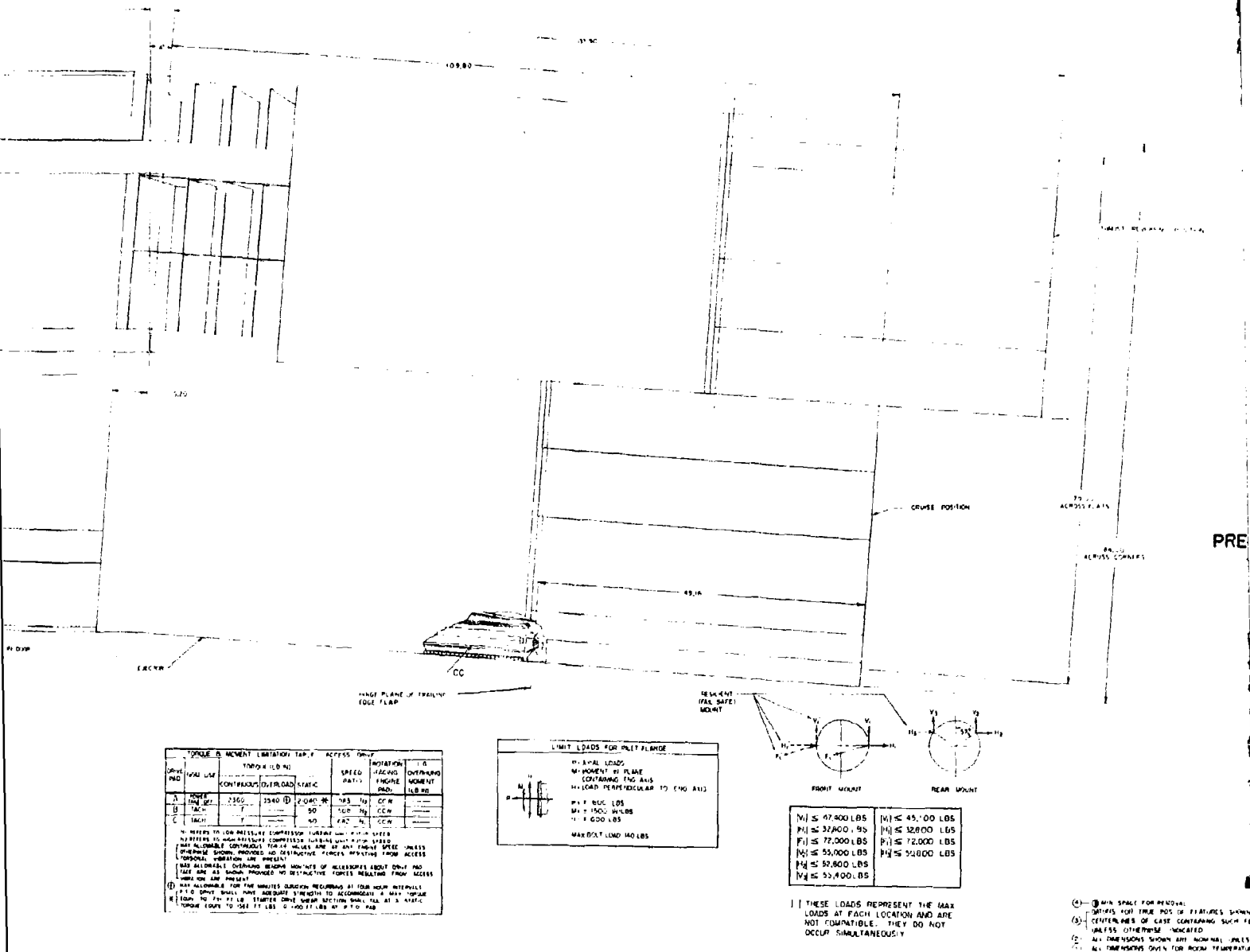
EXAMINATED AT 5 YEAR INTERVALS
REPLACEMENT AFTER 10 YEARS
1000 (M 8840) 10

THIS DOCUMENT CONTAINS UNCLASSIFIED INFORMATION
EXCEPT WHERE SHOWN OTHERWISE. IT IS THE POLICY
OF THE AIR FORCE TO MAKE AVAILABLE TO THE PUBLIC
ALL INFORMATION IN THIS DOCUMENT UNLESS IT IS
DETERMINED THAT DISCLOSURE WOULD BE PREJUDICIAL
TO THE NATIONAL DEFENSE.

3

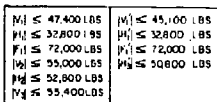
[illegible]





PROPOSED TURBOFAN ACCESSORY ARRANGEMENT

Figure 1-30



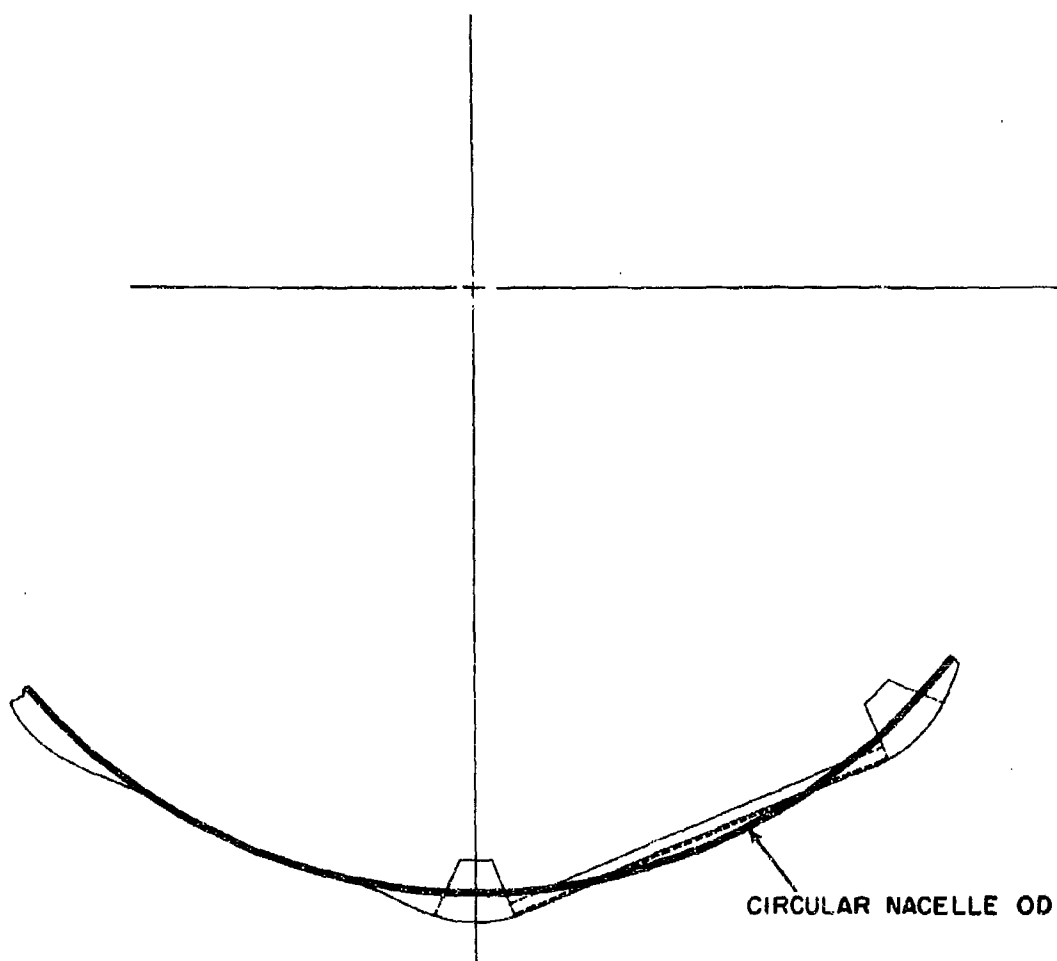
- ④ ① MIN SPACE FOR REMOVAL
- ⑤ DIMENSIONS FOR TRUE POS OF FEATURES SHOWN ARE CENTERLINES OF CASE CONTAINING SUCH FEATURES UNLESS OTHERWISE INDICATED
- ⑥ ALL DIMENSIONS SHOWN ARE NOMINAL UNLESS OTHERWISE SPECIFIED
- ⑦ ALL DIMENSIONS GIVEN FOR ROOM TEMPERATURE UNLESS OTHERWISE NOTED

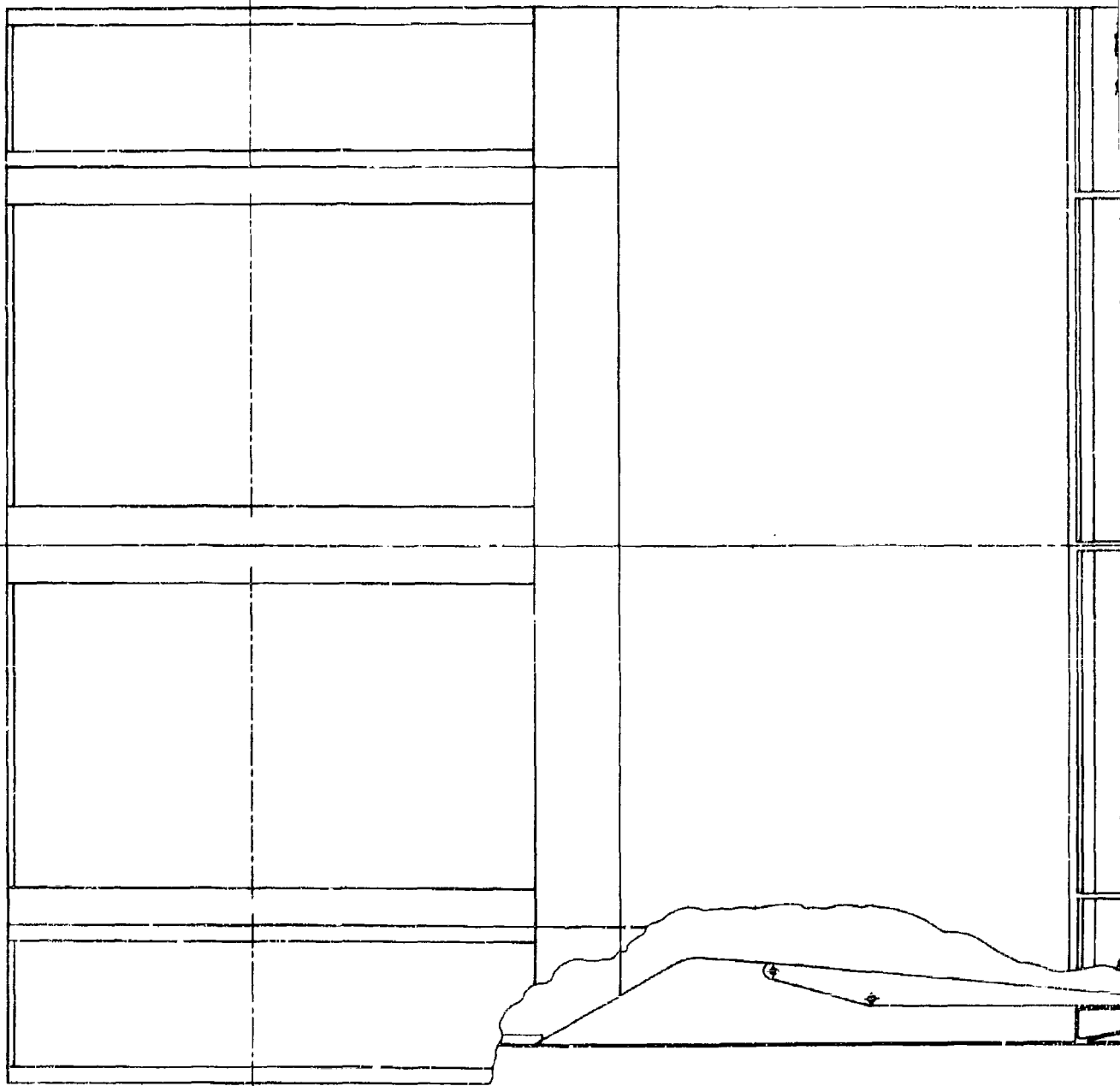
Figure 1-30

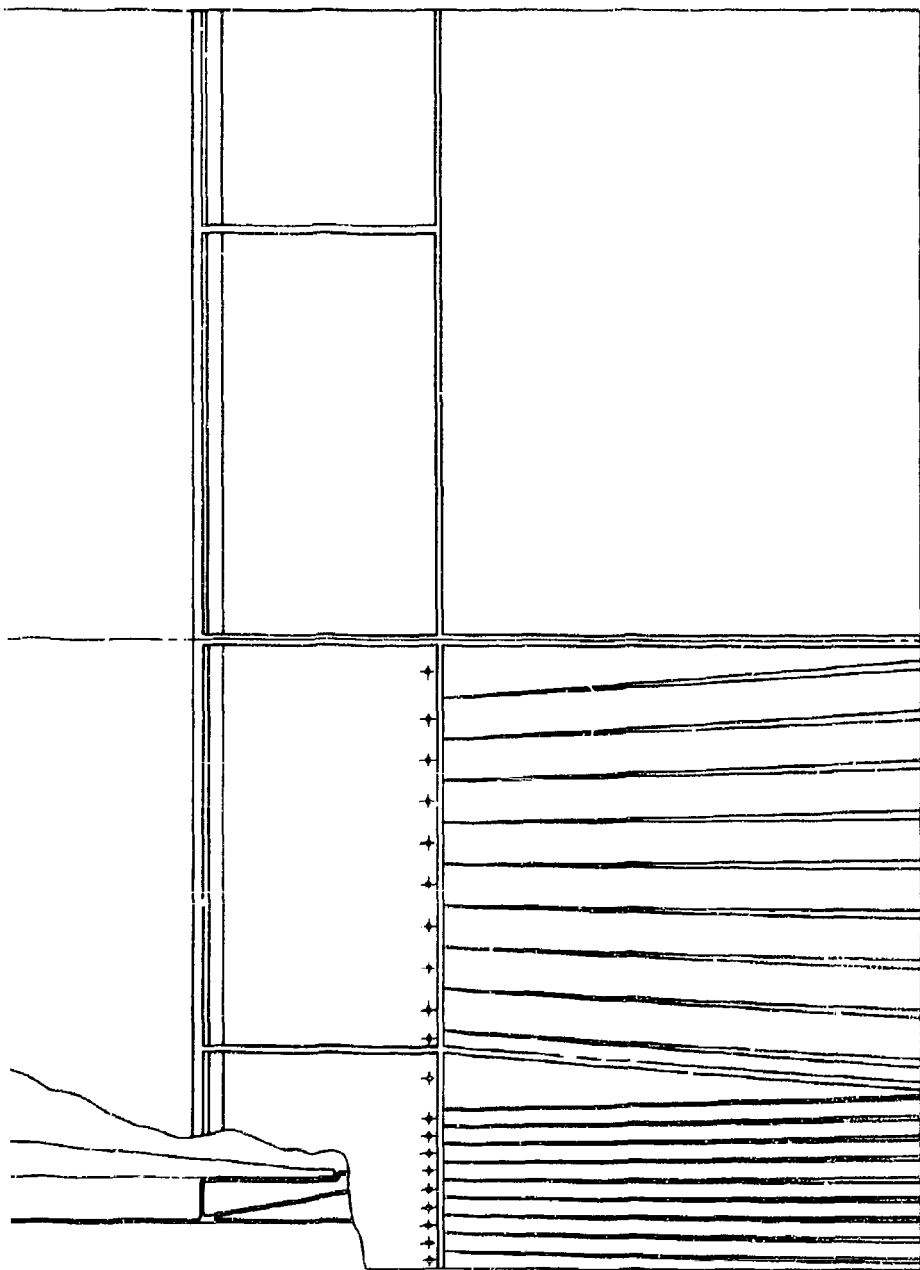
DOWNLOADED AT 3 YEAR INTERVAL
DECLASSIFIED APRIL 17 1998
DOD OF 520010

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS TITLE 18 U.S.C. SECTION 793 AND FOR THE TRANSMISSION OR REVEALATION OF ITS CONTENTS IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

CONFIDENTIAL

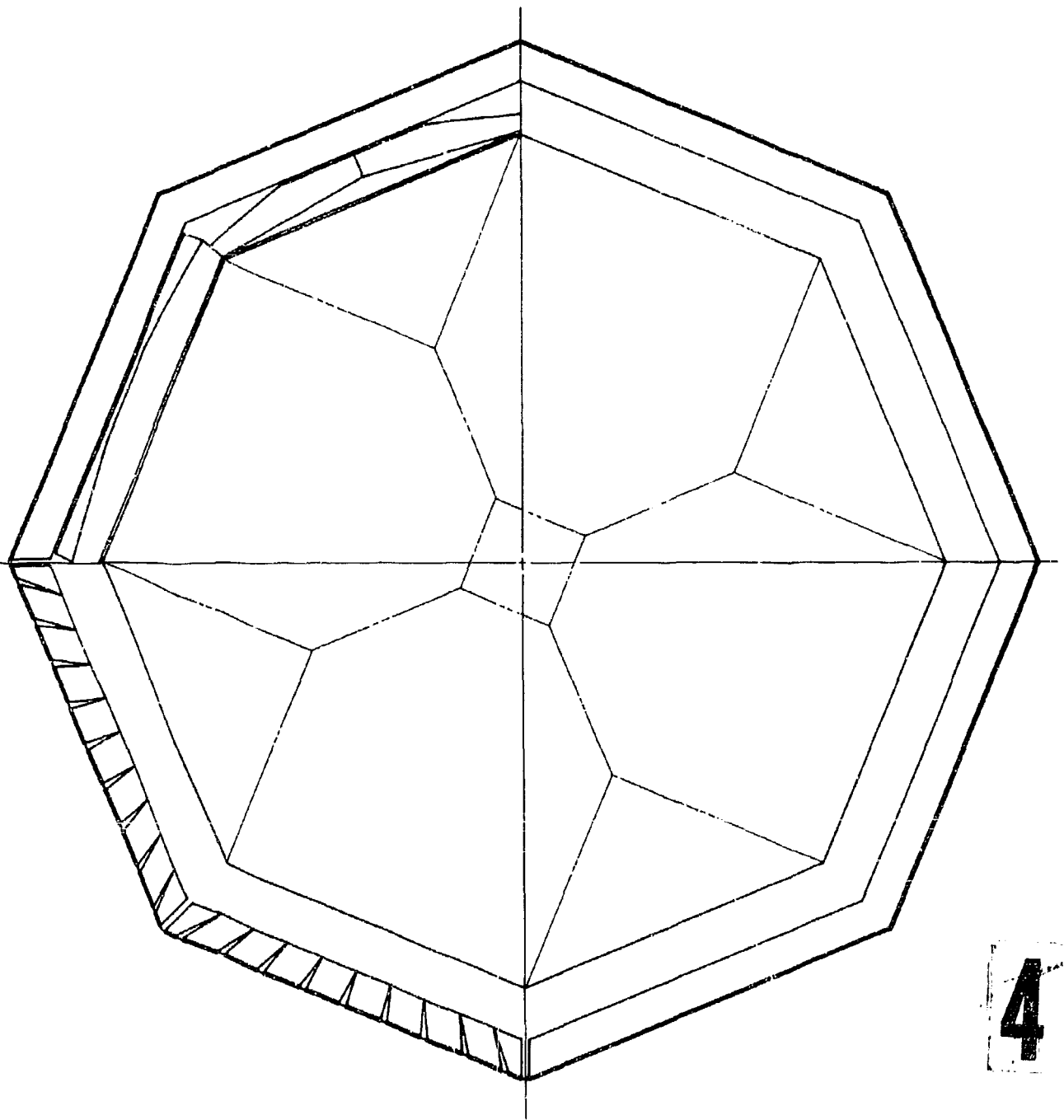






CONFIDENTIAL

PWA-2600



SCHEMATIC OF OCTAGONAL EJECTOR

Figure 1-31

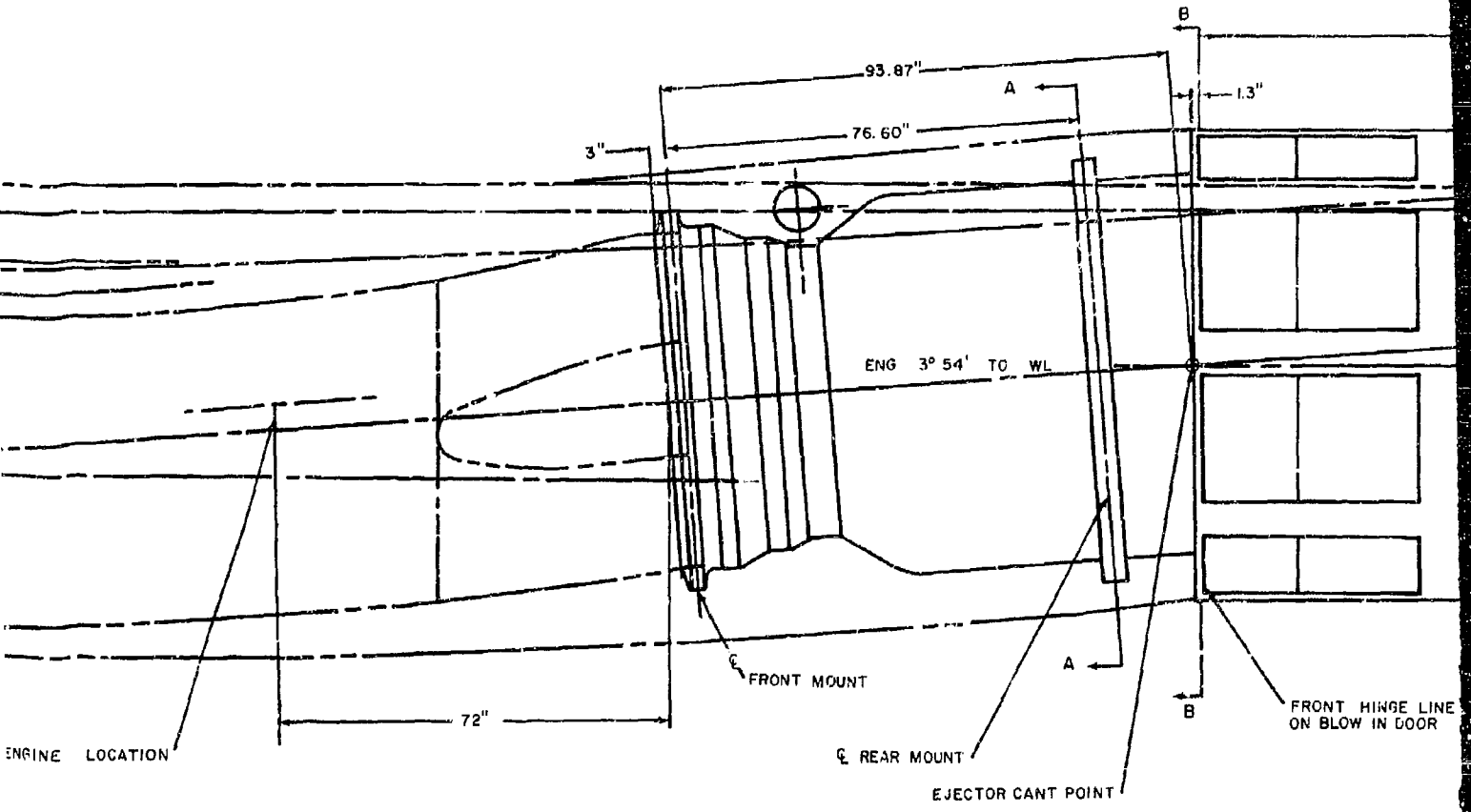
CONFIDENTIAL

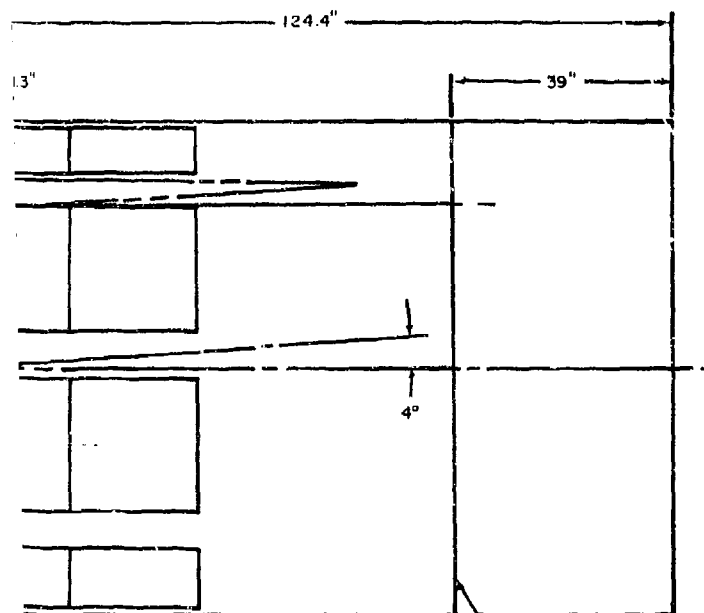
UNCLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS
DATE: 04/09/00
THIS DOCUMENT CONTAINS INFORMATION THAT MAY BE
EXCLUDED FROM THE UNCLASSIFIED STATUS OF THE
DOCUMENT. IT IS THE POLICY OF THE NATIONAL ARCHIVES
TO REVIEW AND, WHERE APPROPRIATE, DECLASSIFY
AND/OR DOWNGRADE INFORMATION IN THIS DOCUMENT
ON A REGULAR BASIS.

