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Report No. PWA-2600 Volume 1
Date: 30 June 1965

SUPERSONIC TRANSPORT AIRCRAFT ENGINE

PHASE II-B DEVELOPMENT PROGRAM

FINAL REPORT (U)

Prepared Under Contract FA-SS-65-18

Period Covered 1 January through 30 June 1965

Approved by C. W. Bristol, Jr.
C. W. Bristol, Jr.
Project Engineer

Pratt & Whitney Aircraft DIVISION OF UNITED AIRCRAFT CORPORATION

EAST HARTFORD • CONNECTICUT

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.

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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.

ITEM 1

INSTALLATION COORDINATION

ITEM 1

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 COMPANIES

1. INSTALLATION COORDINATION WITH THE BOEING AIRPLANE COMPANY

a. 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.

- 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 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 adversly 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.

(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

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, and exhaust gas targeting information, as well as nacelle capture area and a 5° inlet angle relative to the horizontal reference datum, was supplied by Boeing and was used as the basis of these studies. An octagonal ejector was selected to further reduce the base drag area at the wing trailing edge. The flat side on top of the octagonal ejector offered possibilities of closer coupling to the wing surface than the circular section of the Phase IIA ejector.

A number of new arrangements of the engine and the octagonal ejector relative to the wing were investigated. These arrangements are summarized below.

- Figure 1-3 (Phase IIB configuration) shows how an octagonal ejector with the same equivalent area as the 76.00 inch diameter ejector of Phase IIA reduces the base drag area to 398 sq. in.

- As in Figure 1-3 , Figure 1-4 shows an arrangement with the engine and ejector tangent to the wing. The rear engine mount plane was moved aft 40.00 inches (relative to its Phase IIA position) which required a 33.50 inch extension to the engine inlet case. The ejector was canted at an angle of 5° at the rear mount plane. The base drag area was reduced to 337 sq. in.

- Figure 1-5 shows an arrangement with the rear engine mount plane moved aft 30.00 inches. The ejector was canted 8° at the rear mount plane and the engine/ejector was inserted 4.00 inches into the wing. A 31.00 inch extension was required. The base drag area was reduced to 320 sq. in.

- Figure 1-6 shows an arrangement identical to Figure 1-5 except that the ejector cant point was moved aft to the primary nozzle plane. The cant angle is 6°. A 40.00 inch extension was required for this. The base drag area was reduced to 230 sq. in.

- Figure 1-7 shows an arrangement with the rear mount plane moved aft 40.00 inches and the ejector canted at 8° at the primary nozzle plane. The engine/ejector was inserted 4.00 inches into the wing. A 40.00 inch extension was required. The base drag area was reduced to 150 sq. in.

- 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° 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° 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° 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° 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° 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 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° 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. <u>Discussion of STJ 227B Turbojet Installations</u>

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° 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° 40'. This required a 13.80 inch extension.

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[illegible] 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 [illegible].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

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 STJ 227 accessory arrangement.

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 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.

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.

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.

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.

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	298.24	75.00	121.15
2300	Low Flow	56.30	296.26	70.00	121.15

B. ENGINE CYCLE STUDIES

1. INTRODUCTION

Two engines were studied during Phase IIB, 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

	STJ227		STF219	
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	388	388
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 IIA, 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 IIB, the sonic overpressure limits were increased from 2.0 to 2.5 psf for transonic acceleration and from 1.5 to 1.7 psf for supersonic cruise. This shift resulted in the engines 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	62500	67000
Nominal Airflow Size (lb/sec)	525	525	650	650
Cruise Turbine Inlet Temperature (°F)	1900	2200	1900	2200
Transonic Acceleration, Mach 1.2, at 45000 Ft				
Thrust (lb)	20600	21400	18400	19800
Specific Fuel Consumption (lb/ hr /lb)	1.91	1.83	1.88	1.84
Supersonic Cruise Mach 2.7 at 65000 Ft, Thrust = 9800 lb				
Specific Fuel Consumption (lb/ hr /lb)	1.44	1.45	1.54	1.48
Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb				
Specific Fuel Consumption (lb/ hr /lb)	1.05	1.06	0.88	0.91
Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb				
Specific Fuel Consumption (lb/ hr /lb)	1.25	1.25	0.95	0.96

TABLE I-4

Performance of STF-219 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)	56600	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)	9190	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

TABLE 1-5

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

	Basic Engine		Initial Engine
	Phase IIA	Phase IIB	Phase IIB
Supersonic Cruise Turbine Inlet Temperature (°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

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 unaugmented 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 0.008, 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 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 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

Parameter	Accuracy (Percent)
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
$A_{j\,duct}$, 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 $A_{j\,duct}$ 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 at a maximum in the same direction. It is most probable that the error will be that predicted by the root-mean-square method of error summation. The probable error predicted by this method is shown in Table 1-7.

Table 1-7

Probable Error in Inflight Performance Measurement

Flight Condition	Power Setting	Probable Error (Percent) Total Thrust	TSFC
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

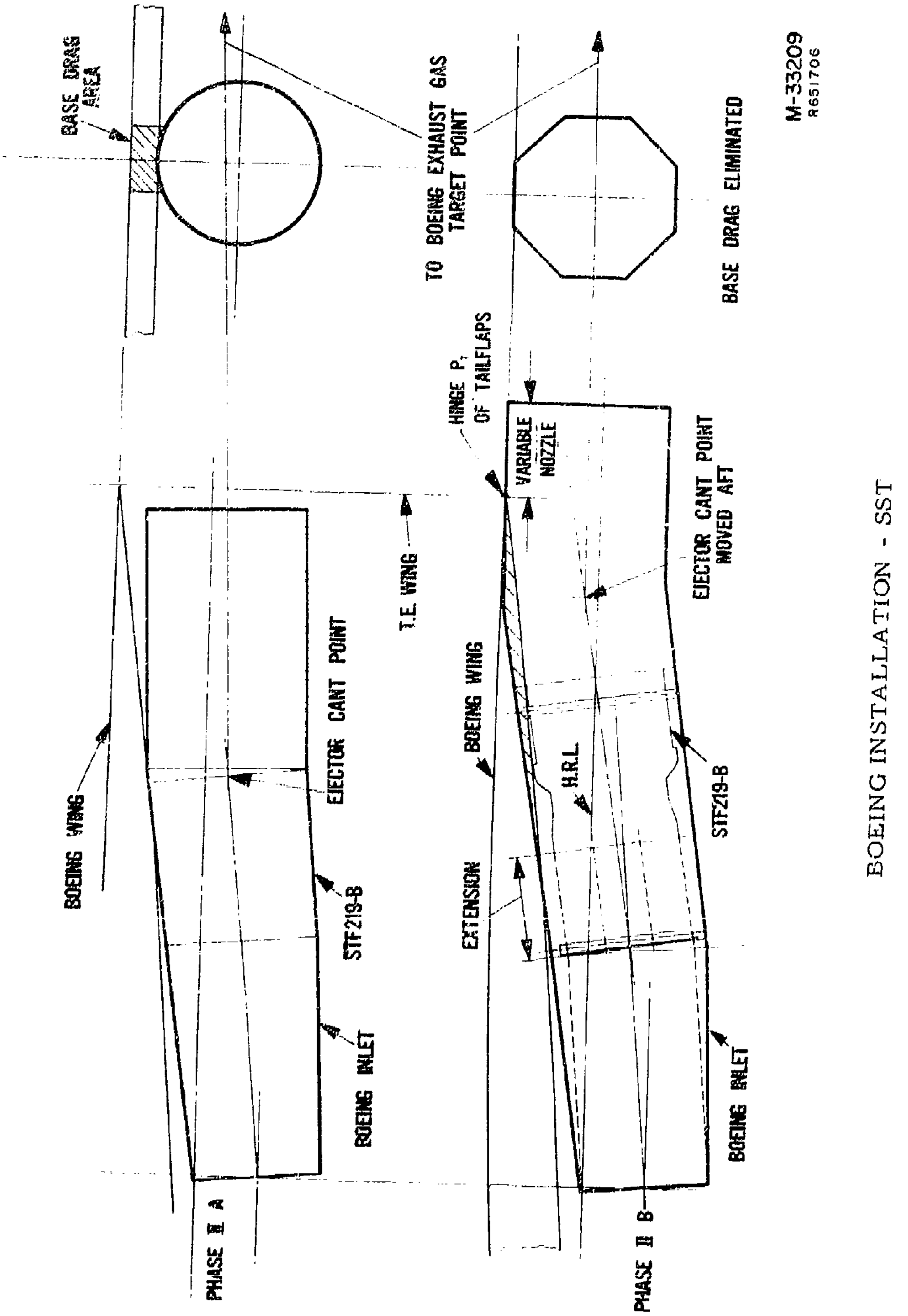
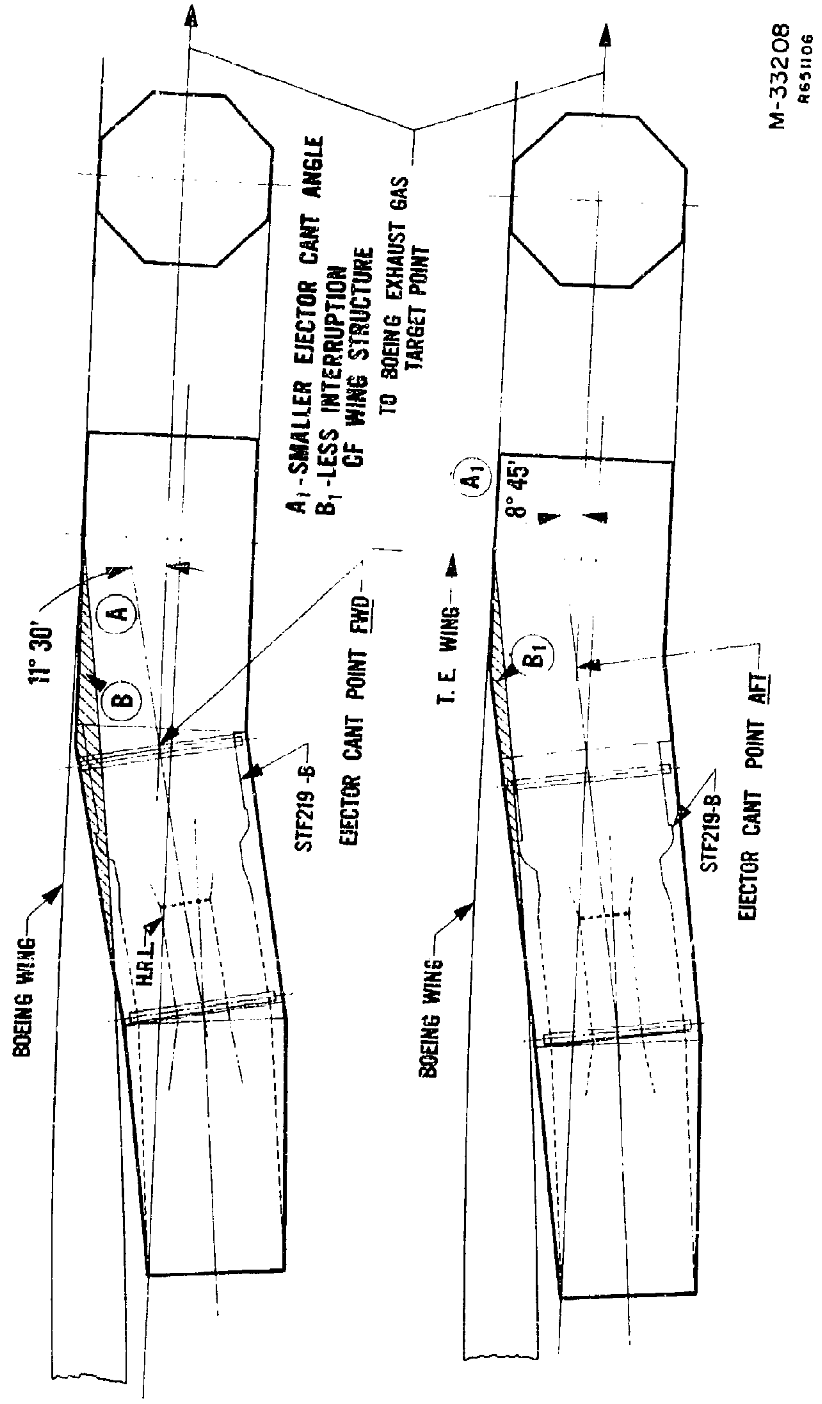
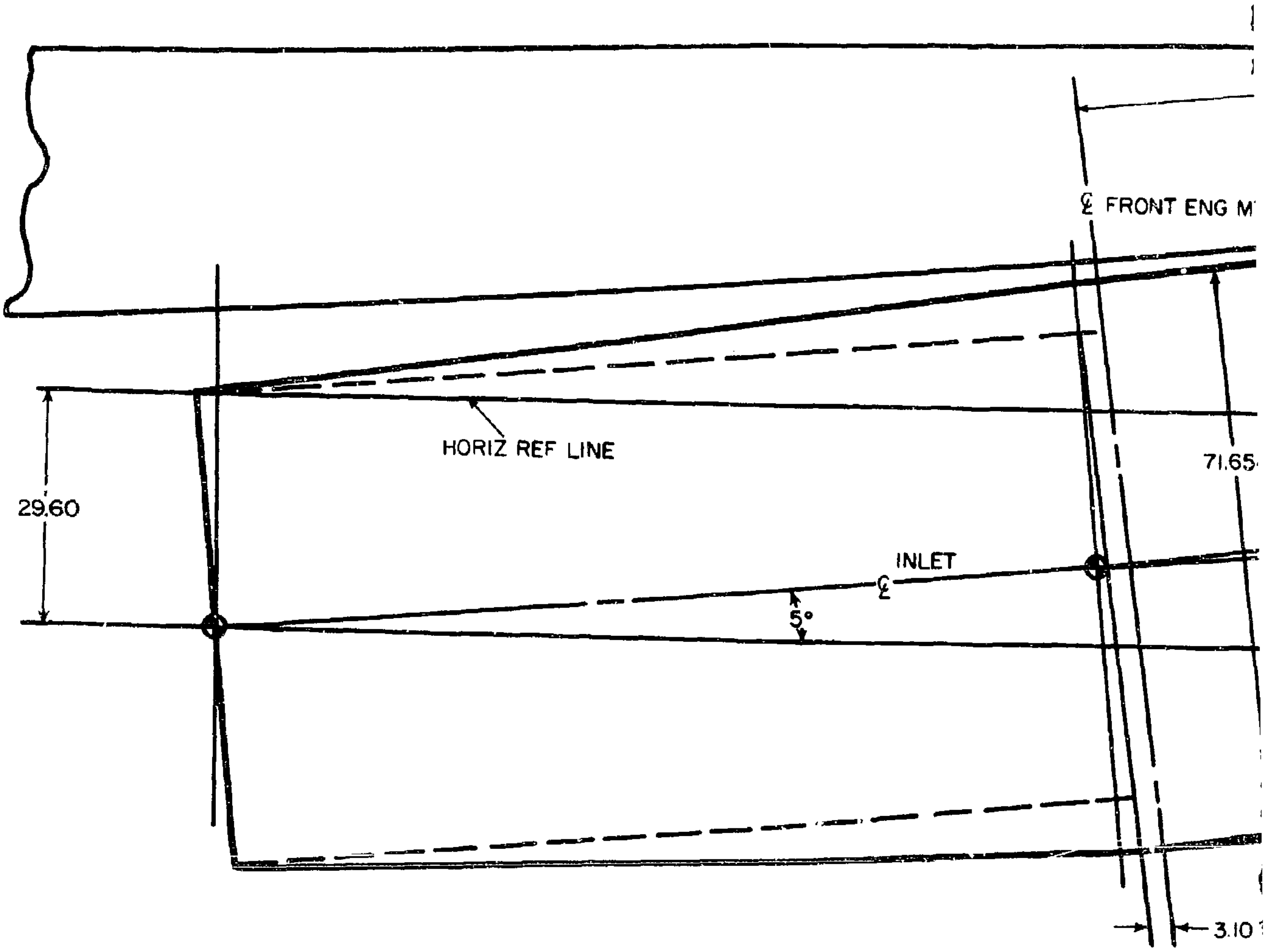

M-33209
R651706
BASE DRAG AREA
TO BOEING EXHAUST GAS TARGET POINT
BASE DRAG ELIMINATED
HINGE P_T OF TAILFLAPS
VARIABLE NOZZLE
EJECTOR CANT POINT MOVED AFT
T.E. WING
EJECTOR CANT POINT
BOEING WING
BOEING WING
H.R.L.
STF219-B
STF219-B
EXTENSION
BOEING INLET
BOEING INLET
PHASE II A
PHASE II B
BOEING INSTALLATION - SST

Figure 1-1

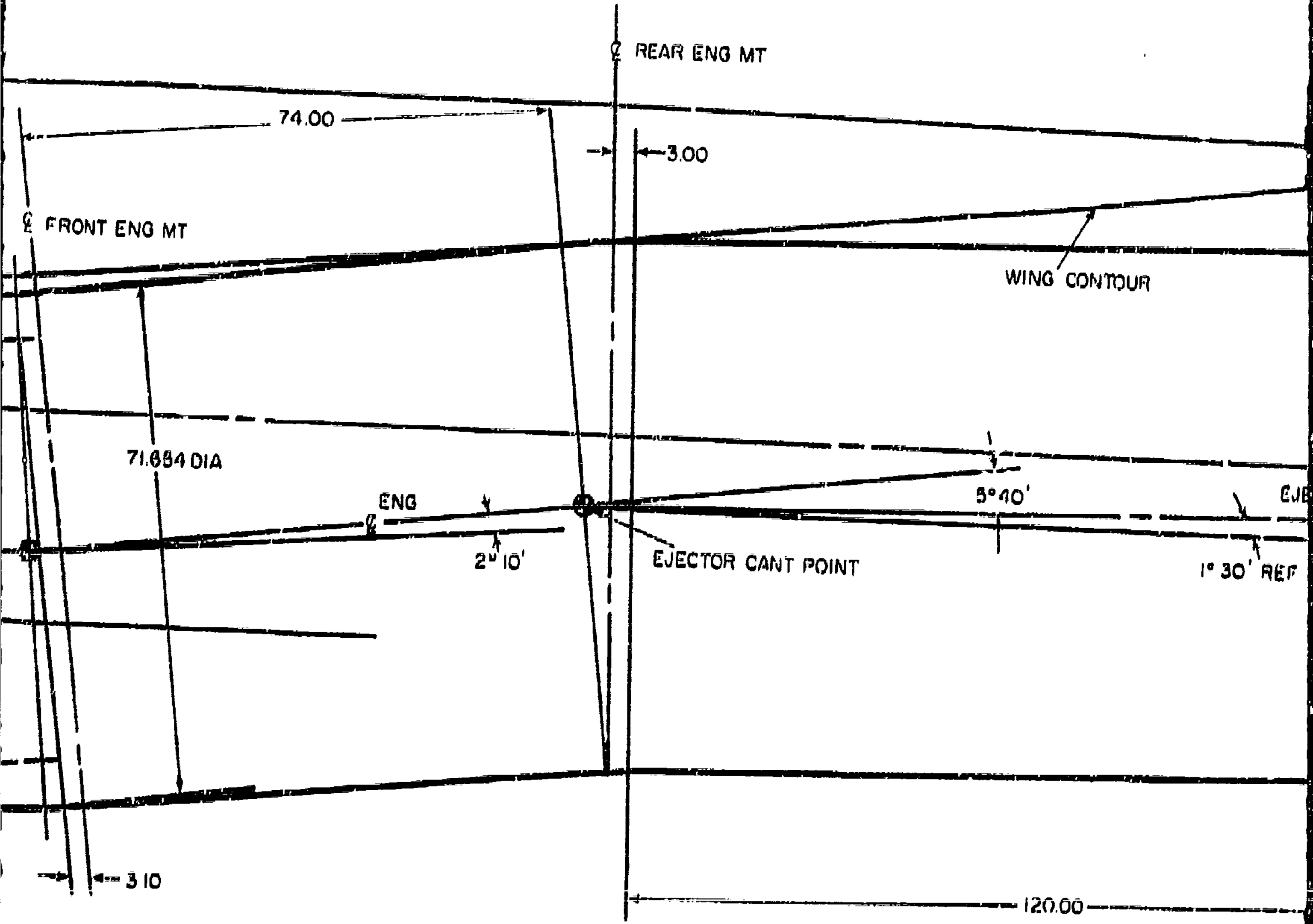


EFFECT OF MOVING THE CANT POINT AFT

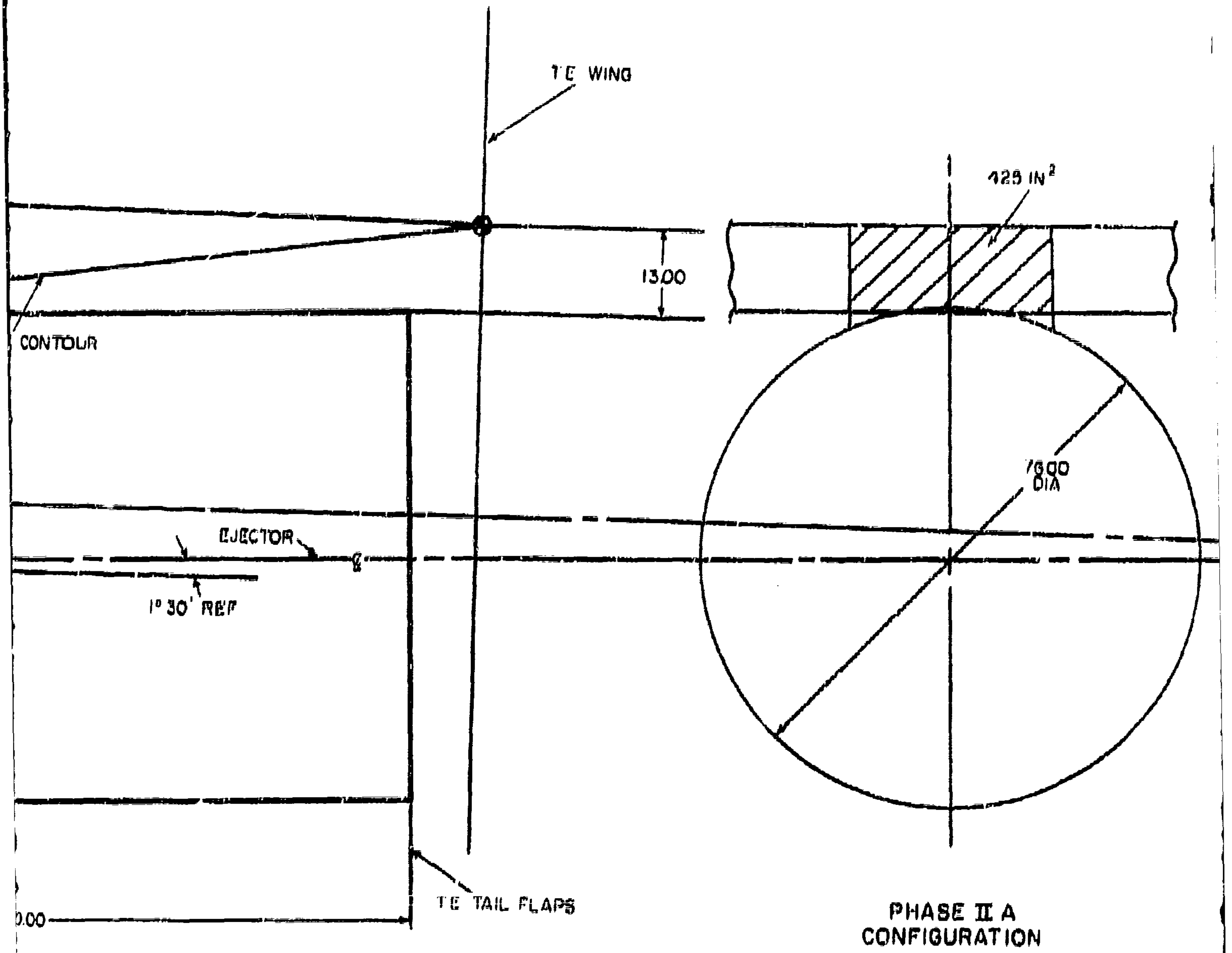
Figure 1-2



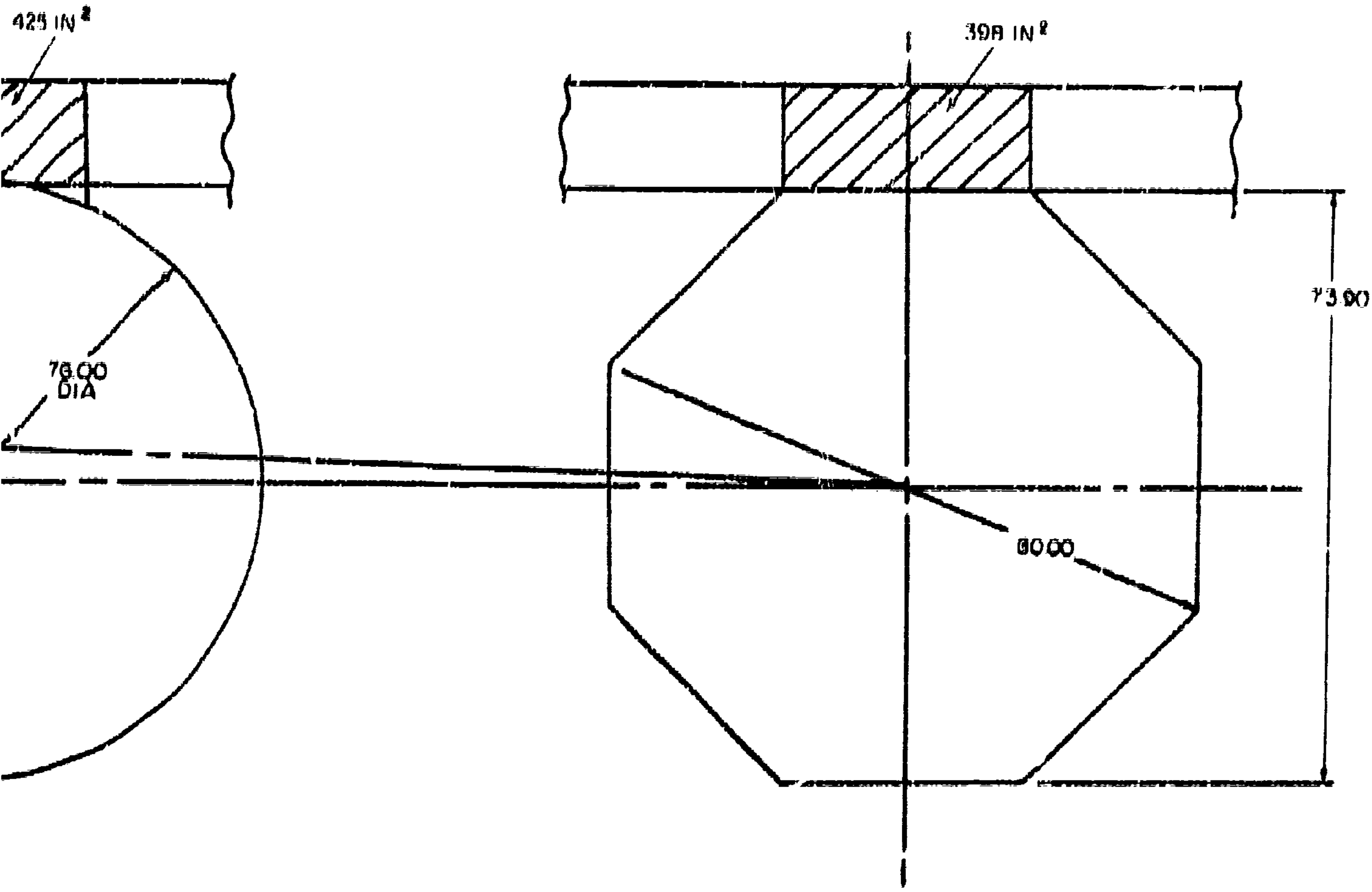
1


℄ REAR ENG MT
74.00
3.00
℄ FRONT ENG MT
WING CONTOUR
71.684 DIA
ENG ℄
5°40'
EJECTOR CANT POINT
2° 10'
1° 30' REF
3.10
120.00

2


T E WING
425 IN²
13.00
CONTOUR
7600
DIA
EJECTOR
1° 30' REF
T E TAIL FLAPS
0.00
PHASE II A
CONFIGURATION

3

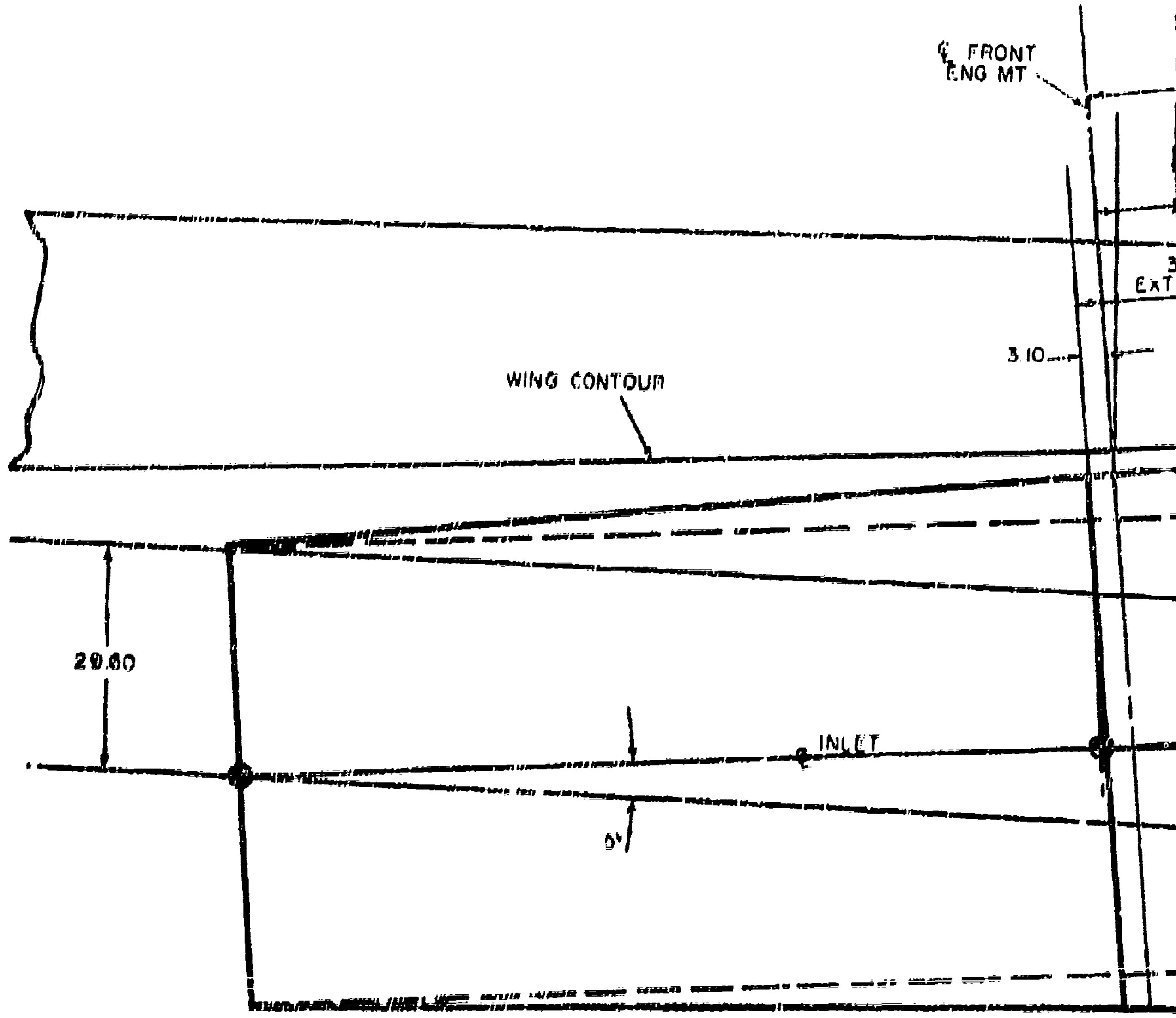


PHASE II B
CONFIGURATION

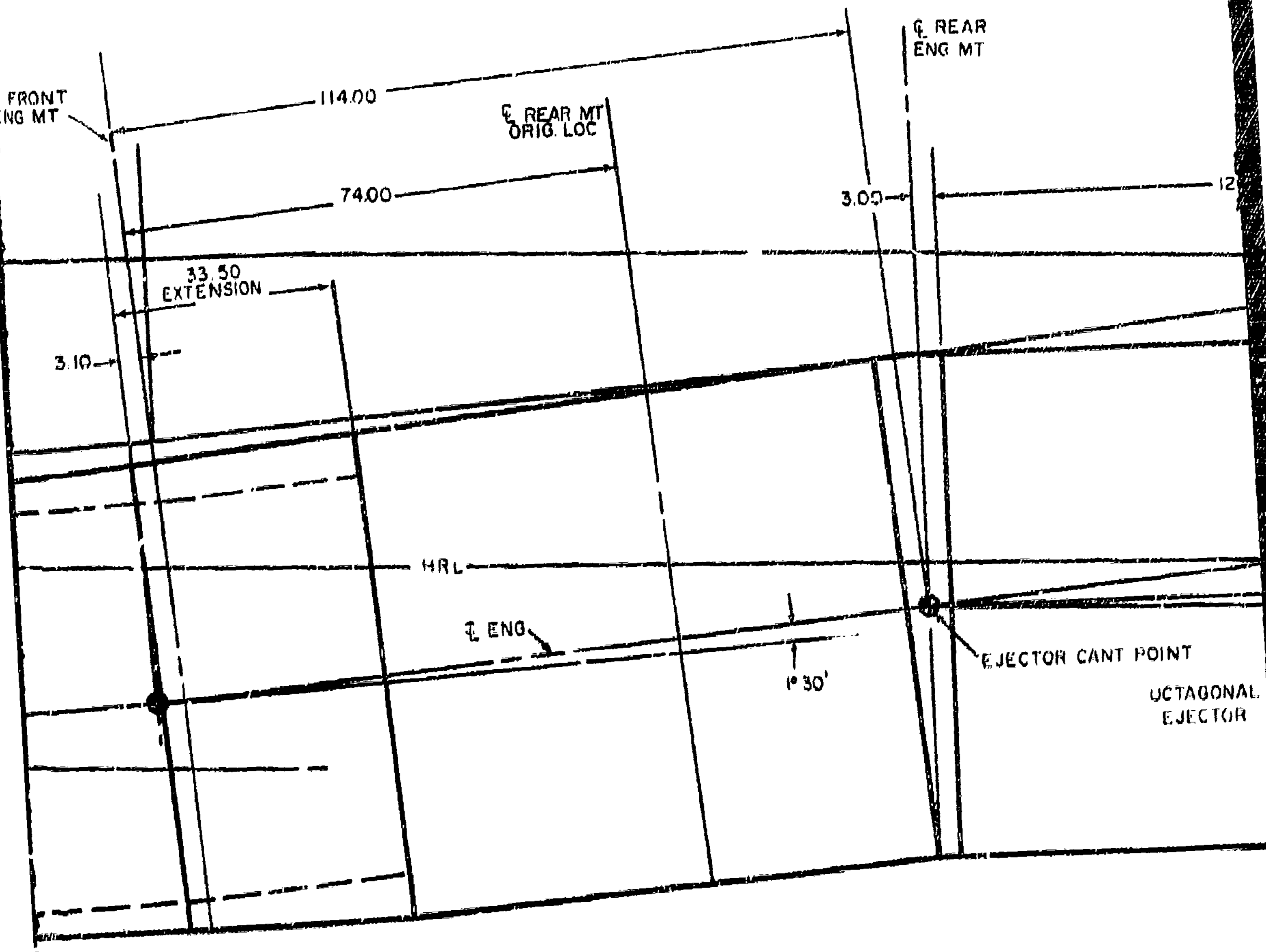
ENGINE + EJECTOR TANGENT TO WING
EJECTOR CANTED AT REAR ENG. MOUNT PLANE
NOTE: ENGINE MOUNT PLANES ARE
NONPARALLEL & T. E. TAIL FLAPS
ARE FWD. OF T. E. WING
BASE DRAG AREA 425 in² (EQUIV. ROUND)
BASE DRAG AREA 398 in² (OCT.)