PRATT & WHITNEY AIRCRAFT

WRP

PRESENT ENGINE LOCATION

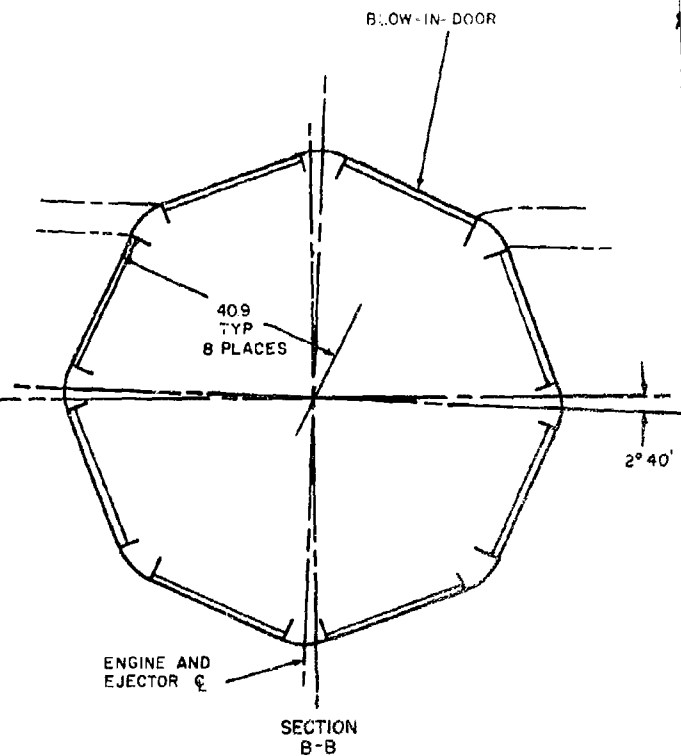




FRONT HINGE LINE
ON BLOW IN DOOR

FRONT HINGE LINE
ON TAIL FEATHERS

EJECTOR CANTED AT
REAR MOUNT PLANE

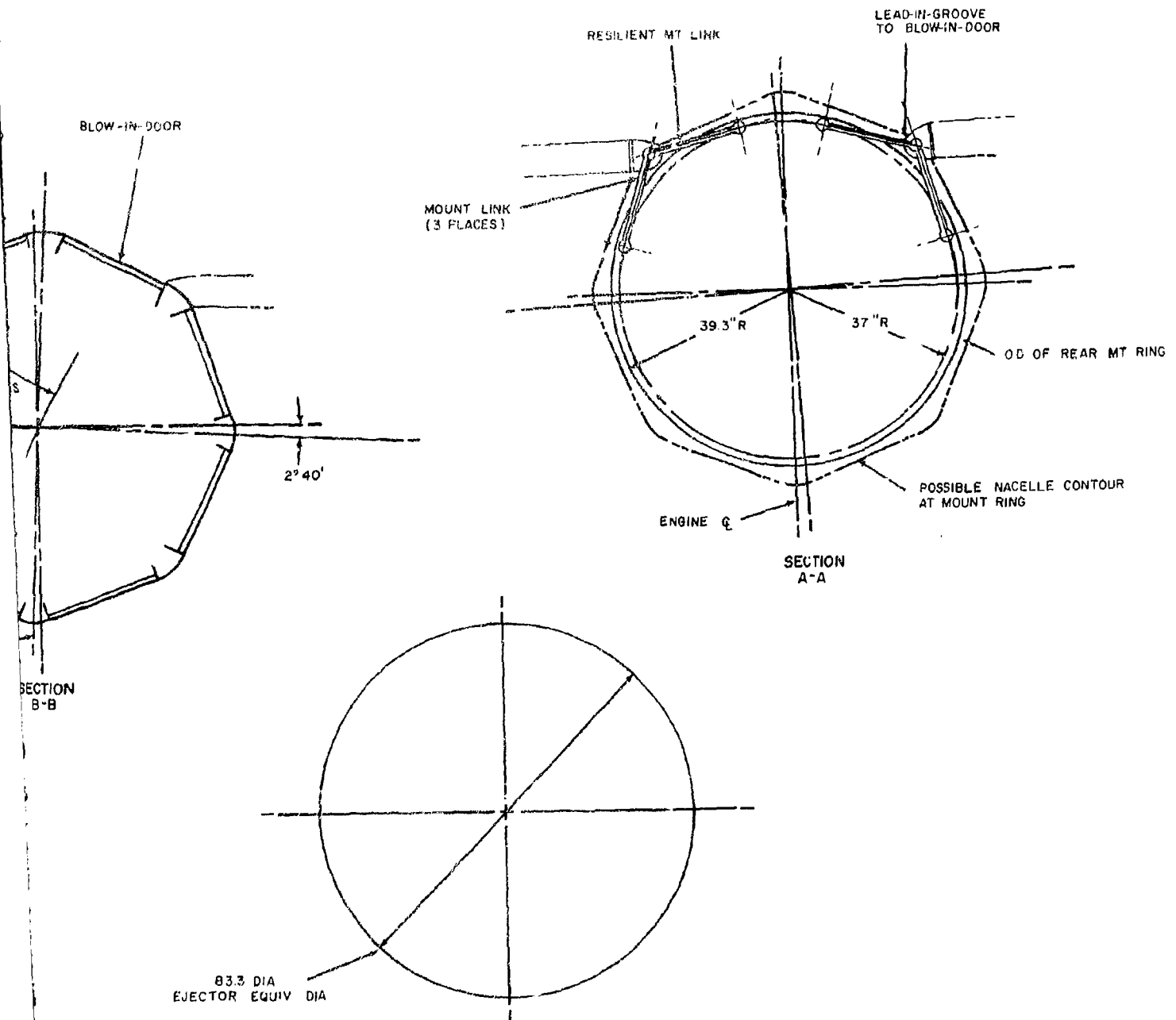


83.3 DIA
EJECTOR EQUIV DIA

3

CONFIDENTIAL

PWA-2600



STF219L 700 LB/SEC TURBOFAN OUTBOARD ENGINE

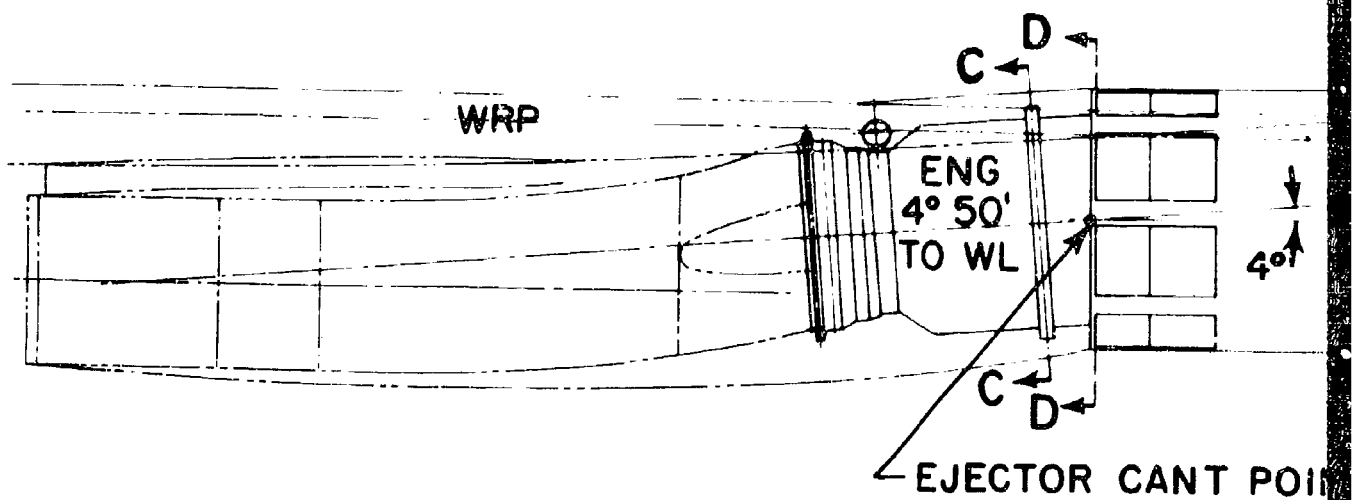
Figure 1-32

CONFIDENTIAL

DO NOT REPRODUCE OR DISSEMINATE THIS INFORMATION
EXCEPT BY AUTHORITY OF THE SECRETARY OF DEFENSE

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
DEFENSE OF THE UNITED STATES OF AMERICA
AND IS THE PROPERTY OF THE UNITED STATES OF AMERICA
IT IS LOANED TO YOUR AGENCY AND IS NOT TO BE
REPRODUCED OR DISSEMINATED IN ANY MANNER
WITHOUT THE WRITTEN PERMISSION OF THE
SECRETARY OF DEFENSE

4



CONFIDENTIAL

PWA-2600

LINE UP BLOW-IN-DOORS
WITH WING LIKE THIS

SECTION C-C

ENGINE AND
EJECTOR ϕ

SECTION D-D

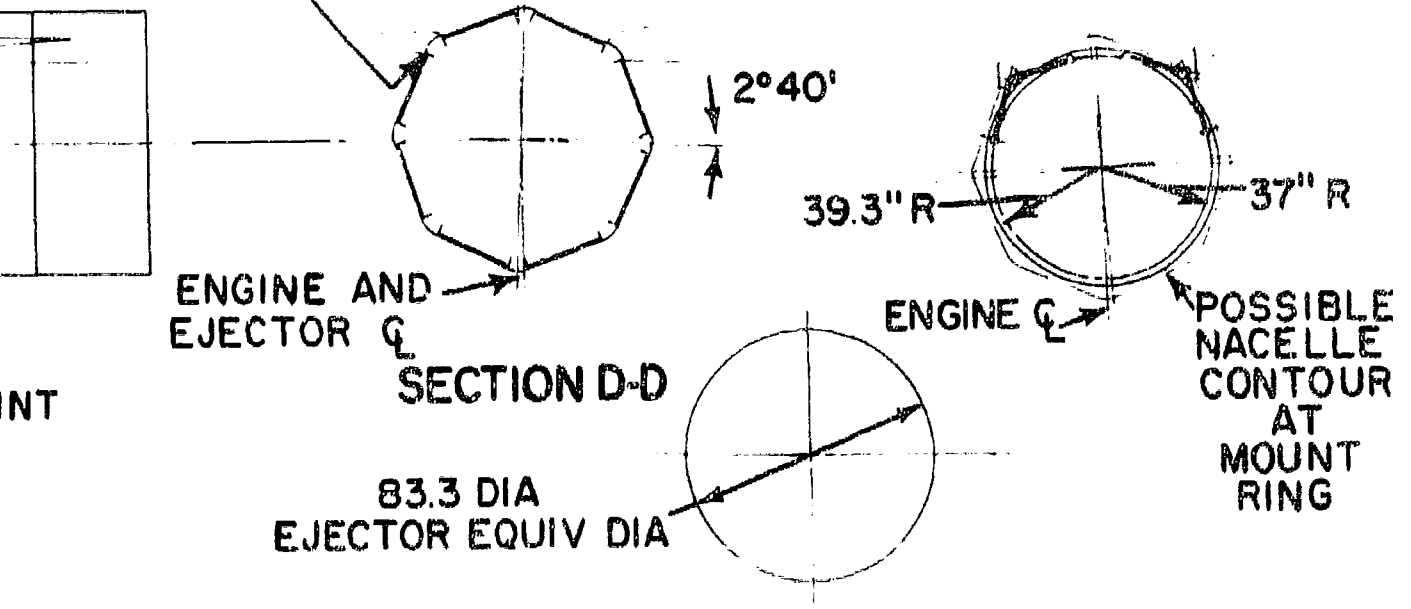
83.3 DIA
EJECTOR EQUIV DIA

39.3" R

37" R

ENGINE ϕ

POSSIBLE
NACELLE
CONTOUR
AT
MOUNT
RING



STF2191, 700 L.B/SEC TURBOFAN INBOARD ENGINE

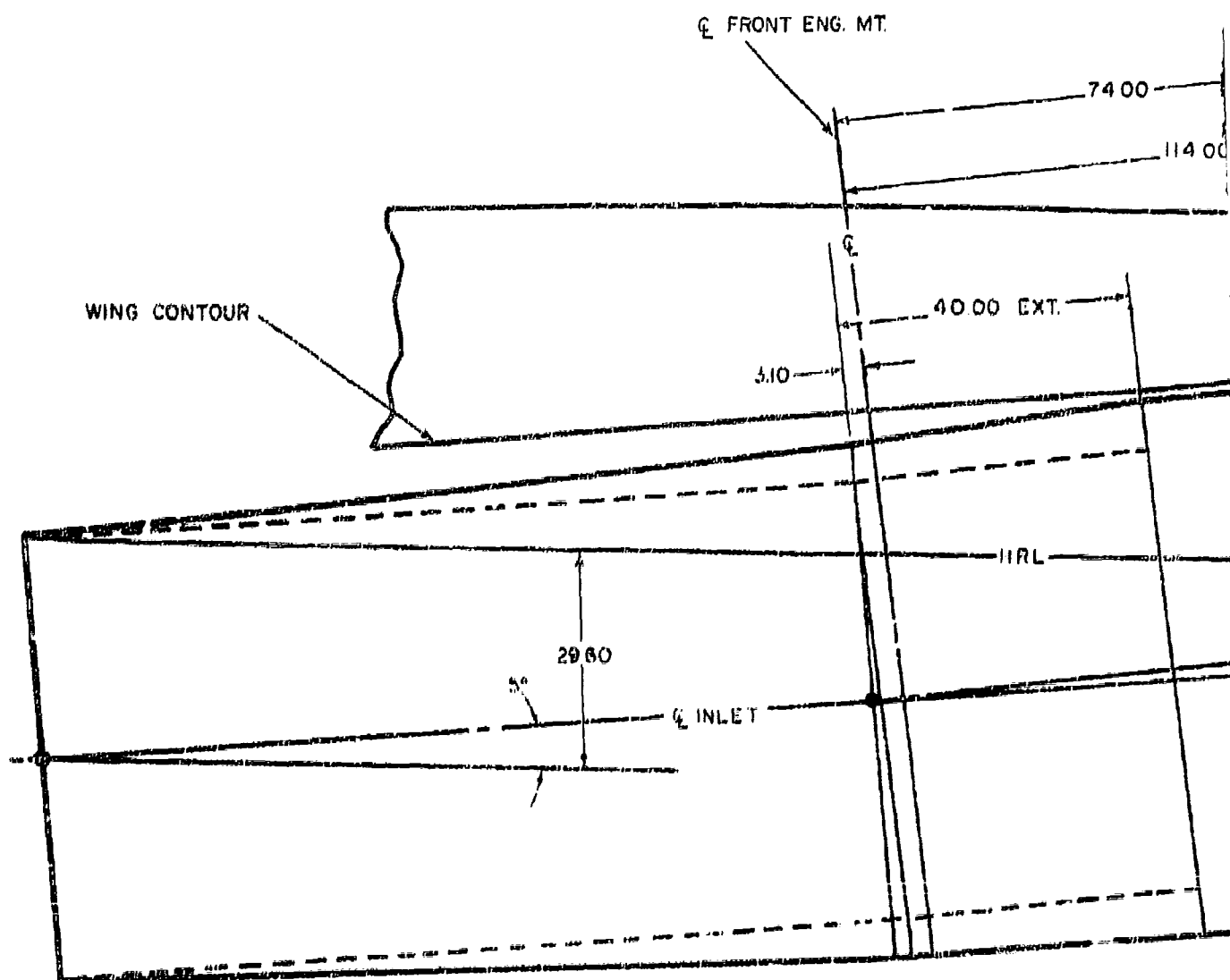
Figure 1-33

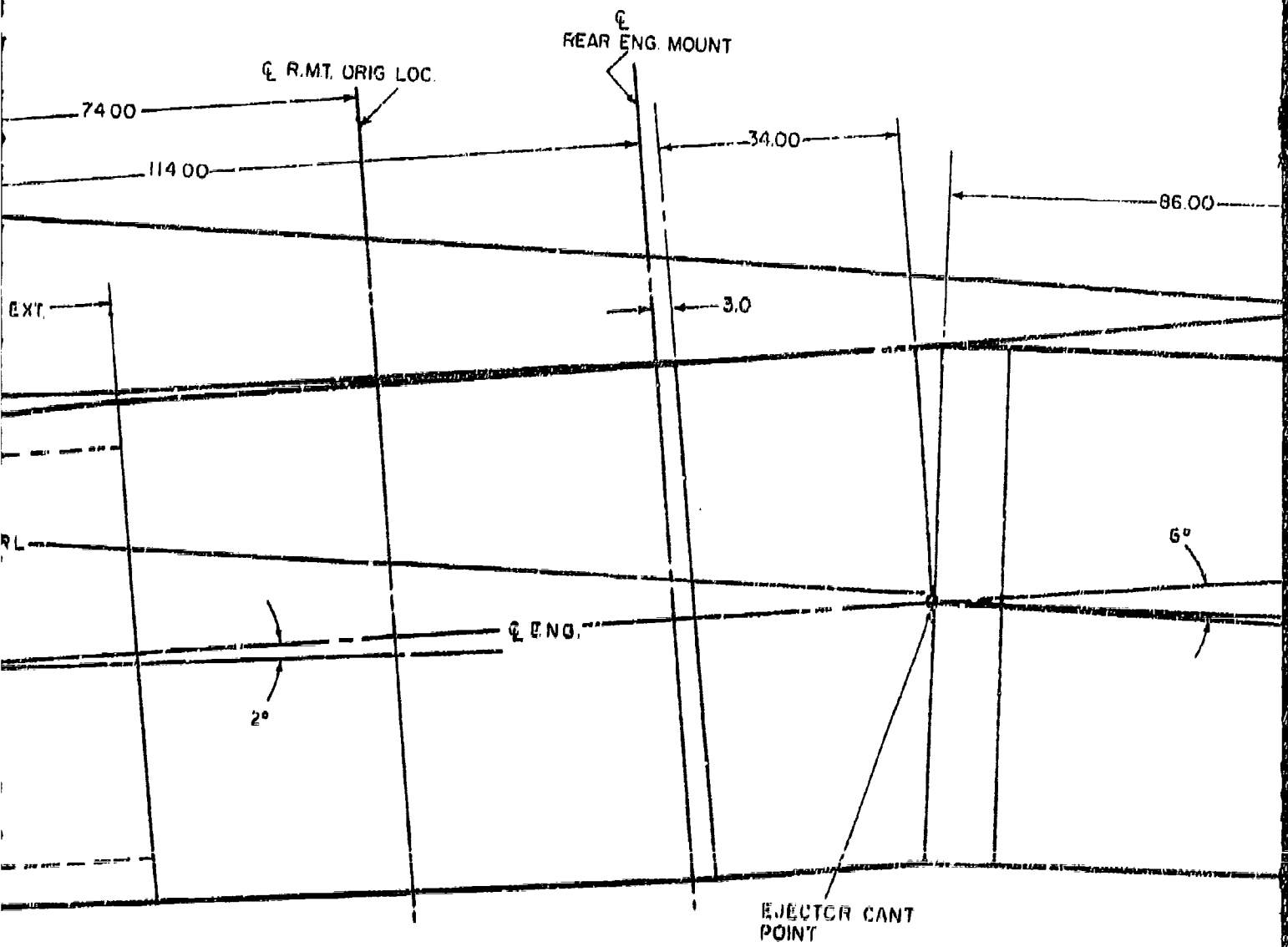
CONFIDENTIAL

2

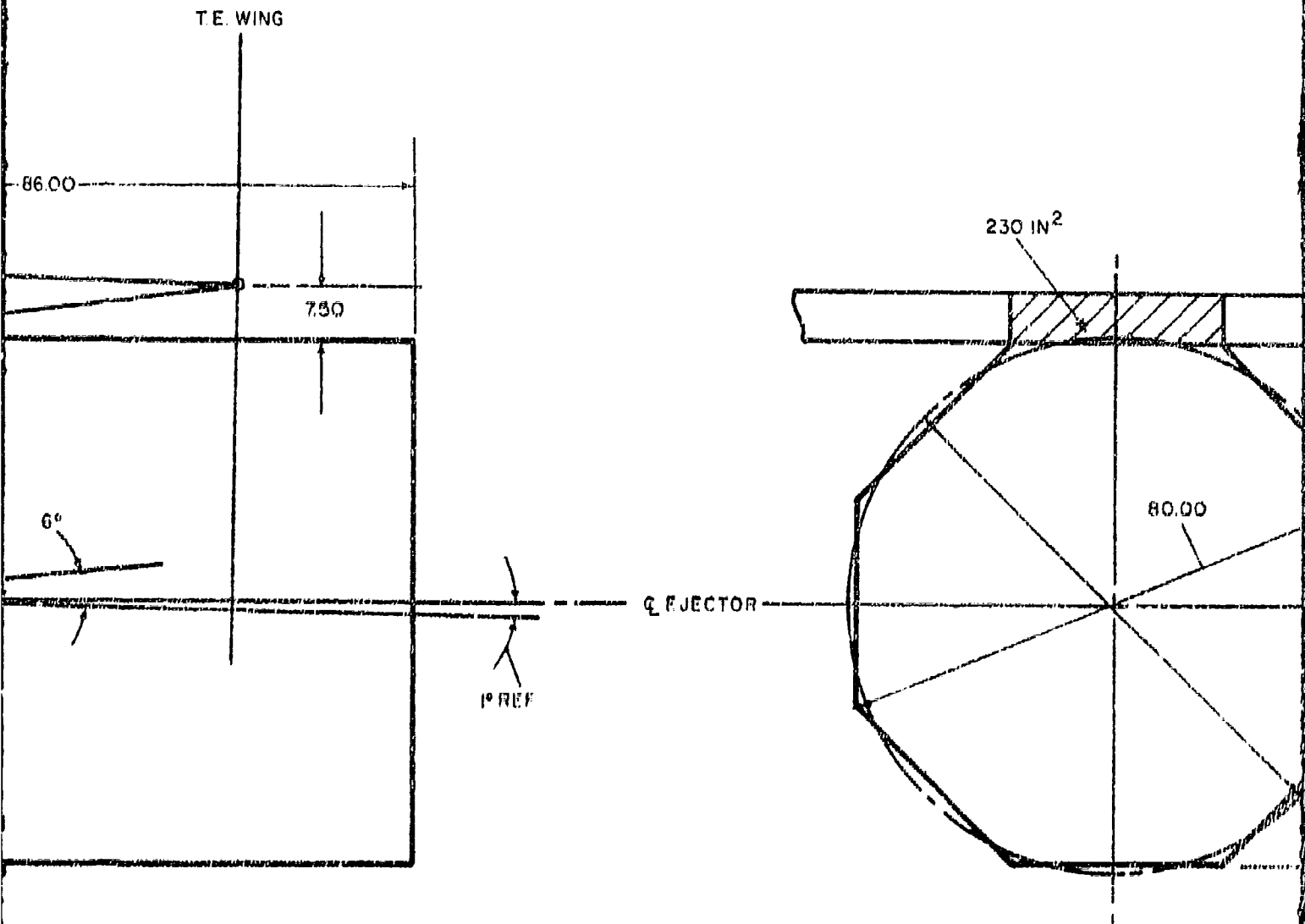
REPRODUCTION BY U.S. AIR FORCE
1.1. JANUARY 1971
COPY OF 100-0

PRATT & WHITNEY AIRCRAFT





CONFIDENTIAL



ARRANGEMENT SHOWING REAR ENGL
MOVED AFT 40.00
ENGINE & EJECTOR ARE TANGENT TO
EJECTOR CANTED AT NOZZLE PLANE
ENGL MOUNT PLANES ARE PARALLEL
BASE DRAG AREA 230 in²

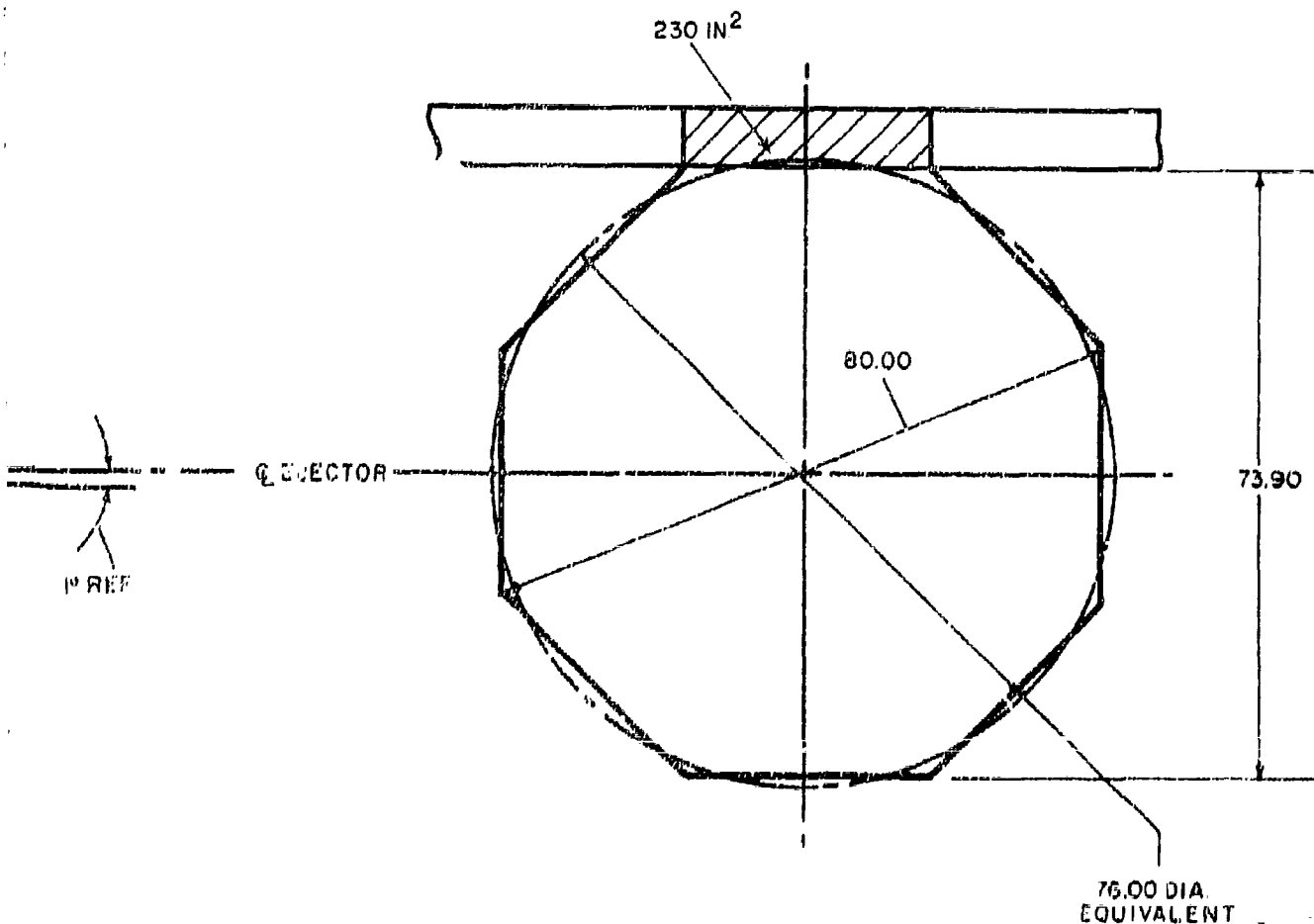
STP219L 650 LB/SEC TURBOFAN

Figure 1-14

CONFIDENTIAL

CONFIDENTIAL

PWA-2600



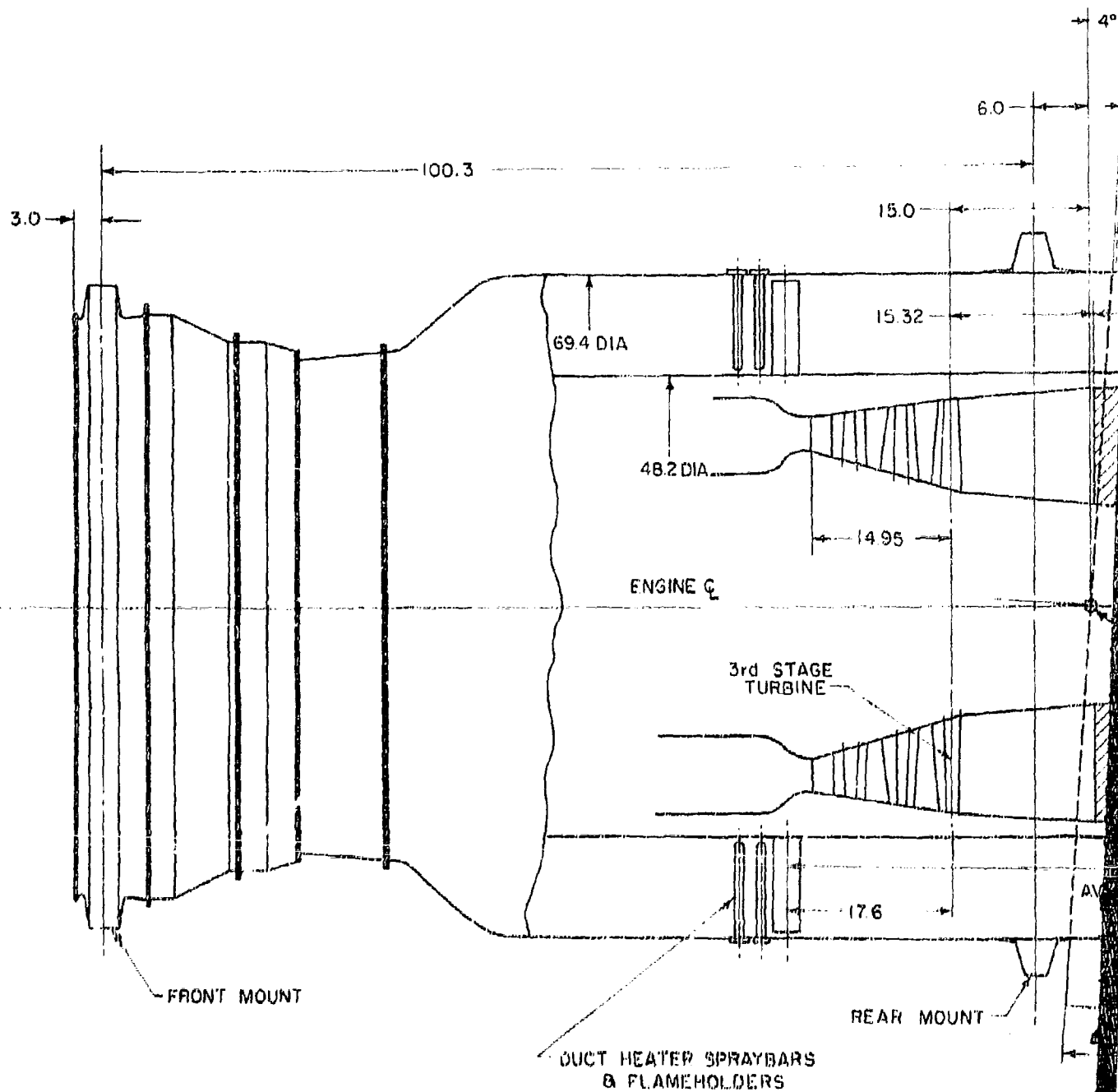
ARRANGEMENT SHOWING REAR ENCL. MOUNT
MOVED AFT. 40.00
ENGINE & EJECTOR ARE TANGENT TO WING
EJECTOR CANTED AT NOZZLE PLANE
ENG. MOUNT PLANES ARE PARALLEL
BASE DRAG AREA 230 in²

STE219L 650 LB/SEC TURBOFAN

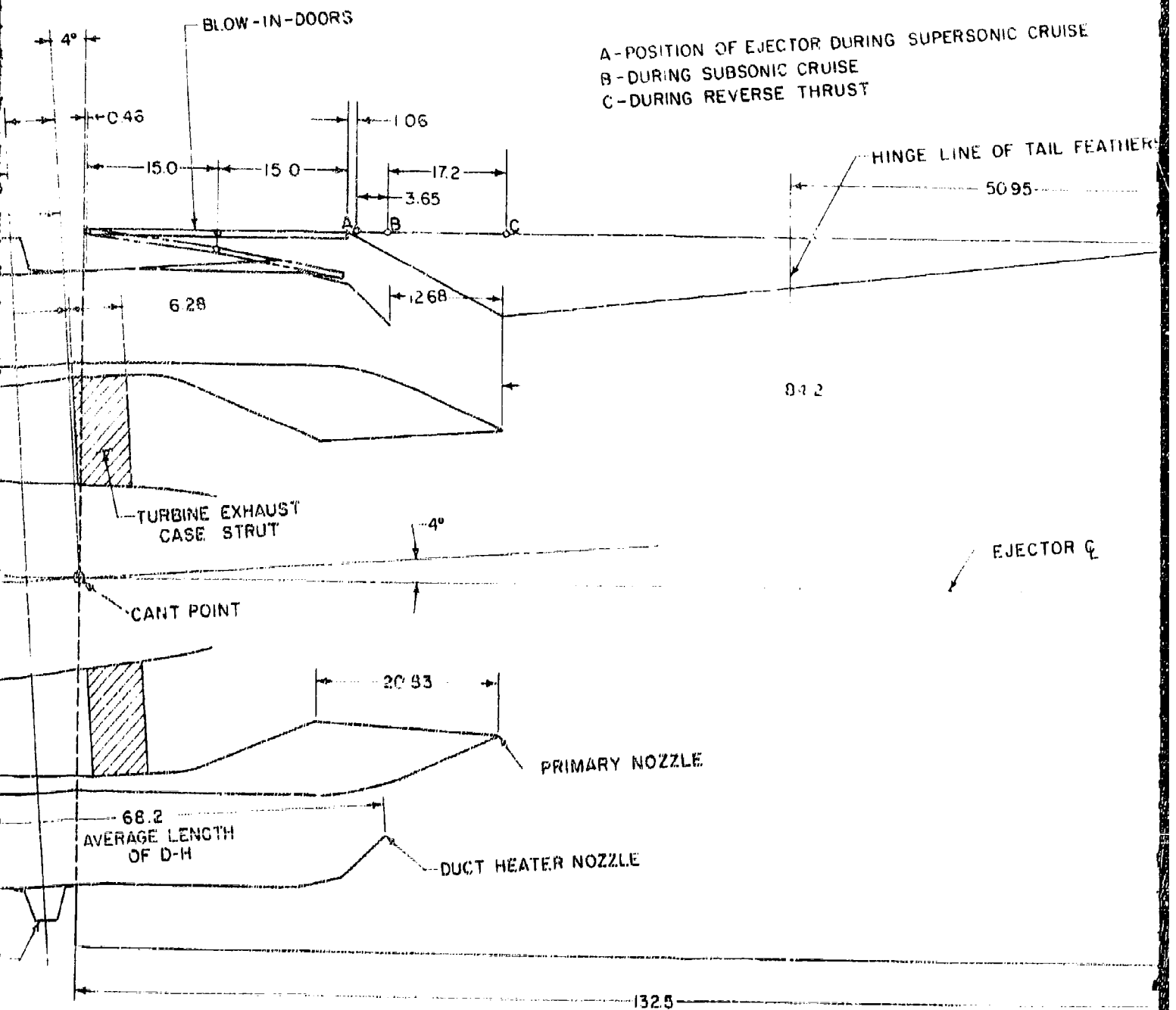
Figure 1-34

CONFIDENTIAL

REVISIONS
1. 10/1/60
2. 10/1/60
3. 10/1/60
4. 10/1/60
5. 10/1/60
6. 10/1/60
7. 10/1/60
8. 10/1/60
9. 10/1/60
10. 10/1/60
11. 10/1/60
12. 10/1/60
13. 10/1/60
14. 10/1/60
15. 10/1/60
16. 10/1/60
17. 10/1/60
18. 10/1/60
19. 10/1/60
20. 10/1/60
21. 10/1/60
22. 10/1/60
23. 10/1/60
24. 10/1/60
25. 10/1/60
26. 10/1/60
27. 10/1/60
28. 10/1/60
29. 10/1/60
30. 10/1/60
31. 10/1/60
32. 10/1/60
33. 10/1/60
34. 10/1/60
35. 10/1/60
36. 10/1/60
37. 10/1/60
38. 10/1/60
39. 10/1/60
40. 10/1/60
41. 10/1/60
42. 10/1/60
43. 10/1/60
44. 10/1/60
45. 10/1/60
46. 10/1/60
47. 10/1/60
48. 10/1/60
49. 10/1/60
50. 10/1/60
51. 10/1/60
52. 10/1/60
53. 10/1/60
54. 10/1/60
55. 10/1/60
56. 10/1/60
57. 10/1/60
58. 10/1/60
59. 10/1/60
60. 10/1/60
61. 10/1/60
62. 10/1/60
63. 10/1/60
64. 10/1/60
65. 10/1/60
66. 10/1/60
67. 10/1/60
68. 10/1/60
69. 10/1/60
70. 10/1/60
71. 10/1/60
72. 10/1/60
73. 10/1/60
74. 10/1/60
75. 10/1/60
76. 10/1/60
77. 10/1/60
78. 10/1/60
79. 10/1/60
80. 10/1/60
81. 10/1/60
82. 10/1/60
83. 10/1/60
84. 10/1/60
85. 10/1/60
86. 10/1/60
87. 10/1/60
88. 10/1/60
89. 10/1/60
90. 10/1/60
91. 10/1/60
92. 10/1/60
93. 10/1/60
94. 10/1/60
95. 10/1/60
96. 10/1/60
97. 10/1/60
98. 10/1/60
99. 10/1/60
100. 10/1/60



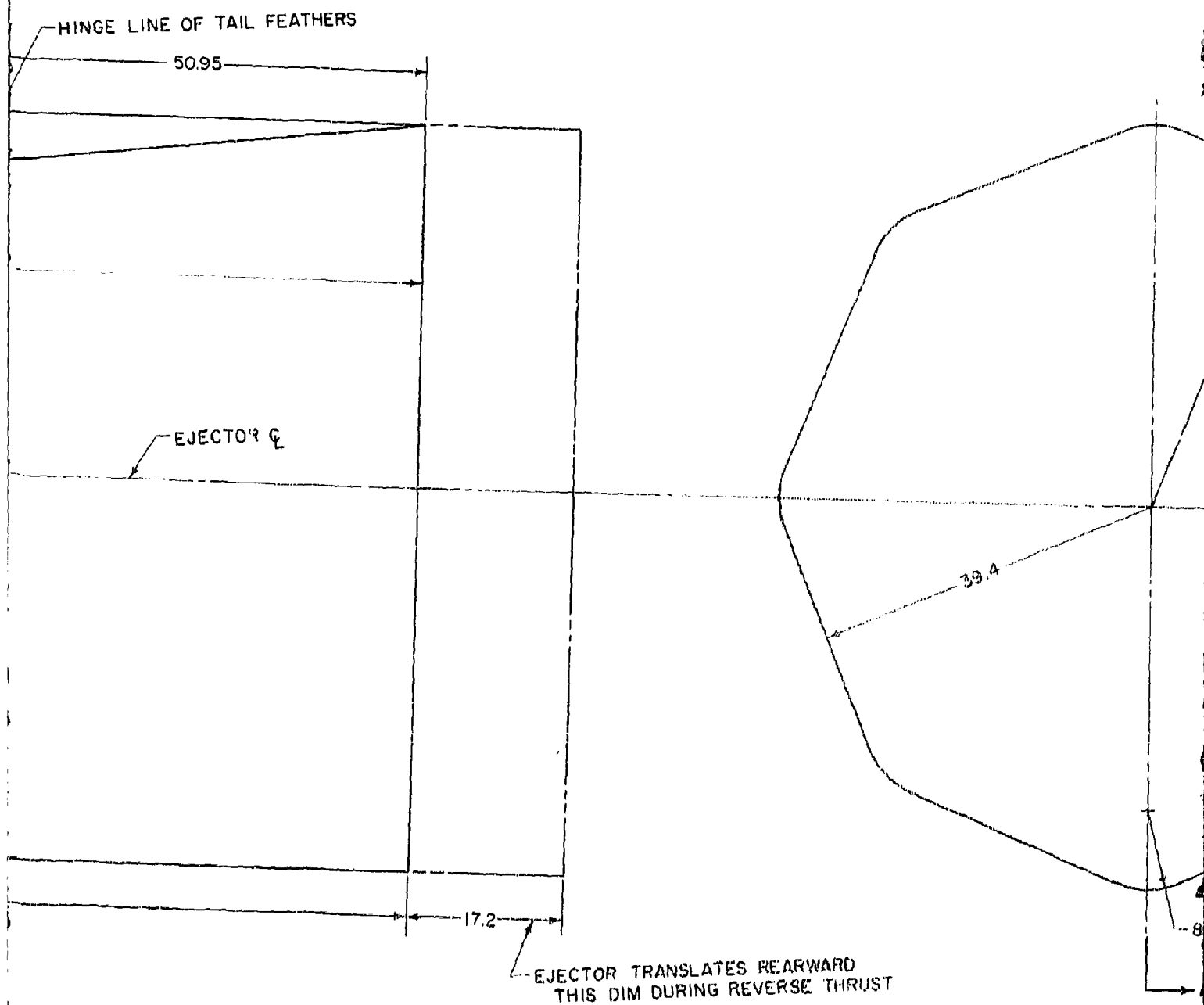
SECTION A-A



SECTION A-A

DURING SUPERSONIC CRUISE
CRUISE
CRUISE

CONFIDENTIAL



STF219L 650 LB/SEC TURBOFAN
ENGINE/EJECTOR RELATION

Figure 1-35

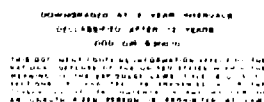
CONFIDENTIAL

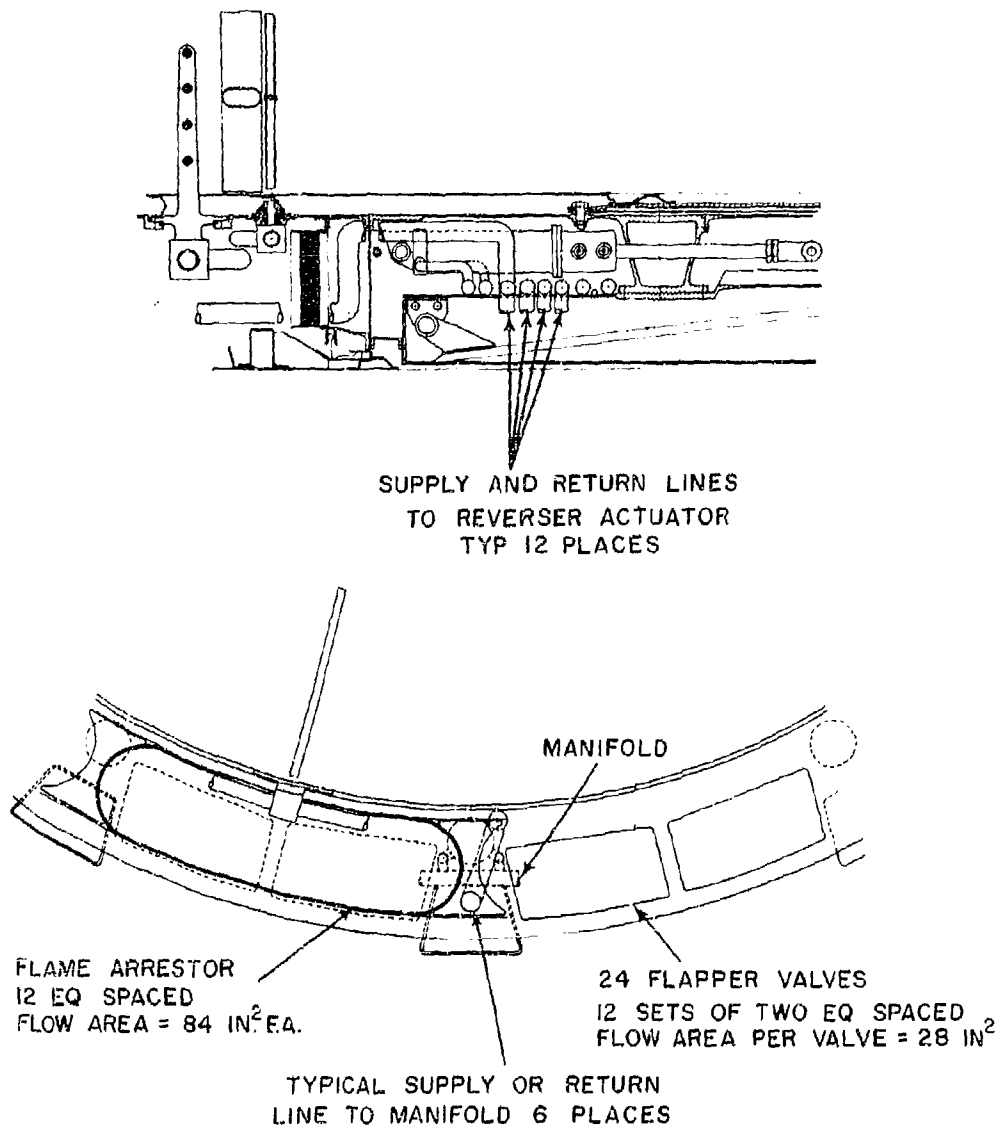
PWA-2600



Figure 1-35

CONFIDENTIAL



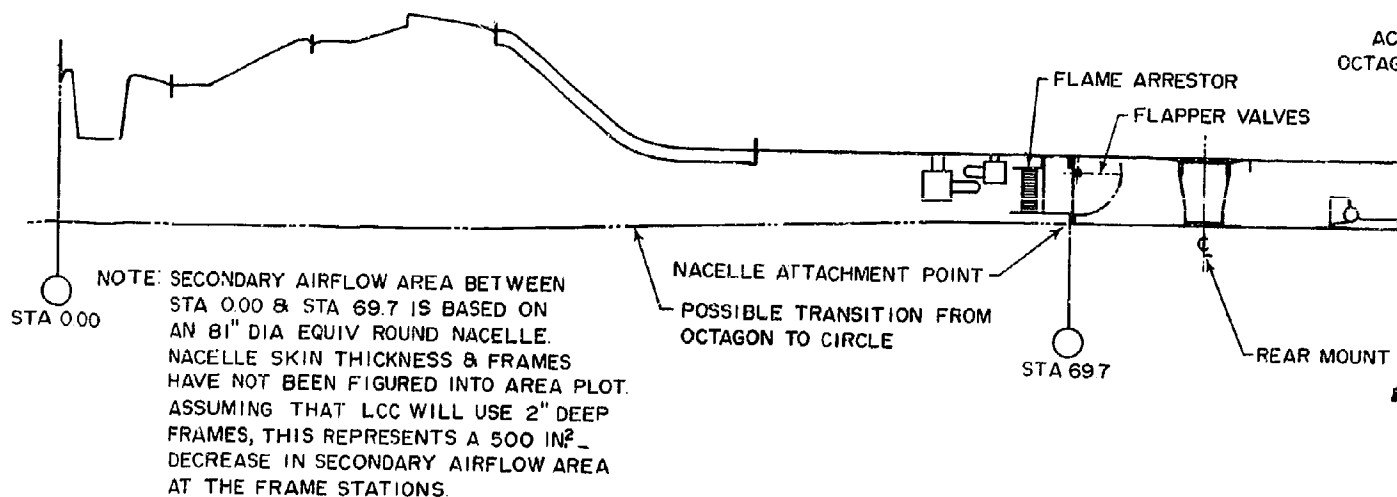
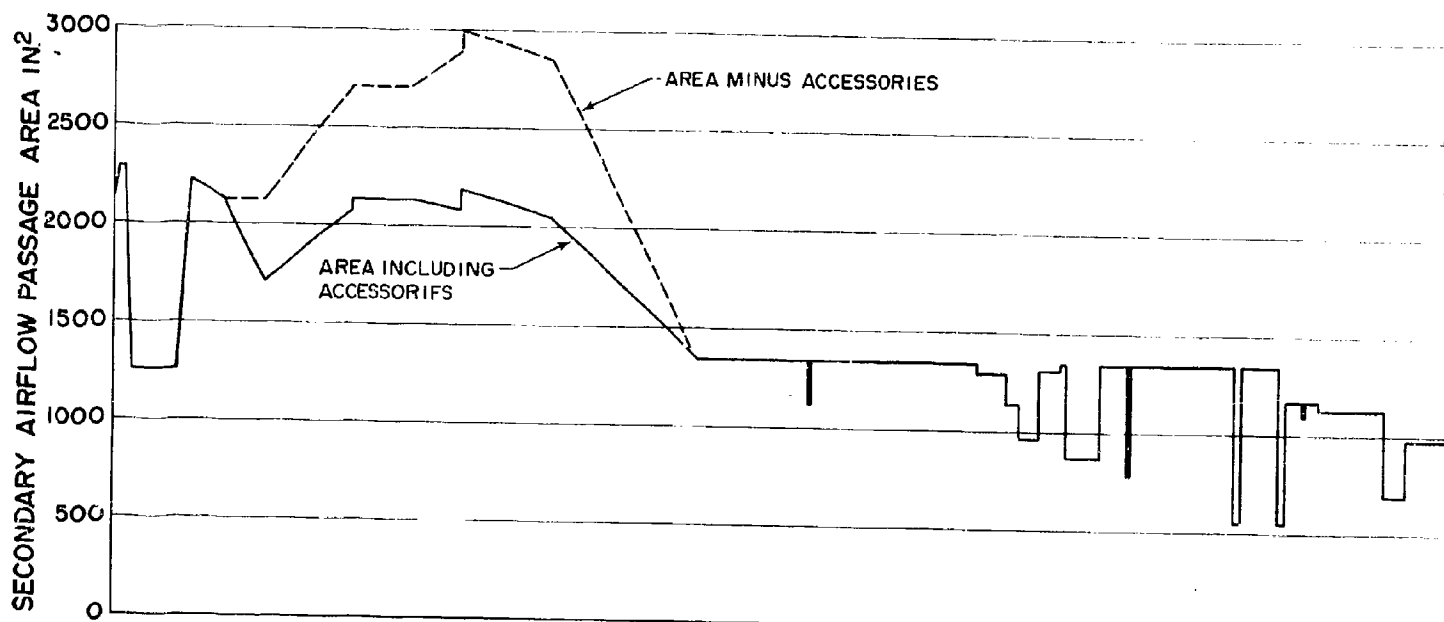


ARRANGEMENT OF FLAME ARRESTORS
IN SECONDARY AIR PASSAGE

Figure 1-37

RECLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
GPO: 1964 O-350-0

THIS DOCUMENT CONTAINS INFORMATION ON WHICH THE
NATIONAL DEFENSE OF THE UNITED STATES MAY BE
DEPENDENT. IT IS THE POLICY OF THE UNITED STATES
GOVERNMENT TO RECLASSIFY AND DECLASSIFY INFORMATION
IN ACCORDANCE WITH THE NATIONAL DEFENSE
AUTHORITY ACT, 50 U.S.C. 3605, AND
EXECUTIVE ORDER 11652, APRIL 2, 1966.