STF219B 600 LBS. /SEC. TURBOFAN

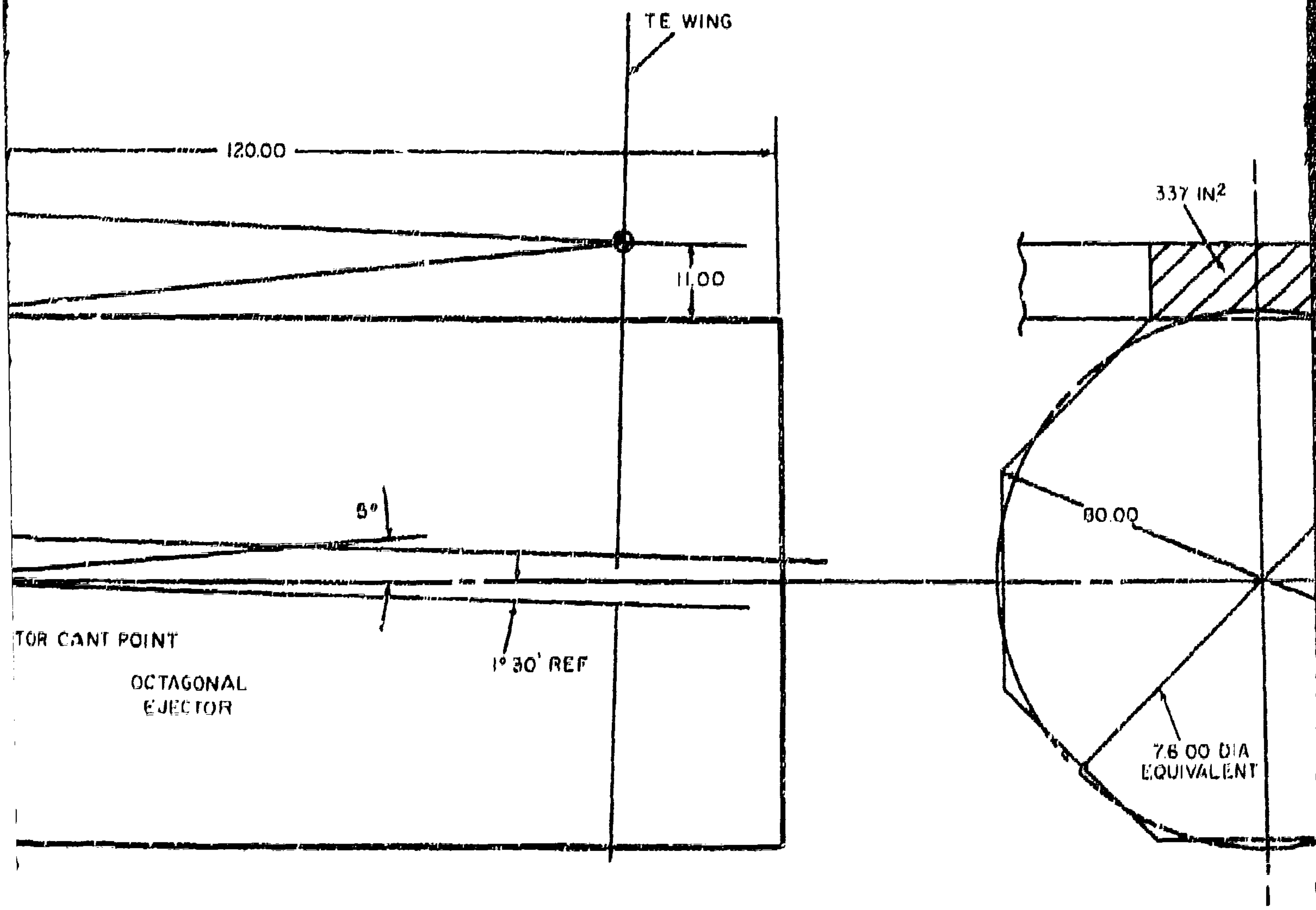
Figure 1-3



1


FRONT
ENG MT
114.00
℄ REAR MT
ORIG. LOC
℄ REAR
ENG MT
74.00
3.00
12
33.50
EXTENSION
3.10
HRL
℄ ENG
EJECTOR CANT POINT
1° 30'
OCTAGONAL
EJECTOR

2

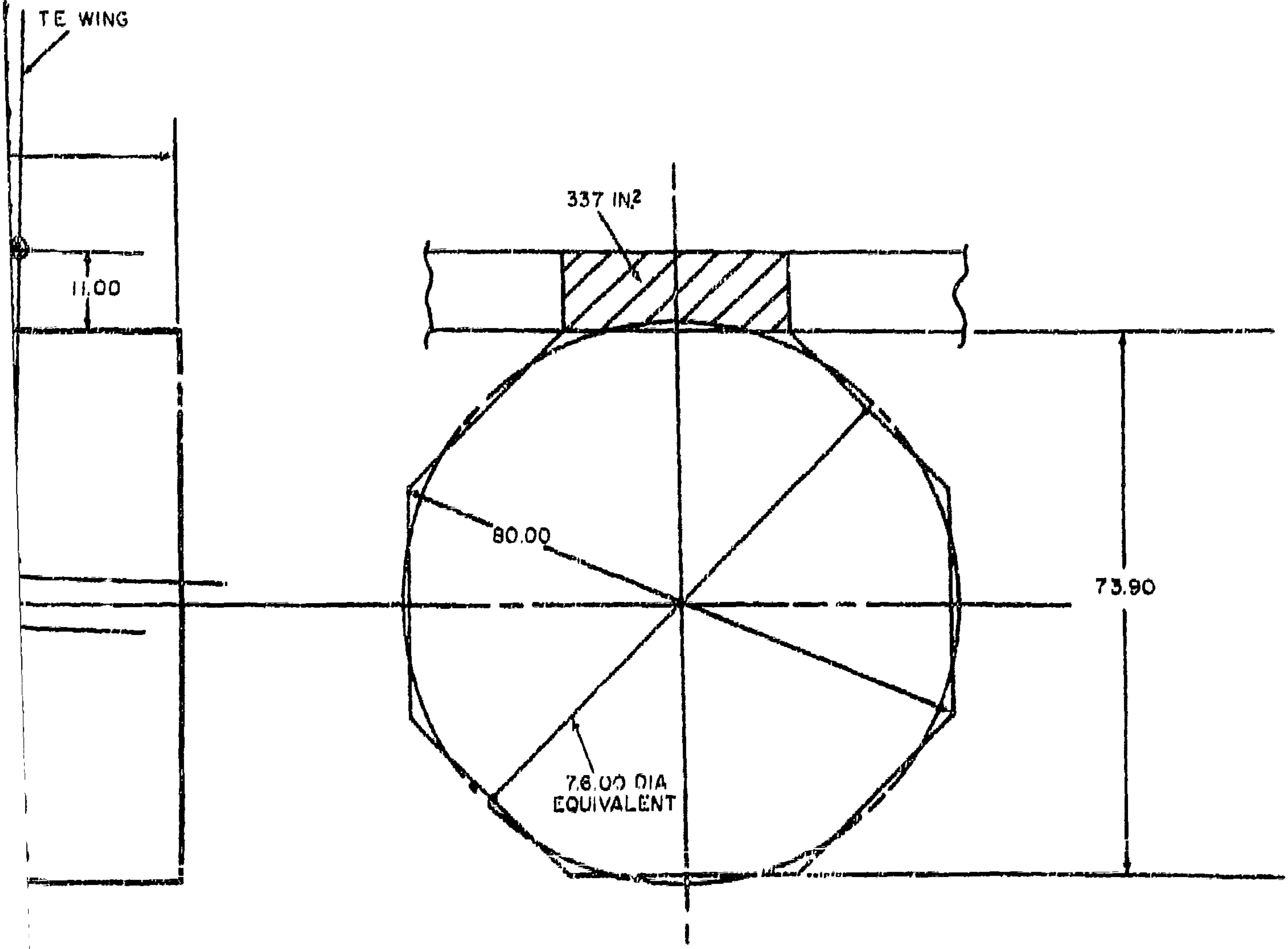


ARRANGEMENT SHOWING R
MOVED AFT 40.00
ENGINE & EJECTOR ARE TA
EJECTOR CANTED AT REA
PLANE
ENGINE, MOUNT PLANES
BASE DRAG AREA 337 in^2

STF219B 600 LBS./SE

Figure 1

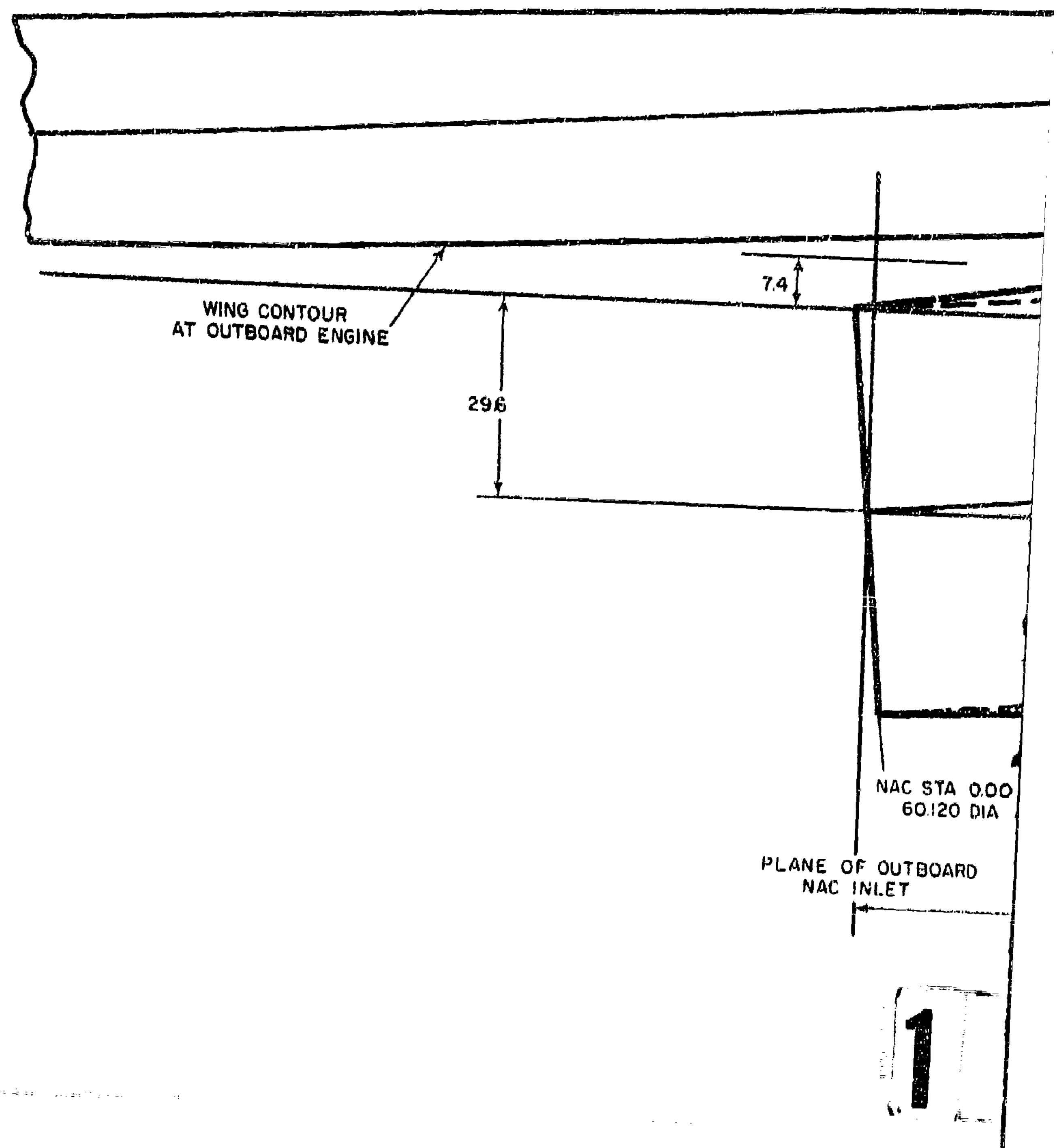
3

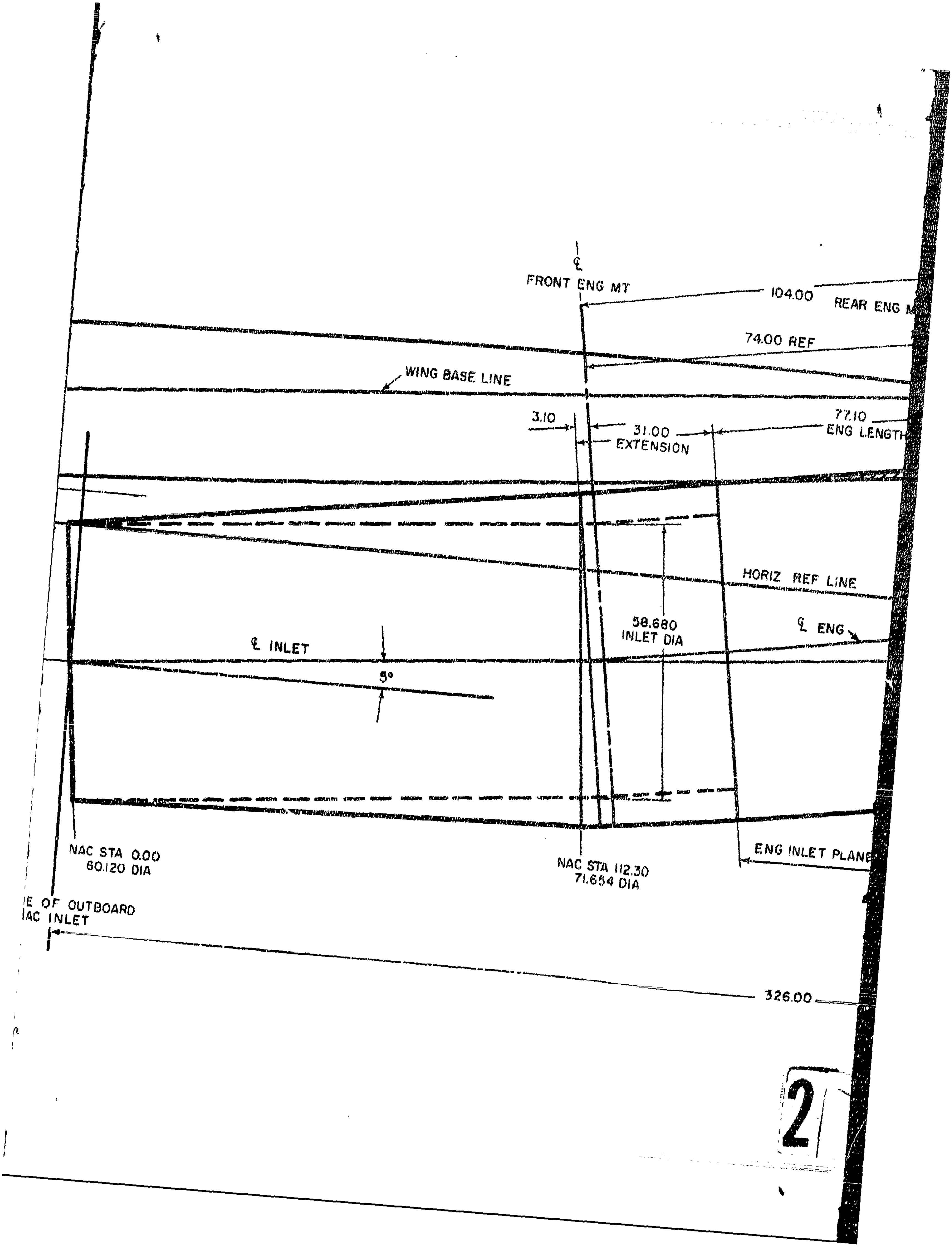


ARRANGEMENT SHOWING REAR ENG. MOUNT
MOVED AFT 40.00
ENGINE & EJECTOR ARE TANGENT TO WING
EJECTOR CANTED AT REAR ENG. MOUNT
PLANE
ENGINE. MOUNT PLANES ARE NONPARALLEL
BASE DRAG AREA 337 in^2

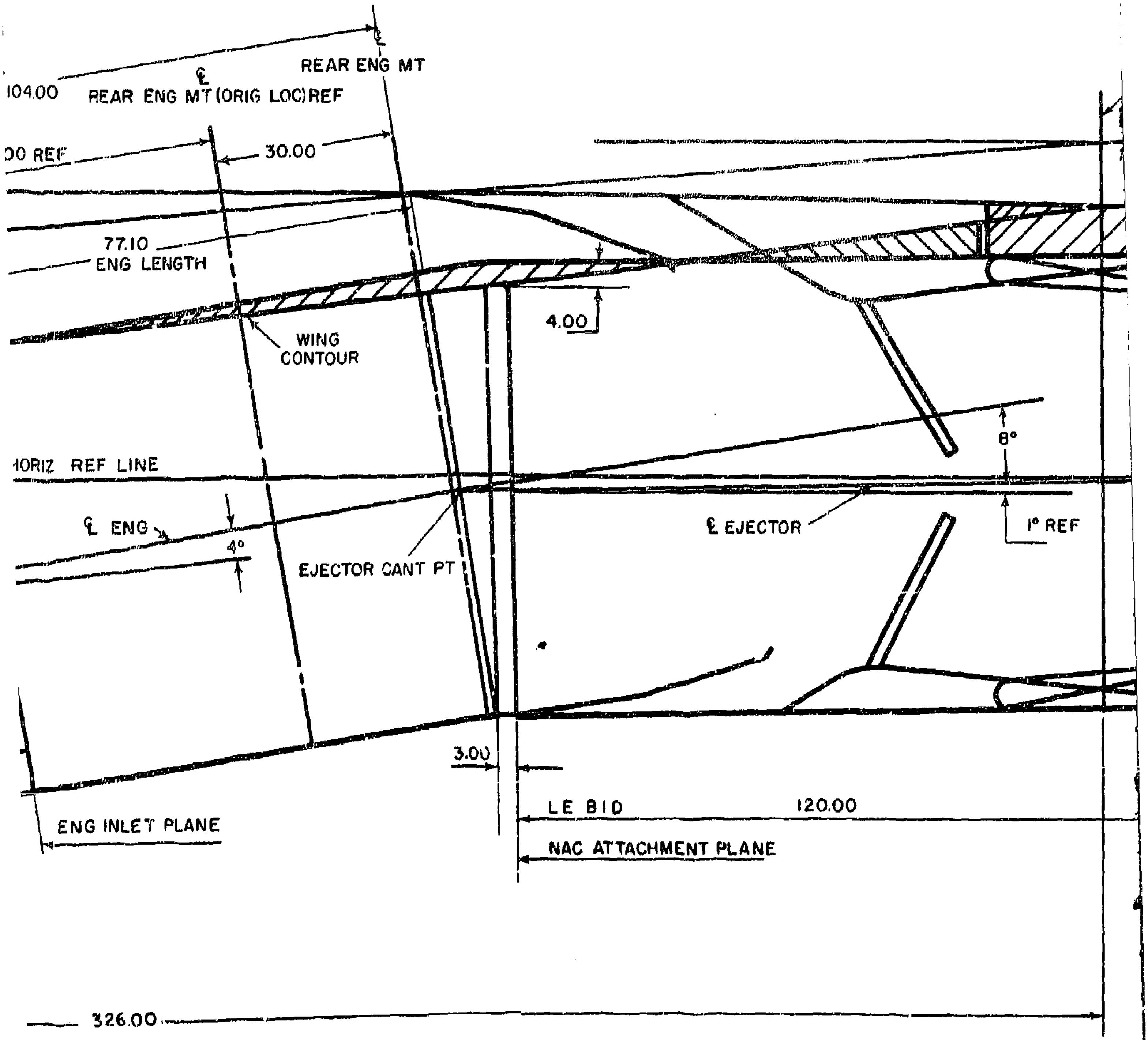
STF219B 600 LBS./SEC. TURBOFAN

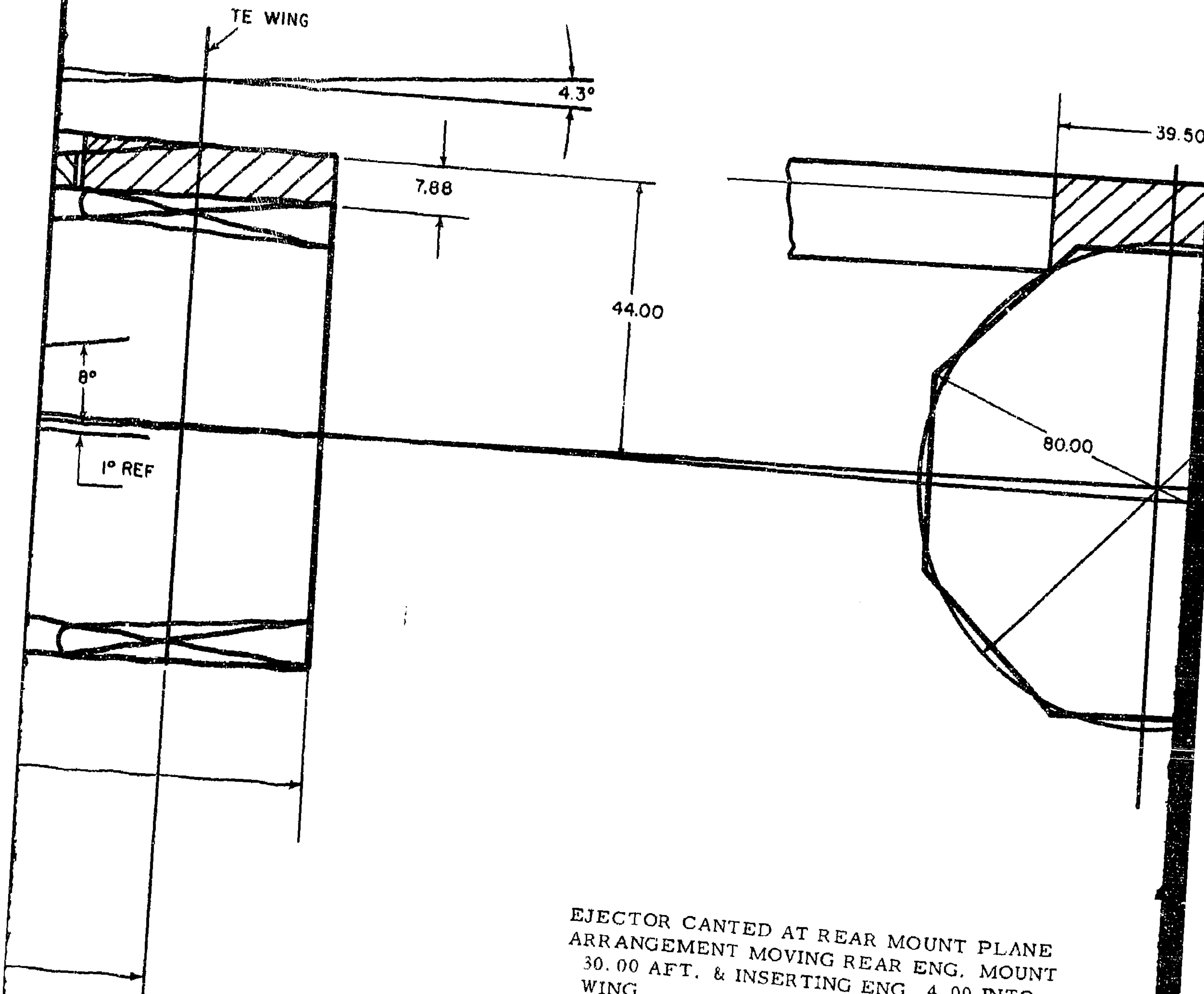
Figure 1-4


WING CONTOUR
AT OUTBOARD ENGINE
7.4
29.6
NAC STA 0.00
60.120 DIA
PLANE OF OUTBOARD
NAC INLET
1



FRONT ENG MT
104.00
REAR ENG M
74.00 REF
WING BASE LINE
3.10
31.00
EXTENSION
77.10
ENG LENGTH
HORIZ REF LINE
58.680
INLET DIA
℄ ENG
℄ INLET
5°
NAC STA 0.00
60.120 DIA
NAC STA 112.30
71.654 DIA
ENG INLET PLANE
IE OF OUTBOARD
IAC INLET
326.00

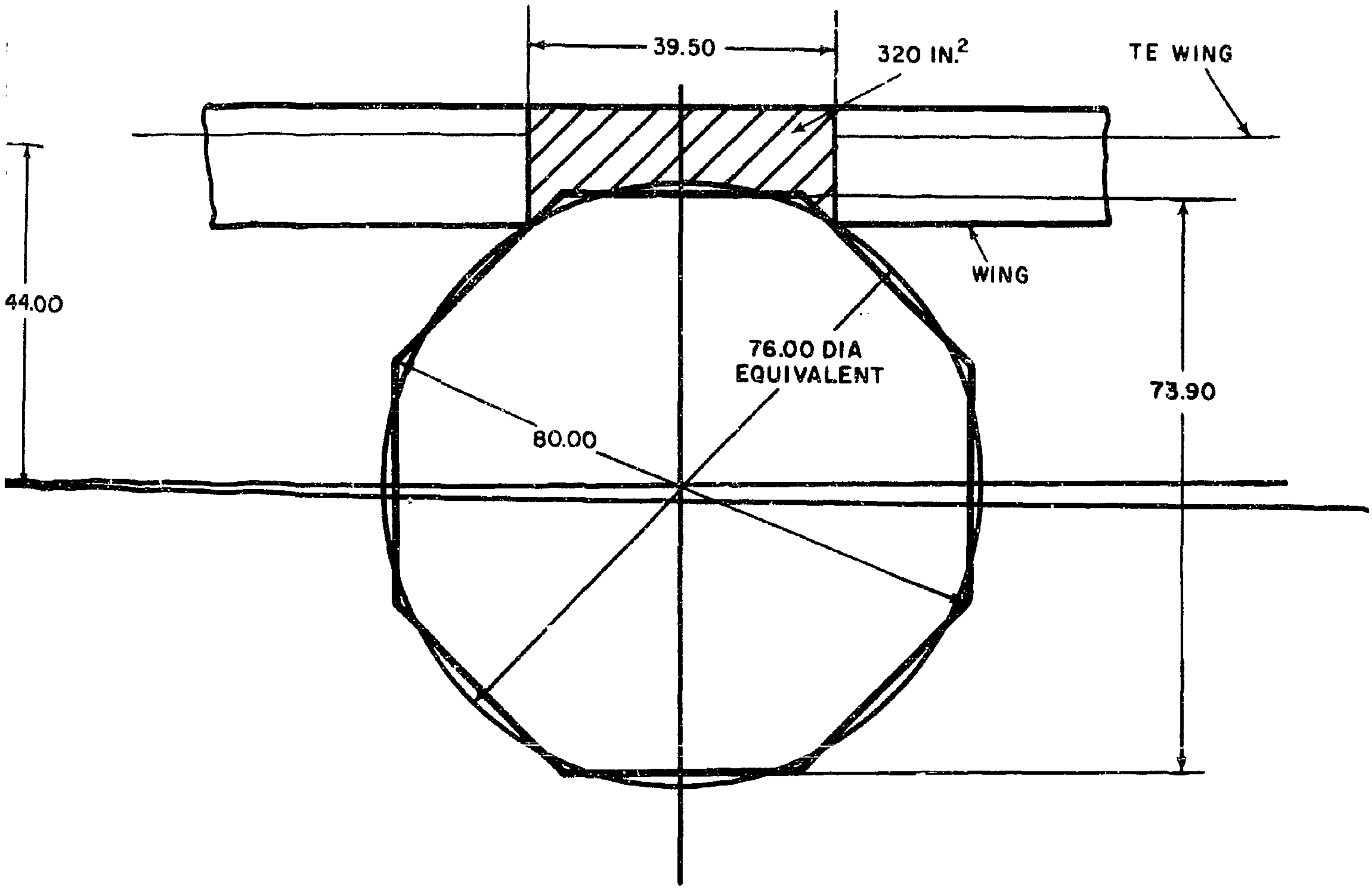

REAR ENG MT
104.00
℄ REAR ENG MT (ORIG LOC) REF
00 REF
30.00
77.10
ENG LENGTH
4.00
WING
CONTOUR
8°
HORIZ REF LINE
℄ ENG
4°
EJECTOR CANT PT
℄ EJECTOR
1° REF
3.00
LE BID
120.00
ENG INLET PLANE
NAC ATTACHMENT PLANE
326.00