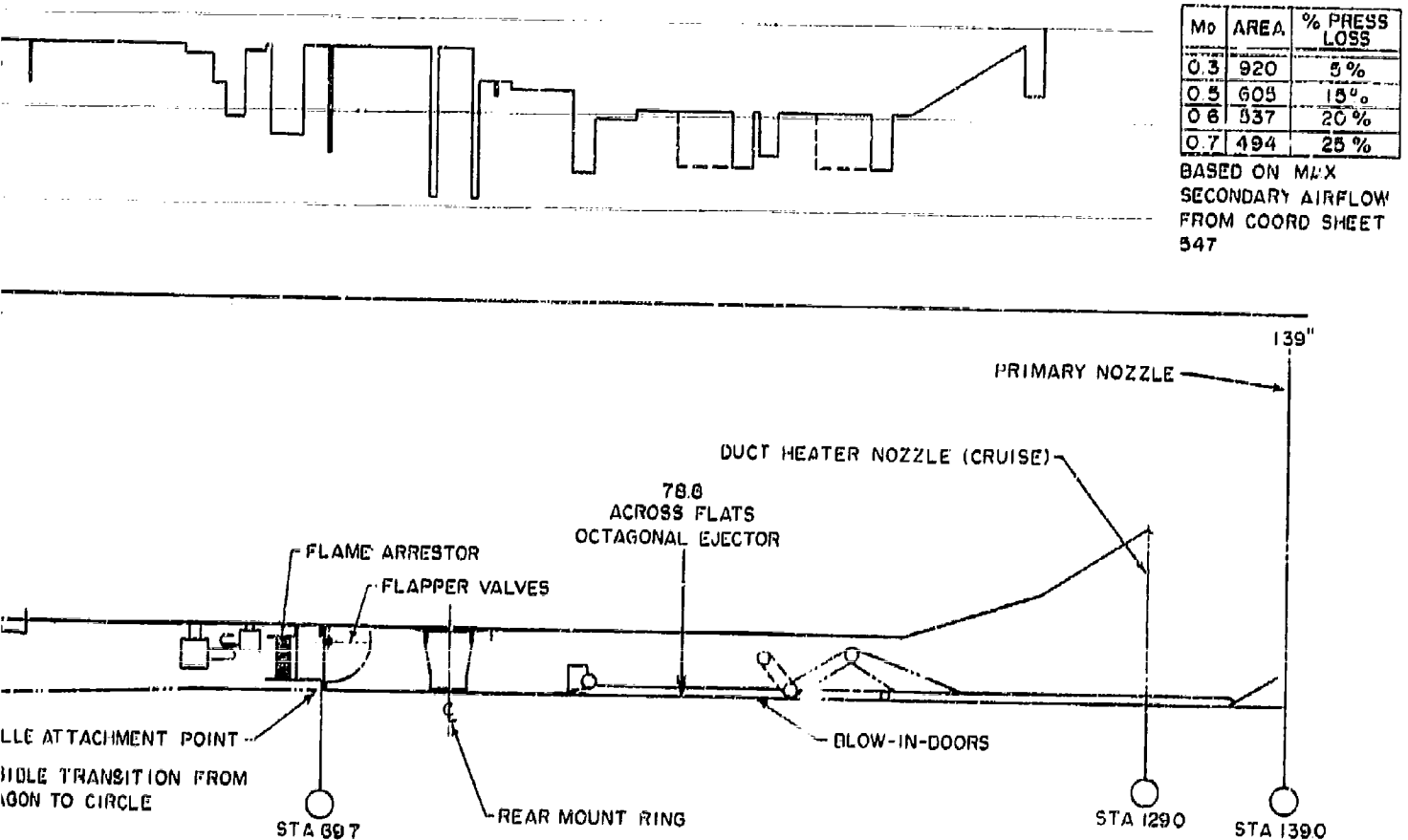


SECONDARY AIRFL

CONFIDENTIAL

PWA-2600

ACCESSORIES



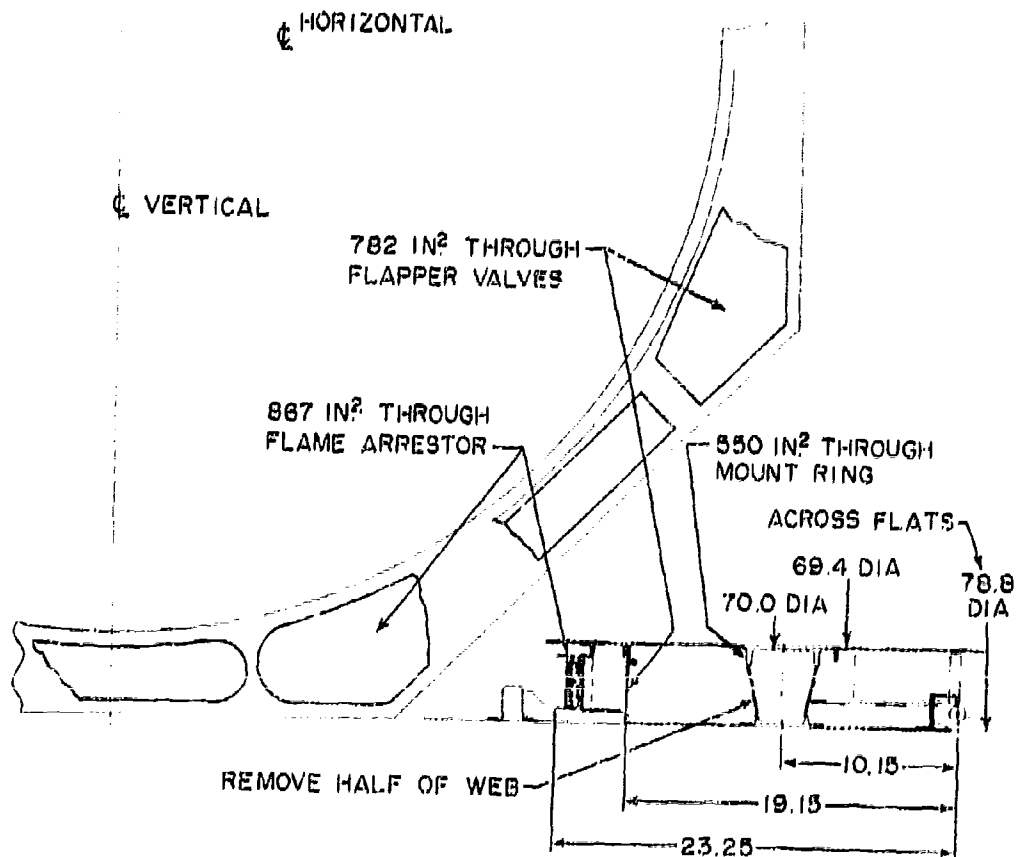
SECONDARY AIRFLOW PASSAGE GENERAL ARRANGEMENT

Figure 1-38

CONFIDENTIAL

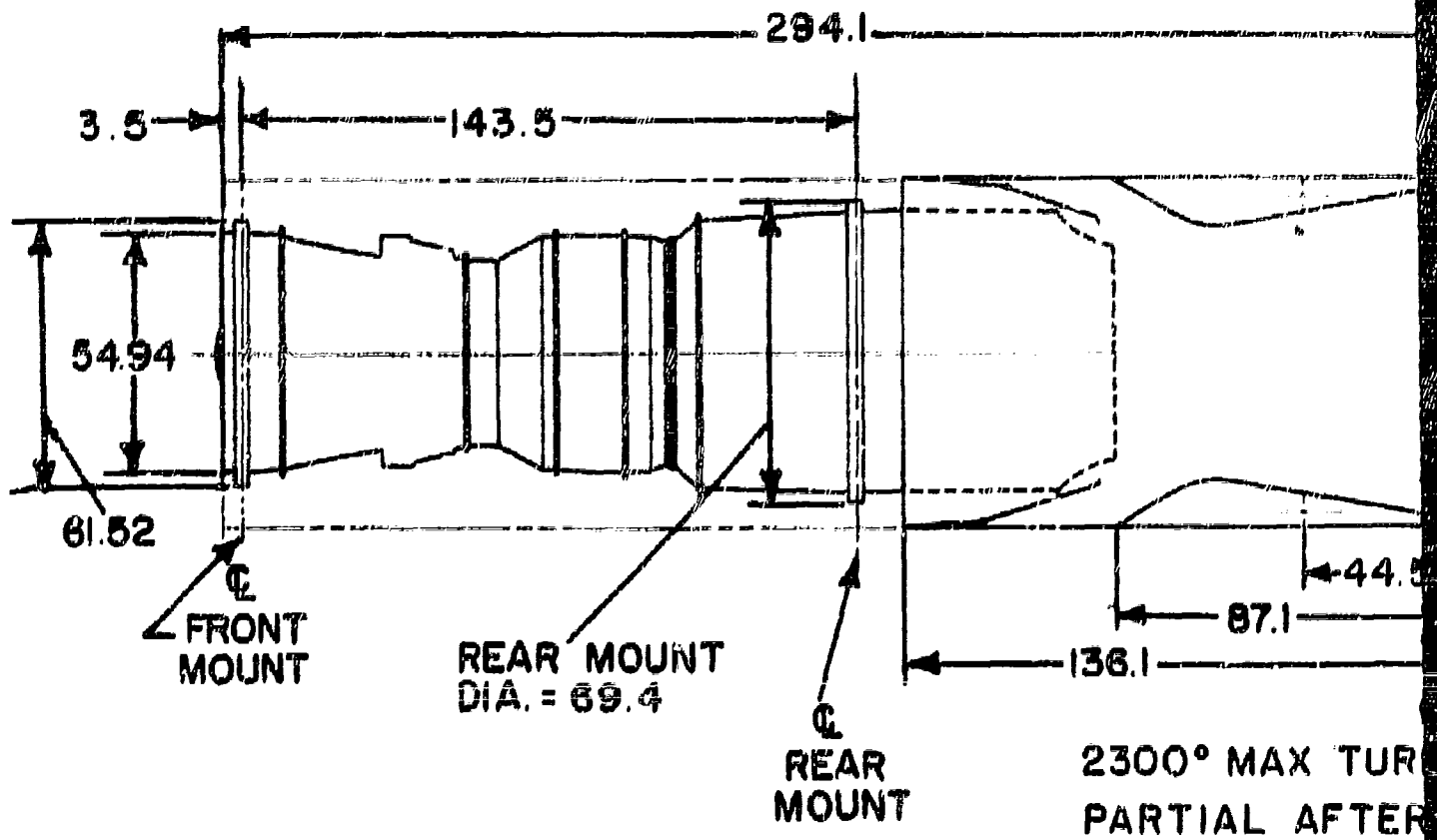
CLASSIFIED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
END FOR 1995-15
THIS DOCUMENT CONTAINS INFORMATION THAT AFFECTS THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE INTELLIGENCE INFORMATION ACT OF 1950 (50 USC 3605) AND IS HEREBY CLASSIFIED "CONFIDENTIAL" TO PREVENT THE DISSEMINATION OF THIS INFORMATION TO ANY PERSON OR ENTITY WHOSE DISSEMINATION COULD BE DETRIMENTAL TO THE NATIONAL DEFENSE OF THE UNITED STATES.

2



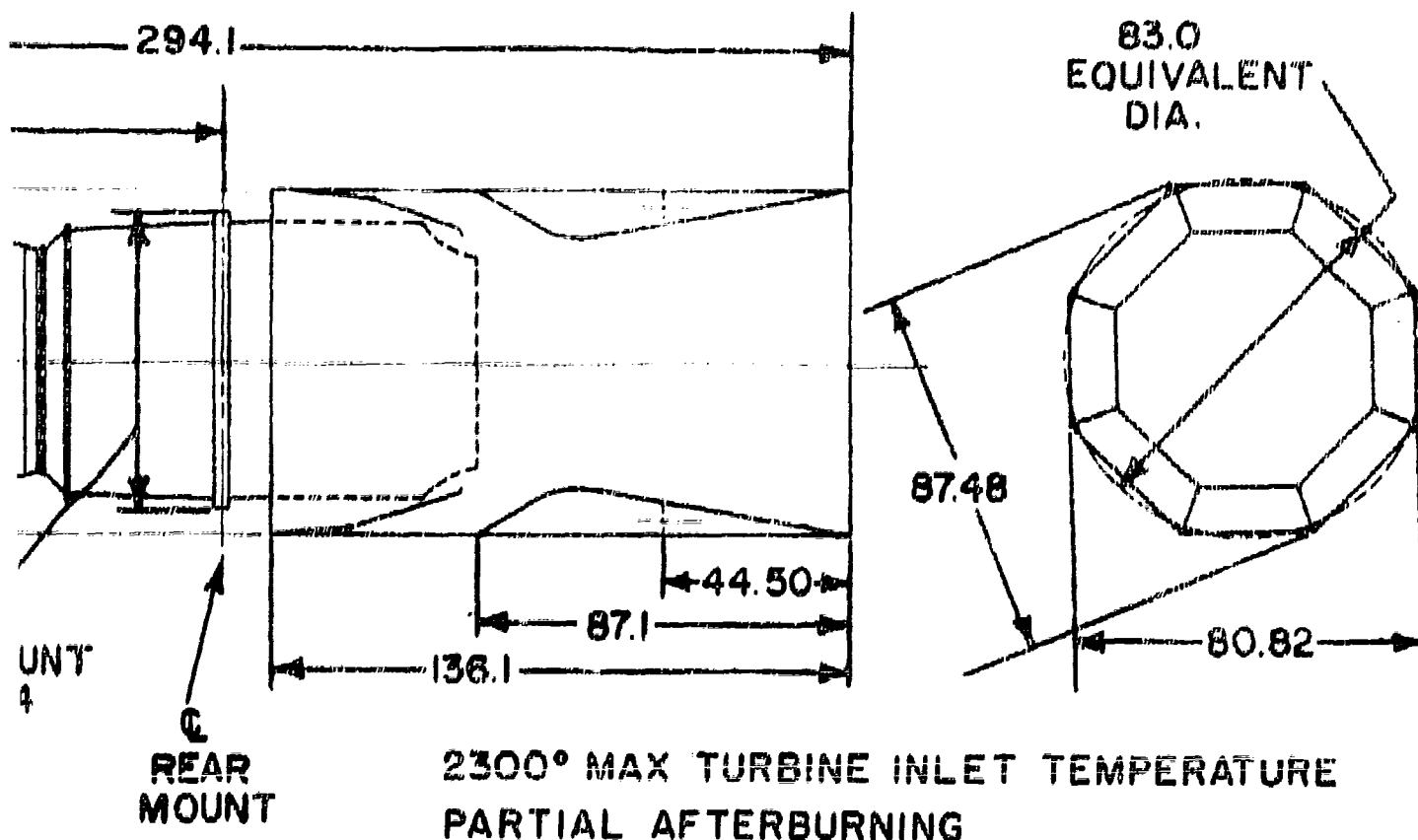
DETAILS OF SECONDARY AIRFLOW PASSAGE

Figure 1-39



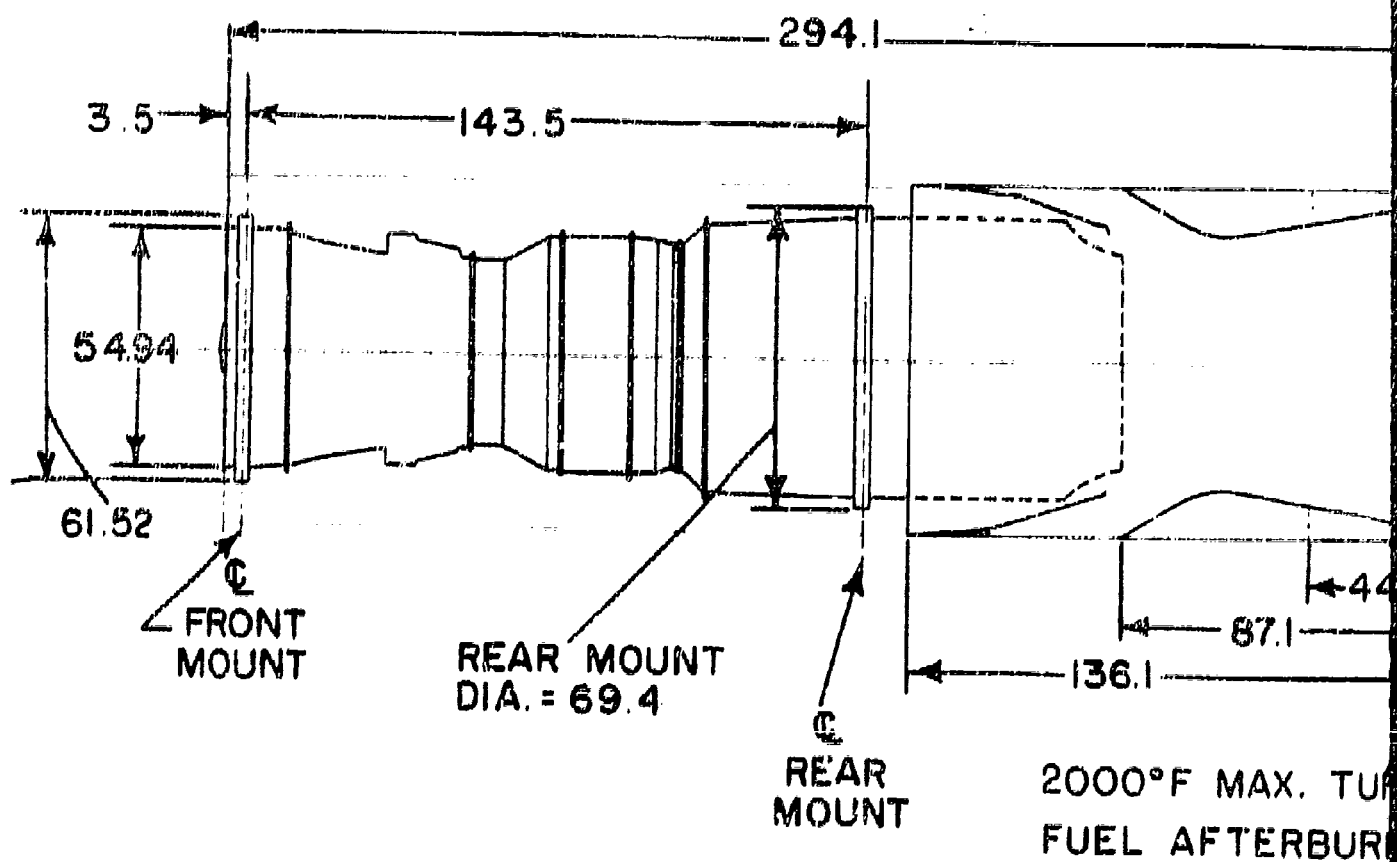
STJ227 525 LB

Fig.



STJ227 525 LB/SEC TURBOJET

Figure 1-40



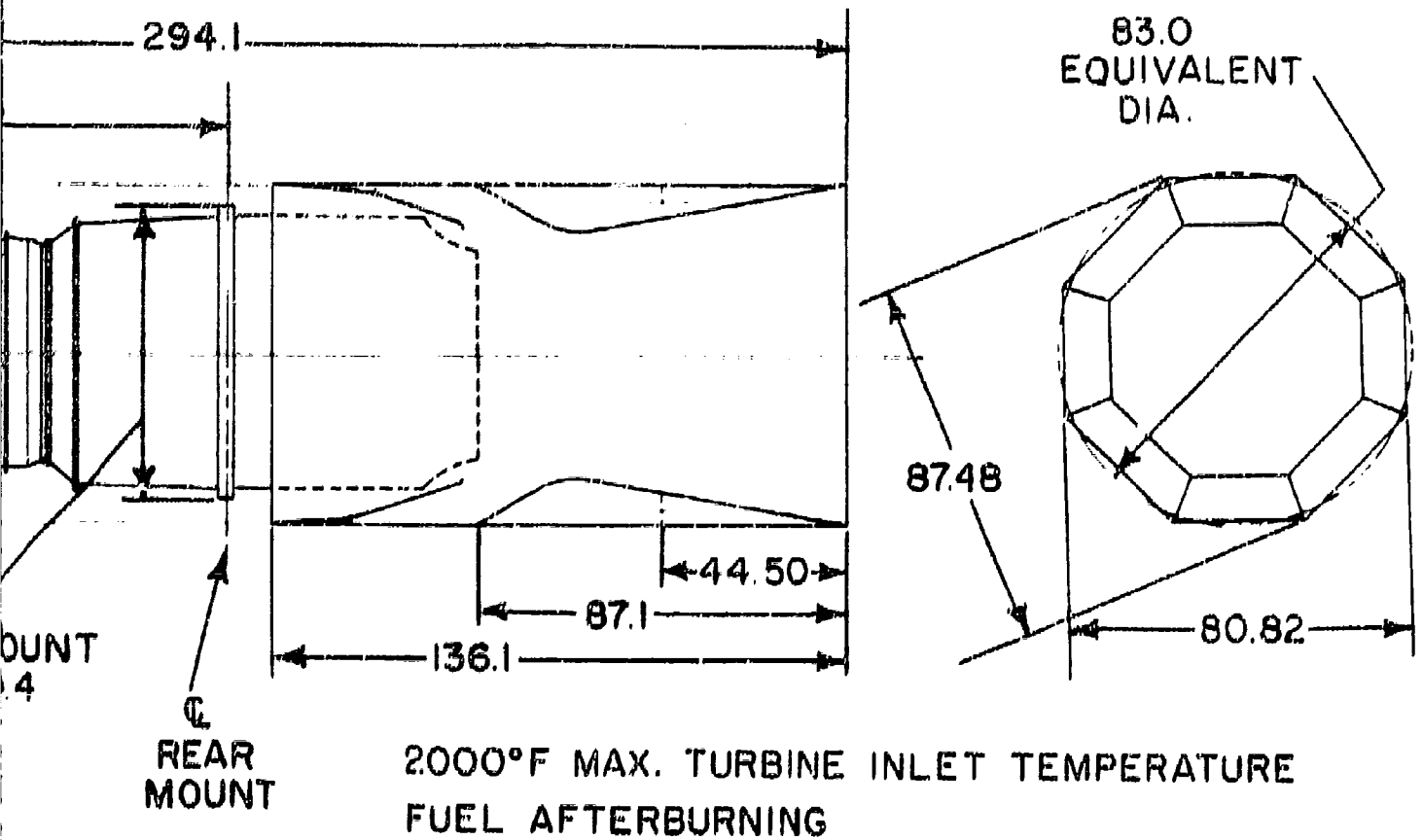
STJ227 525 1

PL

CO

CONFIDENTIAL

PWA-2600



2

STJ227 525 LB/SEC TURBOJET

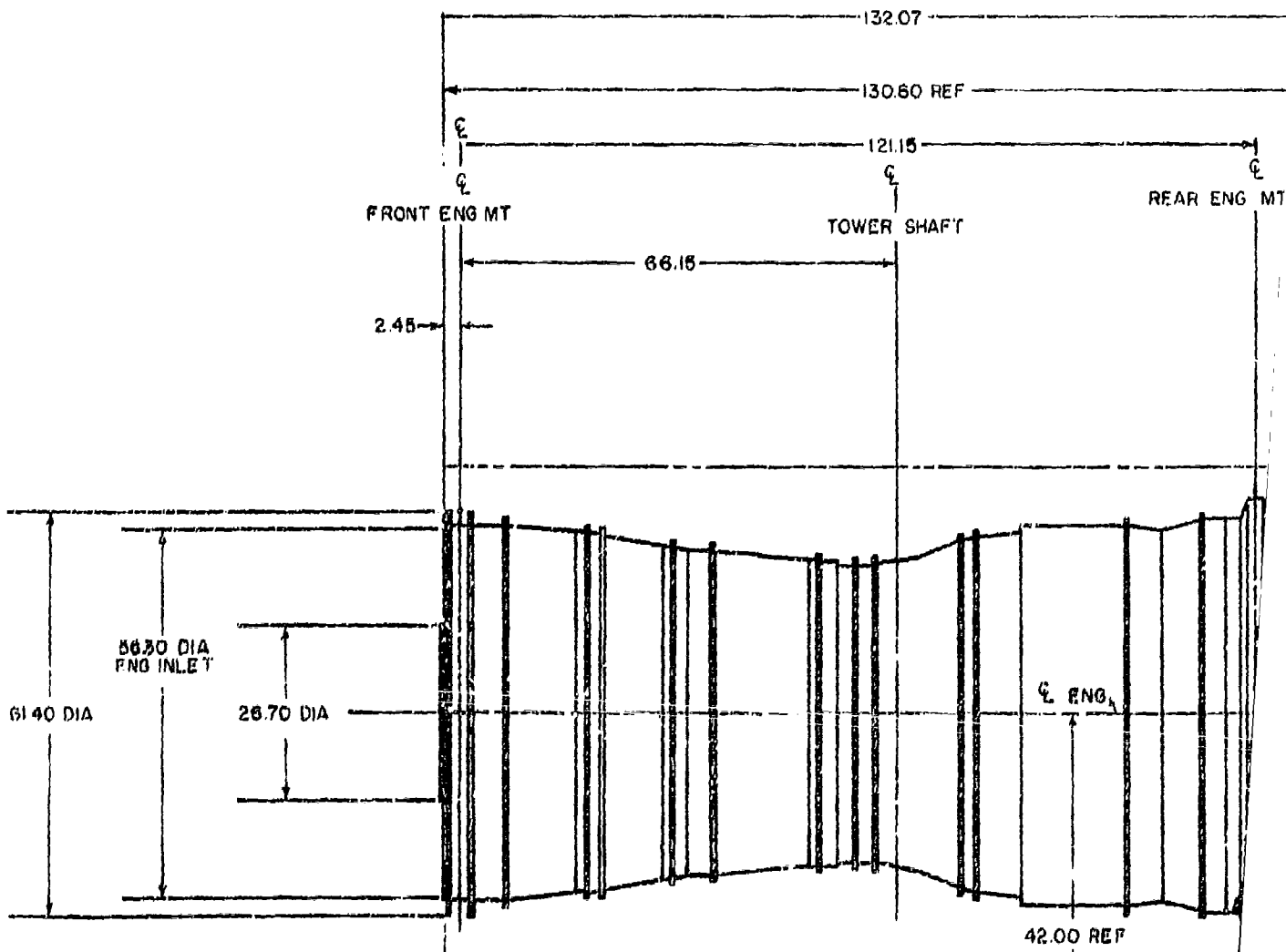
Figure 1-41

CONFIDENTIAL

EXAMINED BY 3 TECH DIVISION
ALL INFORMATION IS TRUE
DATE 10/10/61

THIS DOCUMENT CONTAINS INFORMATION
RELATIVE TO THE NATIONAL DEFENSE
UNIVERSITY OF CALIFORNIA, BERKELEY
RESEARCH CENTER FOR THE STUDY OF
PROGRESSIVE TECHNOLOGICAL
DEVELOPMENT

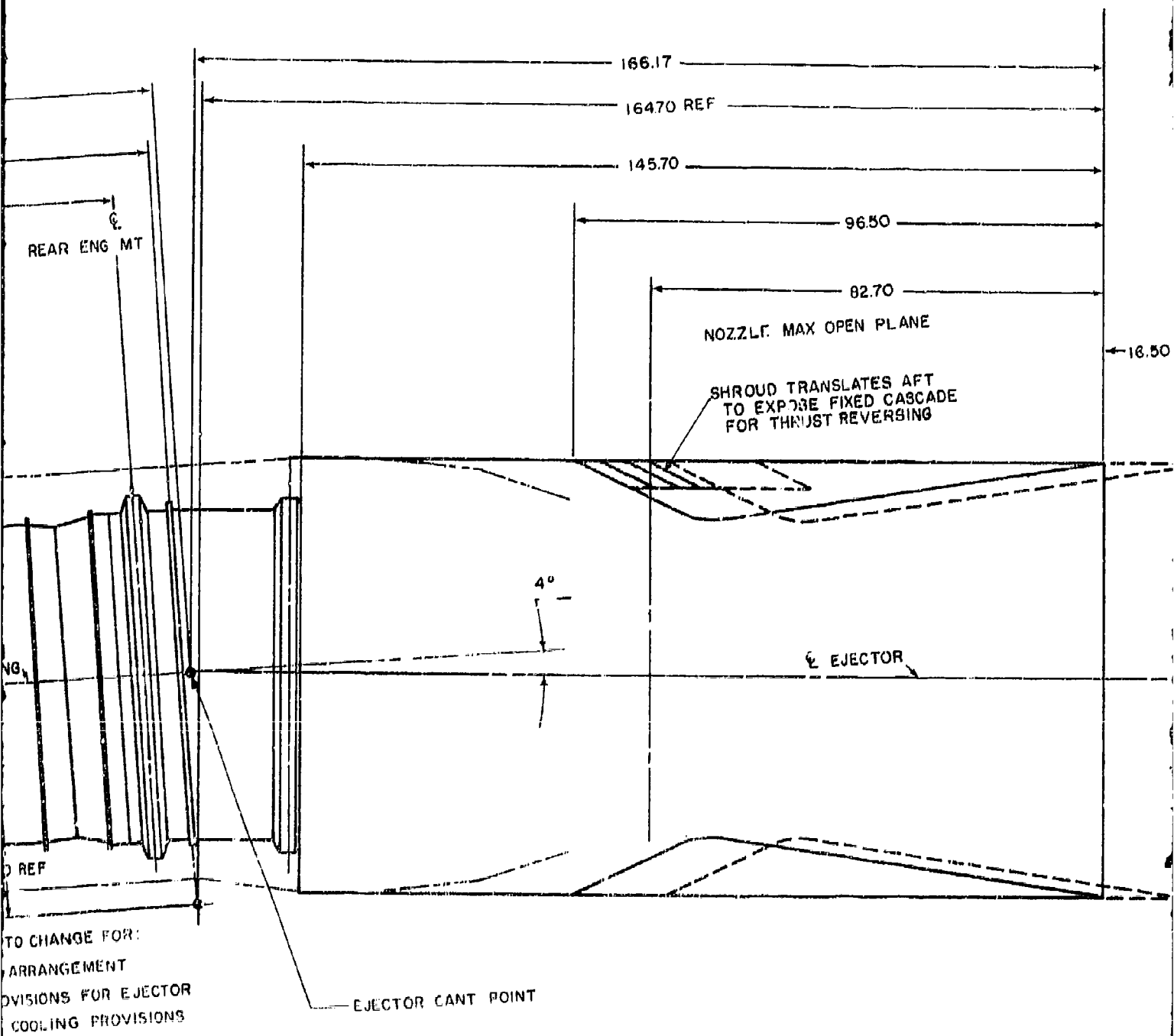
PRATT & WHITNEY AIRCRAFT



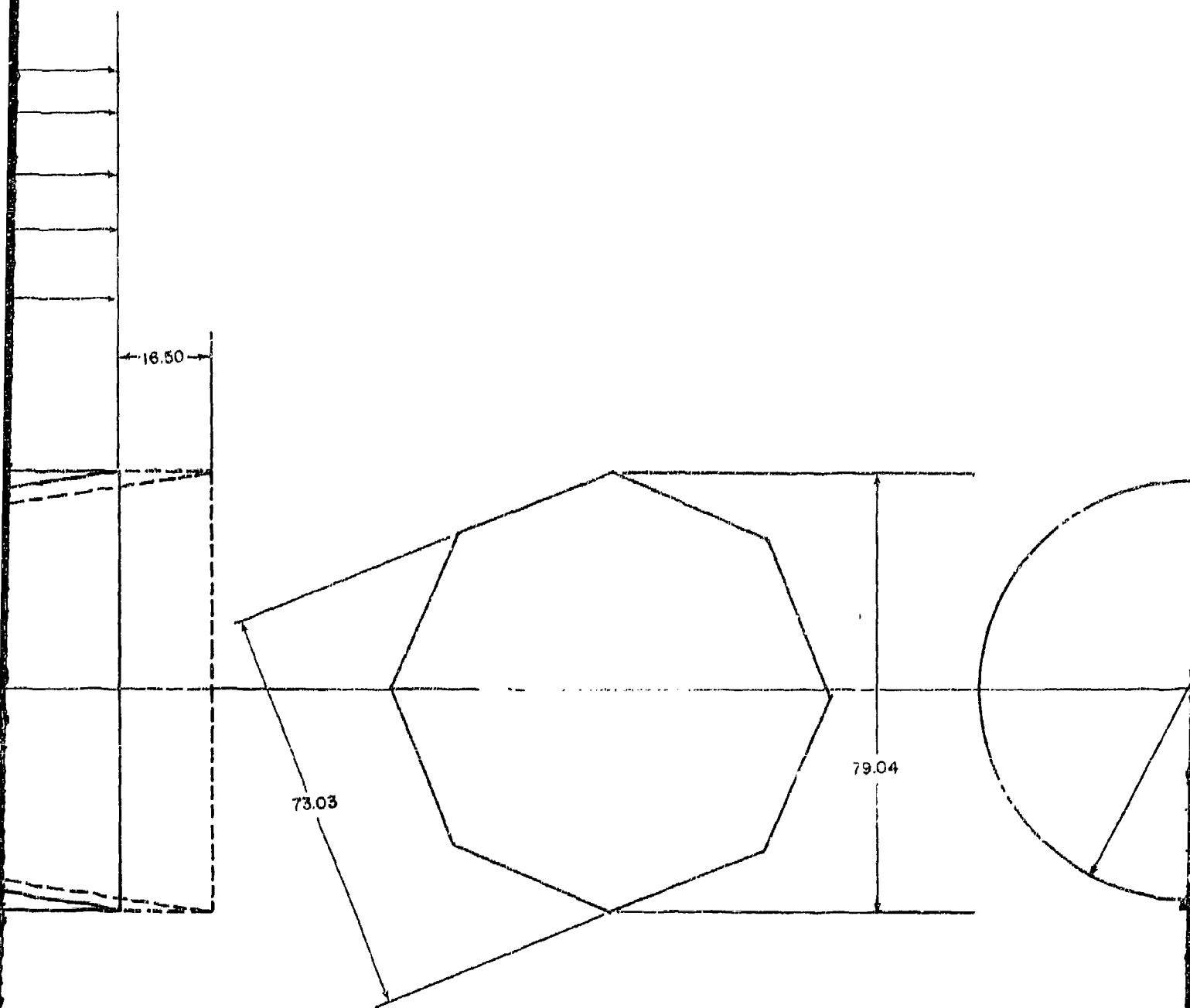
THIS ENVELOPE SUBJECT TO CHANGE FOR:

1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR ENGINE
3. REAR CASE & A/B COOLING PROVISIONS

1



CONFIDENTIAL



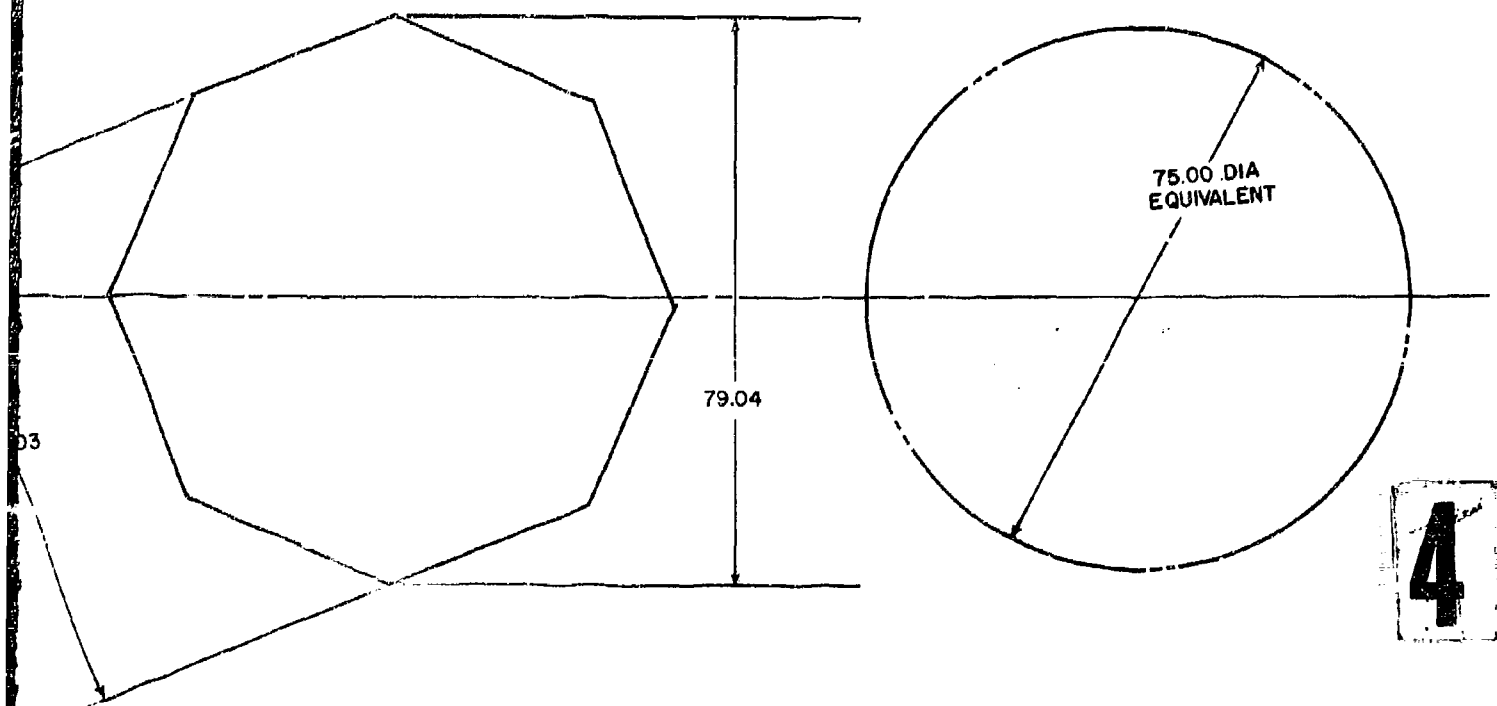
TYPICAL TURBOJET ENGINE. FULL AFTERBU

Figure 1-42

CONFIDENTIAL

CONFIDENTIAL

PWA-2600



TYPICAL TURBOJET ENGINE. FULL AFTERBURNING.

Figure 1-42

CONFIDENTIAL

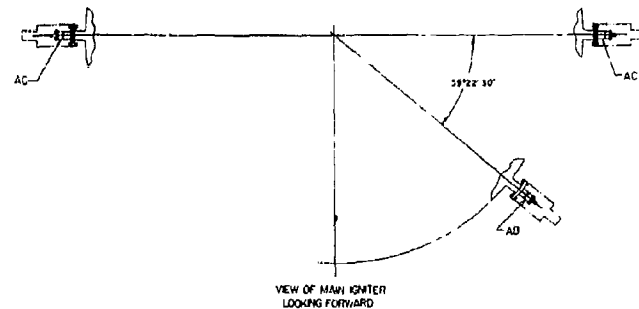
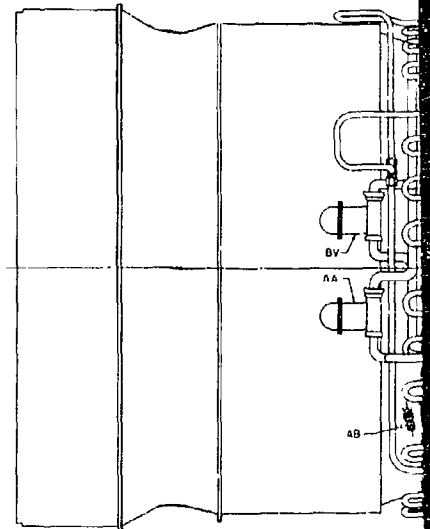
DOWNGRADED AT 3 YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS
DOWNGRADE TO SECRET
THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.
SECTION 793 AND 794, OR TRANSMISSION OF THE
DISSEMINATION OF ITS CONTENTS IS NOT PERMITTED
BY EXECUTIVE ORDER 12958, APRIL 2, 1976

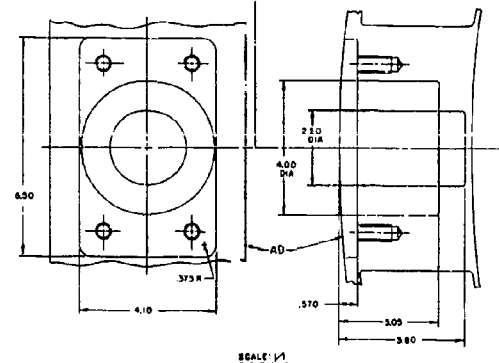
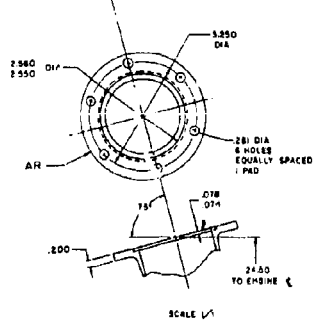
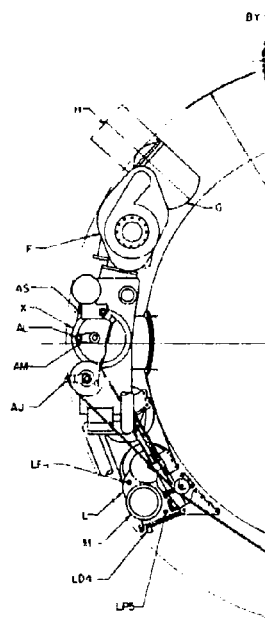
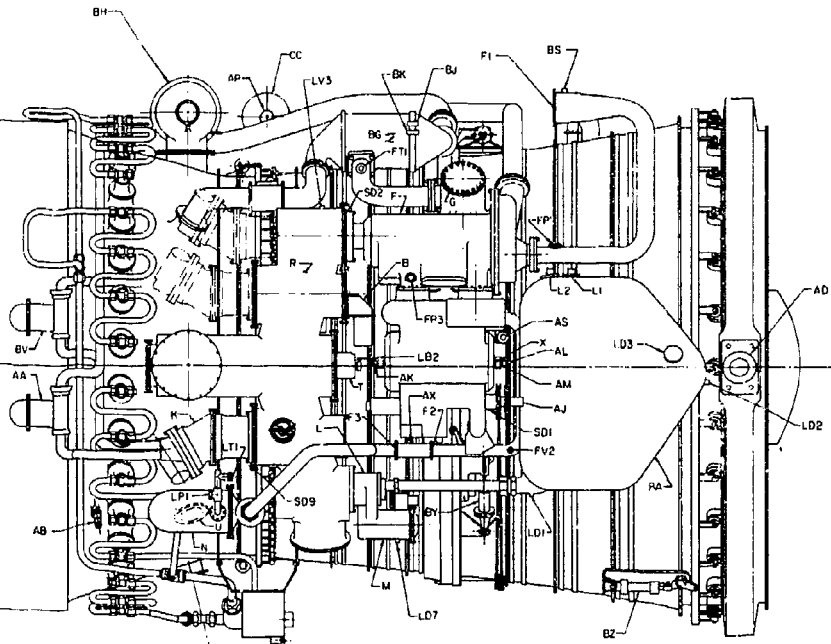
4