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

STF219B 600 I

4

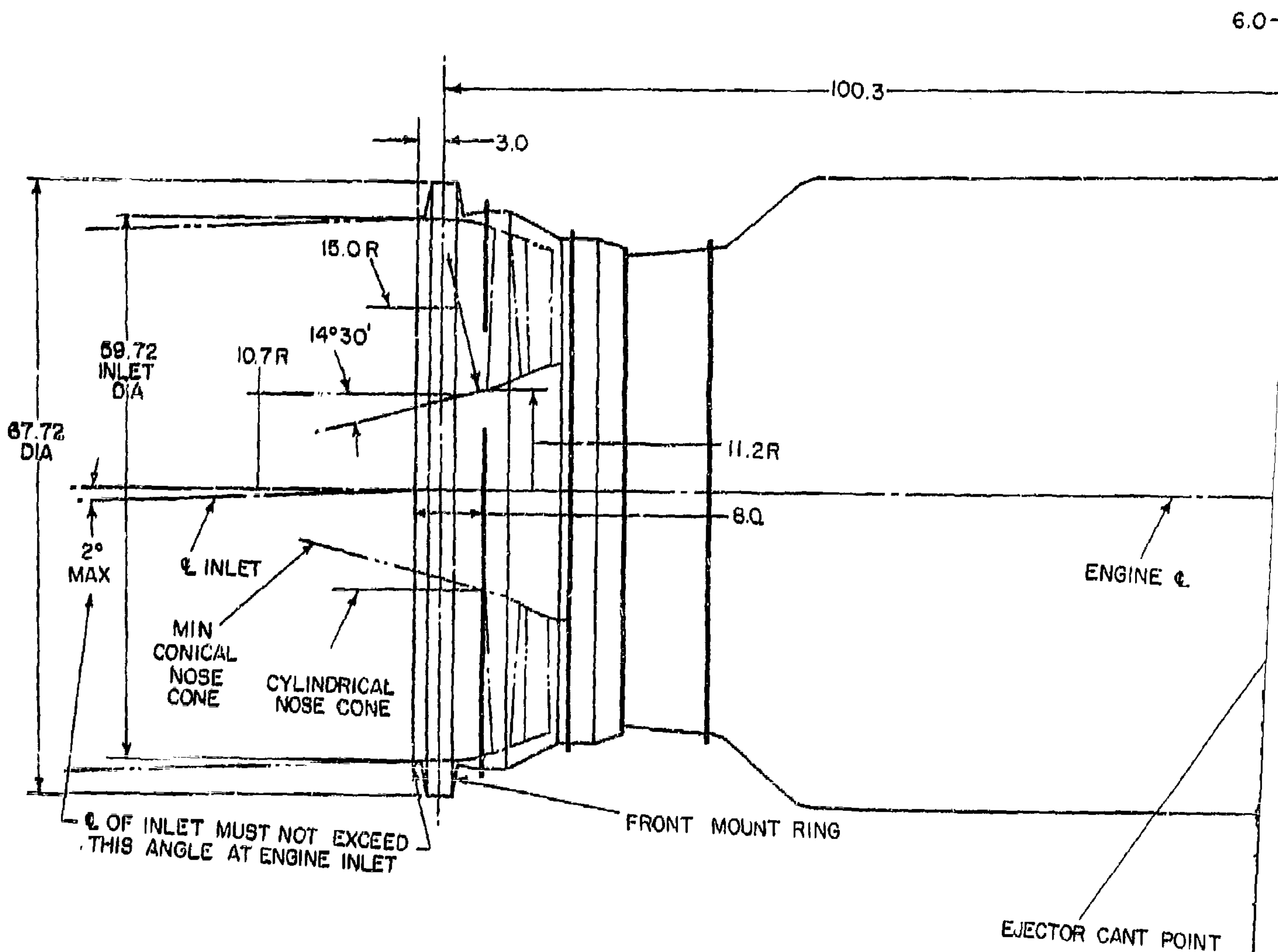


EJECTOR CANTED AT REAR MOUNT PLANE
ARRANGEMENT MOVING REAR ENG. MOUNT
30.00 AFT. & INSERTING ENG. 4.00 INTO
WING

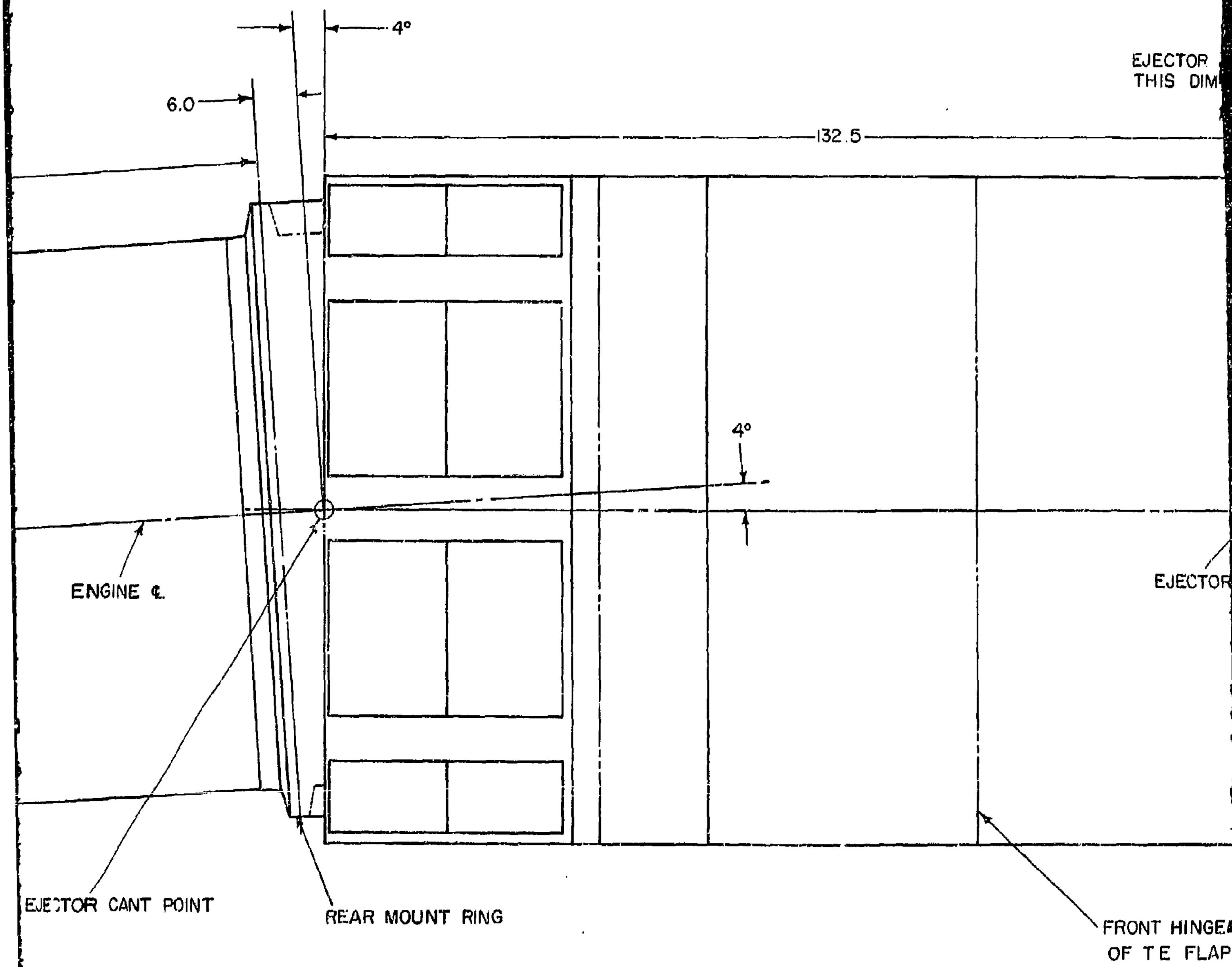
STF219B 600 LBS./SEC. TURBOFAN

Figure 1-5

5

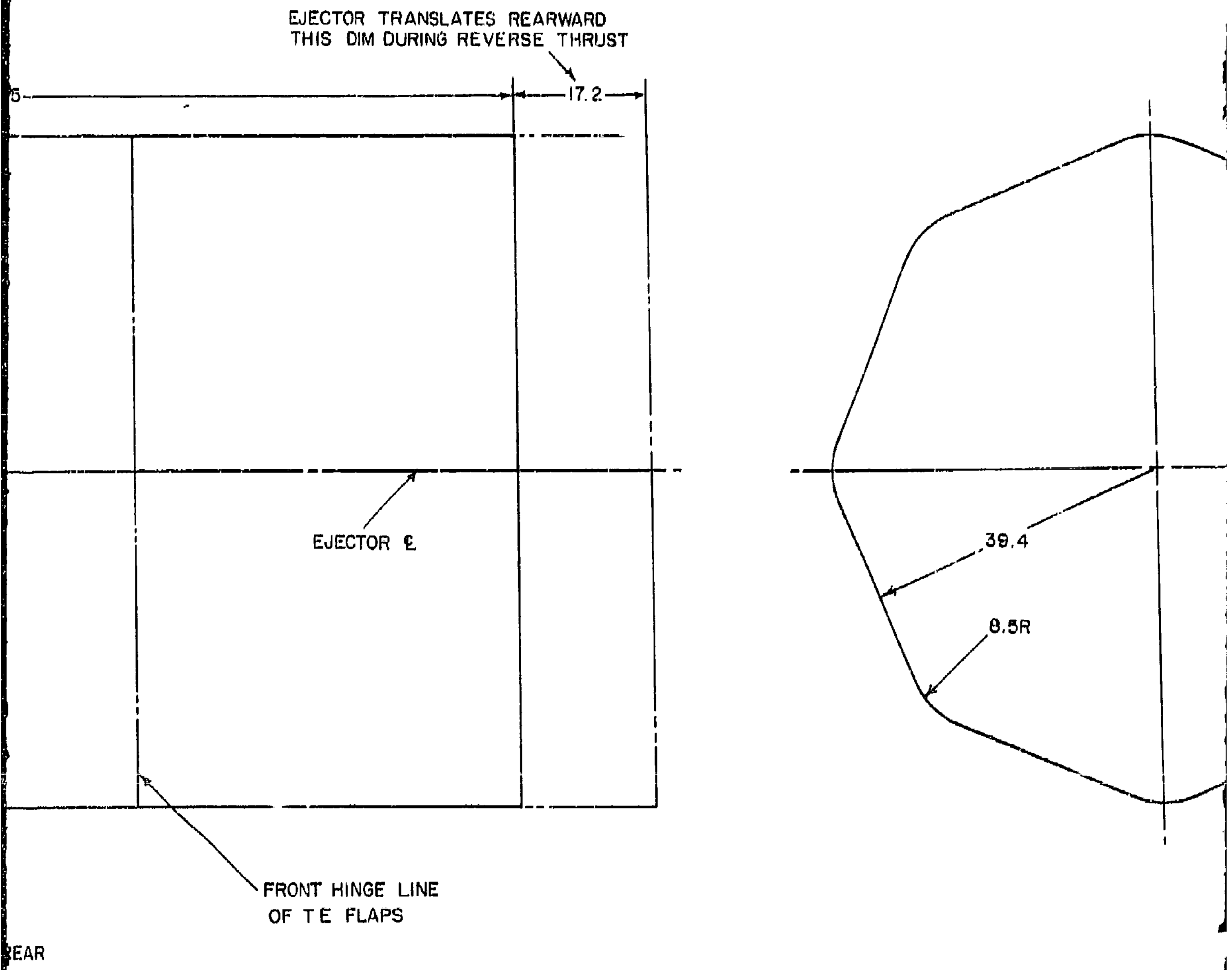

6.0
100.3
3.0
15.0 R
14°30'
59.72 INLET DIA
10.7 R
67.72 DIA
11.2R
8.0
2° MAX
℄ INLET
ENGINE ℄
MIN CONICAL NOSE CONE
CYLINDRICAL NOSE CONE
℄ OF INLET MUST NOT EXCEED THIS ANGLE AT ENGINE INLET
FRONT MOUNT RING
EJECTOR CANT POINT

1



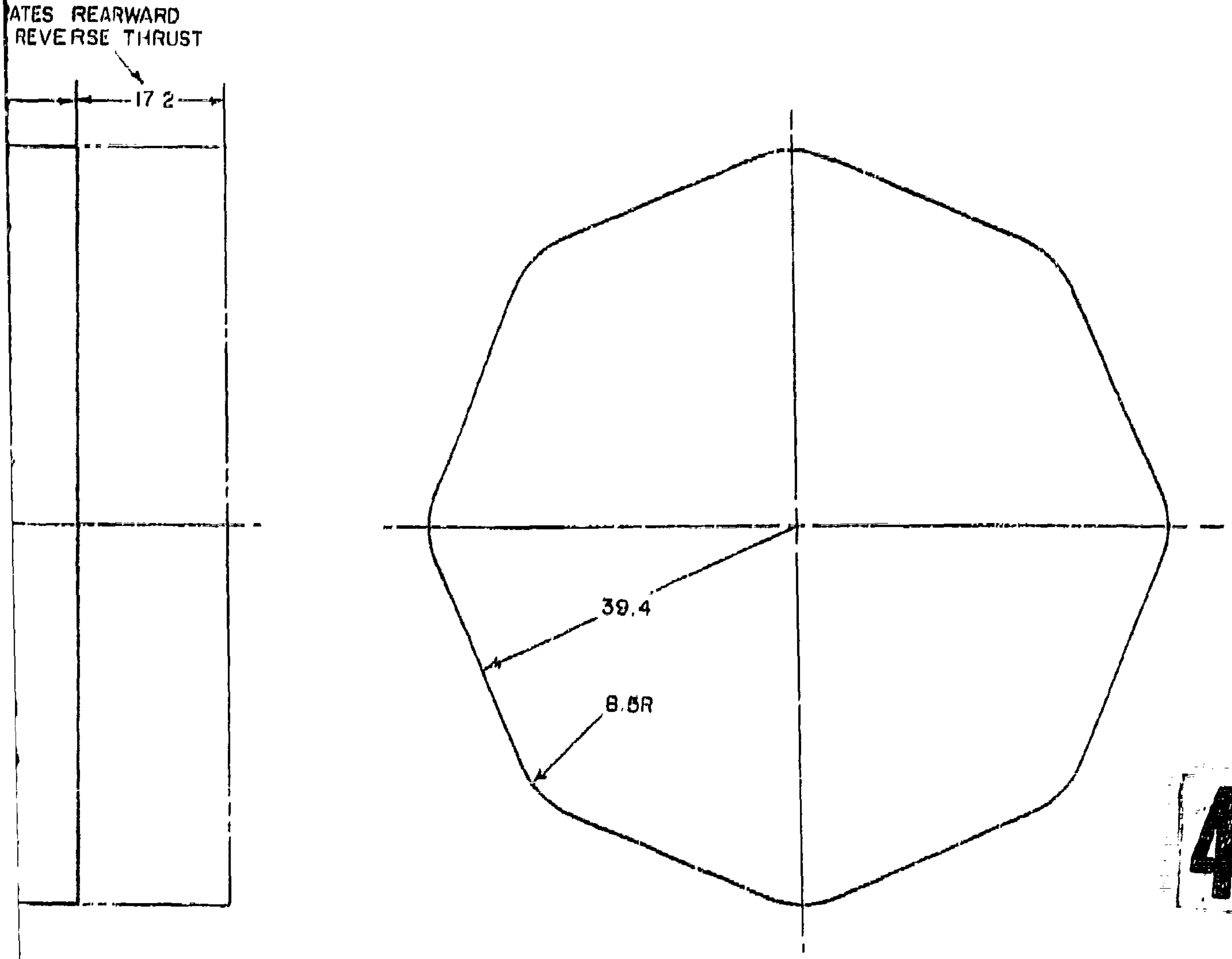
EJECTOR CANTED AT REAR
MOUNT PLANE

2



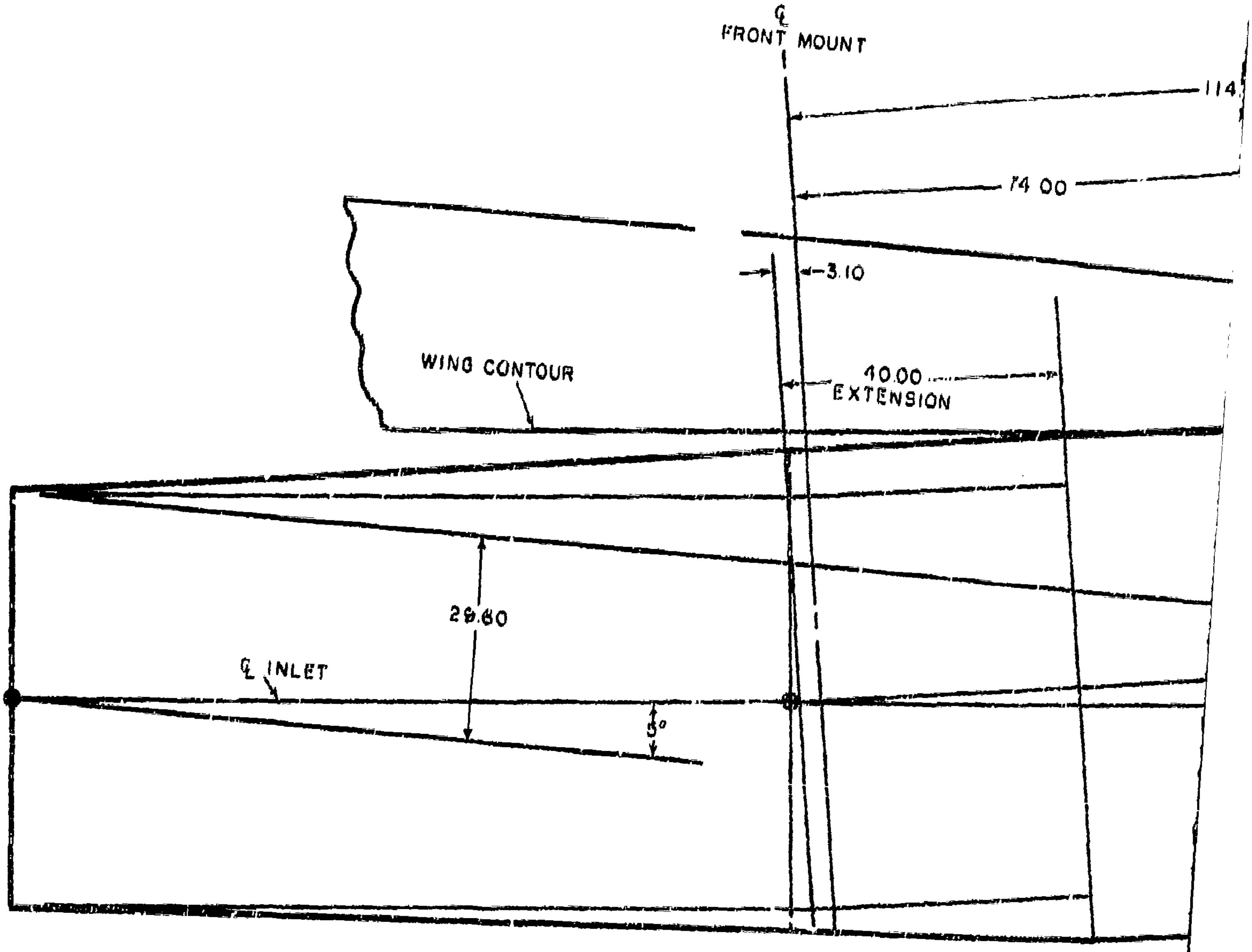
STF219B 600 LBS./SEC. TURBOFAN

Figure 1-6

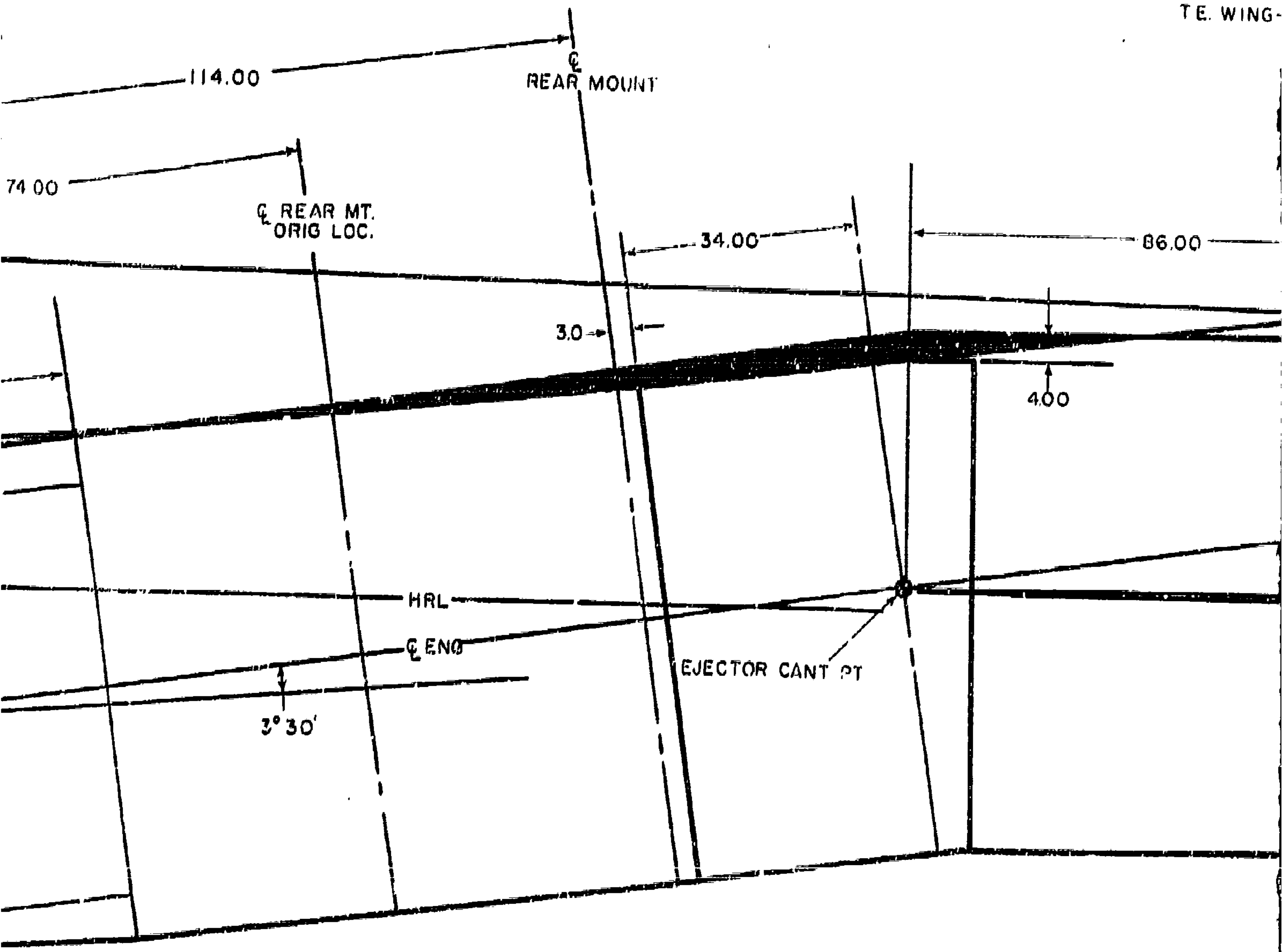


STF219B 600 LBS./SEC. TURBOFAN

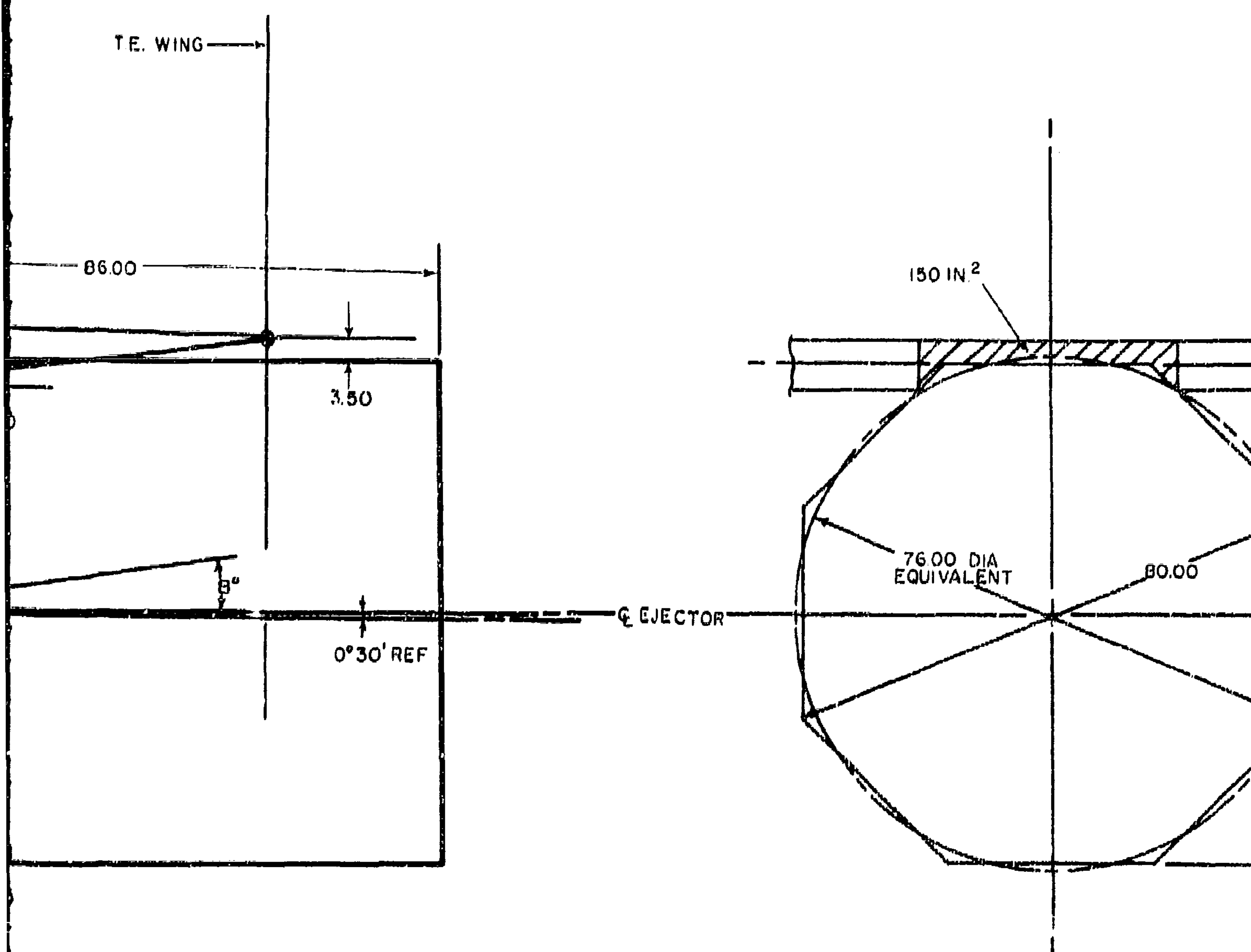
Figure 1-6


℄
FRONT MOUNT
114
74.00
3.10
WING CONTOUR
40.00
EXTENSION
29.60
℄ INLET
3°

1


T.E. WING
114.00
℄ REAR MOUNT
74.00
℄ REAR MT. ORIG LOC.
34.00
86.00
3.0
4.00
HRL
℄ ENG
EJECTOR CANT PT
3° 30'

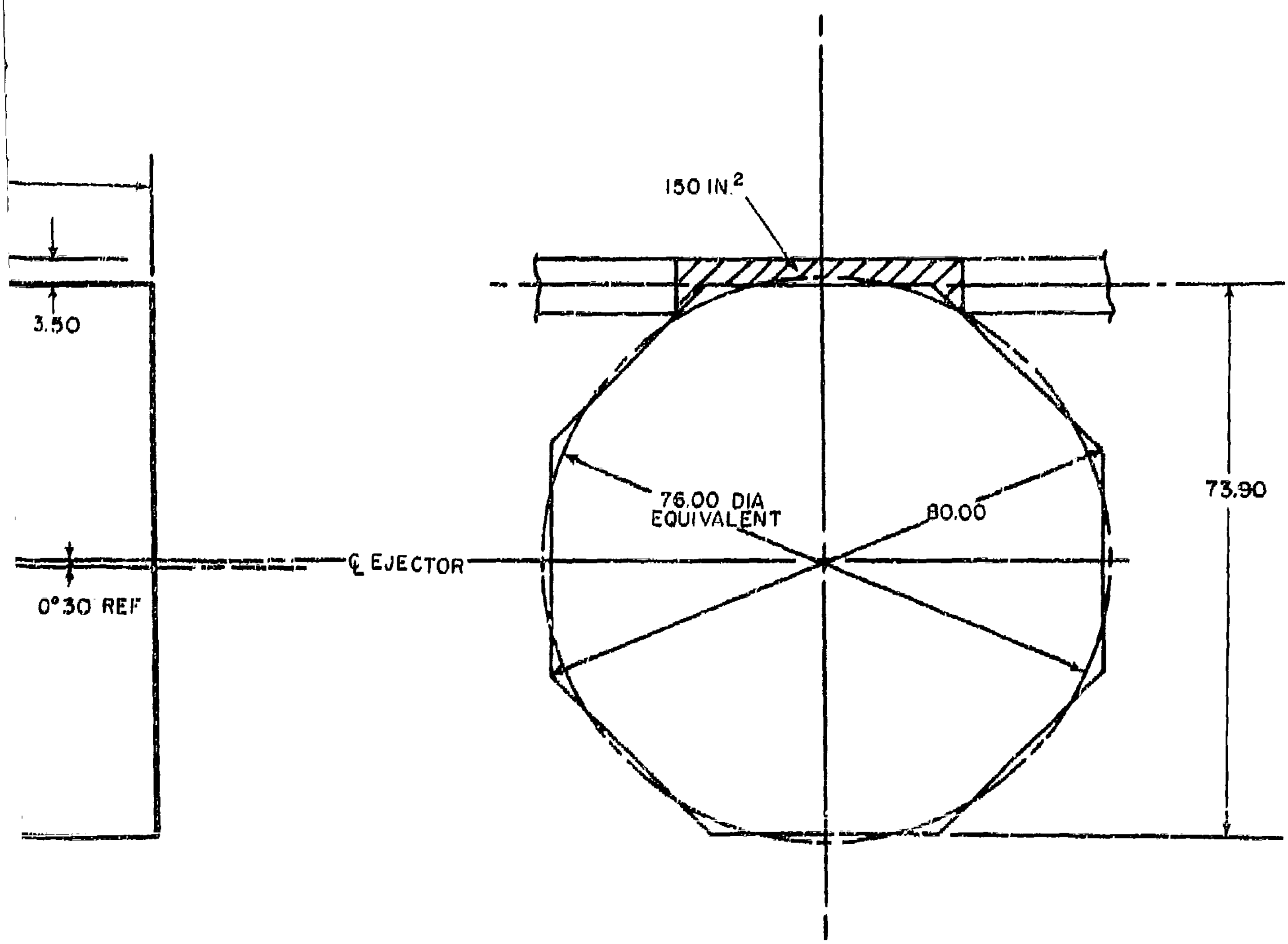
2



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 in^2

STF219B 600 LBS./SEC. TURBOFAN

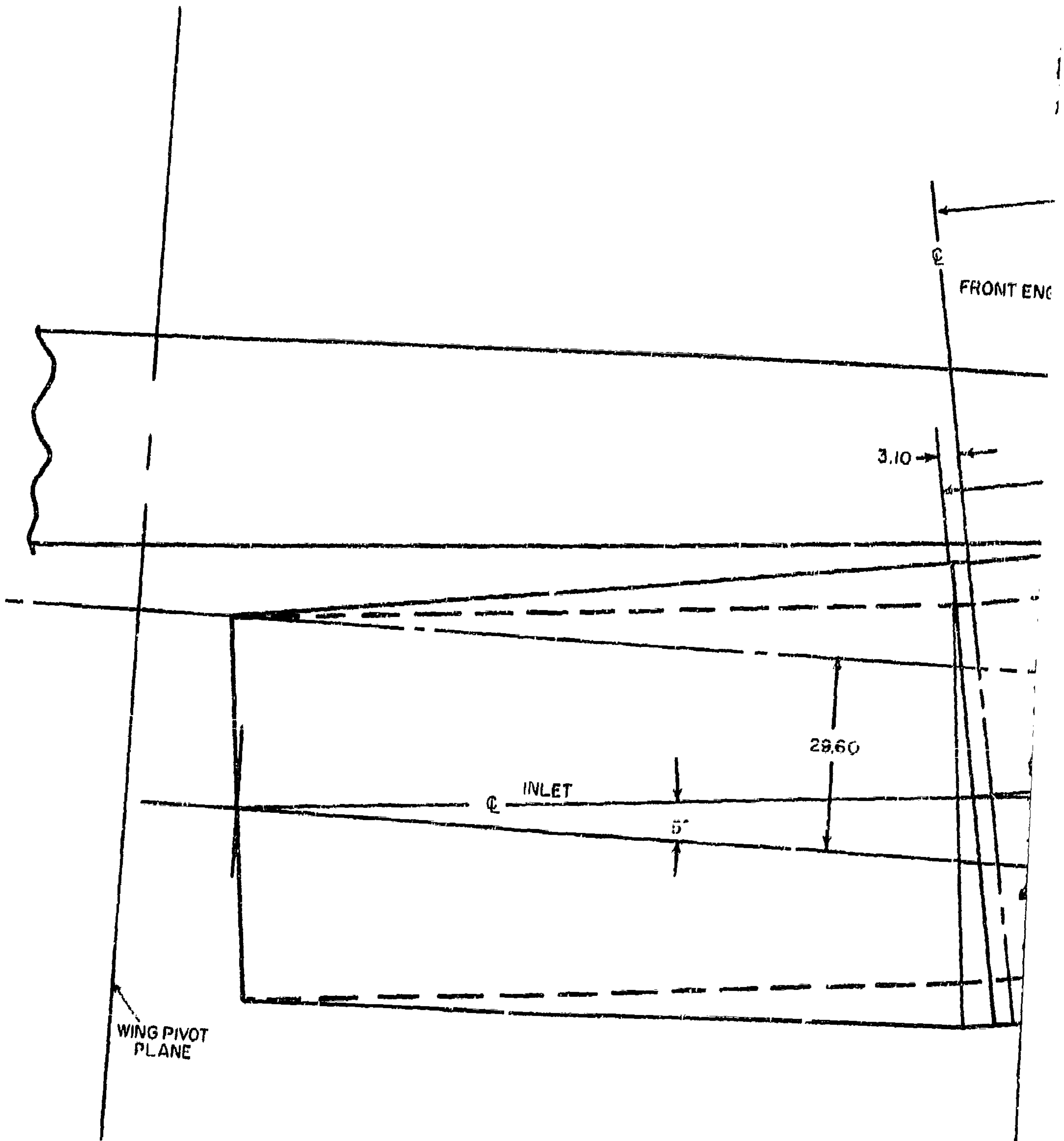
Figure 1-7



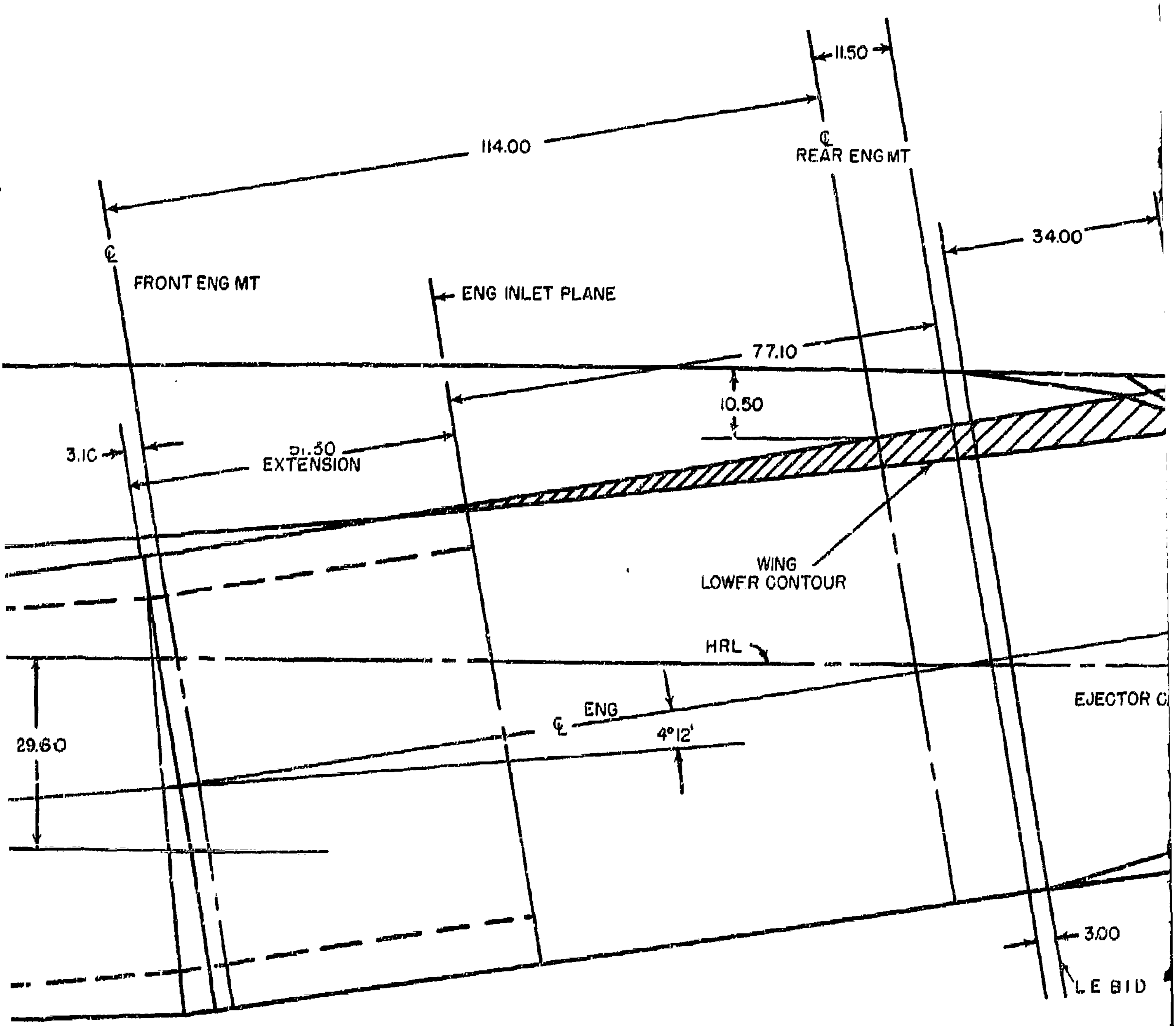
ARRANGEMENT MOVING REAR ENG. MT AFT 40.00 & INSERTING ENG. 4.00 INTO WING
EJECTOR CANTED AT NOZZLE PLANE
ENG. MOUNT PLANES ARE PARALLEL
BASE DRAG AREA = 150 in^2