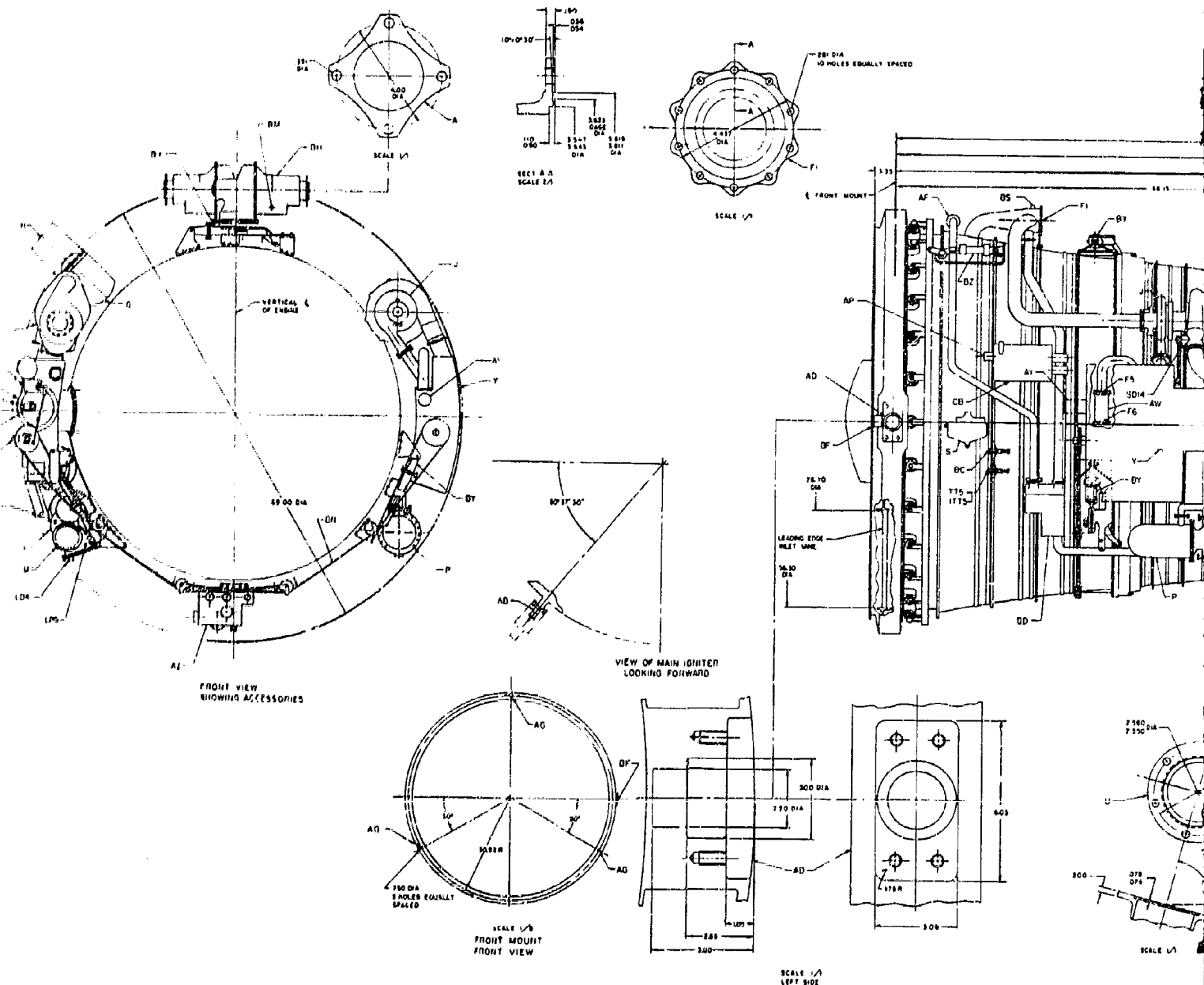
PRATT & WHITNEY AIRCRAFT

ZONE		ACCESSORY DRIVE PADS	
257, 277, 306	A	POWER TAKEOFF	TACHOMETER MOUNTING AND WINDMILL BOOSTER
FUEL DRAIN			
17C	FD1	COMBUSTION CHAMBER FUEL DRAIN	
18C	FD2	DUMP VALVE DRAIN (MAIN)	
18C	FD3	DUMP VALVE DRAIN (AFTERBURNER ZONE 1)	
18C	FD4	DUMP VALVE DRAIN (AFTERBURNER ZONE 2)	
18C	FD5	AFTERBURNER COMBUSTION CHAMBER FUEL DRAIN	
FUEL PRESSURE			
291	FP1	FUEL PUMP INLET PRESSURE	
302	FP3	CHECK VALVE FUEL PRESSURE (OUTLET PRESSURE)	
FUEL FLOW			
187, 307	F1	FUEL PUMP & AFTERBURNER PUMP SUPPLY INLET	
300	F2	MAIN FUEL FLOWMETER SUPPLY INLET	
300	F3	MAIN FUEL FLOWMETER SUPPLY OUTLET	
187	F4	AFTERBURNER FLOWMETER SUPPLY INLET	
180	F5	AFTERBURNER FLOWMETER SUPPLY OUTLET	
FUEL VENT			
290	FV2	FUEL PUMP OUTLET VENT	
FUEL TEMPERATURE			
306	FT1	HEATER OUTLET FUEL TEMPERATURE	
OIL BREATHER			
300	LB2	MAIN OIL OVERBOARD BREATHER	
OIL DRAIN			
28C	LD1	OIL TANK DRAIN	
28D	LD2	OIL TANK OVERFLOW DRAIN	
290	LD3	OIL CUP OVERFLOW DRAIN	
290	LD4	GEARBOX MAIN OIL DRAIN	
300	LD7	OIL STRAINER DRAIN	
OIL FLOW			
28E	LI	OIL TANK NEARBY FILLER	
28E	LI2	OIL TANK MANUAL FILL	
OIL PRESSURE			
210	LP1	PRESSURE FOR TRANSDUCER	
220	LP4	OIL FILTER INLET PRESSURE—PROVISIONS FOR PRESSURE	
230	LP5	OIL FILTER OUTLET PRESSURE—DIFFERENTIAL	
OIL TEMPERATURE			
310	LT1	MAIN OIL TEMPERATURE	
OIL VENT			
307	LV3	OIL PRESSURE TRANSDUCER VENT	
SEAL DRAIN			
280	SD1	FUEL CONTROL SEAL DRAIN	
300	SD2	FUEL PUMP SEAL DRAIN	
300	SD3	HYDRAULIC PUMP SEAL DRAIN	
180	SD14	AFTERBURNER FUEL PUMP SEAL DRAIN	
TEMPERATURE SENSING			
200	TT5	EXHAUST EXIT TEMPERATURE (IN)	
200	TTT5	TURBINE EXIT TEMPERATURE (INDIVIDUAL)	
PRESSURE SENSING			
187	PTS	TURBINE EXIT PRESSURE	
MISCELLANEOUS			
246, 306	F	MAIN FUEL PUMP	
236, 306	G	FUEL PUMP FUEL FILTER DRAIN	
246	H	FUEL PUMP FILTER (MIN SPACE FOR REMOVAL)	
181, 236	J	AFTERBURNER FUEL PUMP	
310	K	HYDRAULIC PUMP	
246, 300	L	OIL PUMP	
246, 300	M	OIL FILTER	
310	N	FUEL OIL COOLER (MAIN)	
180, 230	P	FUEL OIL COOLER (AFTERBURNER)	
300	Q	GEARBOX	
300	R	AUTOMATIC RESTART SWITCH	
181, 189, 210	T	BREATHING PRESSURIZING VALVE	
240, 290	U	HIGH PRESSURE BLEED PAD	
180, 210	V	MAIN FUEL CONTROL	
170	W	AFTERBURNER FUEL CONTROL	
310	AA	EXHAUST NOZZLE CONTROL	
300, 310, 320, 330	AB	HYDRAULIC DISCHARGE FILTER	
300, 310, 320, 330	AC	IGNITER PLUG (MAIN)	
310, 320, 330, 340	AD	IGNITER PLUG (AFTERBURNER)	
180, 189, 190	AE	ENGINE FRONT MOUNTING PROVISIONS	
207	AF	ENGINE REAR MOUNTING PROVISIONS	
		FUEL RETURN TO SUPPLEMENTARY CLOSING	

ZONE		GROUND HANDLING HOLES (FRONT MOUNT)	
21A	AG	GROUND HANDLING HOLES (FRONT MOUNT)	
110	AH	TURN HANDLING AREA (REAR)	
280	AJ	POWER CONTROL LEVER (90° ANGLE OF TRAVEL)	
300	AK	SHUT OFF LEVER (90° ANGLE OF TRAVEL)	
280	AL	APPROACH VELOCITY CONTROL LEVER (90° ANGLE OF TRAVEL)	
280, 290	AM	WACH NO ON SHOCK POSITION RESET LEVER (90° ANGLE OF TRAVEL)	
180	AN	AIR TURBINE VALVE (AFTERBURNER FUEL PUMP)	
310	AP	IGNITION EXCITER ELECTRICAL CONN.	
310	AR	THERMAL ANTI-ICE PAD	
246, 280	AS	AEROHYDRAULIC BRAKE CONTROL AIR SUPPLY CONN.	
170	AT	EXHAUST NOZZLE FEEDBACK	
180	AU	CHECK & DUMP VALVE (AFTERBURNER ZONE 1)	
180	AV	CHECK & DUMP VALVE (AFTERBURNER ZONE 2)	
180	AW	MAIN FUEL FLOWMETER MOUNTING PROVISIONS	
300	AX	AFTERBURNER FUEL FLOWMETER MOUNTING PROVISIONS	
210, 180	AY	FUEL CONTROL FUEL FILTER (MIN SPACE FOR REMOVAL)	
180, 230	AZ	WINDMILL BYPASS CHECK & DUMP LOW OIL VALVE (MAIN)	
290	BA	OIL TANK	
170	BB	NOZZLE POSITION INDICATOR MOUNTING PROVISIONS	
200	BC	REVERSE POSITION INDICATOR CONN. (ELECTRICAL)	
180	BD	THERMOSTAT FUEL BYPASS VALVE	
180, 220, 300, 310	BF	VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFG)	
200	BG	FUEL HEATER	
170, 240, 310	BH	POWER TAKEOFF DEARBON	
307	BI	FUEL HEATER VALVE ELECTRICAL CONN.	
307	BJ	FUEL HEATER VALVE POSITION INDICATOR (ELECTRICAL)	
180	BK	POWER TAKEOFF DECOUPLER	
180, 240	BL	POWER TAKEOFF DECOUPLER ACTUATING SHAFT	
280	BM	CONTROL CABLE—FUEL CONTROLS	
180, 280	BS	FUEL INLET TEMPERATURE SENSOR	
180	BU	AFTERBURNER FUEL OIL COOLER THERMOSTATIC BYPASS VALVE	
310	BV	HYDRAULIC RETURN FILTERS	
180	BW	EXHAUST NOZZLE ACTUATORS	
210, 280, 300, 310, 320	BY	STARTING BLEED DOOR ACTUATORS	
200, 290	BZ	INLET GUIDE VANE ACTUATORS	
180, 189	CA	AEROHYDRAULIC BRAKE ACTUATORS	
200	CB	EXCITER (MAIN)	
180, 210	CC	EXCITER (AFTERBURNER)	

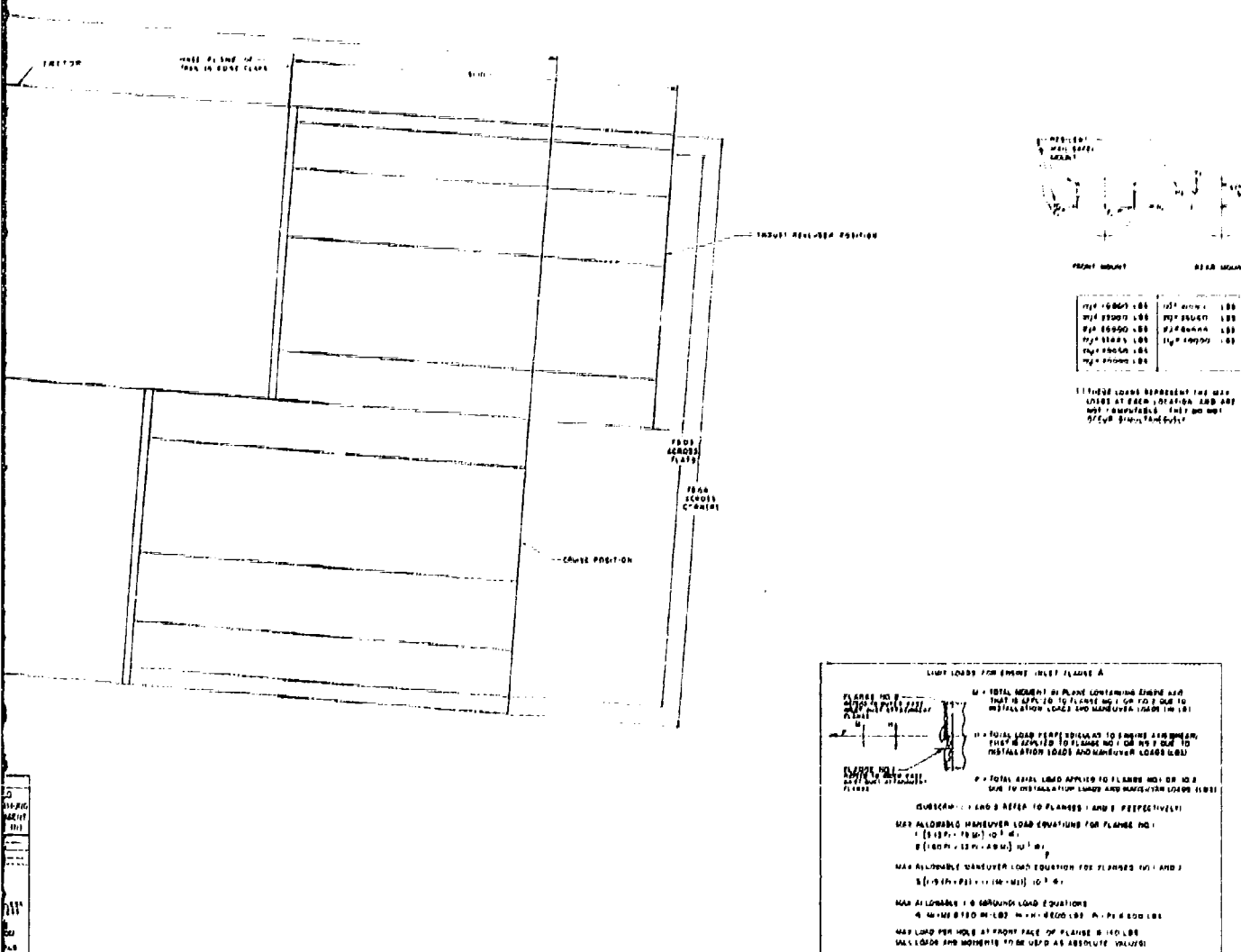






CONFIDENTIAL

FWA-2600



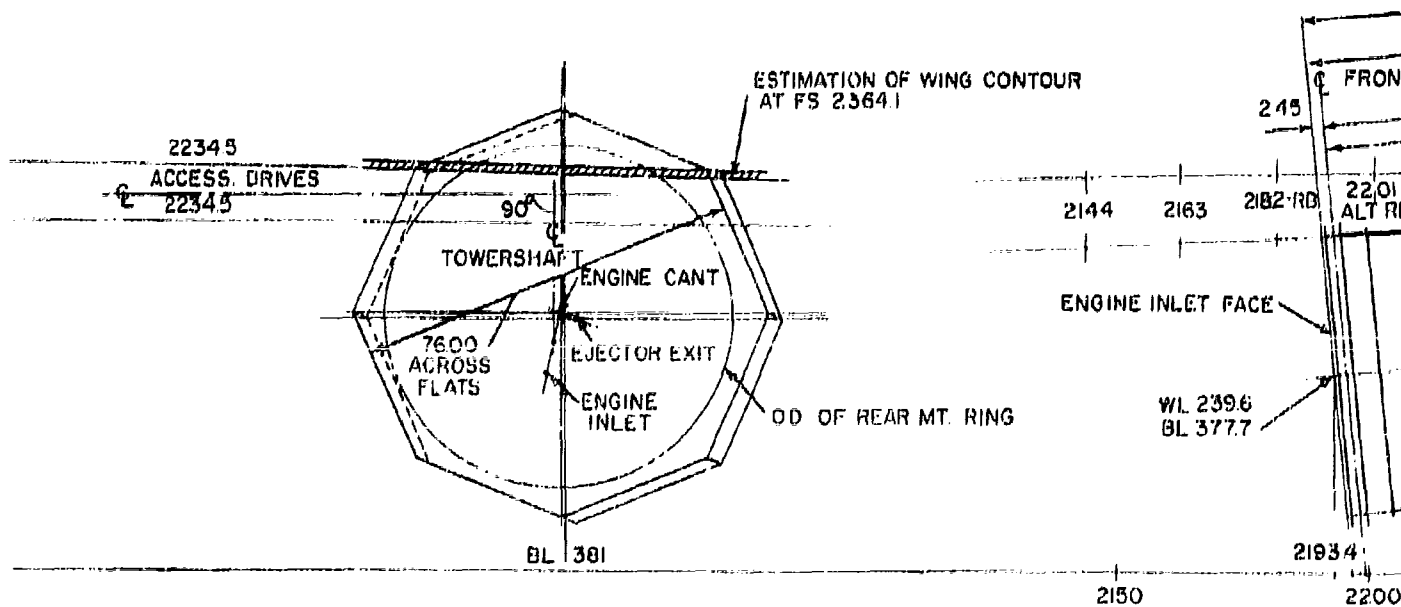
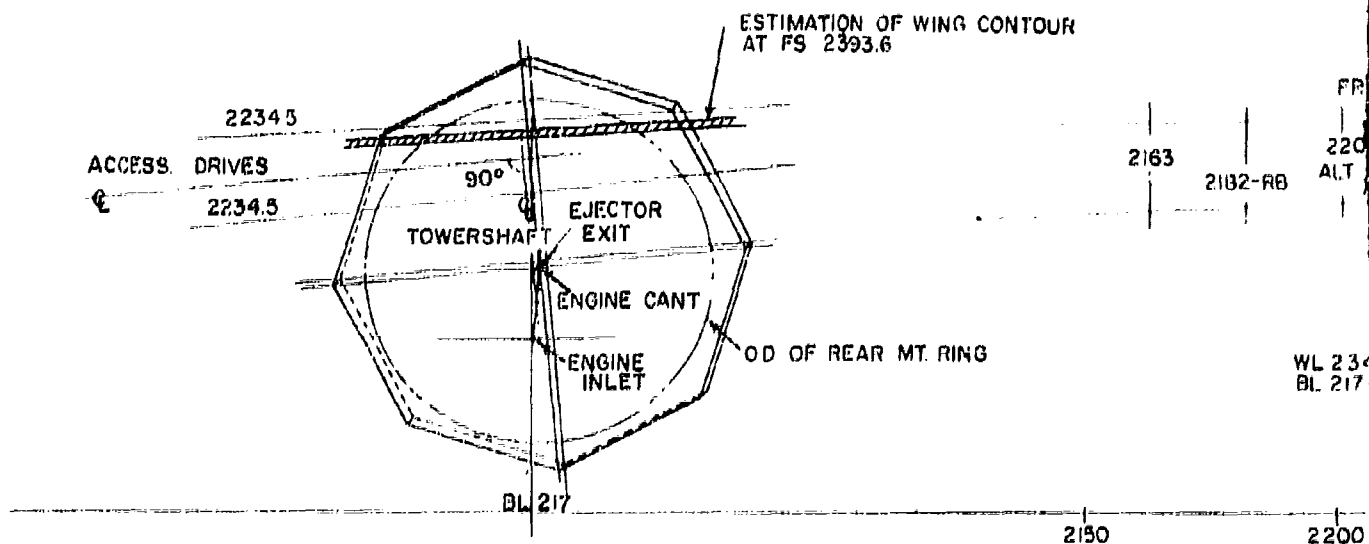
PROPOSED TURBOJET ACCESSORY ARRANGEMENT

Figure 1-43

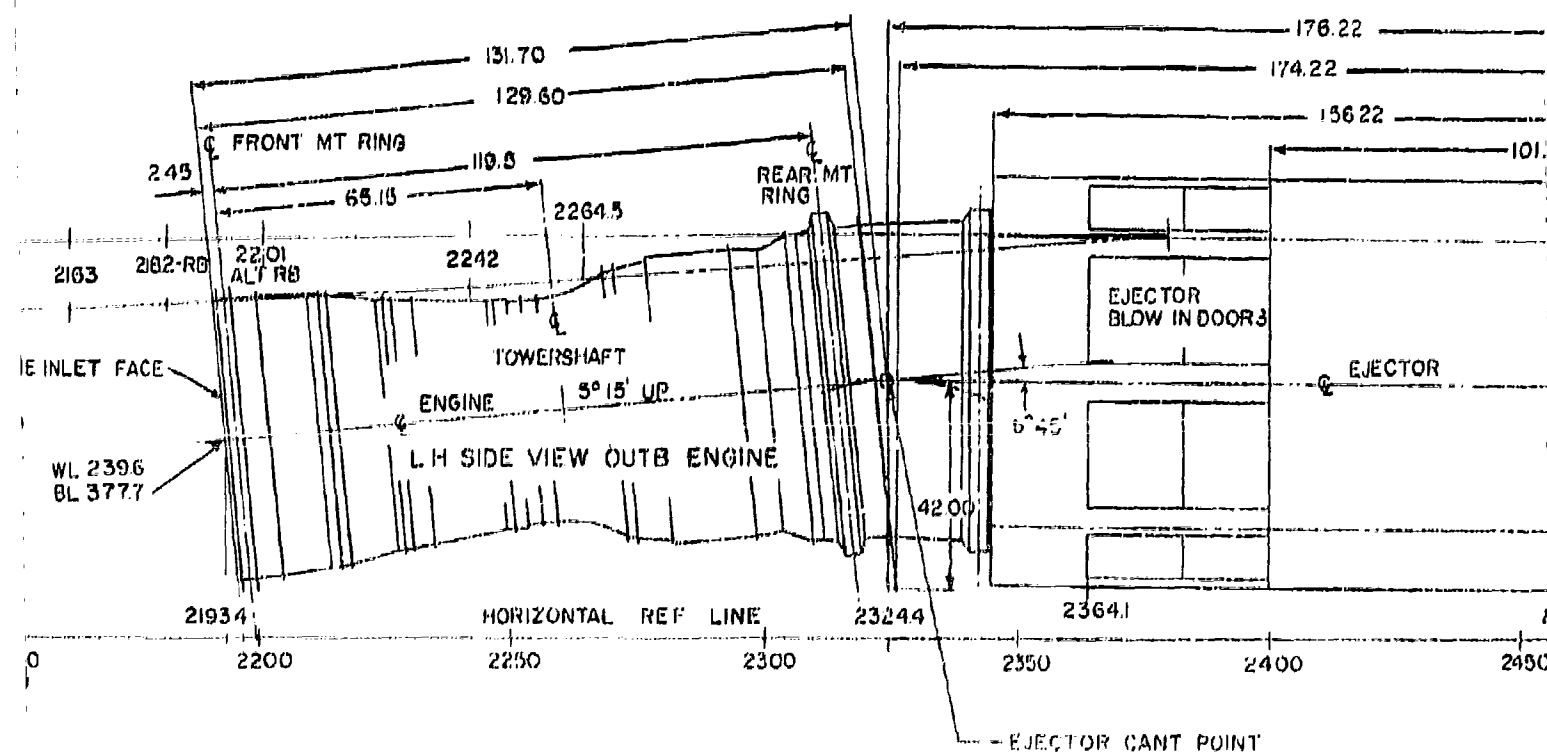
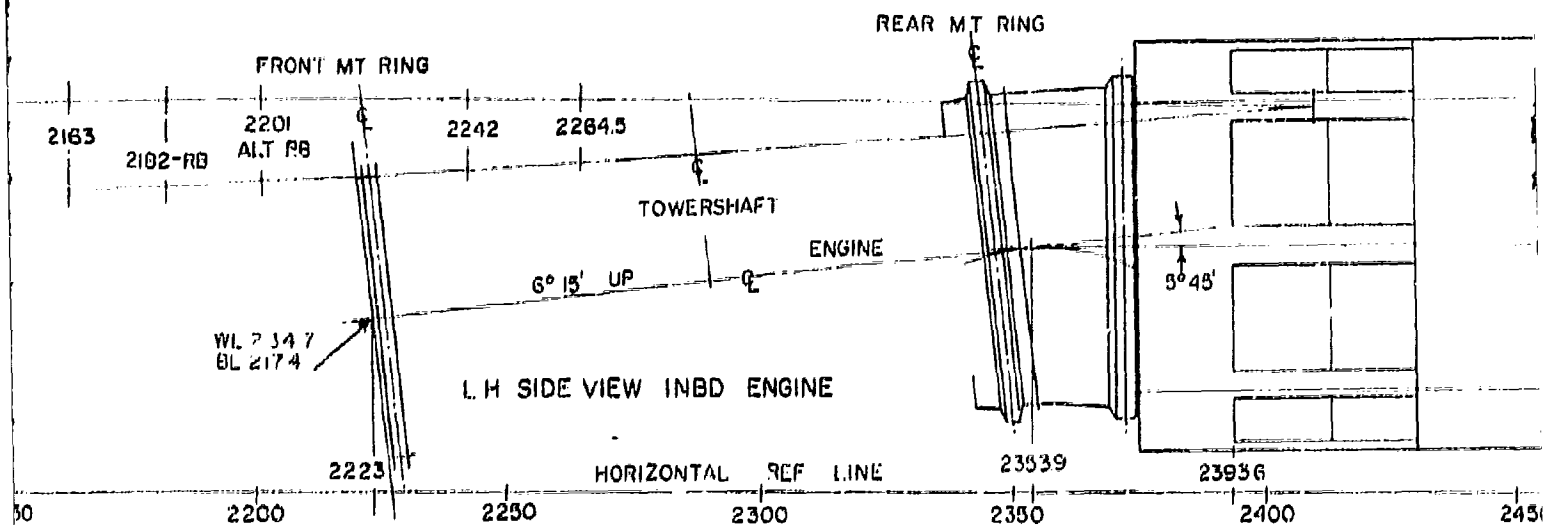
CONFIDENTIAL

(1) 2014 年 12 月 31 日 2015 年 12 月 31 日
 2016 年 12 月 31 日 2017 年 12 月 31 日
 2018 年 12 月 31 日 2019 年 12 月 31 日

[illegible]