STF219B 600 LBS./SEC. TURBOFAN

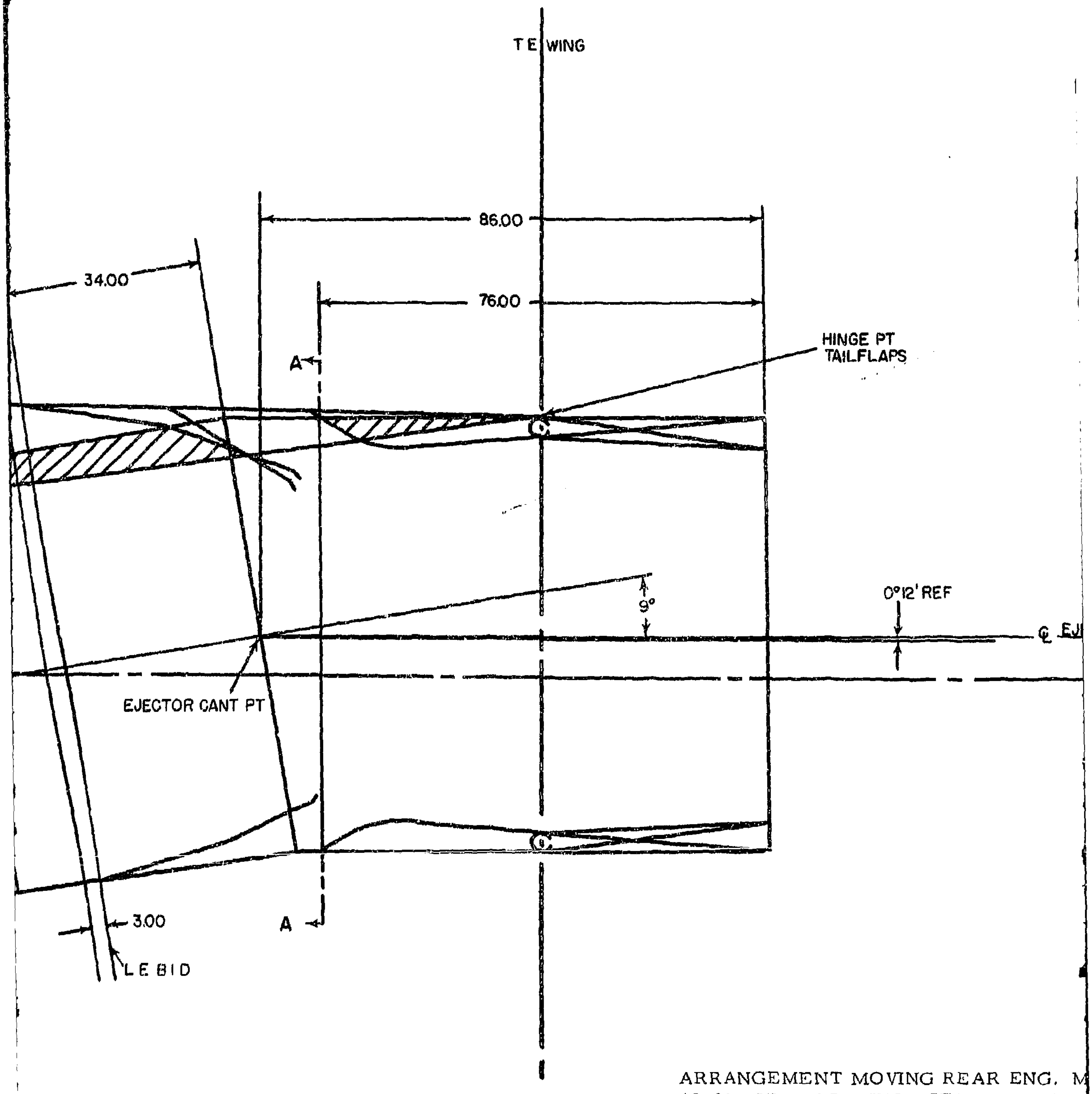
Figure 1-7



1

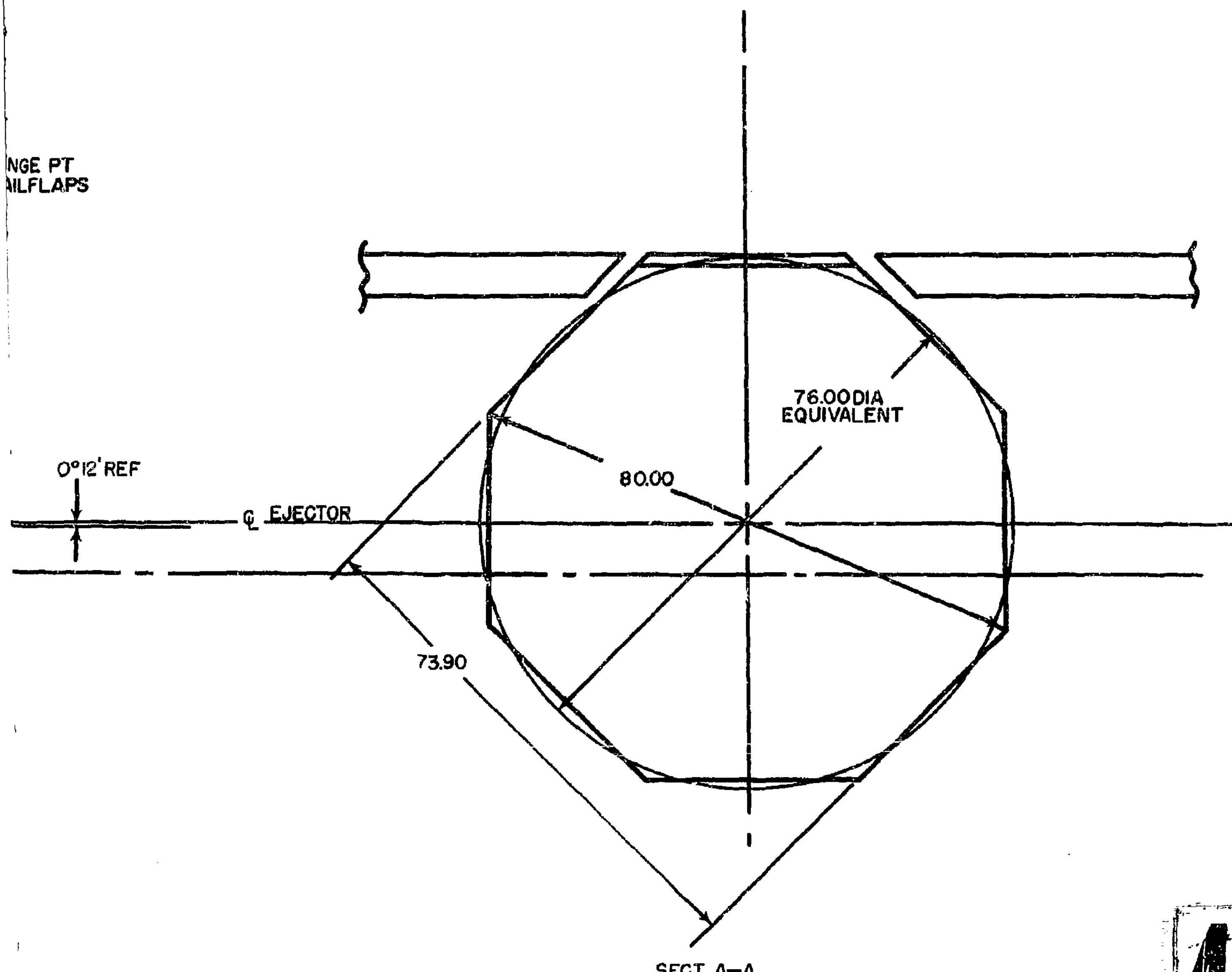

11.50
114.00
℄
REAR ENG MT
34.00
℄
FRONT ENG MT
ENG INLET PLANE
77.10
10.50
3.10
51.50
EXTENSION
WING
LOWER CONTOUR
HRL
ENG
4°12'
29.60
3.00
L E BID

2



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

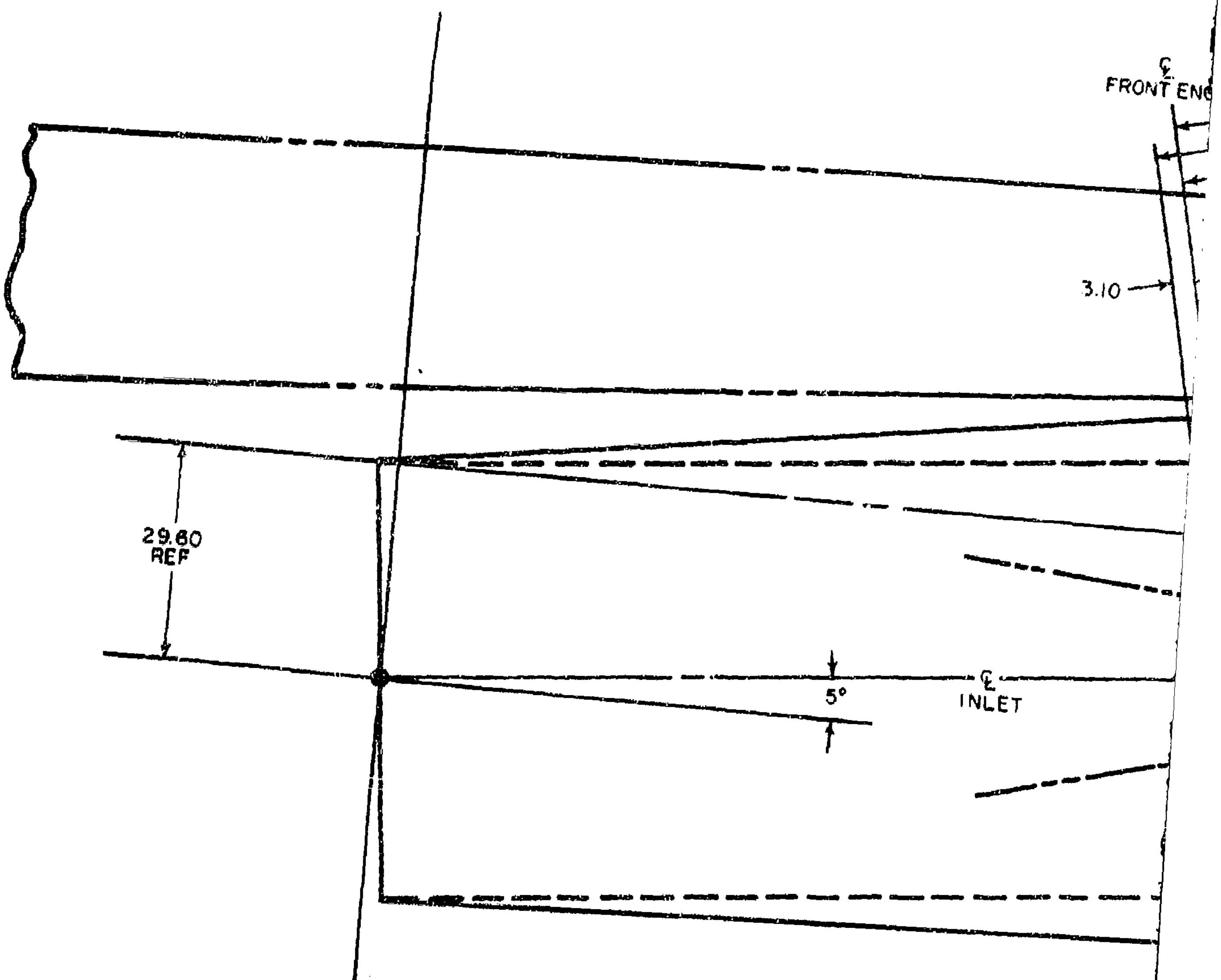
3

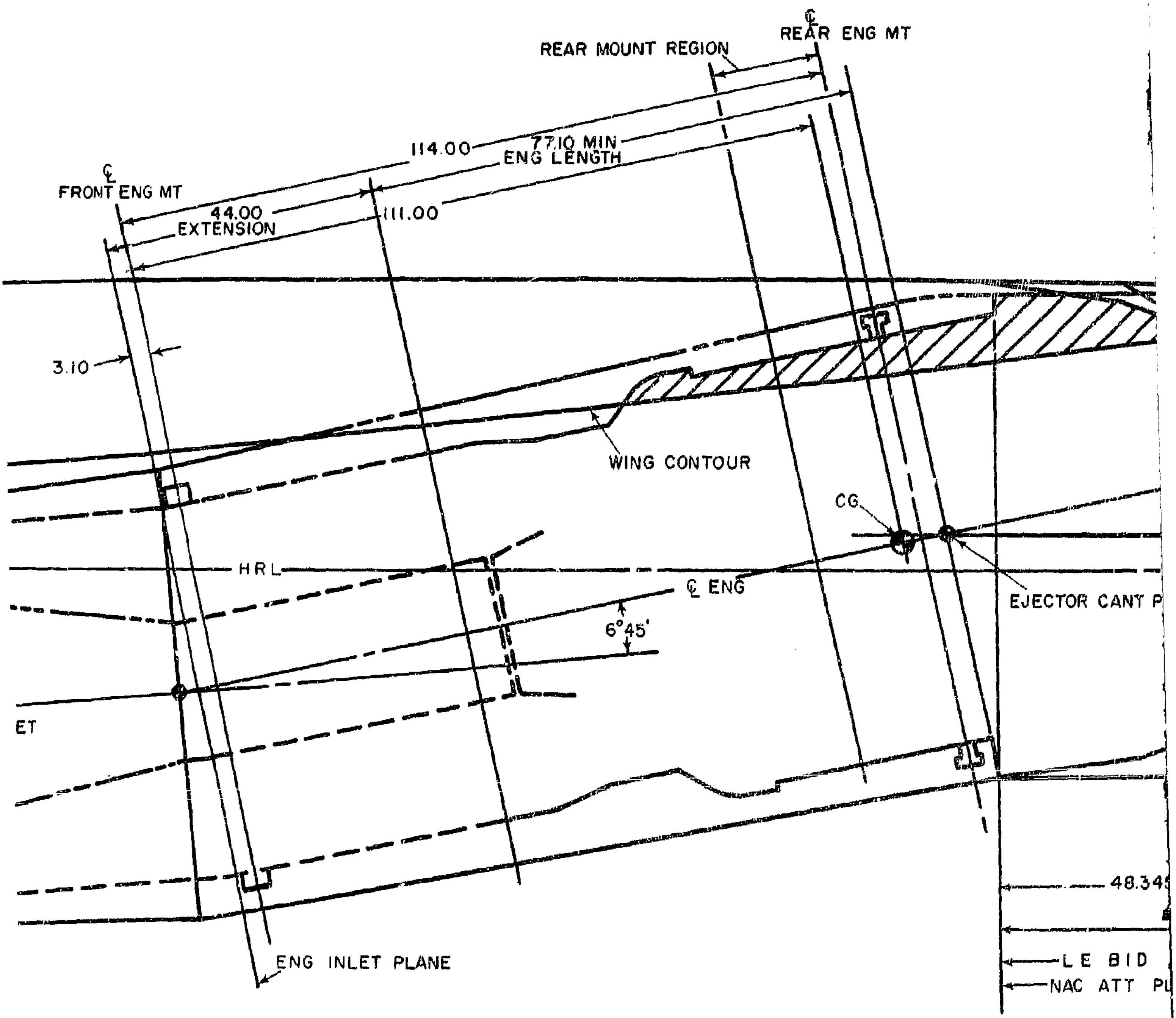


MOVING REAR ENG. MT.
TING EJECTOR TAIL FLAPS
NG
ED AT NOZZLE PLANE

STF219B 600 LBS. /SEC. TURBOFAN

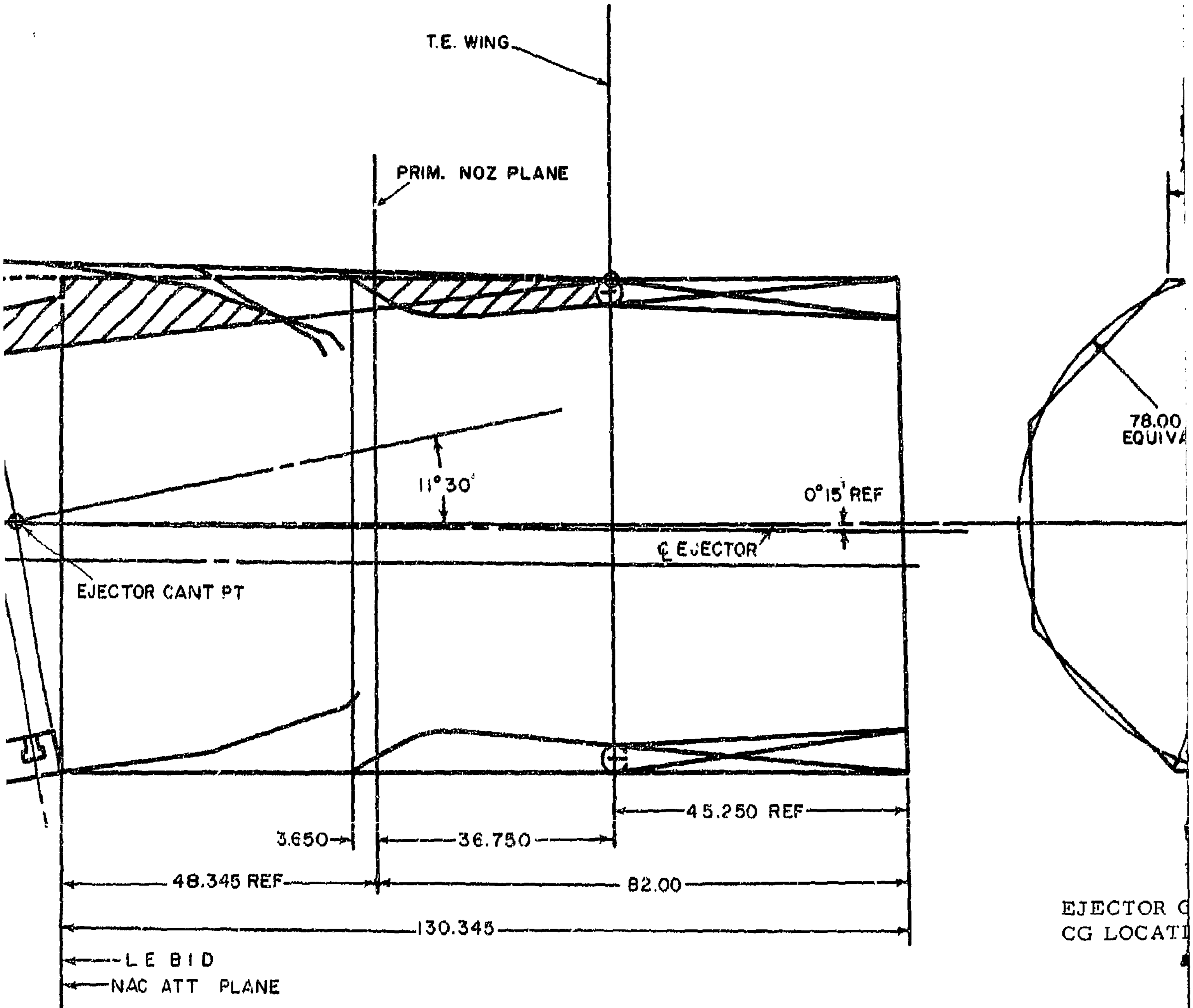
Figure 1-8


FRONT ENG
3.10
29.60
REF
5°
INLET



℄ REAR ENG MT
REAR MOUNT REGION
114.00
77.10 MIN
ENG LENGTH
℄ FRONT ENG MT
44.00
EXTENSION
111.00
3.10
WING CONTOUR
CG
HRL
℄ ENG
EJECTOR CANT P
6°45'
ET
48.34
ENG INLET PLANE
L E BID
NAC ATT PL

2

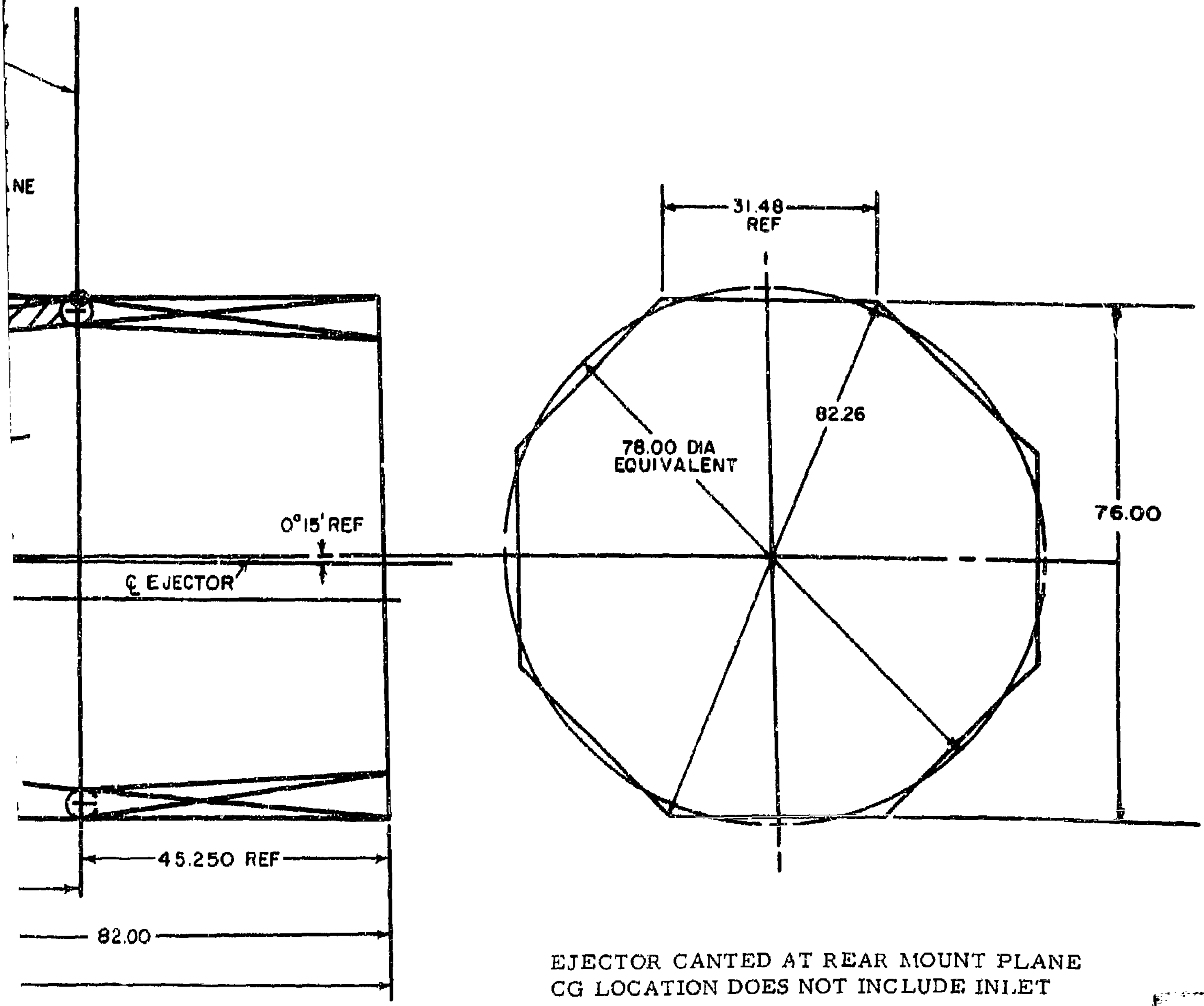


EJECTOR G
CG LOCATI

STF219B 600 LBS./S

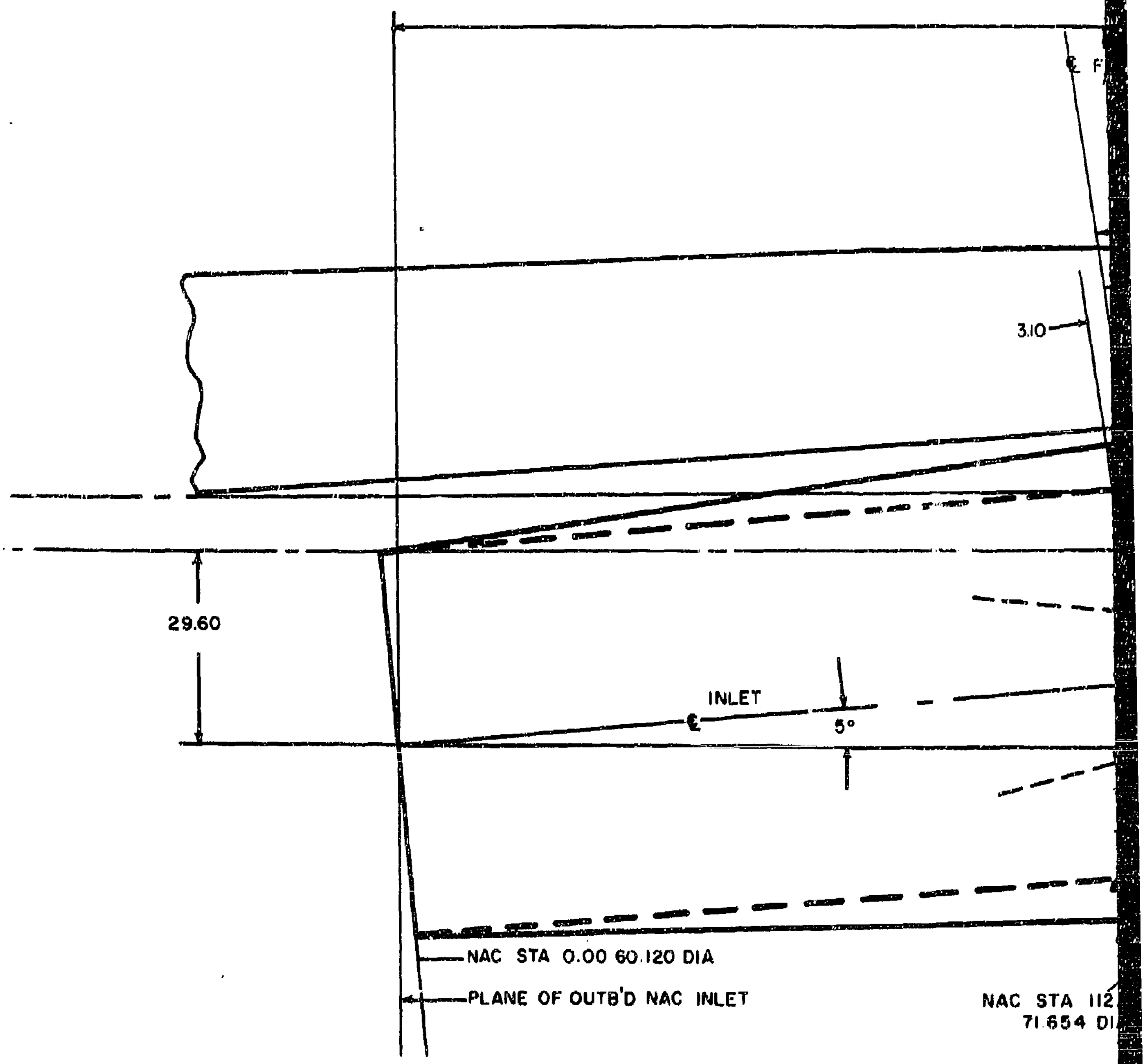
Figure

3



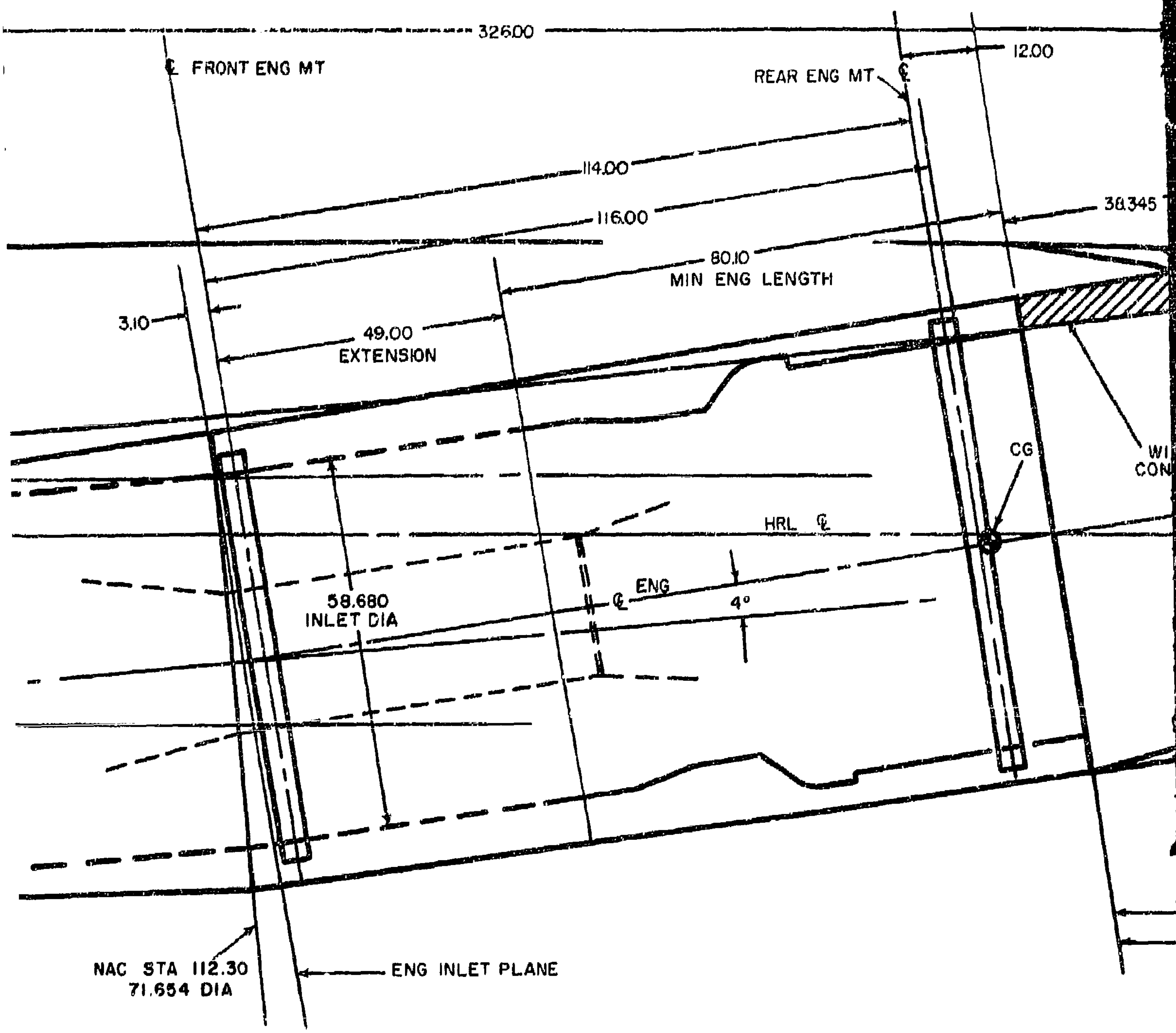
STF219B 600 LBS./SEC. TURBOFAN

Figure 1-9



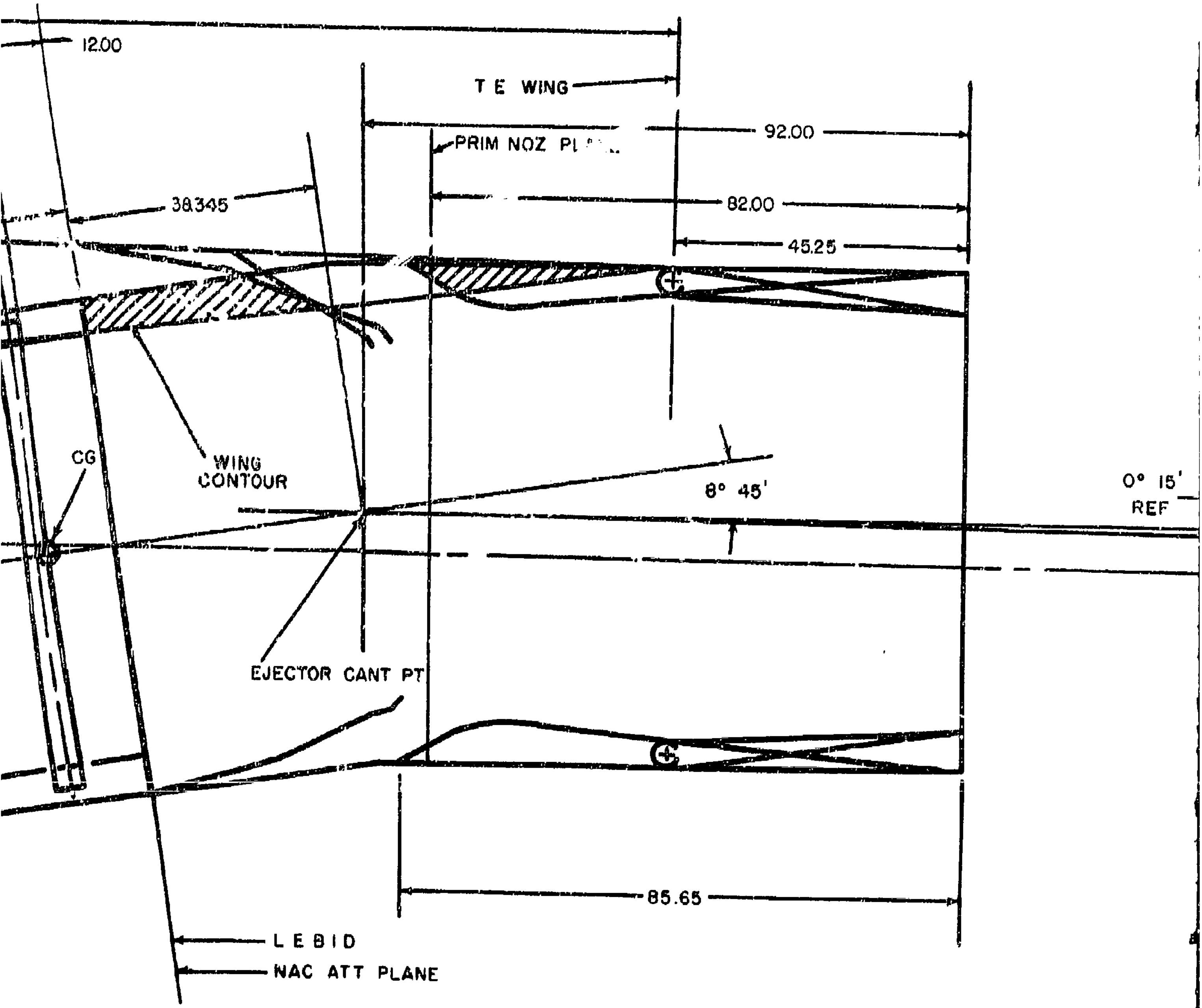
℄ F
3.10
29.60
INLET
℄
5°
NAC STA 0.00 60.120 DIA
PLANE OF OUTB'D NAC INLET
NAC STA 112
71.654 DI

1



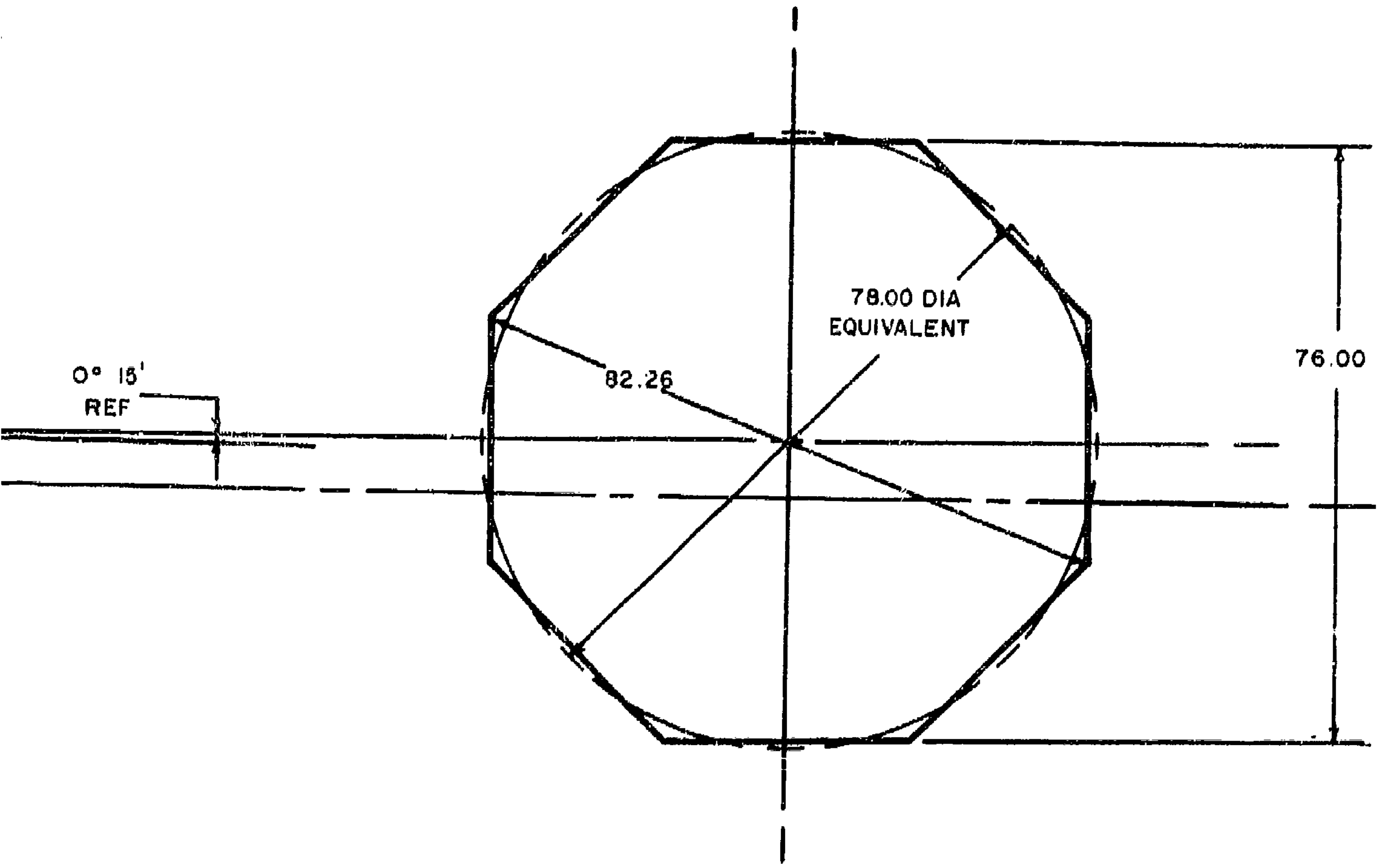
326.00
℄ FRONT ENG MT
12.00
REAR ENG MT ℄
114.00
116.00
38.345
80.10
MIN ENG LENGTH
3.10
49.00
EXTENSION
CG
HRL ℄
℄ ENG
4°
58.680
INLET DIA
NAC STA 112.30
71.654 DIA
ENG INLET PLANE

2



12.00
T E WING
92.00
PRIM NOZ PL
82.00
38.345
45.25
WING CONTOUR
CG
8° 45'
0° 15' REF
EJECTOR CANT PT
85.65
L E B I D
NAC ATT PLANE

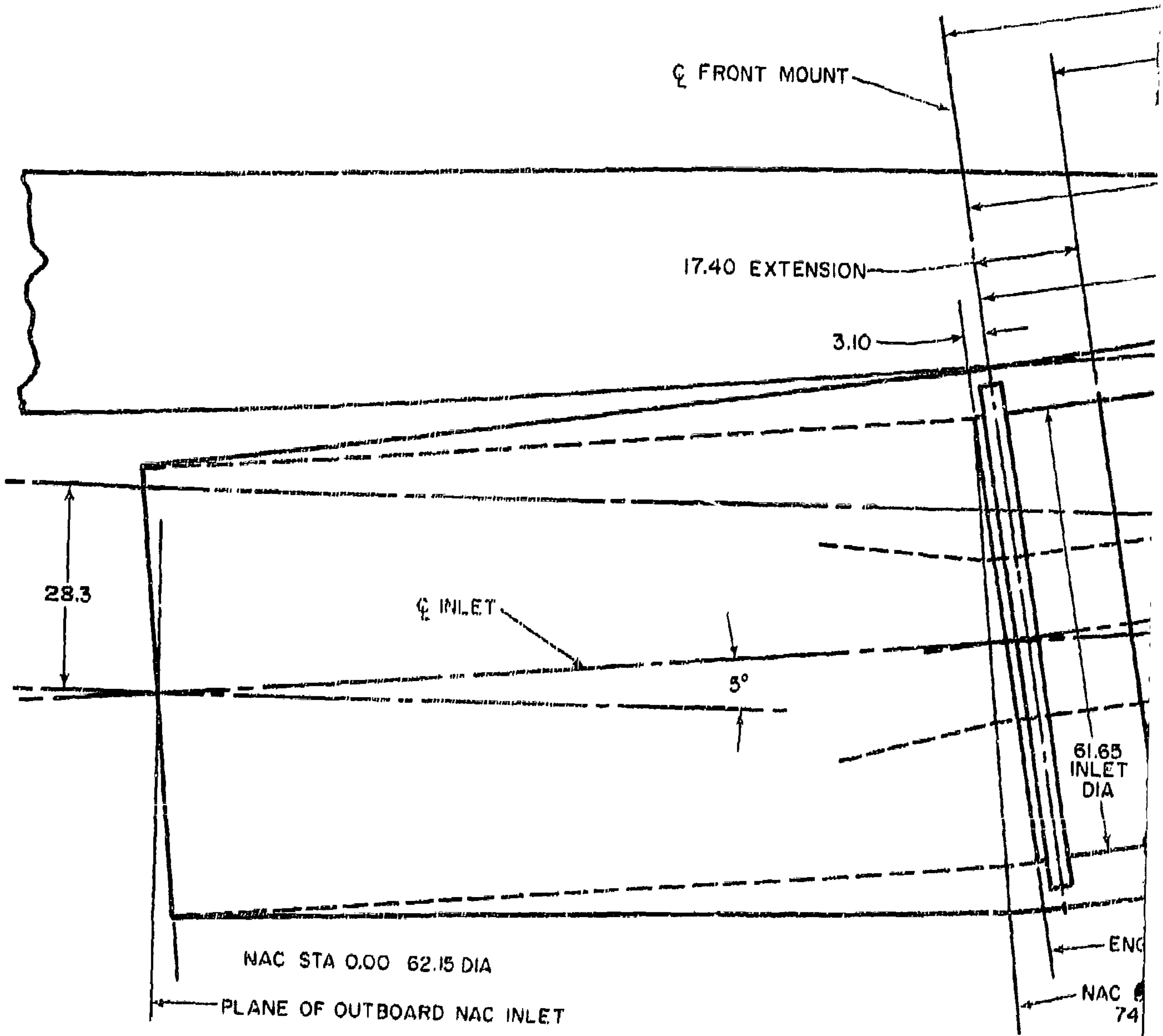
3



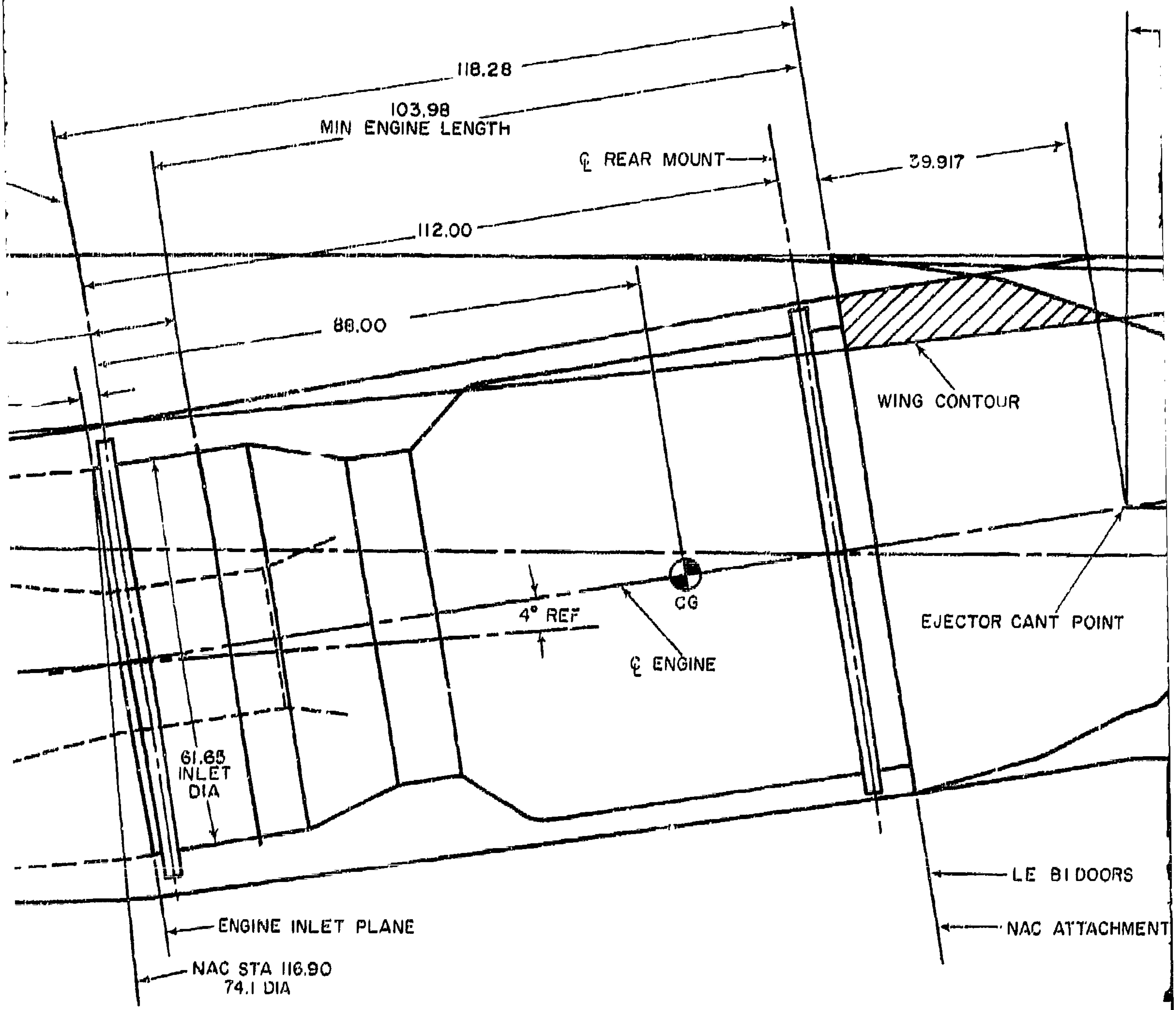
EJECTOR CANTED 10.00 FWD. OF PRIMARY
NOZZLE PLANE
CG LOCATION DOES NOT INCLUDE INLET

STF219B 600 LBS./SEC. TURBOFAN

Figure 1-10

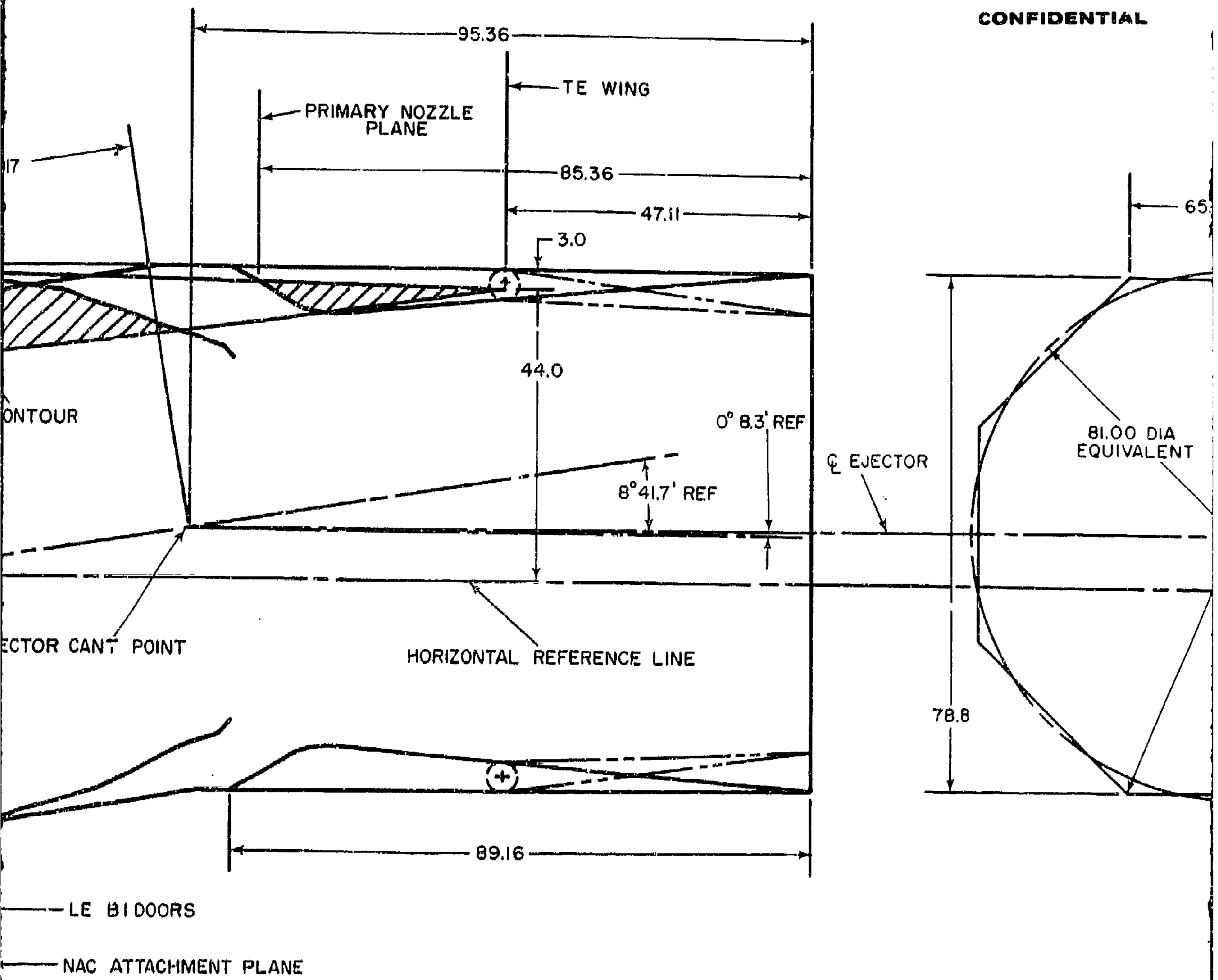


℄ FRONT MOUNT
17.40 EXTENSION
3.10
28.3
℄ INLET
5°
61.65
INLET
DIA
NAC STA 0.00 62.15 DIA
PLANE OF OUTBOARD NAC INLET
ENG
NAC
74


118.28
103.98
MIN ENGINE LENGTH
℄ REAR MOUNT
39.917
112.00
88.00
WING CONTOUR
4° REF
CG
℄ ENGINE
EJECTOR CANT POINT
61.65
INLET
DIA
LE BI DOORS
NAC ATTACHMENT
ENGINE INLET PLANE
NAC STA 116.90
74.1 DIA