CONFIDENTIAL



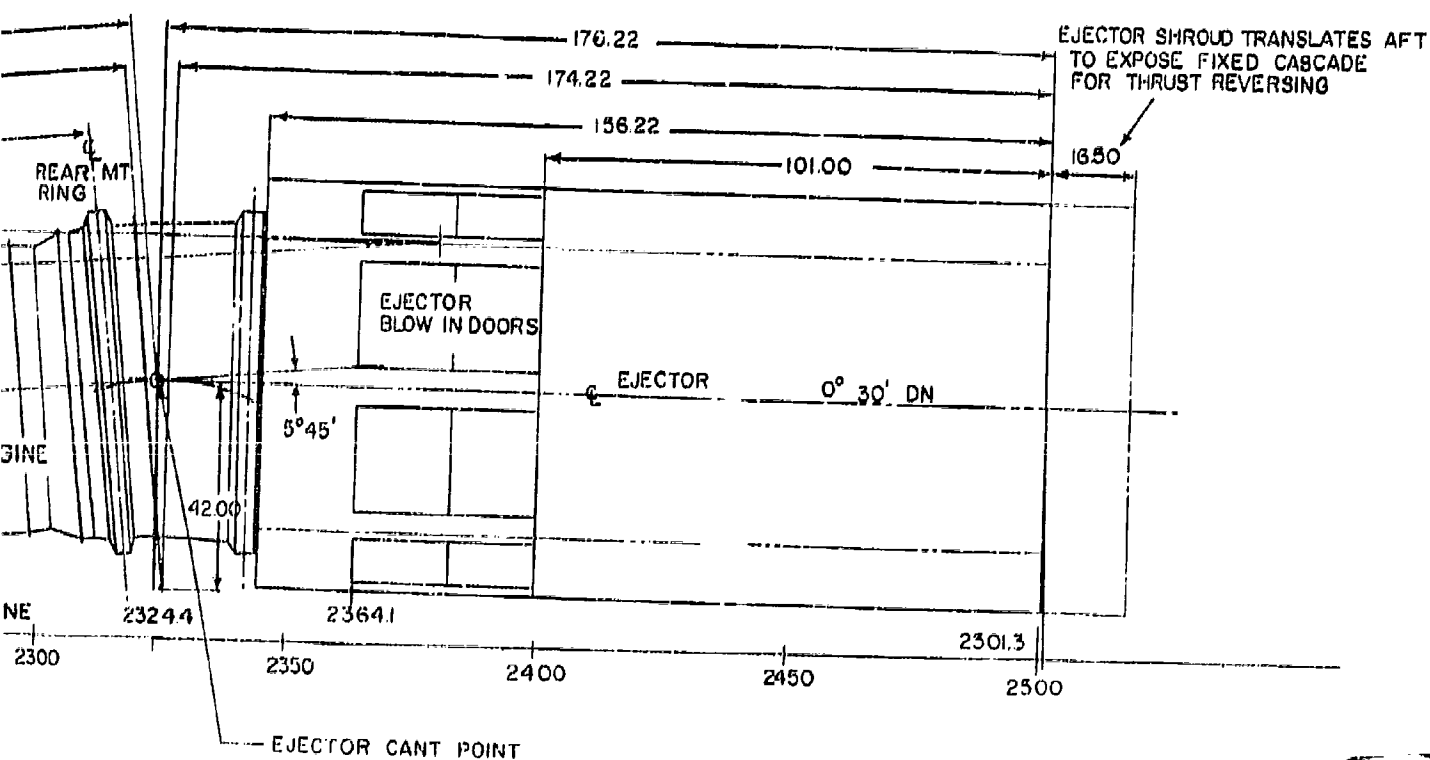
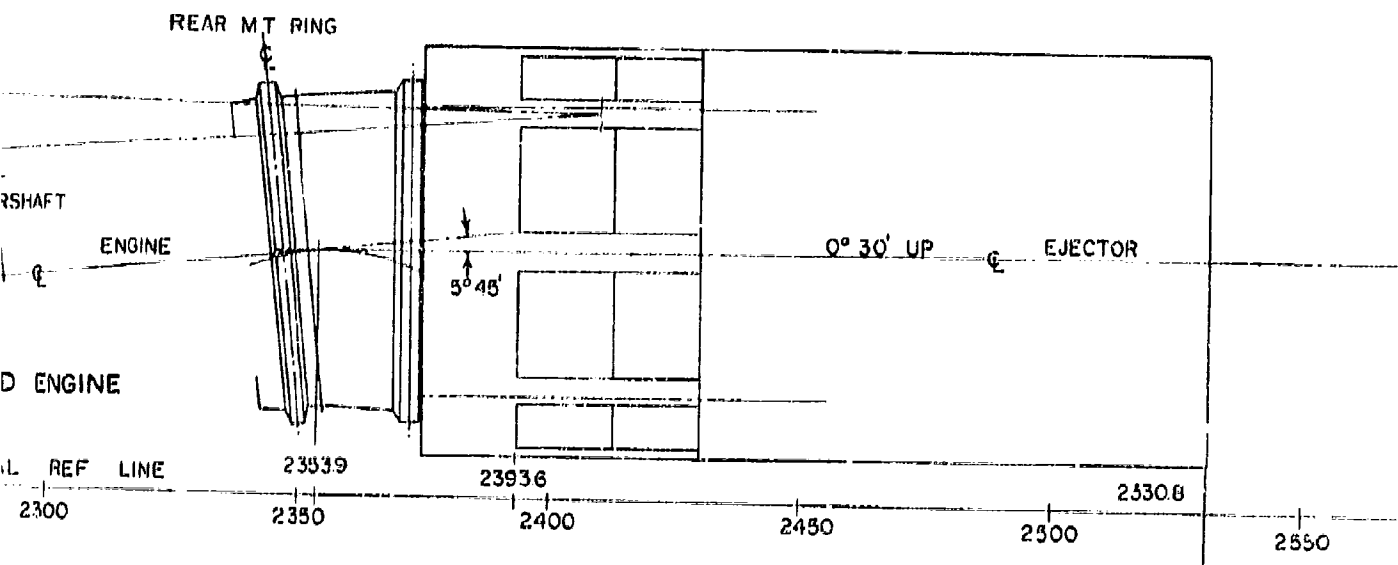
TYPICAL TURBOJET INST
EFFECT OF INCREASE

Figure 1

CONFIDENTIAL

CONFIDENTIAL

PWA-2600



TYPICAL TURBOJET INSTALLATION SHOWING
EFFECT OF INCREASED CANT ANGLE

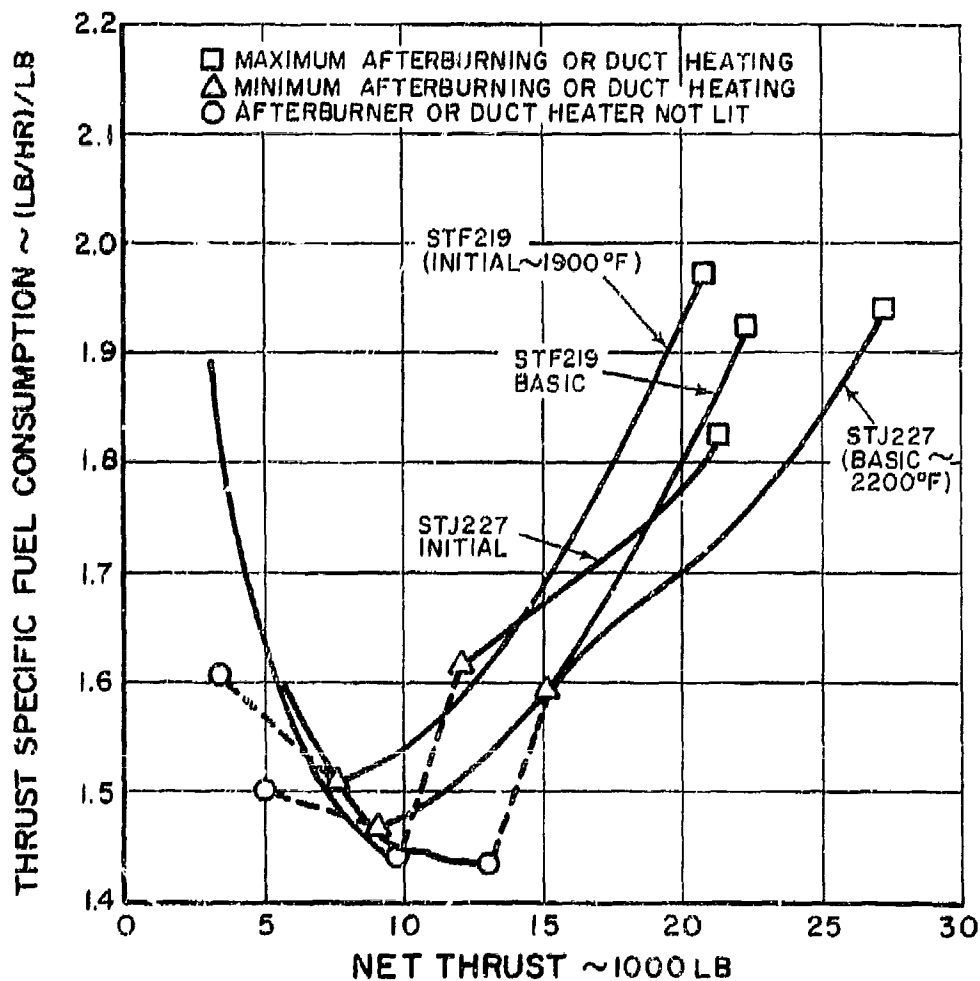
Figure 1-44

3

CONFIDENTIAL

UNCLASSIFIED AT 5 YEAR INTERVAL
DECLASSIFIED AFTER 15 YEARS
DATE 01-08-2000 BY 6032
THIS DOCUMENT CONTAINS INFORMATION THAT IS THE
PROPERTY OF THE UNITED STATES OF AMERICA
AND IS NOT TO BE RELEASED OR DISCLOSED TO THE
PUBLIC OR TO ANY OTHER PERSON OR ENTITY
WITHOUT THE EXPRESS WRITTEN PERMISSION OF THE
SECRETARY OF THE DEPARTMENT OF DEFENSE

US STANDARD ATMOSPHERE - 1962 (GEOM)
 RAM RECOVERY PER MIL-E-5008B
 65,000 FT PHASE II B MN 2.7
 BASE AIRFLOW



ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES
 AT MACH 2.7 AT 65,000 FEET

Figure 1-45

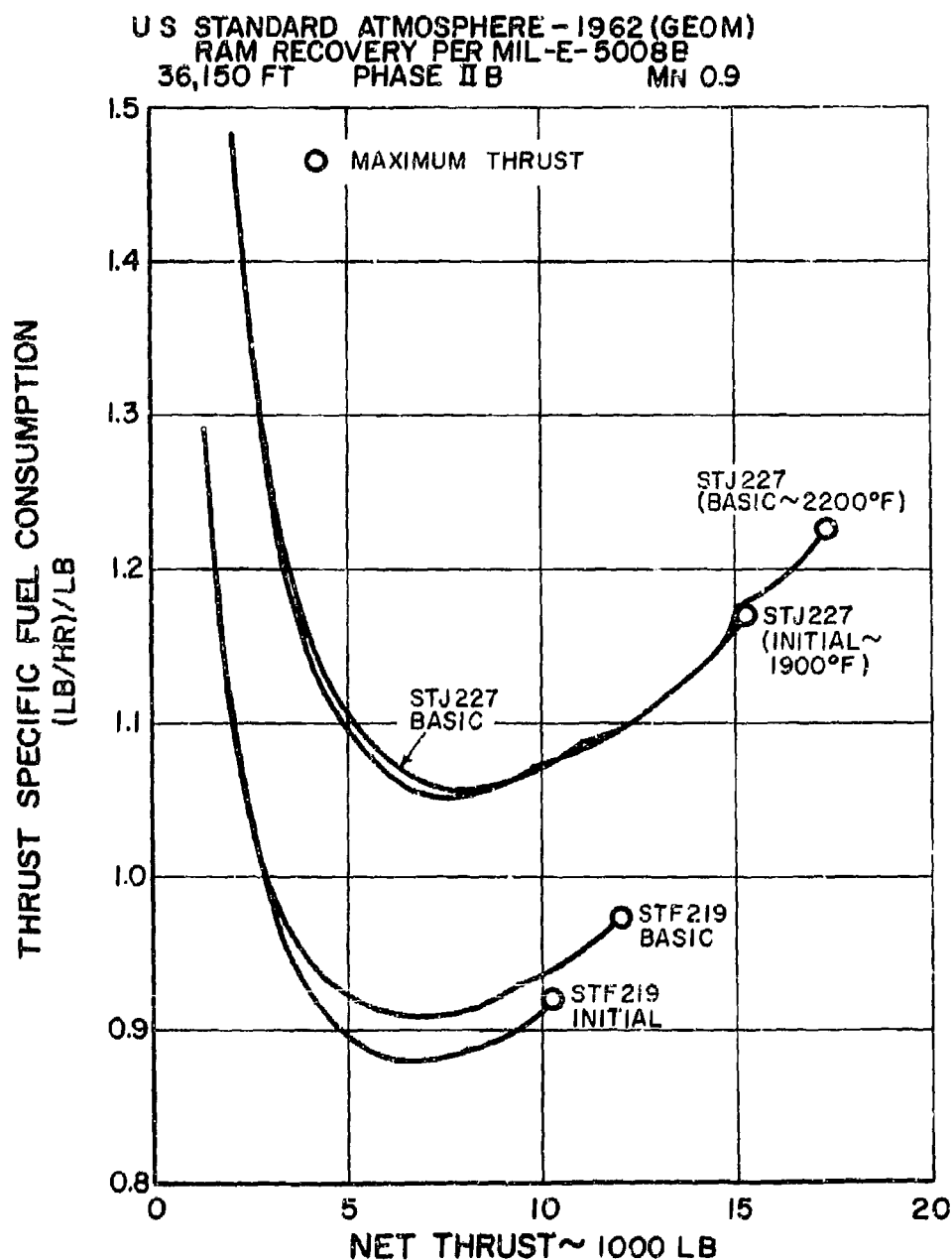
CONFIDENTIAL

DISSEMINATED AT 5 YEAR INTERVALS
 MAY 1970, AFTER 15 YEARS
 DATE: 05M 1985

THIS DOCUMENT CONTAINS INFORMATION RELATING TO
 NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
 MEANING OF THE Espionage Laws, TITLE 18, U.S.C.
 SECTION 793 AND 794, TRANSMISSION OF THE
 INFORMATION TO AN UNAUTHORIZED PERSON OR
 DISCLOSURE OF THE INFORMATION TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

CONFIDENTIAL

PWA-2600



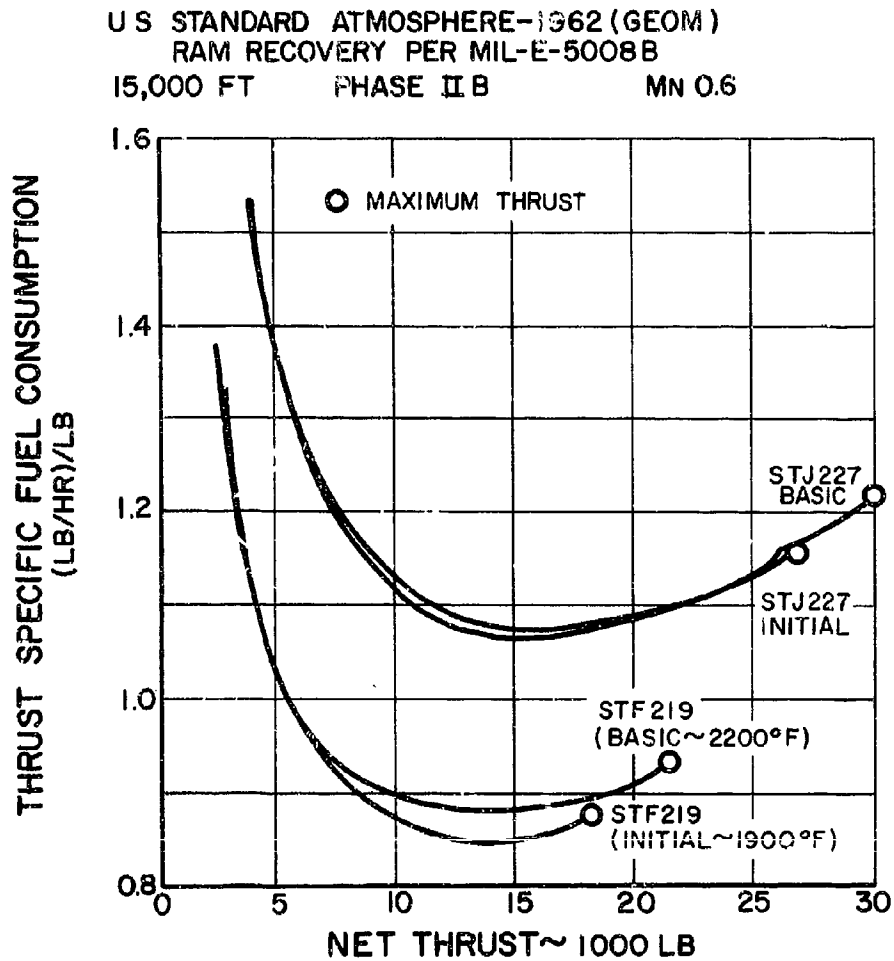
ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES
 AT MACH 0.9 AT 36,150 FEET

Figure 1-46

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
 DECLASSIFIED AFTER 15 YEARS
 DOD DPM 800010

THIS DOCUMENT CONTAINS INFORMATION SPECIFIC TO
 NATIONAL DEFENSE OF THE UNITED STATES WHICH IS
 HEREIN RELEASED TO THE PUBLIC BY THE NATIONAL ARCHIVES
 PURSUANT TO THE NATIONAL DEFENSE AUTHORITY ACT OF 1950
 AND THE NATIONAL DEFENSE INFORMATION ACT OF 1950
 AS AMENDED BY THE NATIONAL DEFENSE INFORMATION ACT OF 1976



ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES
AT MACH 0.6 AT 15,000 FEET

Figure 1-47

CONFIDENTIAL

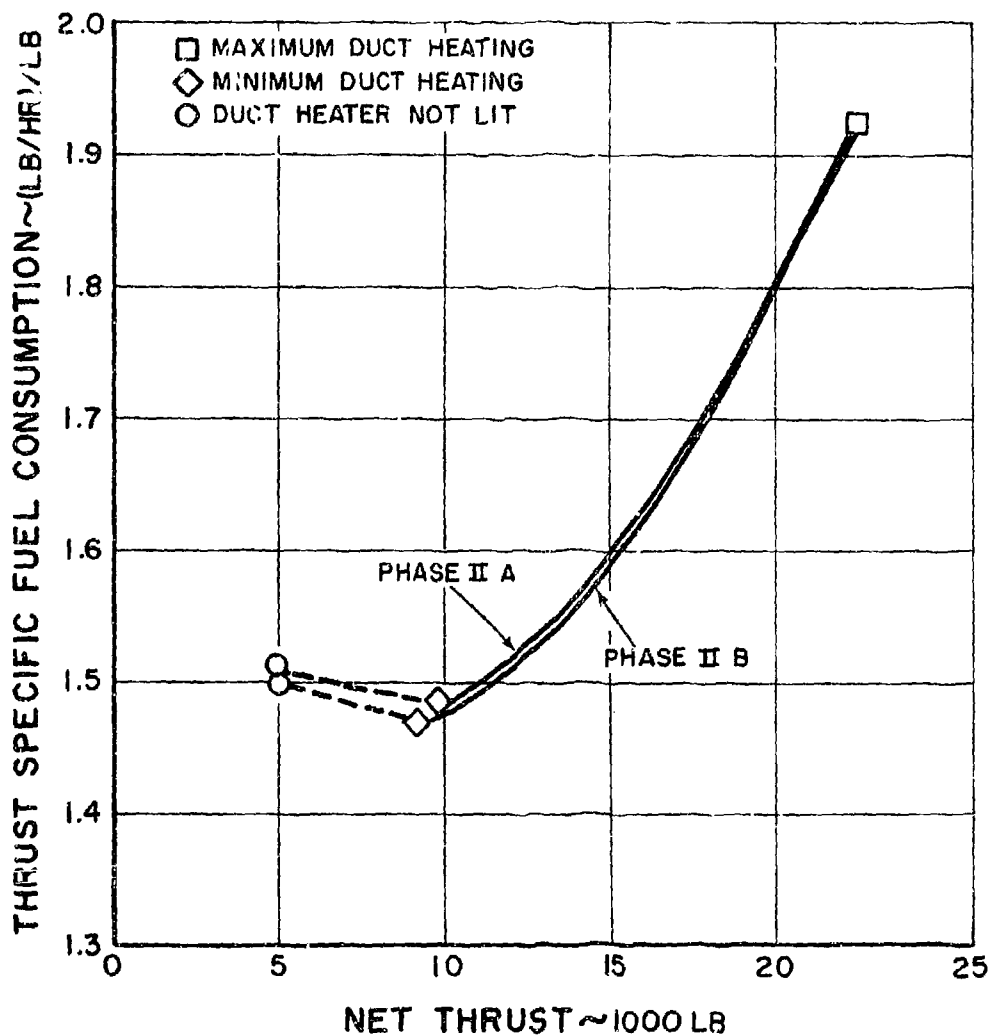
DOWNGRADED AT 3 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS
DOD DM 8200-10

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE SPY-GLASS LAWS, TITLE 18, U.S.C., SECTIONS 793 AND 794. TRANSMISSION OF THE REVELATION OF ITS CONTENTS TO ANY PERSON IS AN UNLAWFUL PERSON IS PROHIBITED BY LAW.

CONFIDENTIAL

PWA-2600

US STANDARD ATMOSPHERE - 1962 (GEOM)
 RAM RECOVERY PER MIL-E-5008B
 DESIGN TURBINE INLET TEMPERATURE 2200°F
 65,000 FT MN 2.7



ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA
 AND AFTER PHASE IIB AT MACH 2.7 AT 65,000 FEET

Figure 1-48

CONFIDENTIAL

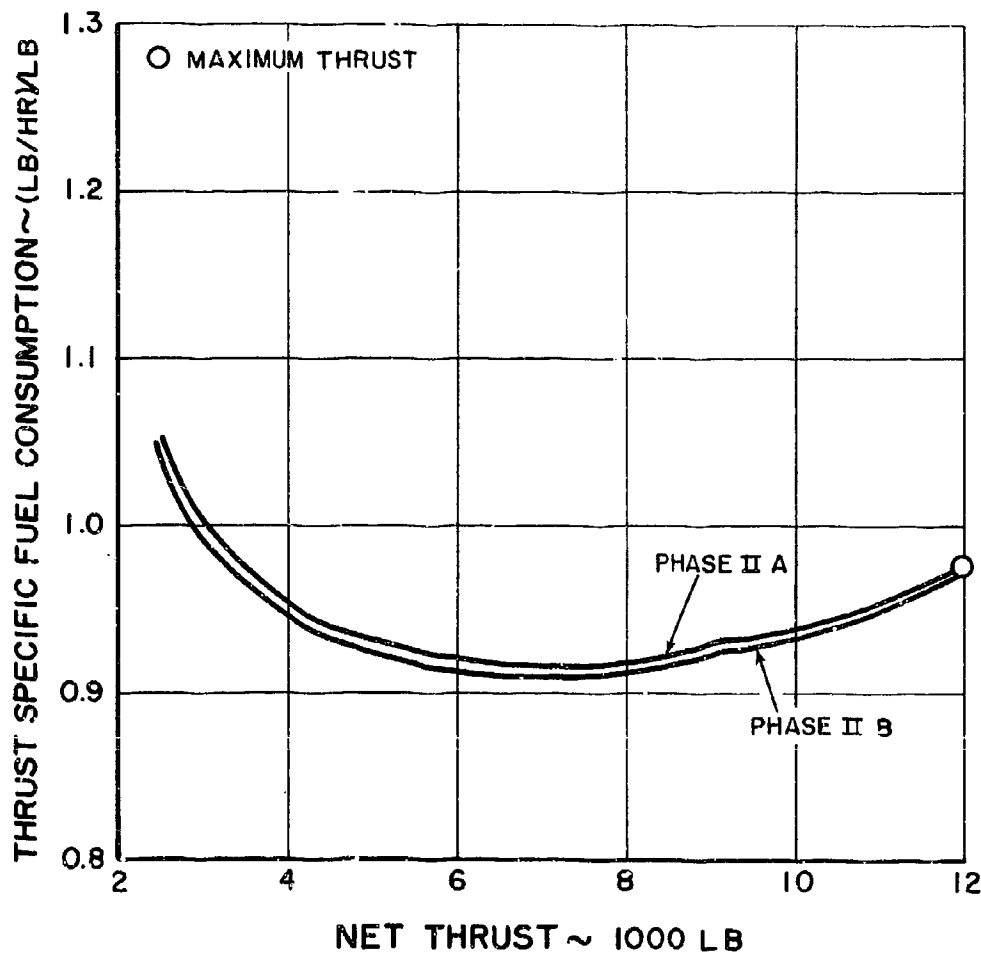
DOWNGRADED AT 3 YEAR INTERVALS
 DECLASSIFIED AFTER 15 YEARS
 GPO: 1990-10

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C. SECTION 793 AND 794, AND THE PROSECUTION OF THE REVELATION OF ITS CONTENTS IN ANY MANNER IS AN UNLAWFUL ACT, IS PROHIBITED BY LAW.