2

CONFIDENTIAL



CG LOCATION DOES NOT INCLUDE INLET

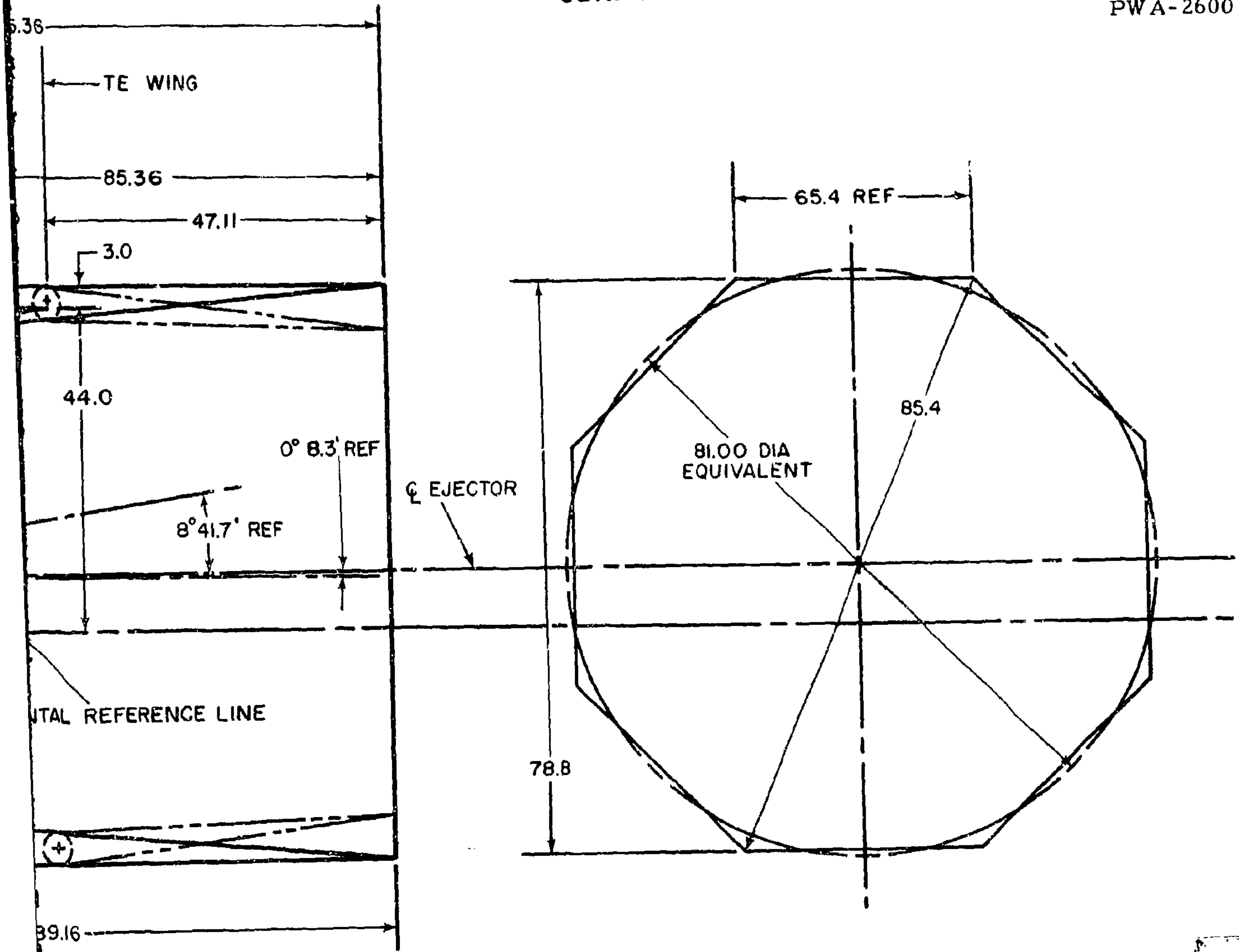
EJECTOR CAN
NOZZLE PL
CG LOCATION

STF219 650 LBS./SEC. TUR

Figure 1-11

CONFIDENTIAL

3

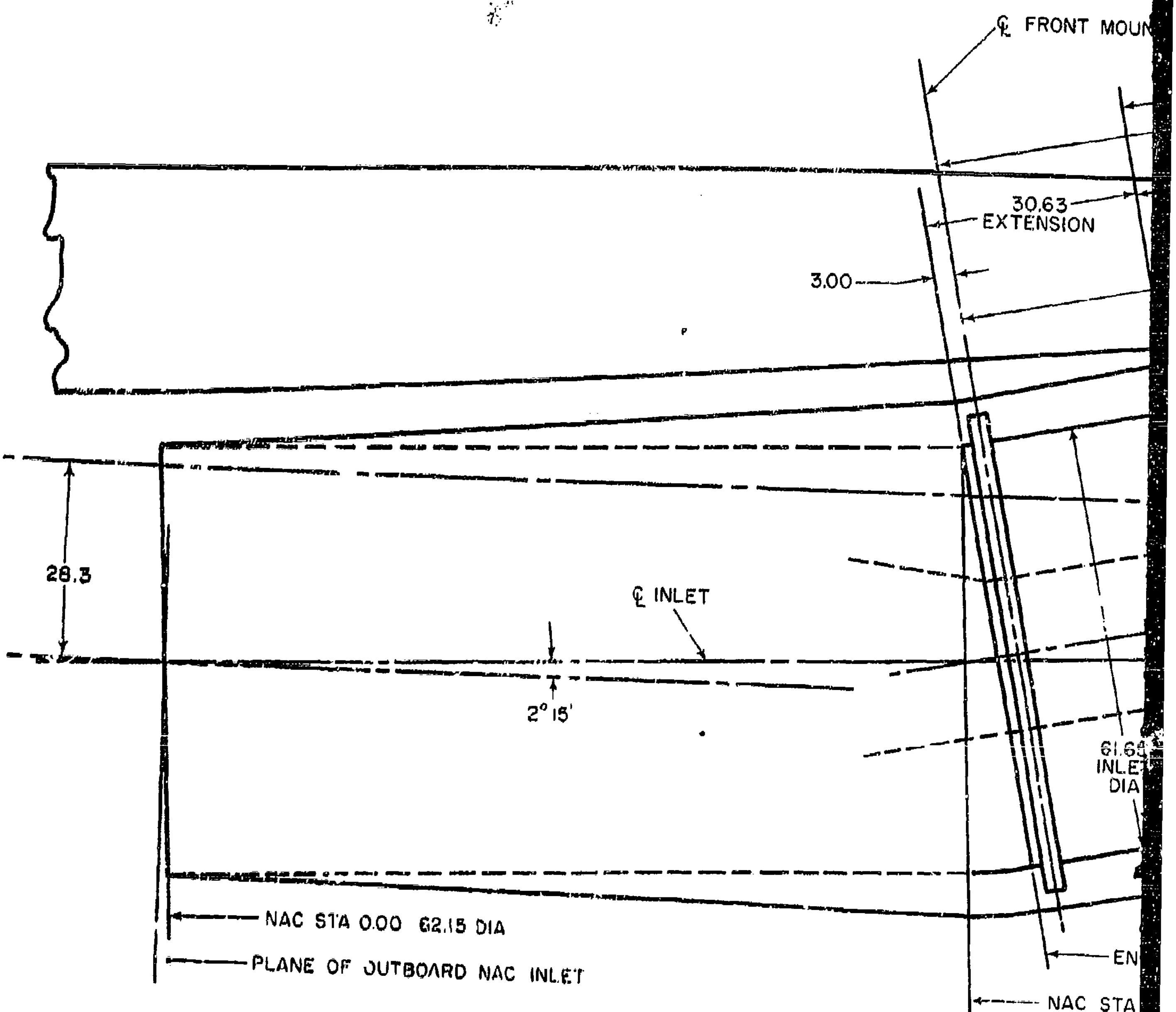


CG LOCATION DOES NOT INCLUDE INLET

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

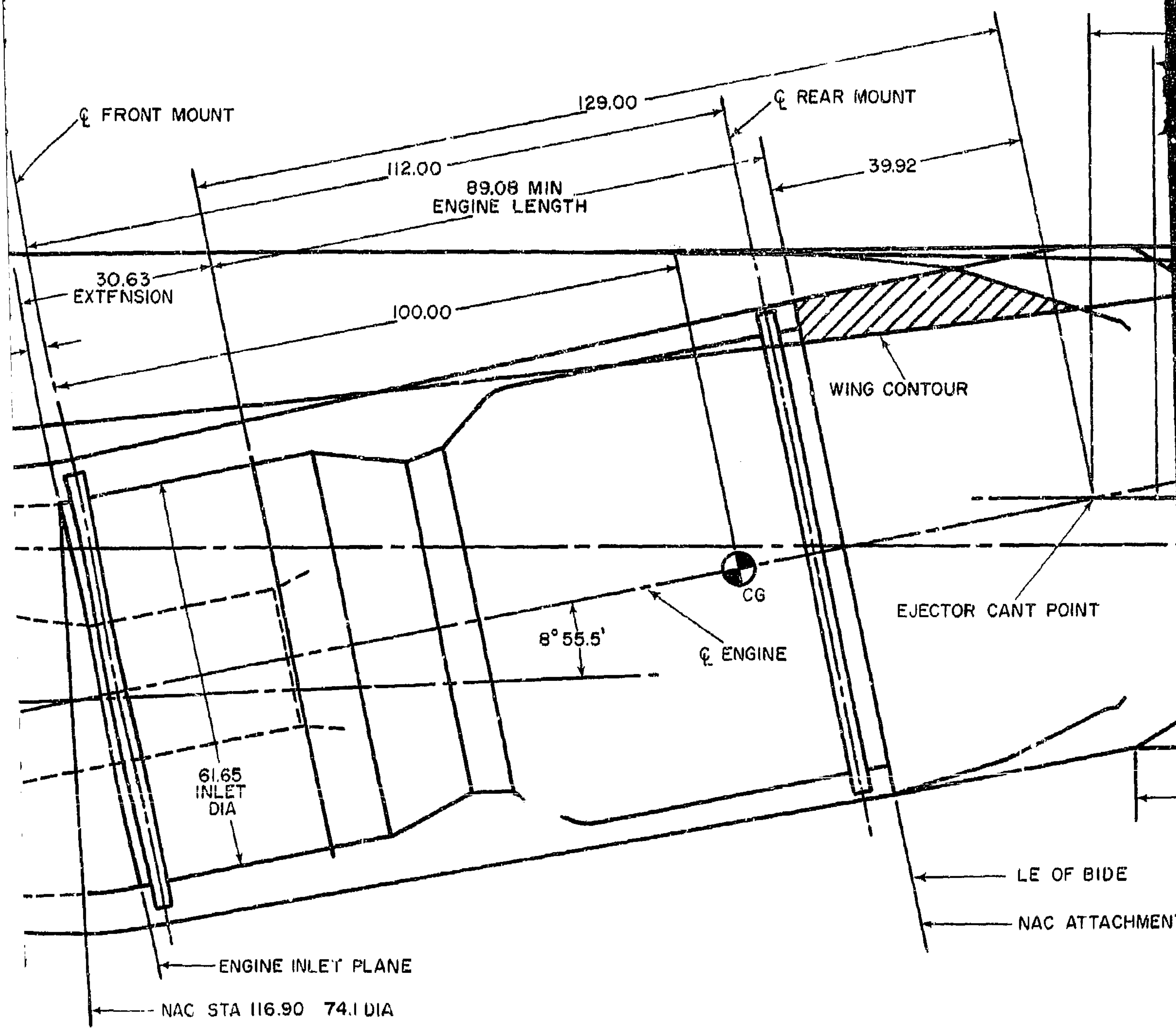
STF219 650 LBS./SEC. TURBOFAN

Figure 1-11



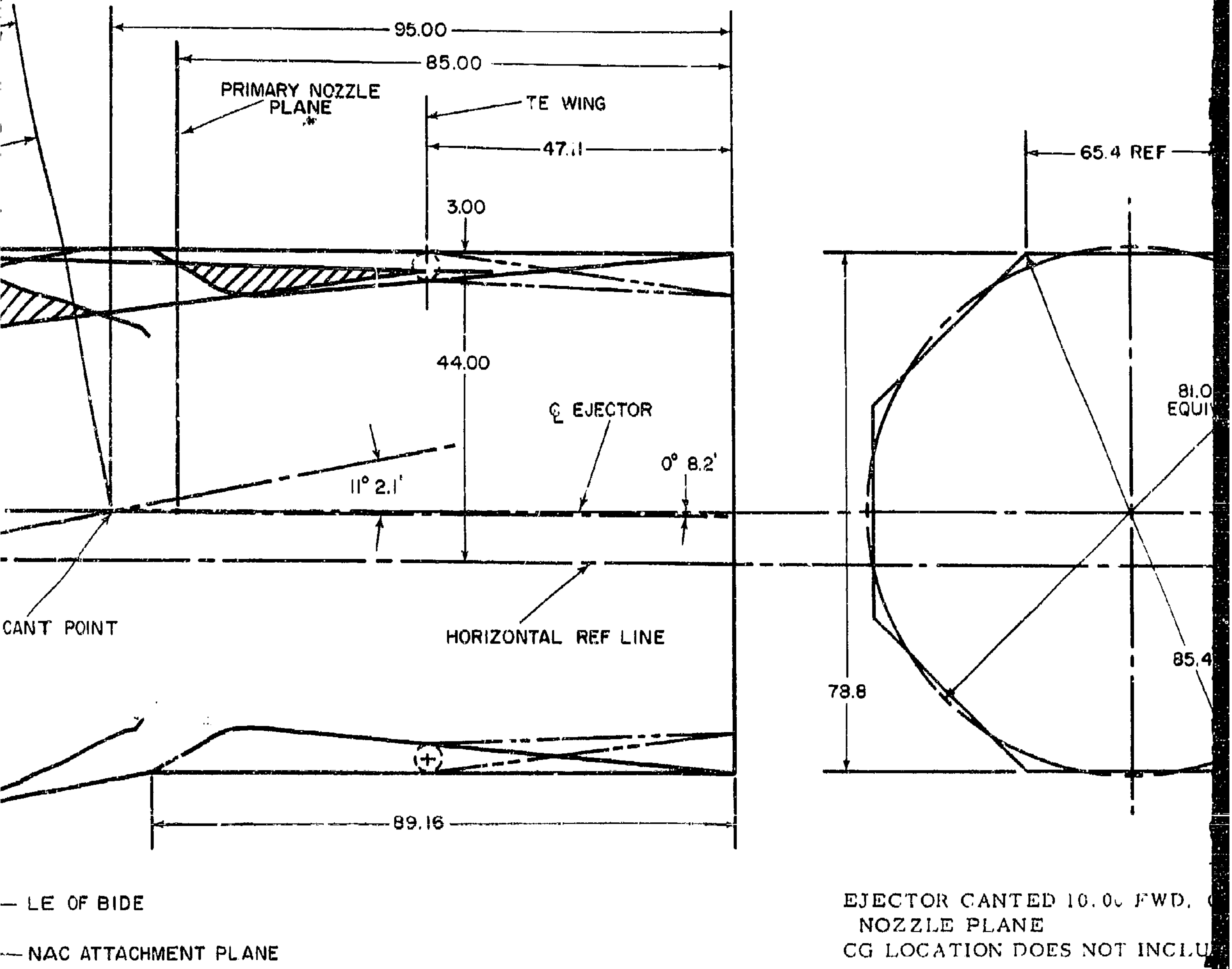
℄ FRONT MOUN
30.63
EXTENSION
3.00
28.3
℄ INLET
2° 15'
61.65
INLET
DIA
NAC STA 0.00 62.15 DIA
PLANE OF OUTBOARD NAC INLET
EN
NAC STA

1



℄ FRONT MOUNT
129.00
℄ REAR MOUNT
112.00
39.92
89.08 MIN
ENGINE LENGTH
30.63
EXTENSION
100.00
WING CONTOUR
CG
EJECTOR CANT POINT
8° 55.5'
℄ ENGINE
61.65
INLET
DIA
LE OF BIDE
NAC ATTACHMENT
ENGINE INLET PLANE
NAC STA 116.90 74.1 DIA

2

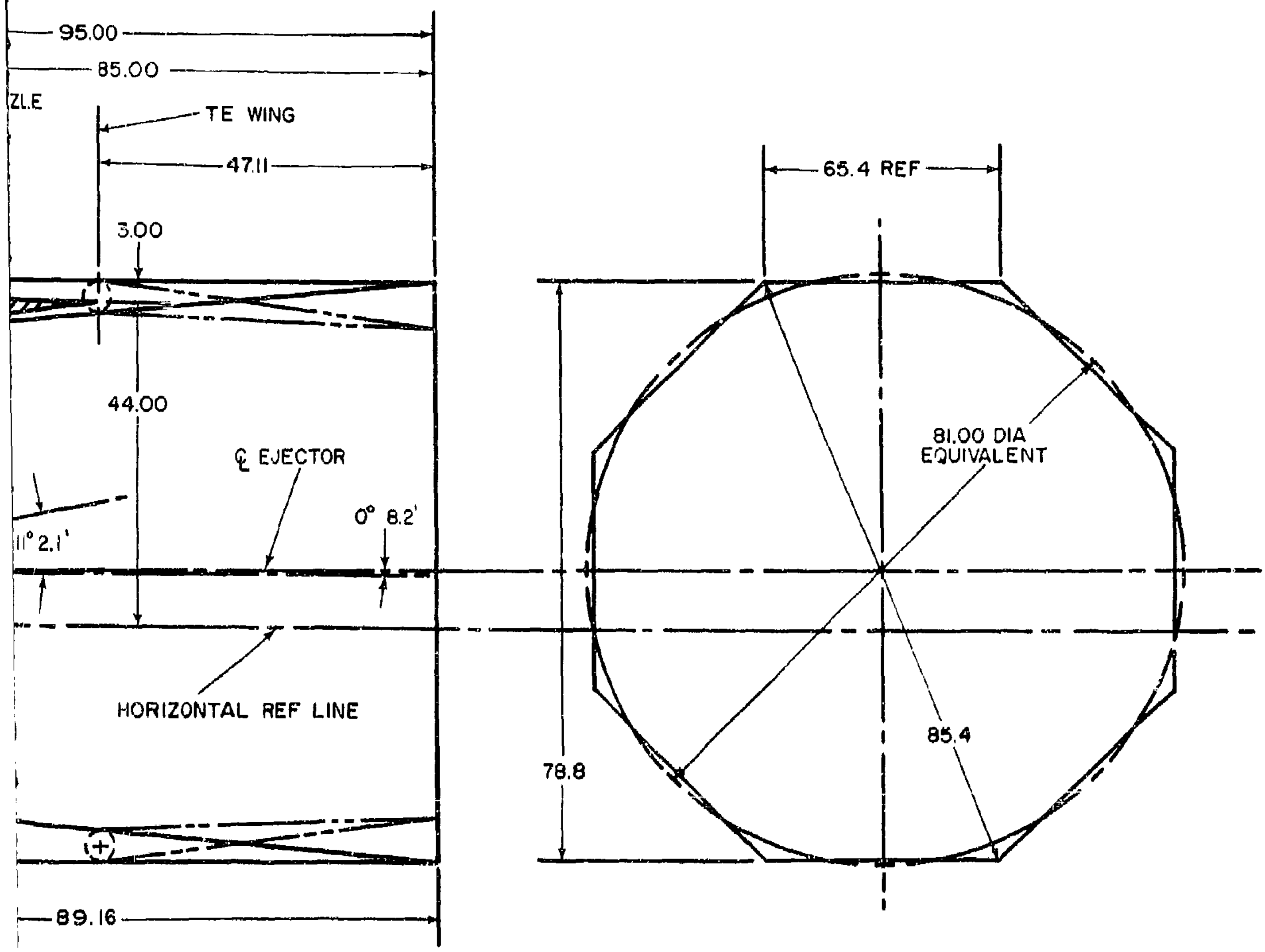


EJECTOR CANTED 10.0 FWD.
NOZZLE PLANE
CG LOCATION DOES NOT INCLU

CG LOCATION DOES NOT INCLUDE INLET

STF219 650 LBS./SEC. TURBOFAN

Figure 1-12

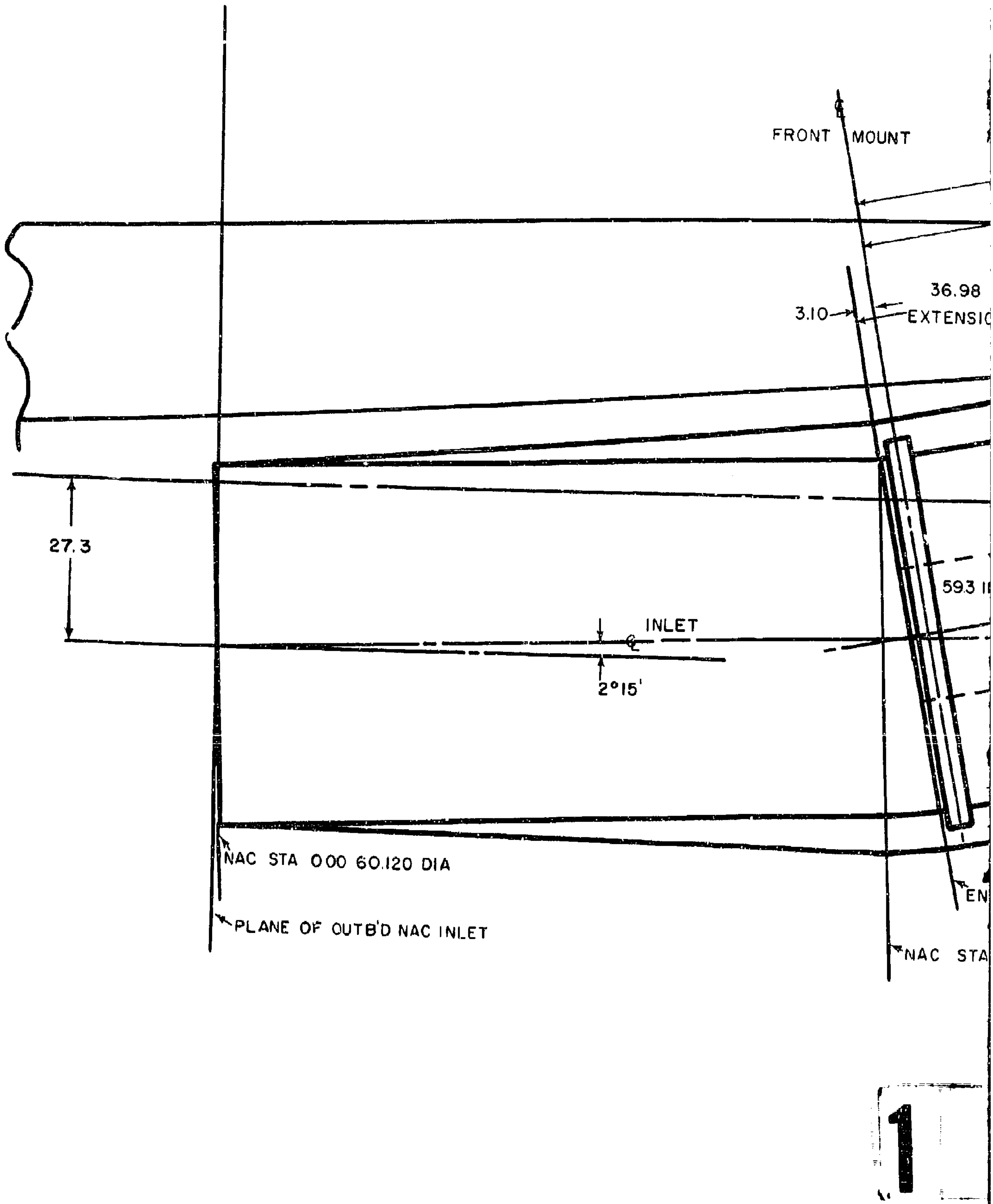


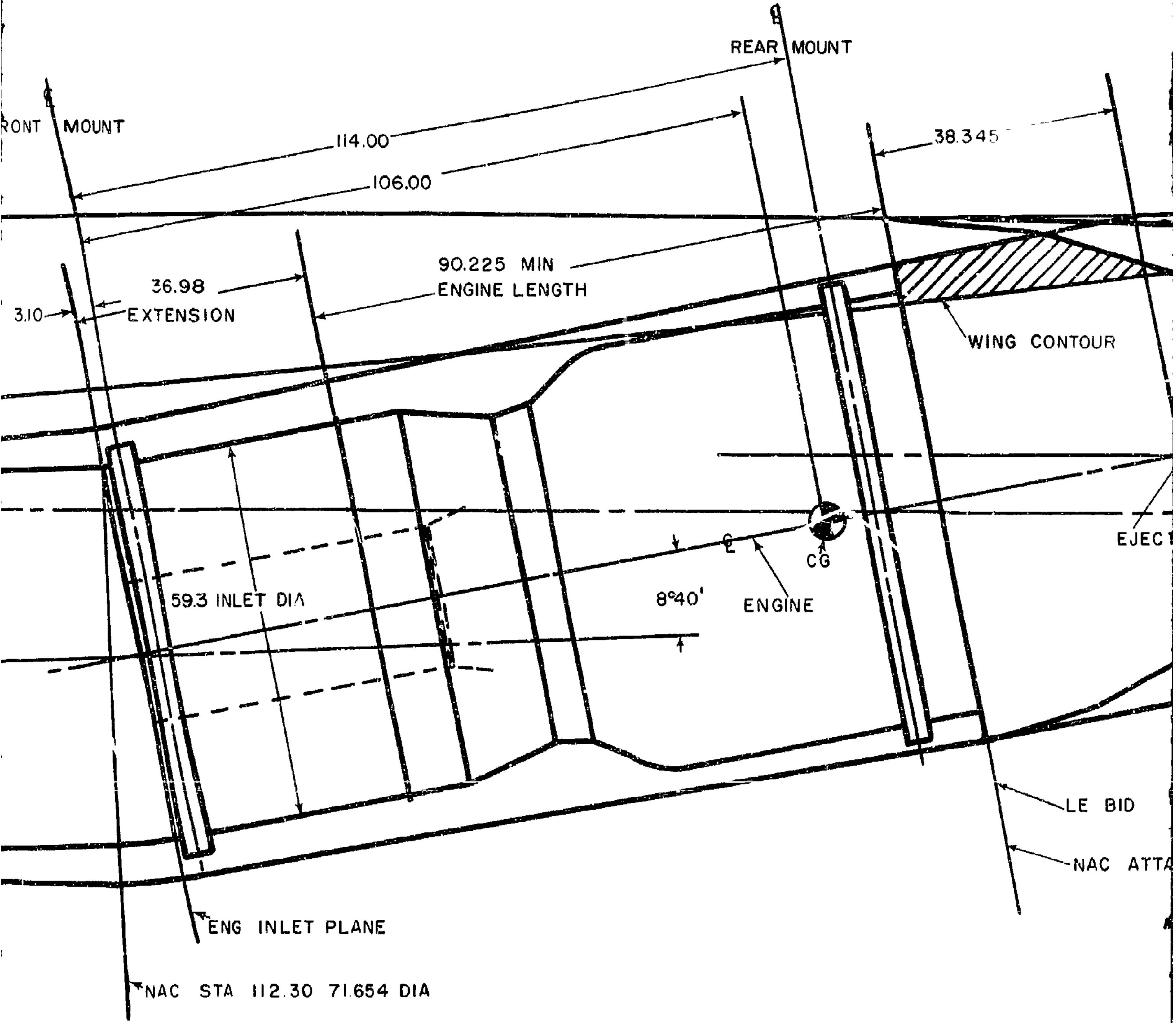
EJECTOR CANTED 10.00 FWD. OF PRIMARY
NOZZLE PLANE
CG LOCATION DOES NOT INCLUDE INLET

CG LOCATION DOES NOT INCLUDE INLET

STF219 650 LBS./SEC. TURBOFAN

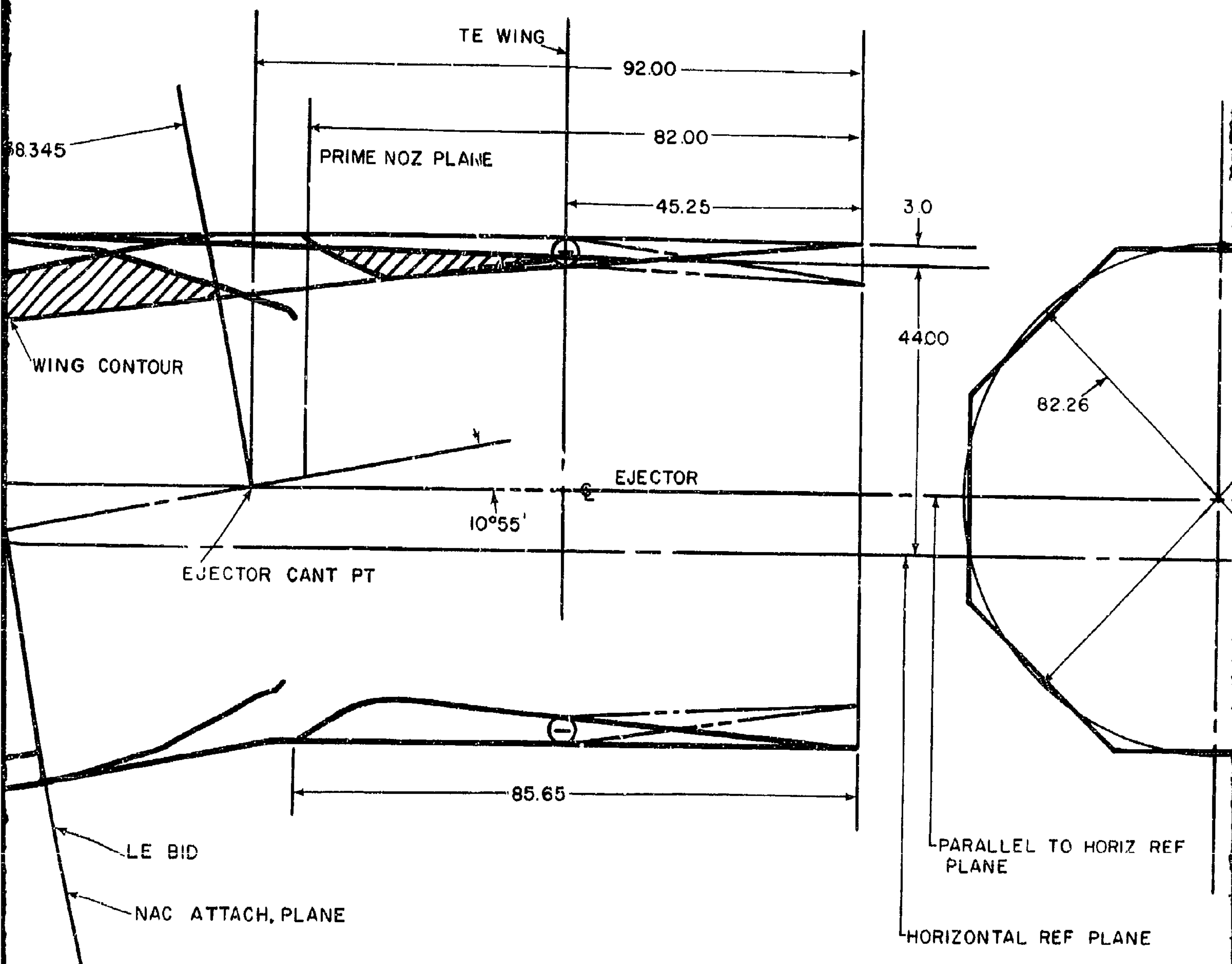
Figure 1-12


FRONT MOUNT
36.98
3.10
EXTENSIO
27.3
59.3 I
INLET
2°15'
NAC STA 0.00 60.120 DIA
PLANE OF OUTB'D NAC INLET
EN
NAC STA
1



℄ REAR MOUNT
℄ FRONT MOUNT
114.00
106.00
38.345
90.225 MIN ENGINE LENGTH
36.98 EXTENSION
3.10
WING CONTOUR
59.3 INLET DIA
8°40'
℄ ENGINE
CG
EJECT
LE BID
NAC ATTA
ENG INLET PLANE
NAC STA 112.30 71.654 DIA

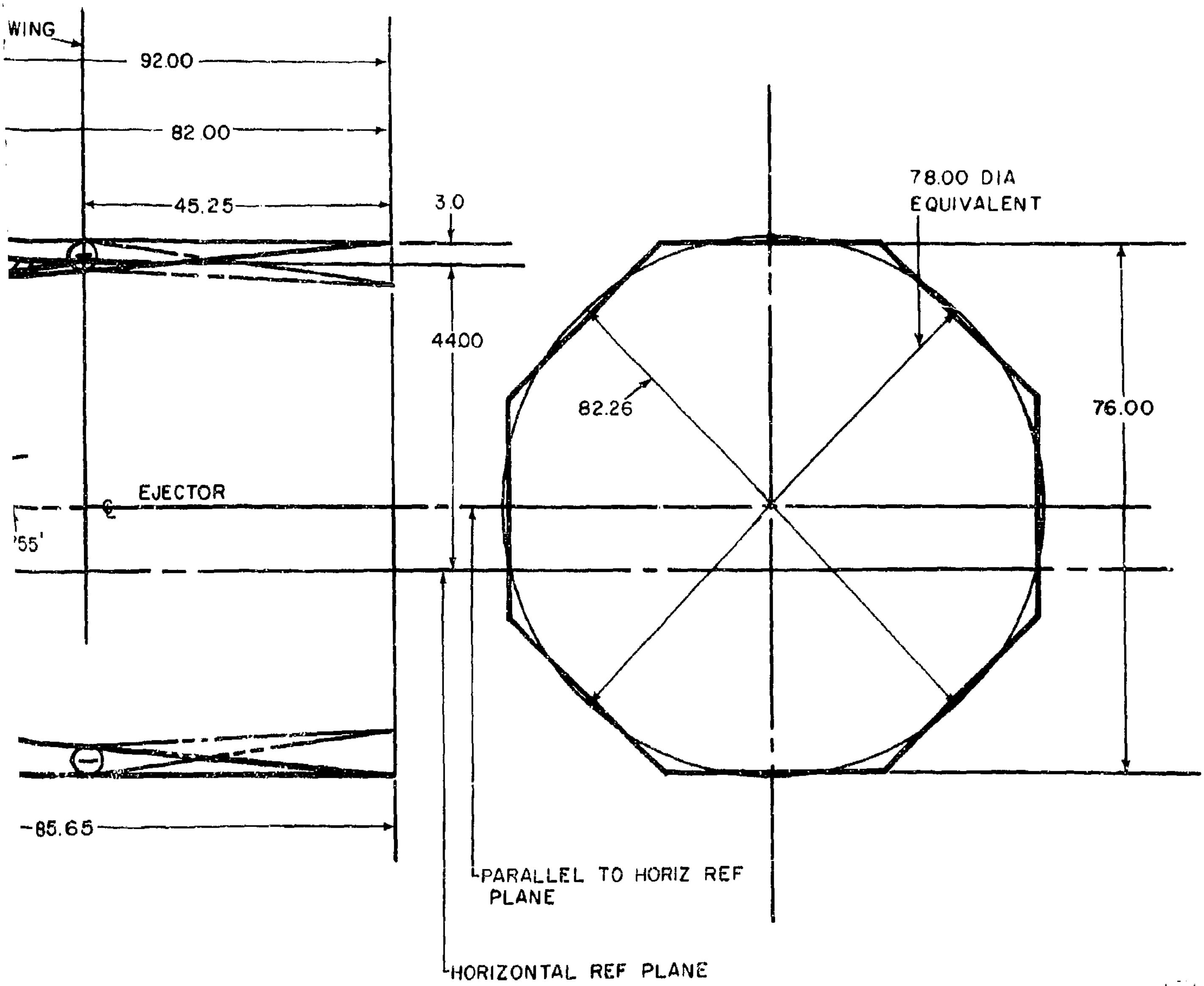
2



EJECTOR CANTED 10
NOZZLE PLANE
CG LOCATION DOES N

STF219 600 LBS./SEC. TUR

Figure 1-13



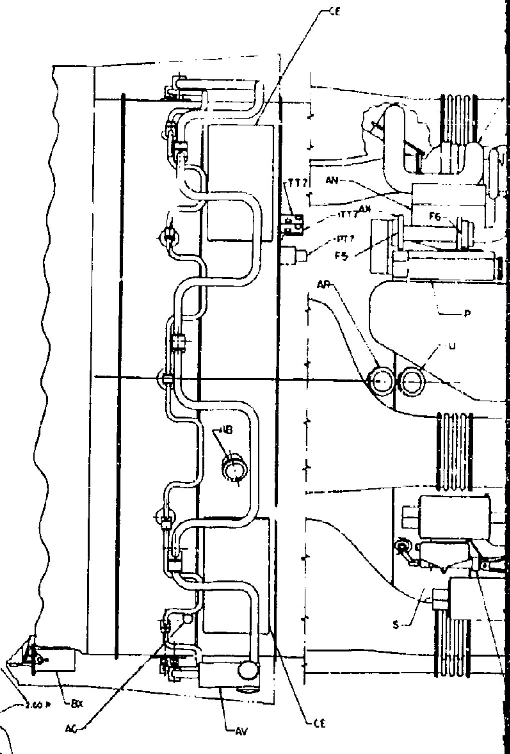
EJECTOR CANTED 10.00 FWD. OF PRIMARY NOZZLE PLANE
CG LOCATION DOES NOT INLCUDE INLET

STF219 600 LBS./SEC. TURBOFAN

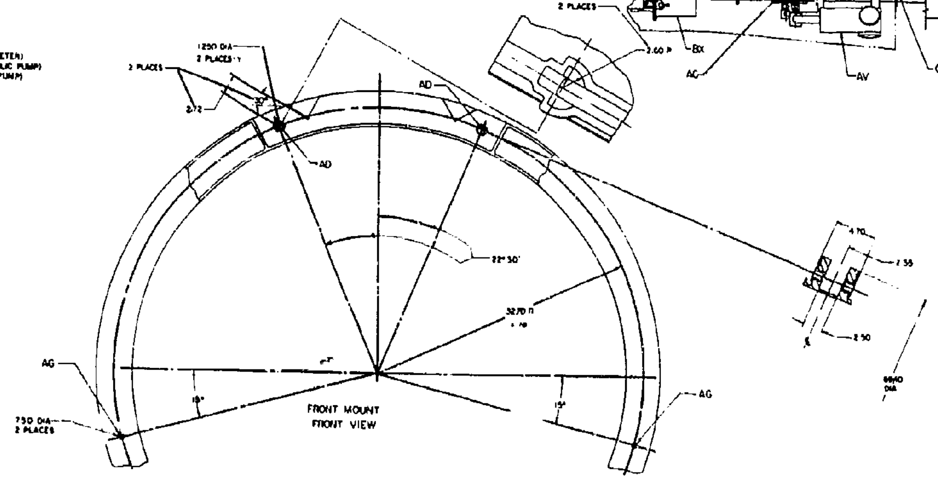
Figure 1-13

ZONES		
		ACCESSORY DRIVE PADS
23-F,32F	A	POWER TAKEOFF
23-D,19-C	B	TACHOMETER MTG PAD (N1) (AND 20005 x VIII)
18-E,22-E	C	TACHOMETER MTG PAD (N2) (AND 20006 x VIII)
18-D,27-D	D	HYDRAULIC PUMP (TWO) (AS 4048)
		FUEL DRAIN
17-C	FD1	COMBUSTION CHAMBER FUEL DRAIN
23-C,16-C	FD2	DUMP VALVE DRAIN (GAS GENERATOR)
23-C	FD4	DUMP VALVE DRAIN (DUCT HEATER ZONE I)
23-C	FD4A	DUMP VALVE DRAIN (DUCT HEATER ZONE II)
		FUEL PRESSURE
18-D	FP1	FUEL PUMP INLET PRESSURE
23-D	FP3	CHECK VALVE FUEL PRESSURE
		FUEL FLOW
19-F,23-E,27-F	F1	FUEL PUMP & DUCT HEATER PUMP SUPPLY INLET
17-C	F2	MAIN FUEL FLOWMETER SUPPLY INLET
17-C	F3	MAIN FUEL FLOWMETER SUPPLY OUTLET
22-E	F5	DUCT HEATER FLOWMETER SUPPLY INLET
19-E	F6	DUCT HEATER FLOWMETER SUPPLY OUTLET
		FUEL VENT
18-C	FV2	FUEL PUMP OUTLET VENT
		OIL BREATHER
17-E	LB2	MAIN OIL OVERBOARD BREATHER
		OIL DRAIN
24-D,28-D	LD1	OIL TANK DRAIN
24-D,28-D	LD2	OIL TANK OVERFLOW DRAIN
24-E,28-E	LD3	OIL CUP OVERFLOW DRAIN
18-D	LD4	GEARBOX MAIN OIL DRAIN
17-D	LD7	OIL STRAINER DRAIN
		OIL FLOW
24-E,28-E	L1	OIL TANK REMOTE FILLER
25-E,28-E	L2	OIL TANK MANUAL FILL
		OIL PRESSURE
17-F	LP1	PRESSURE FOR TRANSMITTER
17-D	LP4	OIL FILTER INLET PRESSURE — PROVISIONS FOR PRESSURE DIFFERENTIAL
17-D	LP5	OIL FILTER OUTLET PRESSURE — PROVISIONS FOR PRESSURE DIFFERENTIAL
		OIL TEMPERATURE
22-E	LT1	MAIN OIL TEMPERATURE
		OIL VENT
17-E,18-D	LV3	OIL PRESSURE TRANSMITTER VENT
		SEAL DRAIN
18-D	SD1	FUEL CONTROL SEAL DRAIN
18-D,27-D	SD2	FUEL PUMP SEAL DRAIN
18-D	SD3	ACCESSORY DRIVE OVERBOARD DRAIN (TACHOMETER)
17-D	SD4	ACCESSORY DRIVE OVERBOARD DRAIN (HYDRAULIC PUMP)
18-D	SD5	ACCESSORY DRIVE OVERBOARD DRAIN (FUEL PUMP)
17-D	SD9	HYDRULIC PUMP SEAL DRAIN
23-E	SD14	DUCT HEATER FUEL PUMP SEAL DRAIN
		TEMPERATURE SENSING
29-E	TT7	TURBINE EXIT TEMPERATURE (AVERAGE)
29-E	ITT7	TURBINE EXIT TEMPERATURE (INDIVIDUAL)
		PRESSURE SENSING
29-E	PT7	TURBINE EXIT PRESSURE

ZONES		MISCELLANEOUS
25-E	E	OIL TANK
19-D,21-D	F	GAS GENERATOR FUEL PUMP
22-D,19-D	G	FUEL PUMP FUEL FILTER (GAS GENERATOR)
22-C	H	FUEL PUMP FILTER (MIN SPACE FOR REMOVAL)
23-F,19-F	J	DUCT HEATER FUEL PUMP
17-D	K	HYDRAULIC PUMP
17-D	L	OIL PUMP
17-D	M	OIL FILTER
23-F,19-F	N	FUEL-OIL COOLER (GAS GENERATOR)
29-E,23-E	P	FUEL-OIL COOLER (DUCT HEATER)
17-E,21-E	R	GEARBOX
24-E,19-E	S	AUTOMATIC RESTART SWITCH
17-E	T	BREATHER PRESSURIZING VALVE
29-E,17-C,19-F	U	HIGH PRESSURE BLEED PAD
21-E,19-E	V	UNITIZED CONTROL (FUEL, AUGMENTATION & NOZZLE)
25-D,28-D	AA	EXCITER (GAS GENERATOR)
22-D,23-D,30-D,19-C	AB	IGNITER PLUG - (GAS GENERATOR) (MIN SPACE FOR REMOVAL)
31-C,24-D,25-C,19-C	AC	IGNITER PLUG - (DUCT HEATER) (MIN SPACE FOR REMOVAL)
32-B	AD	ENGINE FRONT MOUNTING PROVISIONS
16-F,6-C	AE	ENGINE REAR MOUNT PROVISIONS
17-D	AF	FUEL RETURN TO SUPPLEMENTARY COOLING
34-A,31-A,30-D,20-D	AG	GROUND HANDLING HOLES (FRONT MOUNT)
6-B	AH	GROUND HANDLING AREA (REAR)
19-E,21-E	AJ	POWER CONTROL LEVER (120° ANGLE OF TRAVEL)
19-E,21-E	AK	SHUT OFF LEVER (90° ANGLE OF TRAVEL)
15-E,21-E	AL	APPROACH VELOCITY CONTROL LEVER (90° ANGLE OF TRAVEL)
19-E	AM	MACH NO. OR SHOCK POSITION RESET LEVER (60° ANGLE OF TRAVEL)
24-F,24-F	AN	AIR TURBINE VALVE DUCT HEATER FUEL PUMP
25-D,28-D	AP	IGNITION EXCITER ELECTRICAL CONN (GAS GENERATOR)
29-E	AR	THERMAL ANTI ICING PAD
21-E,19-E	AS	AERODYNAMIC BRAKE CONTROL AIR SUPPLY CONN
19-F,21-F	AT	DUCT NOZZLE POSITION FEEDBACK
22-C,17-B	AU	CHECK & DUMP VALVE (DUCT HEATER ZONE I)
23-C,30-C	AV	CHECK & DUMP VALVE (DUCT HEATER ZONE II)
17-B,21-C	AW	GAS GENERATOR FUEL FLOWMETER MOUNTING PROVISIONS
24-F,29-E	AX	DUCT HEATER FUEL FLOWMETER MOUNTING PROVISIONS
18-E	AY	FUEL CONTROL FUEL FILTER (MIN SPACE FOR REMOVAL)
18-C,22-C	AZ	CHECK & DUMP & LOW IDLE VALVE (GAS GENERATOR FUEL)
19-C	BA	CHECK & DUMP VALVE FILTER
17-F,22-F	BB	NOZZLE POSITION INDICATOR MOUNTING PROVISIONS
18-F	BC	REVERSER POSITION INDICATOR ELECTRICAL CONN
17-E	BD	THERMOSTATIC FUEL BYPASS VALVE
17-E	BE	HYDRAULIC RETURN FILTER
9A,16D,29-D	BF	VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFG)
8-C,29-D	BP	SECONDARY AIR BYPASS DUCTS
19-C,22-D	BR	FUEL FILTER PRESSURE ELECTRICAL CONNECTION
18-D	BS	FUEL INLET TEMPERATURE SENSOR
8-C	BT	OIL SCAVENGE PUMP-BEARING COMPARTMENT
22-C,18-C	BV	HYDRAULIC DISCHARGE FILTER
5-C	BX	DUCT NOZZLE ACTUATORS (8 REQD)
22-C,19-C,26-C	CA	AERODYNAMIC BRAKE ACTUATORS
12-B	CC	THRUST REVERSING ACTUATOR (2 REQD)
17-C,17-F,29-C,29-F	CE	GAS GENERATOR ACCESS DOORS
24-C	CF	EXCITER (DUCT HEATER)
24-C,29-C	CG	IGNITION EXCITER ELECTRICAL CONN (GAS GENERATOR)

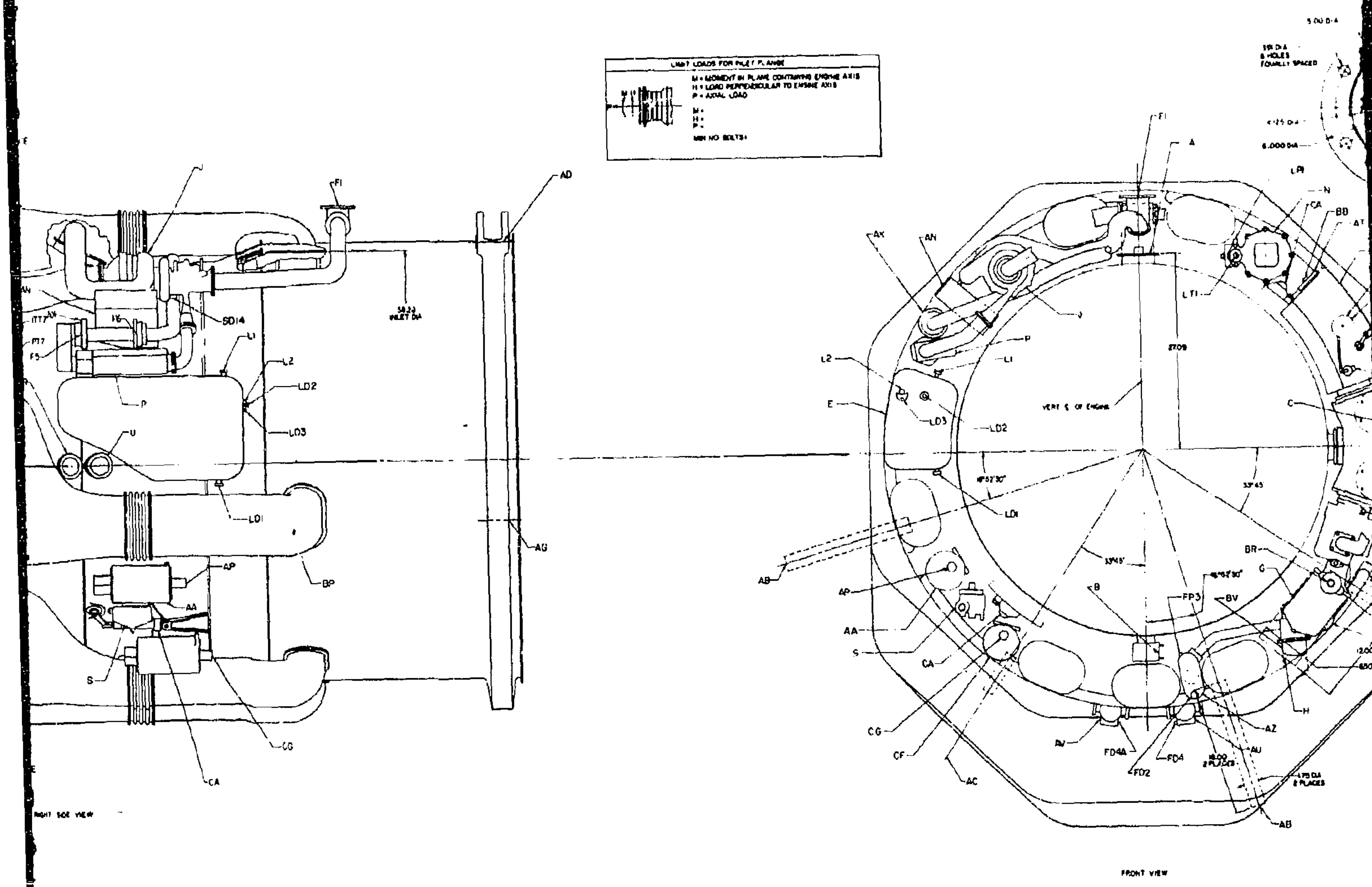


RIGHT SIDE VIEW



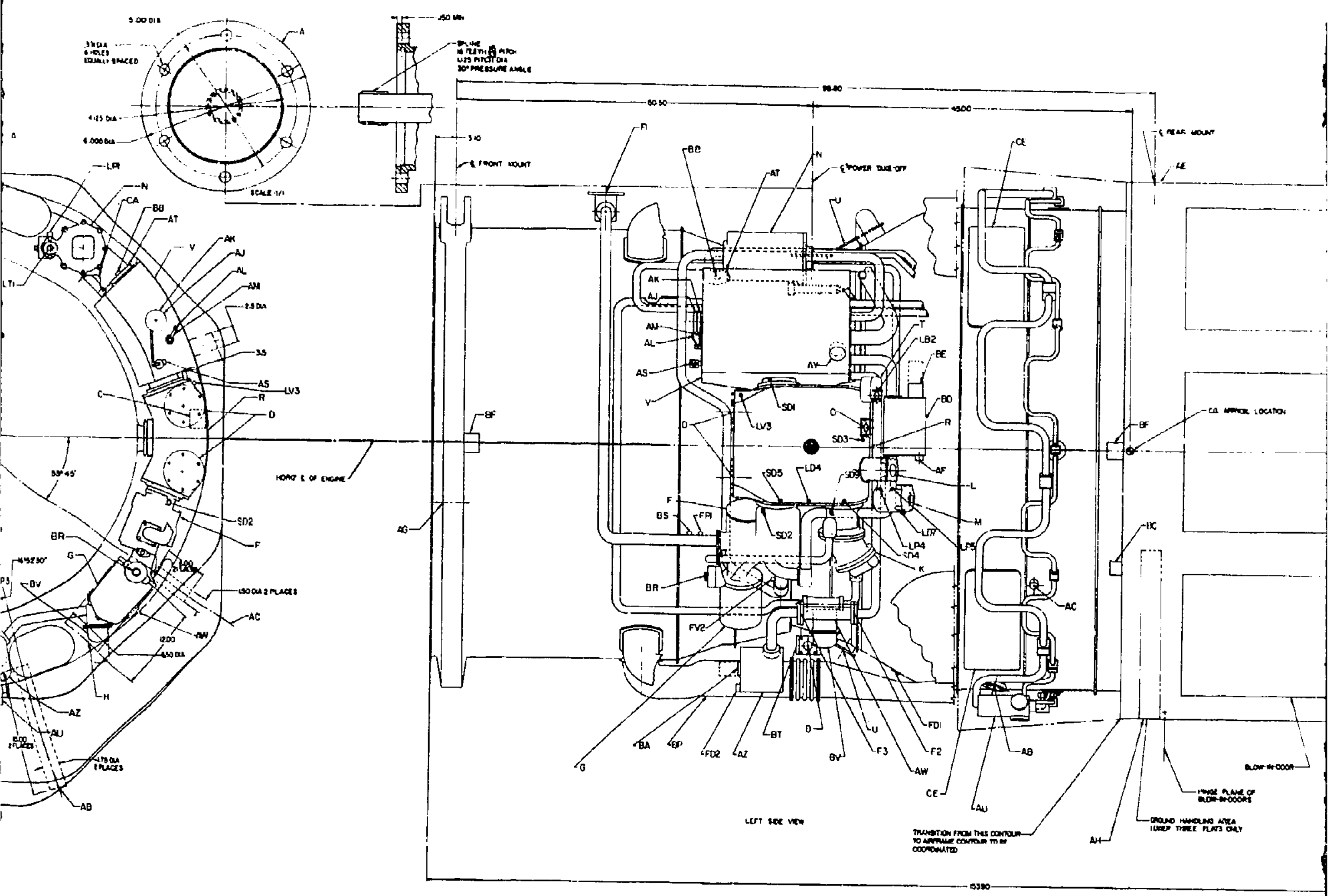
FRONT MOUNT
FRONT VIEW

1

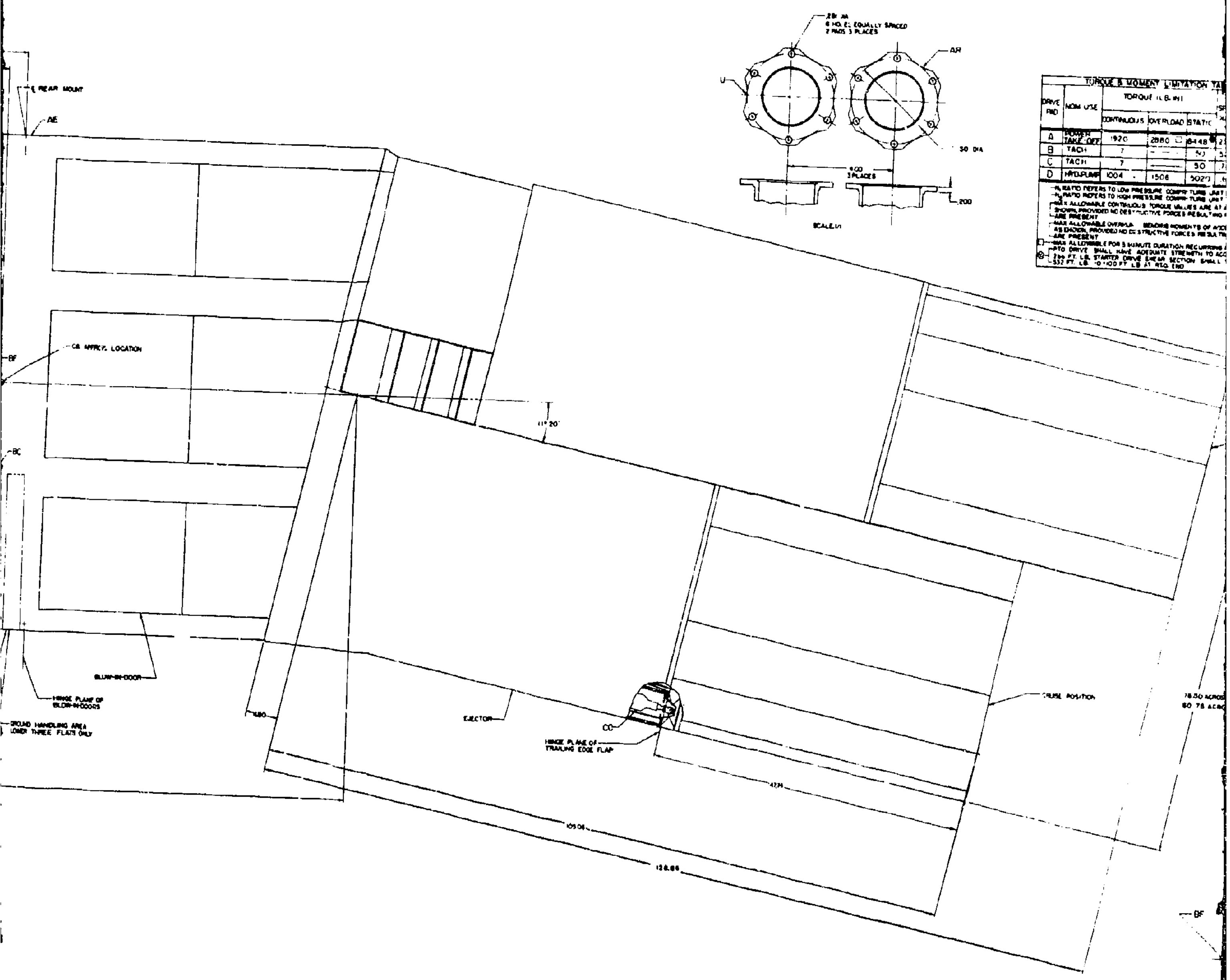


LIMIT LOADS FOR INLET FLANGE
M = MOMENT IN PLANE CONTAINING ENGINE AXIS
H = LOAD PERPENDICULAR TO ENGINE AXIS
P = AXIAL LOAD
M =
H =
P =
MIN NO BOLTS =
AD
FI
J
AN
SD14
L1
L2
LD2
LD3
P
U
LD1
AG
BP
AP
AA
S
CG
CA
58.32 INLET DIA
RIGHT SIDE VIEW
A
AX
N
CA
BB
LT1
C
E
VERT ℄ OF ENGINE
27.09
33°45'
33°45'
BR
G
B
FP3
BV
H
AZ
AU
FD4A
FD4
FD2
CF
AC
AB
FRONT VIEW

2


LEFT SIDE VIEW
SCALE 1/1
HORIZ ℄ OF ENGINE
℄ FRONT MOUNT
℄ REAR MOUNT
℄ POWER TAKE-OFF
C.G. APPROX. LOCATION
BLOW-IN-DOOR
HINGE PLANE OF BLOW-IN-DOORS
GROUND HANDLING AREA (KEEP THREE FLATS ONLY)
TRANSITION FROM THIS CONTOUR TO AIRFRAME CONTOUR TO BE COORDINATED
SPLINE
30° PRESSURE ANGLE
1.50 DIA 2 PLACES
33°45'

3

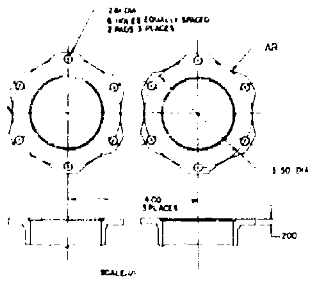


TORQUE & MOMENT LIMITATION TA[illegible]

DRIVE PAD	NOM USE	TORQUE (LB-IN) CONTINUOUS	OVERLOAD	STATIC	[illegible]
A	POWER TAKE OFF	1920	2880	8448	[illegible]
B	TACH	7	—	50	[illegible]
C	TACH	7	—	50	[illegible]
D	HYD-PUMP	1004	1506	5027	[illegible]

PROPOSED TURB[illegible]

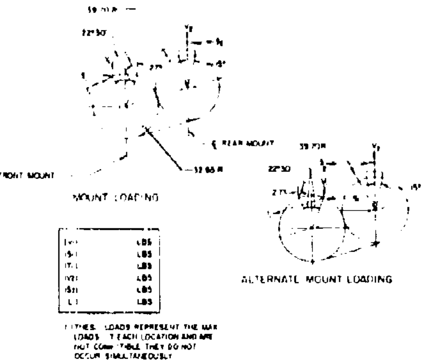
4

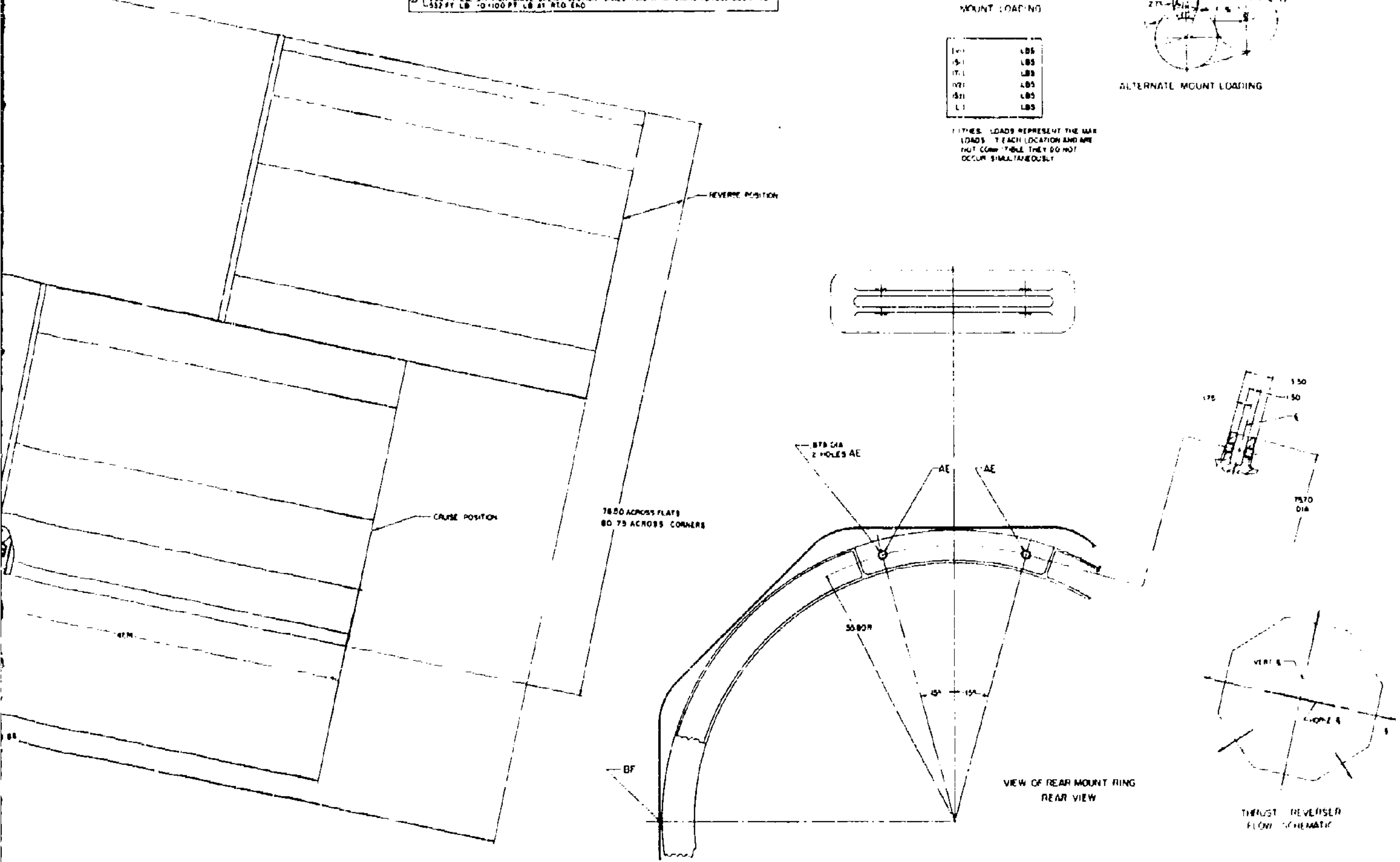


TORQUE & MOMENT LIMITATION TABLE ACCESS DRIVES

DRIVE PAD	NOM USE	TORQUE (LB IN)			SPEED RATIO	N_1 N_2	ROTATION (FACING ENGINE PAD)	1G OVERHUNG MOMENT (LB IN)
		CONTINUOUS	OVERLOAD	STATIC				
A	POWER TAKE-OFF	1920	2880	8448	2.324	N_2	CCW	------
B	TACH	7		50	.508	N_2	CCW	------
C	TACH	7	------	50	.708	N_1	CCW	------
D	HYD PUMP	1004	1506	5020	.600	N_1	CW	------

- N_1 RATIO REFERS TO LOW PRESSURE COMPR-TURB UNIT SPEED
- N_2 RATIO REFERS TO HIGH PRESSURE COMPR TURB UNIT SPEED
- MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT ANY ENGINE SPEED UNLESS OTHERWISE SHOWN, PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS TORSIONAL VIBRATION ARE PRESENT
- MAX ALLOWABLE OVERHUNG BENDING MOMENTS OF ACCESSORIES ABOUT DRIVE PAD FACE ARE AS SHOWN, PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION ARE PRESENT
- MAX ALLOWABLE FOR 5 MINUTE DURATION RECURRING AT 4 HR INTERVALS MIN
- PTO DRIVE SHALL HAVE ADEQUATE STRENGTH TO ACCOMMODATE A MAX TORQUE EQUIV TO 256 FT LB. STARTER DRIVE SHEAR SECTION SHALL FAIL AT A STATIC TORQUE EQUIV TO 532 FT LB 10,100 FT LB AT PTO END