CONFIDENTIAL

PWA-2600

U S STANDARD ATMOSPHERE - 1962 (GEOM)
RAM RECOVERY PER MIL-E-5008 B
DESIGN TURBINE INLET TEMPERATURE 2300°F
36,150 FT Mn 0.9



ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA
AND AFTER PHASE IIB AT MACH 0.9 AT 36,150 FEET

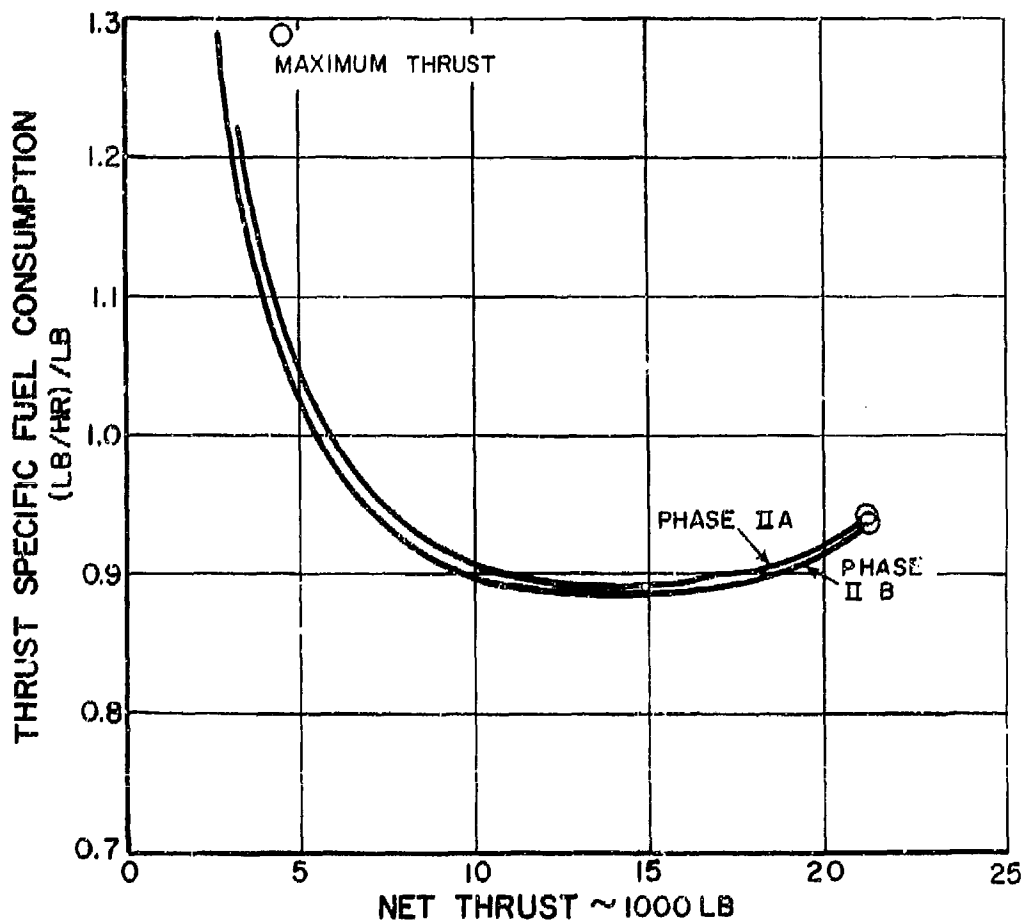
Figure 1-49

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS
DDO DM 820010

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES IN THAT THE
REVEALING OF THE INFORMATION HEREIN TO AN UN-
AUTHORIZED PERSON IS PROHIBITED BY LAW

US STANDARD ATMOSPHERE - 1962 (GEOM)
RAM RECOVERY PER MIL-E-5008 B
DESIGN TURBINE INLET TEMPERATURE - 2300 °F
15,000 FT Mn 0.6



ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA
AND AFTER PHASE IIB AT MACH 0.6 AT 15,000 FEET

Figure 1-50

CONFIDENTIAL

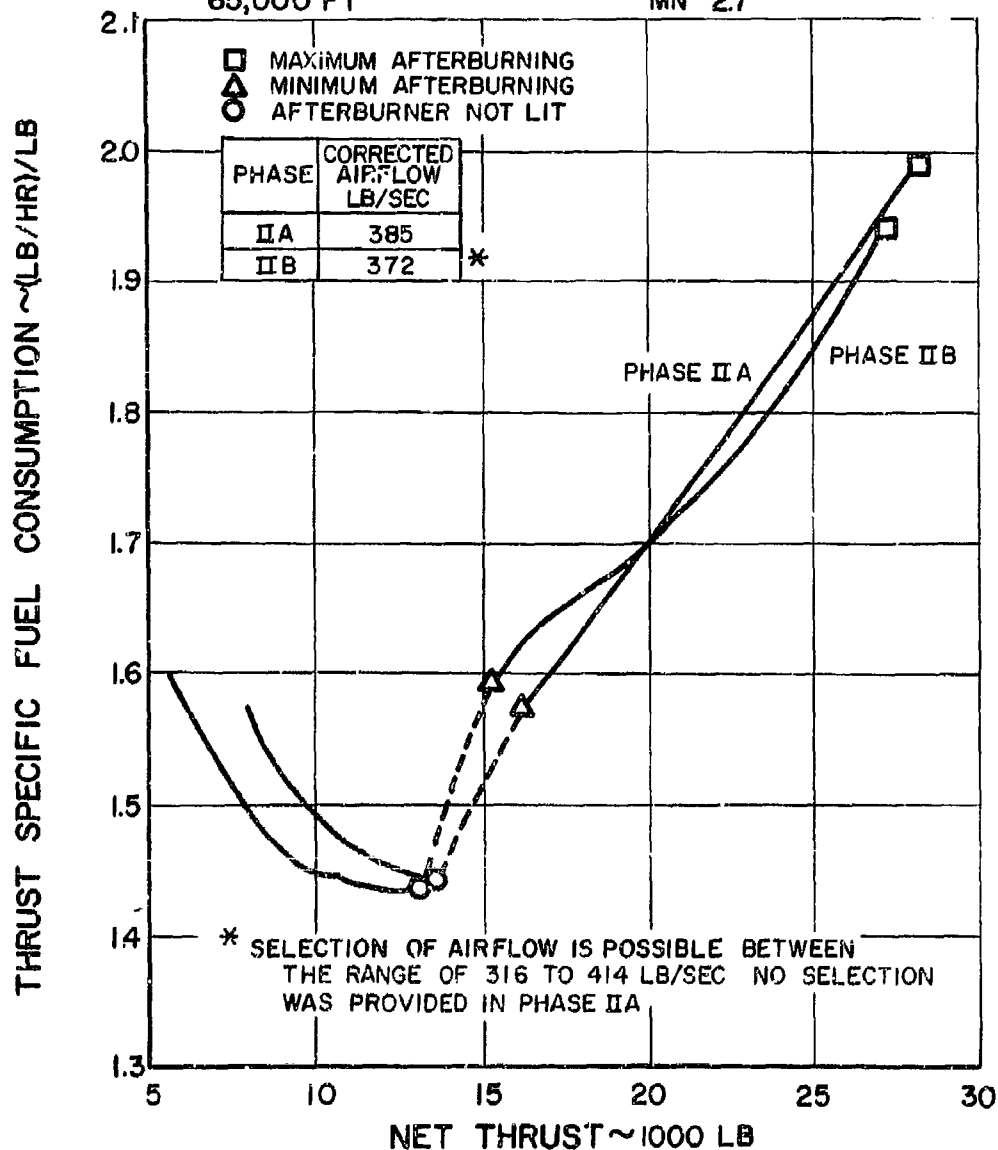
DOWNGRADED AT 2 YEAR INTERVALS
DECLASSIFIED AFTER 10 YEARS
DOD ODT 830010

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.,
SECTION 793 AND 794, THE TRANSMISSION OR THE
REVELATION OF ITS CONTENTS IN ANY MANNER TO
AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

CONFIDENTIAL

PWA-2600

US STANDARD ATMOSPHERE - 1962 (GEOM)
 RAM RECOVERY PER MIL-E-5008B
 DESIGN TURBINE INLET TEMPERATURE 2200°F
 65,000 FT MN 2.7



ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA
 AND AFTER PHASE IIB AT MACH 2.7 AT 65,000 FEET

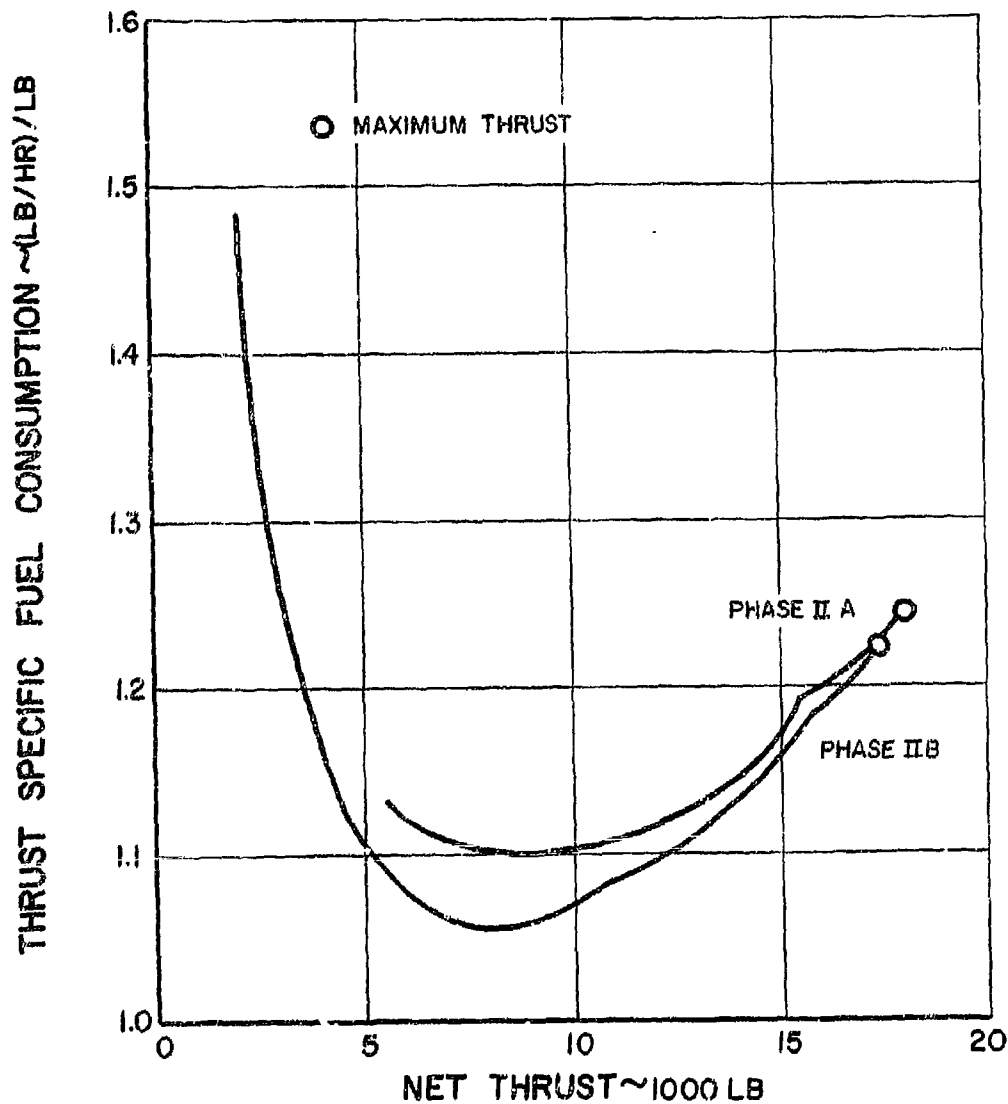
Figure 1-51

CONFIDENTIAL

SHOWGRADED AT 1 YEAR INTERVALS
 DECLASSIFIED AFTER 10 YEARS
 UNLESS OTHERWISE SPECIFIED

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE INTELLIGENCE INFORMATION ACT OF 1950 (50 USC 1701-1705). IT IS HEREBY DECLARED TO BE UNCLASSIFIED EXCEPT WHERE SHOWN OTHERWISE. DATE 10-1-80 BY 1045

U S STANDARD ATMOSPHERE - 1962 (GEOM)
 RAM RECOVERY PER MIL-E-5008B
 DESIGN TURBINE INLET TEMPERATURE 2300°F
 36,150 FT Mn 0.9



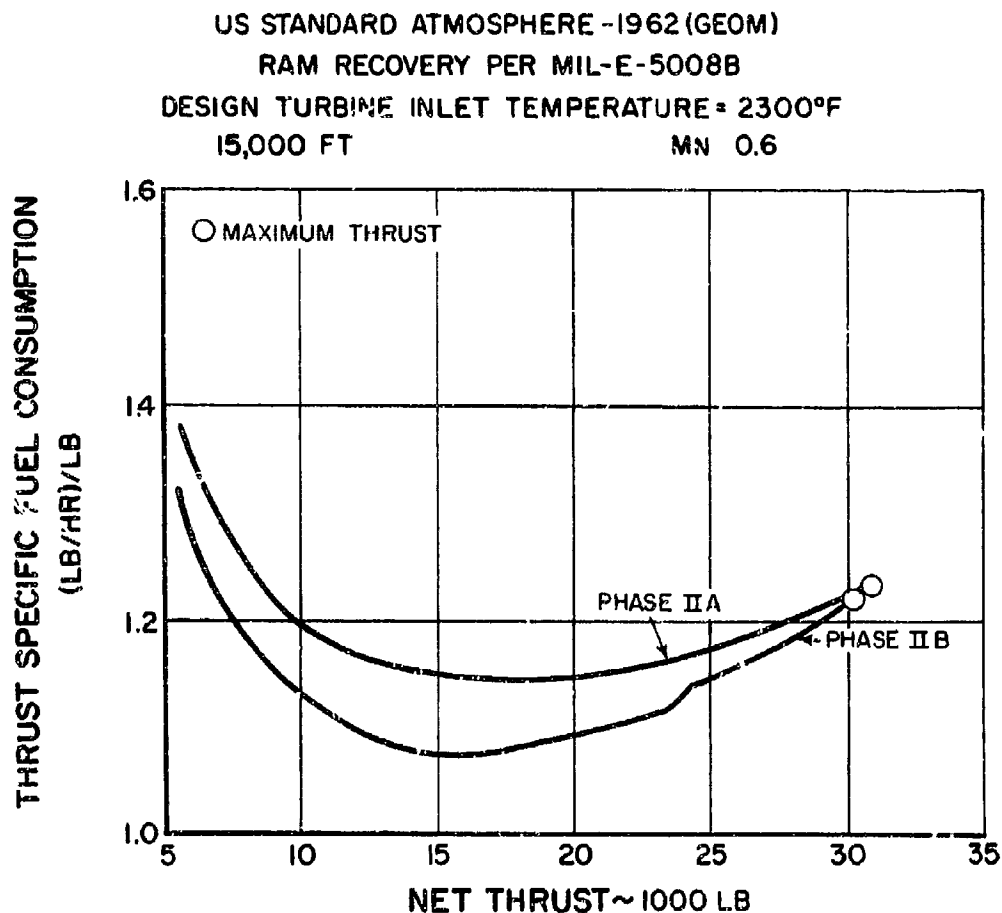
ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA
 AND AFTER PHASE IIB AT MACH 0.9 AT 36,150 FEET

Figure 1-52

CONFIDENTIAL

DOWNGRADED AT 5 YEAR INTERVALS
 DECLASSIFIED AFTER 15 YEARS
 DOWNGRADING 500010

THIS DOCUMENT CONTAINS INFORMATION OF A TECHNICAL NATURE, THE DISCLOSURE OF WHICH TO THE PUBLIC COULD BE DETRIMENTAL TO THE NATIONAL DEFENSE. IT IS THE POLICY OF THE UNITED STATES GOVERNMENT TO MAKE AVAILABLE TO THE PUBLIC INFORMATION OF A TECHNICAL NATURE, THE DISCLOSURE OF WHICH TO THE PUBLIC COULD BE DETRIMENTAL TO THE NATIONAL DEFENSE, AT THE EARLIEST PRACTICABLE DATE.



ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA
AND AFTER PHASE IIB AT MACH 0.6 AT 15,000 FEET

Figure 1-53

CONFIDENTIAL

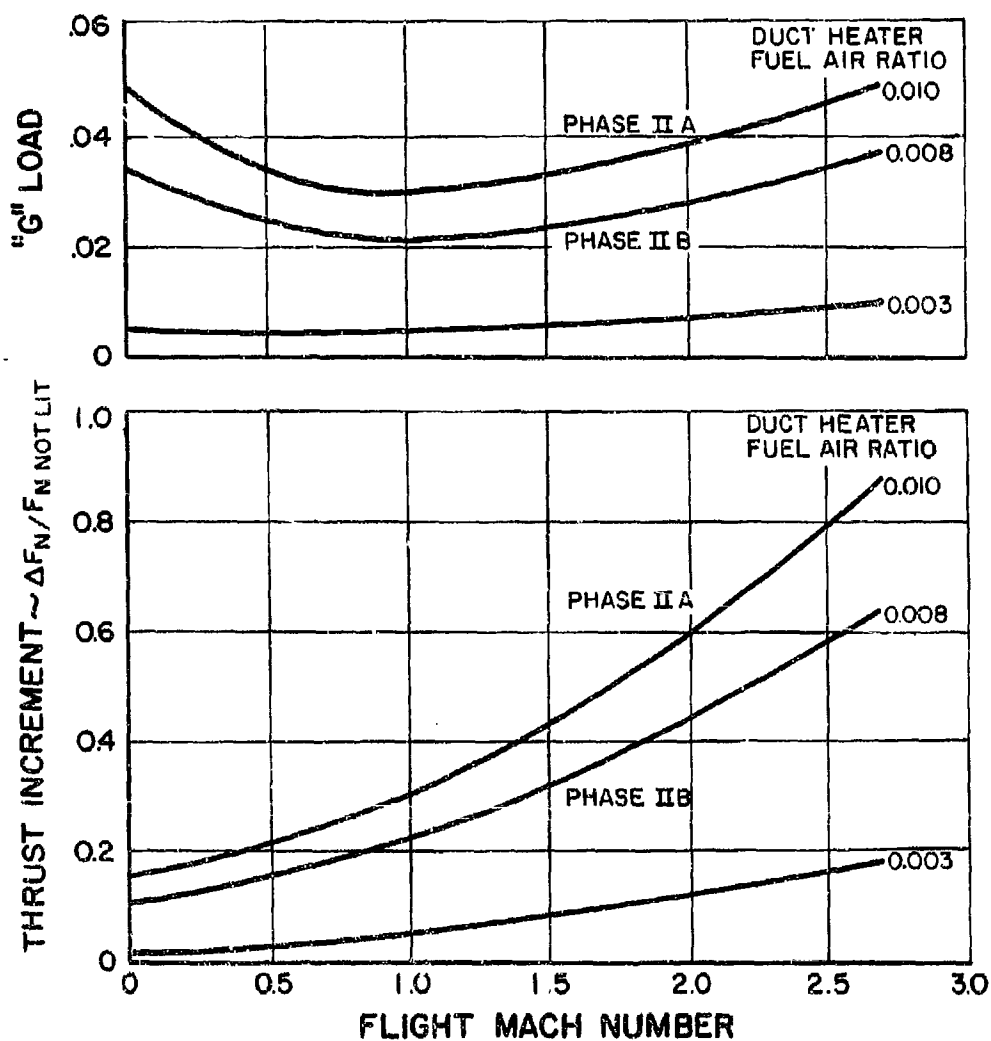
DOWNGRADED AT 5 YEAR INTERVALS
DECLASSIFIED AFTER 15 YEARS

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE
MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C.,
SECTION 793 AND 794, AND IT IS PROHIBITED TO
REPRODUCE OR TO TRANSMIT IN ANY MANNER TO
AN UNAUTHORIZED PERSON OR AGENCY BY ANY

CONFIDENTIAL

PWA-2600

U.S. STANDARD ATMOSPHERE-1962 (GEOM)
 RAM RECOVERY PER MIL--E- 5008B
 TOGW = 450,000 LBS



LOADING AND THRUST INCREMENT PRODUCED BY
 LIGHTING DUCT HEATER OF STF219 ENGINE

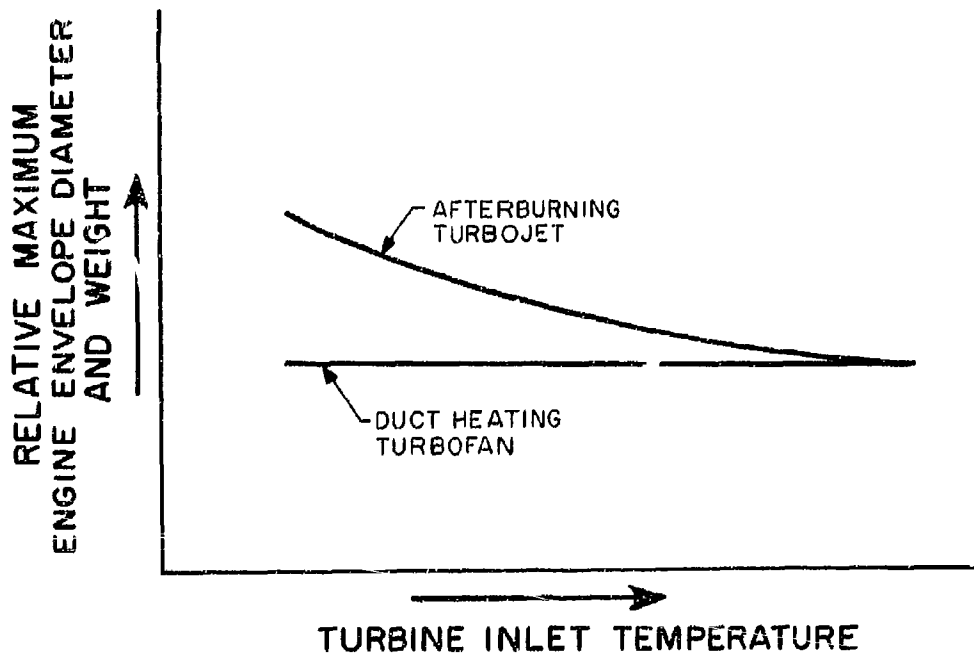
Figure 1-54

CONFIDENTIAL

REPRODUCED AT 5 YEAR INTERVALS
 DECLASSIFIED AFTER 15 YEARS

500 001 0000 12

THIS DOCUMENT CONTAINS INFORMATION RELATING TO THE
 NATIONAL DEFENSE OF THE UNITED STATES AND IS
 HEREIN DESIGNATED AS "CONFIDENTIAL" IN ACCORDANCE WITH
 THE PROVISIONS OF THE NATIONAL DEFENSE AUTHORITY ACT
 OF 1950 (50 USC 4051) AND THE EXECUTIVE ORDER
 OF 1955 (12 USC 1701) WHICH REQUIRE THE
 PROTECTION OF THIS INFORMATION FROM DISCLOSURE TO THE
 PUBLIC.

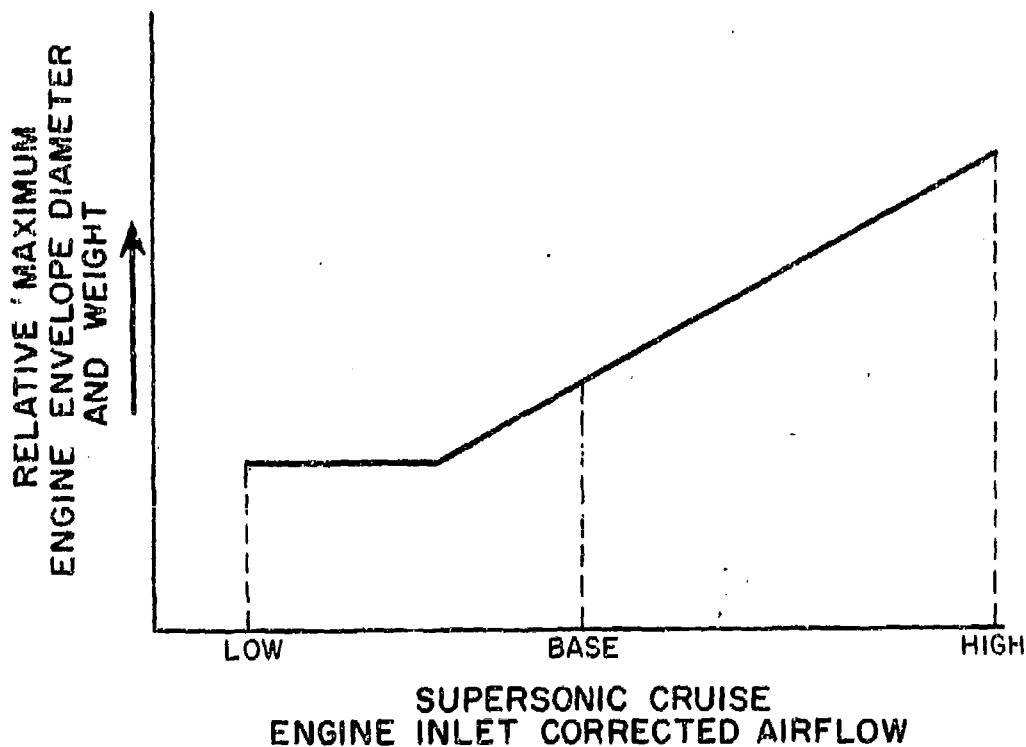


EFFECT OF TURBINE INLET TEMPERATURE ON ENGINE WEIGHT AND MAXIMUM DIAMETER

Figure 1-55

CONFIDENTIAL

PWA-2600



EFFECT OF ENGINE SUPERSONIC CRUISE INLET AIRFLOW
ON ENGINE WEIGHT AND MAXIMUM DIAMETER

Figure 1-56

CONFIDENTIAL

CLASSIFIED BY 3-14-1980
DECLASSIFIED BY 3-14-1980
DATE 10-10-80

THIS DOCUMENT CONTAINS INFORMATION THAT IS UNCLASSIFIED EXCEPT WHERE SHOWN OTHERWISE. IT IS THE POLICY OF THE NATIONAL ARCHIVES TO MAKE AVAILABLE TO THE PUBLIC INFORMATION THAT IS UNCLASSIFIED.