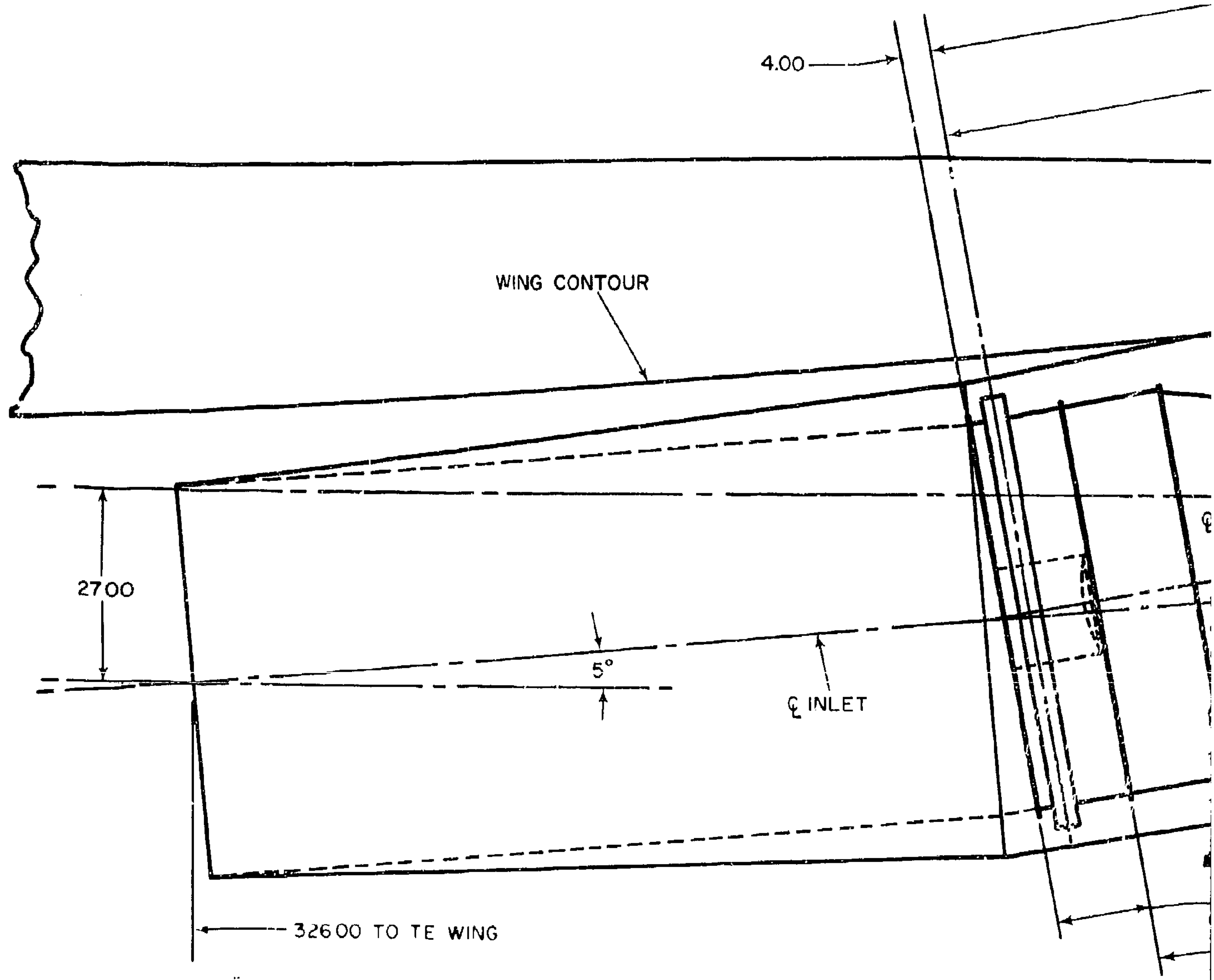




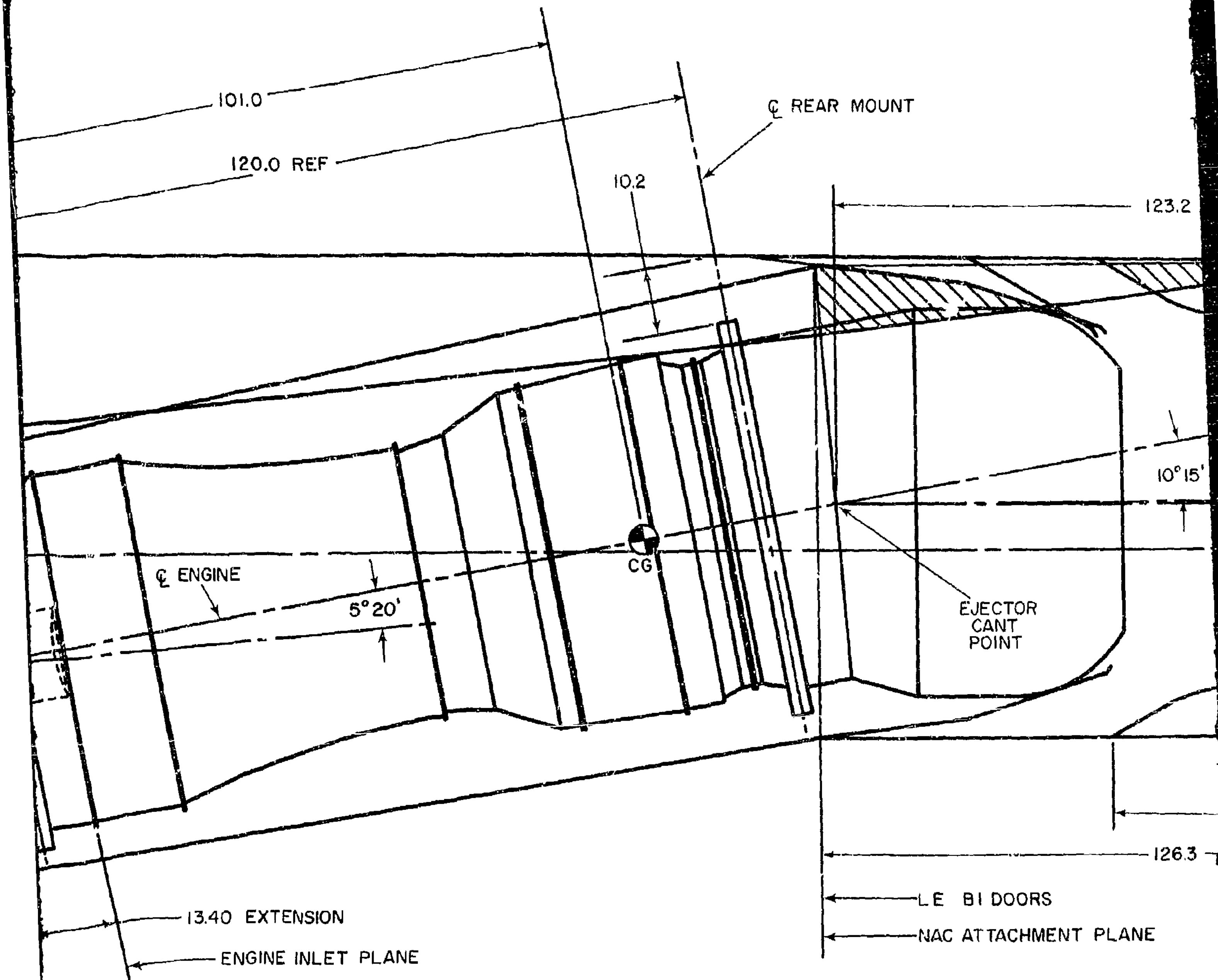
PROPOSED TURBOFAN ACCESSORY ARRANGEMENT

Figure 1-14

5

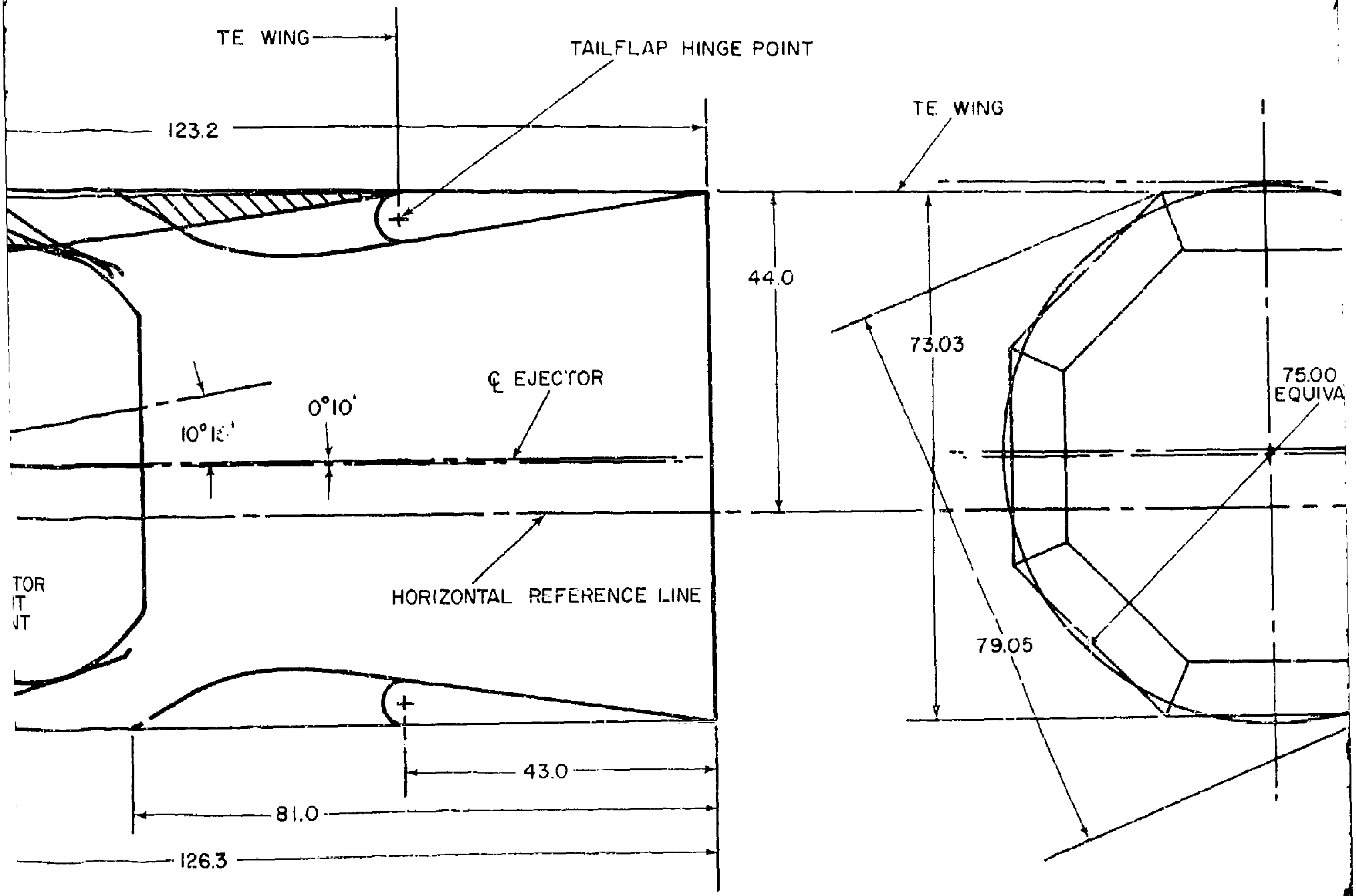


1



2300°F MAX. TURBINE I
EJECTOR CANTED AT R
CG LOCATION DOES NO

2



DORS

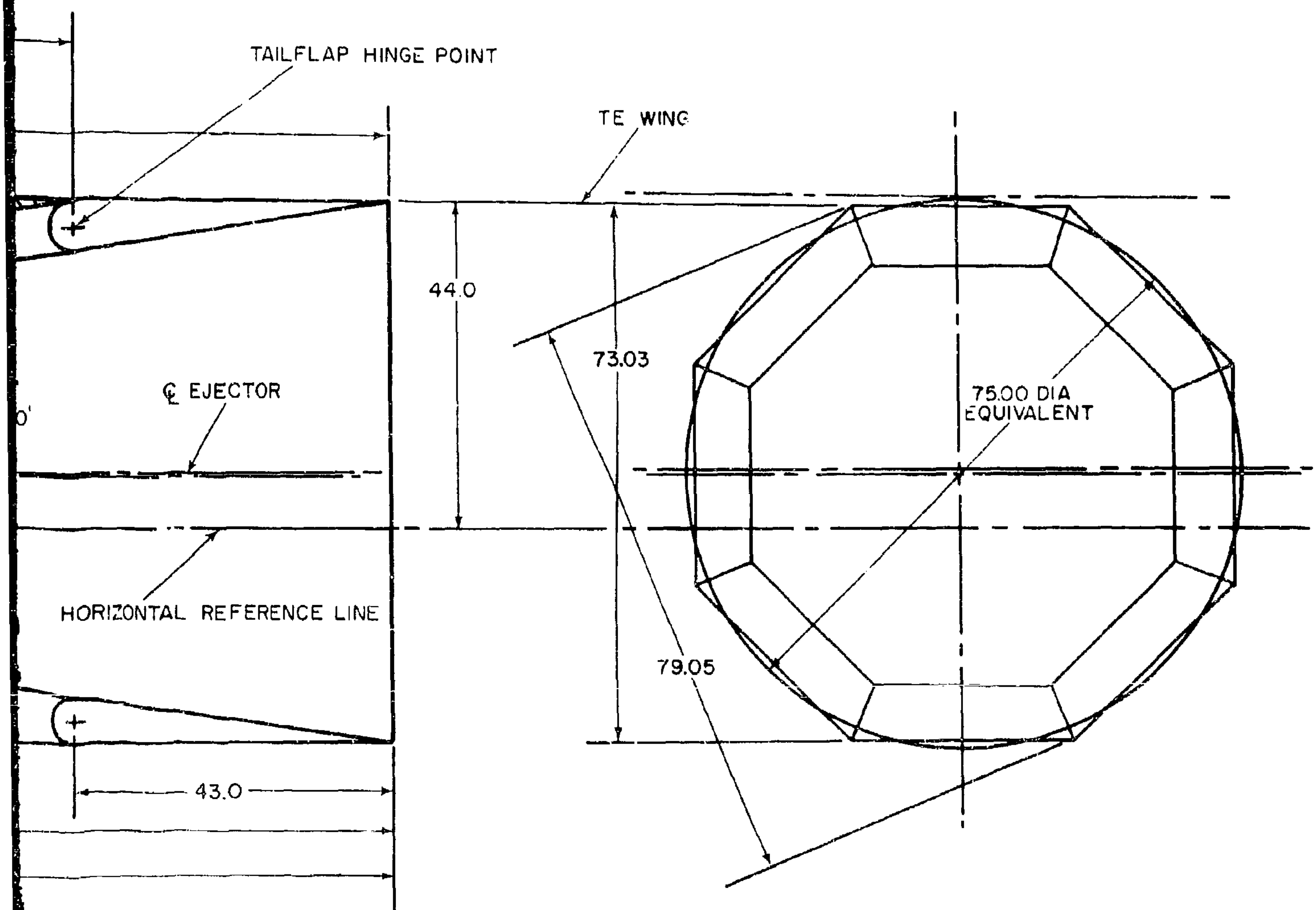
CHMENT PLANE

CG LOCATION DOES NOT INCLUDE THE INLET

F MAX. TURBINE INLET TEMP.
TOR CANTED AT REAR MOUNT PLANE
OCATION DOES NOT INCLUDE THE INLET

STJ227 500 LBS./SEC. TURBOJET

Figure 1-15



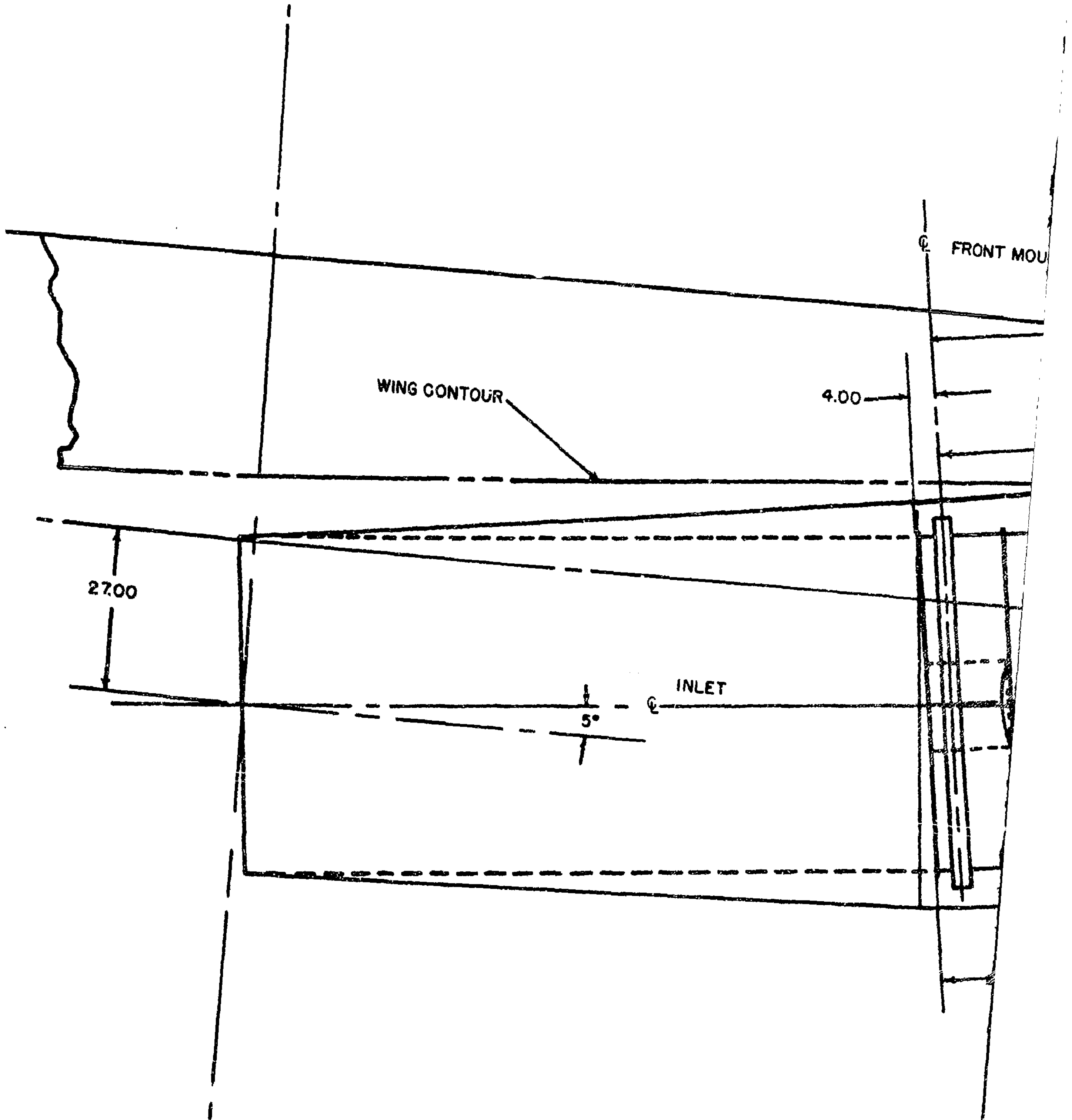
CG LOCATION DOES NOT INCLUDE THE INLET

TEMP.
OUNT PLANE
DE THE INLET

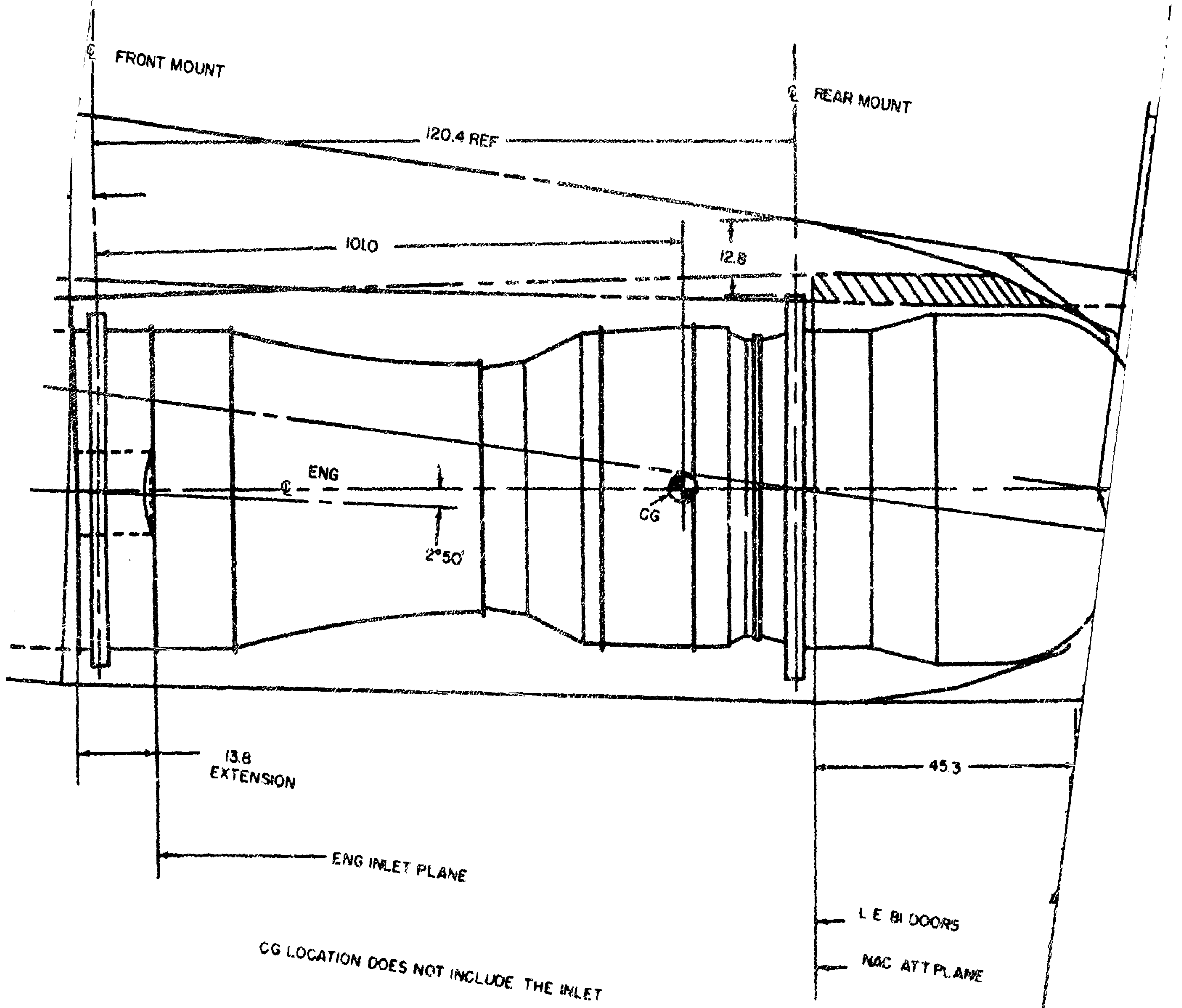
STJ227 500 LBS./SEC. TURBOJET

Figure 1-15

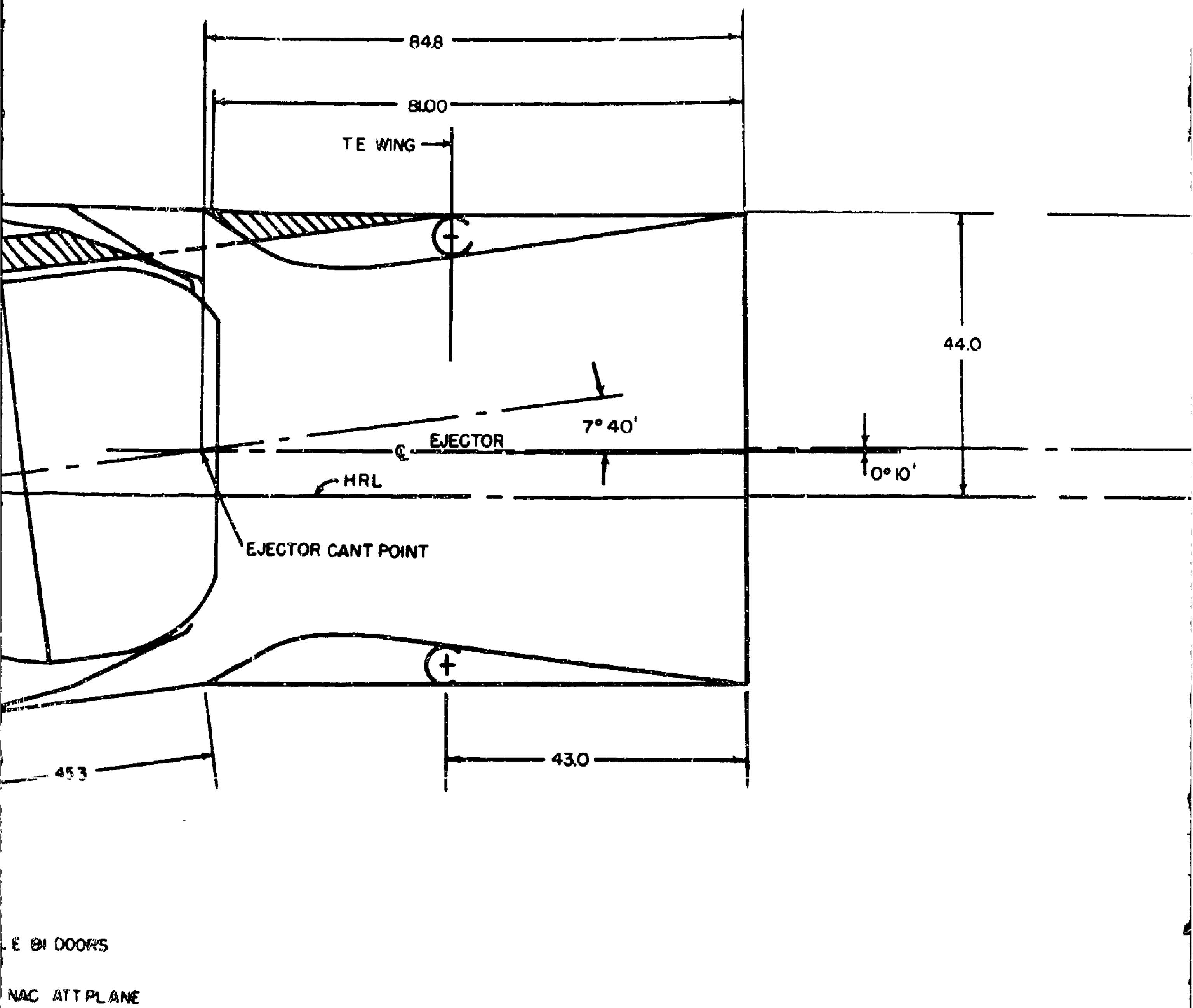
4



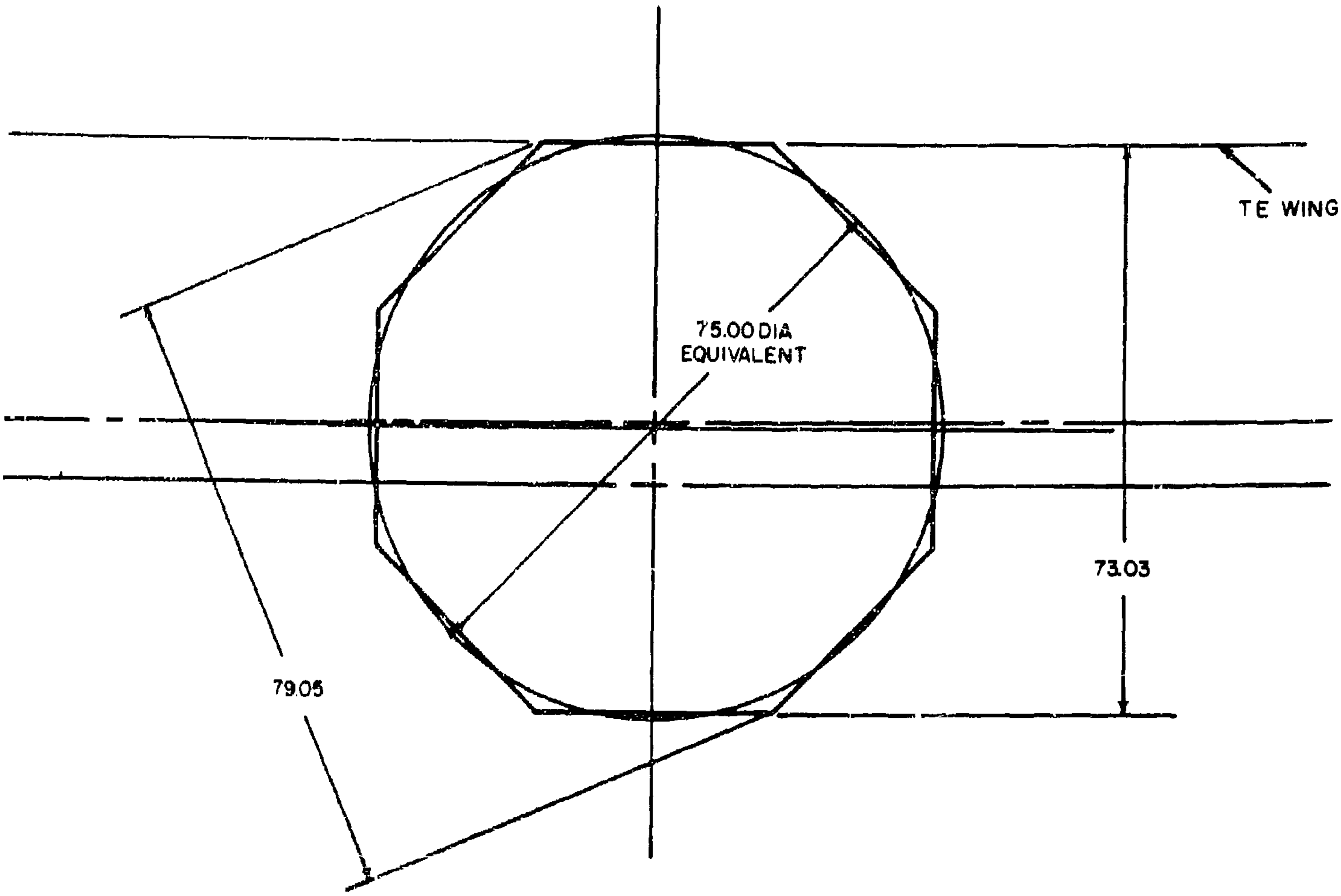
FRONT MOU
WING CONTOUR
4.00
27.00
INLET
5°



℄ FRONT MOUNT
℄ REAR MOUNT
120.4 REF
101.0
12.8
℄ ENG
CG
2°50'
13.8
EXTENSION
45.3
ENG INLET PLANE
L E BI DOORS
NAC ATT PLANE
CG LOCATION DOES NOT INCLUDE THE INLET


84.8
81.00
TE WING
44.0
7° 40'
℄ EJECTOR
0° 10'
HRL
EJECTOR CANT POINT
45.3
43.0
E 81 DOORS
NAC ATT PLANE

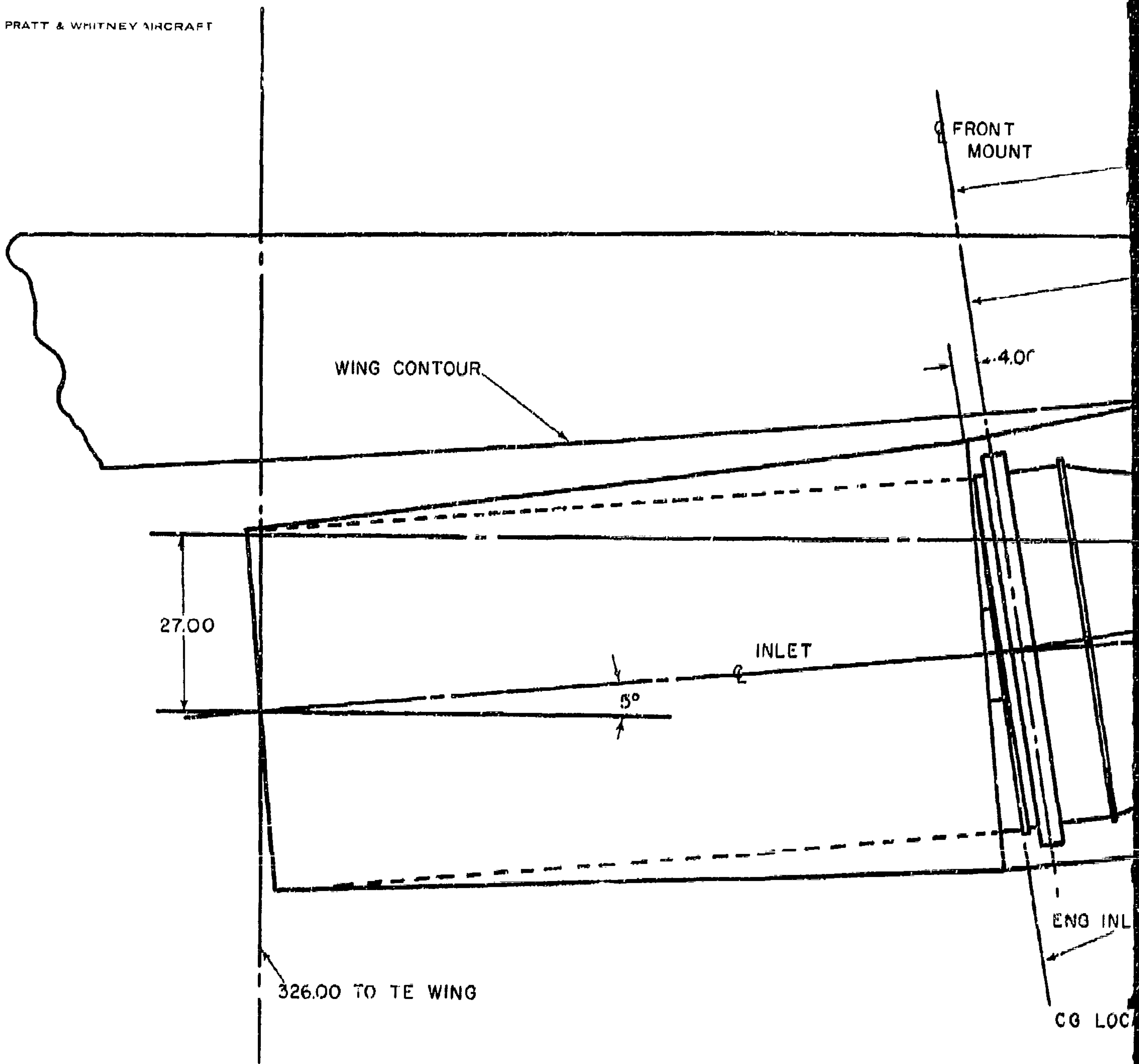
3



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

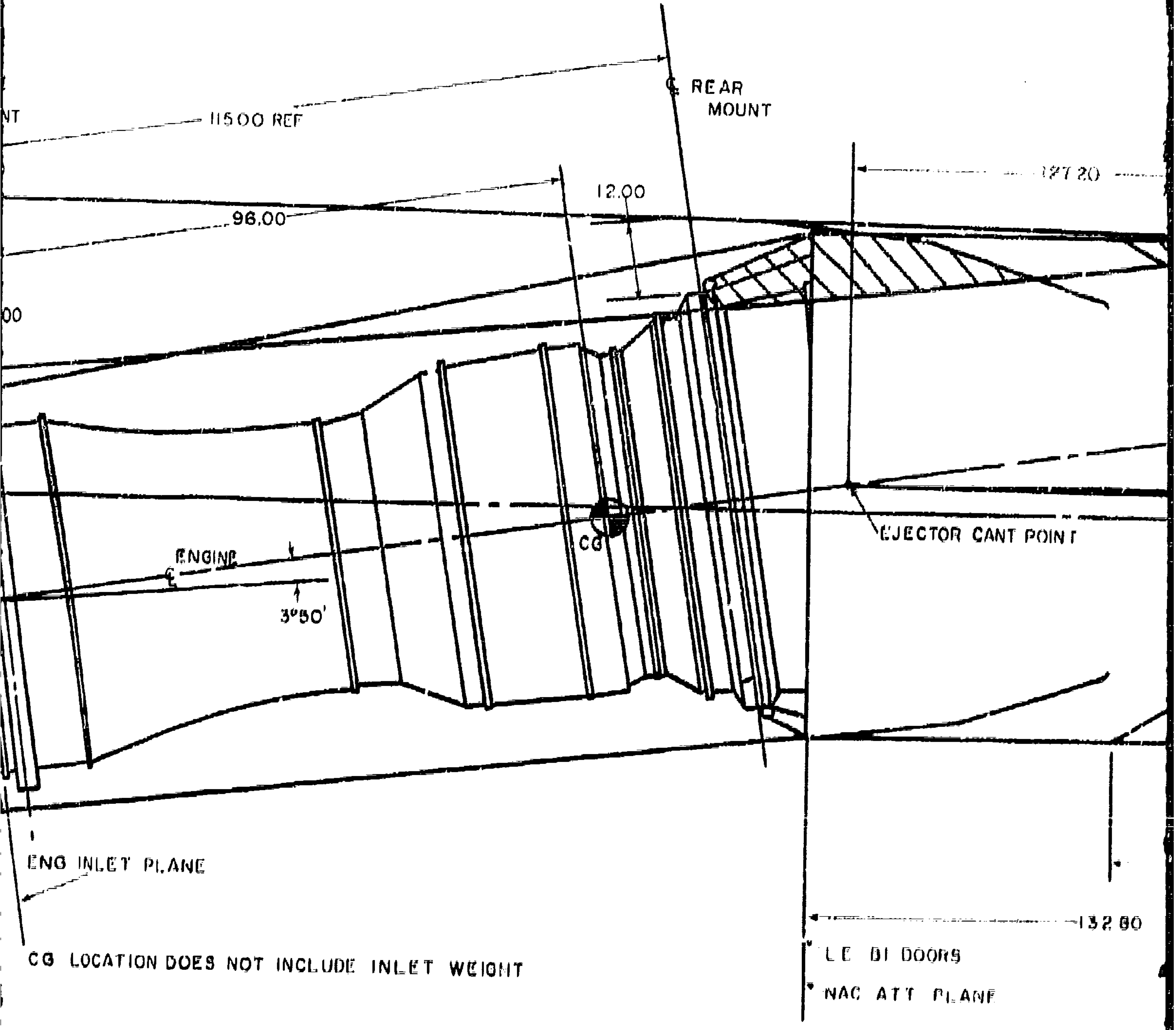
STJ227 500 LBS./SEC. TURBOJET

Figure 1-16



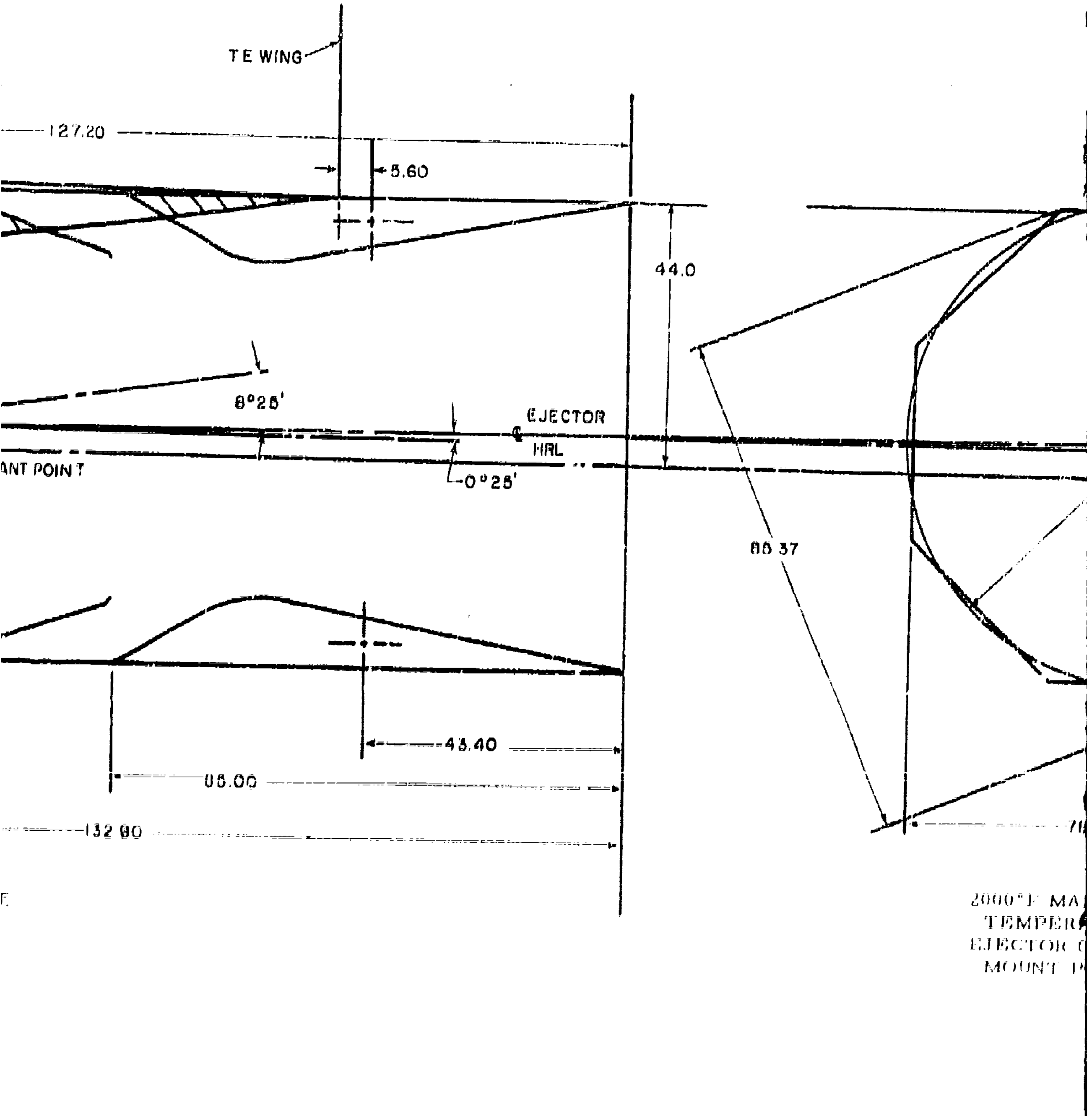
FRONT MOUNT
WING CONTOUR
4.0
27.00
INLET
5°
ENG INL
326.00 TO TE WING
CG LOC

1



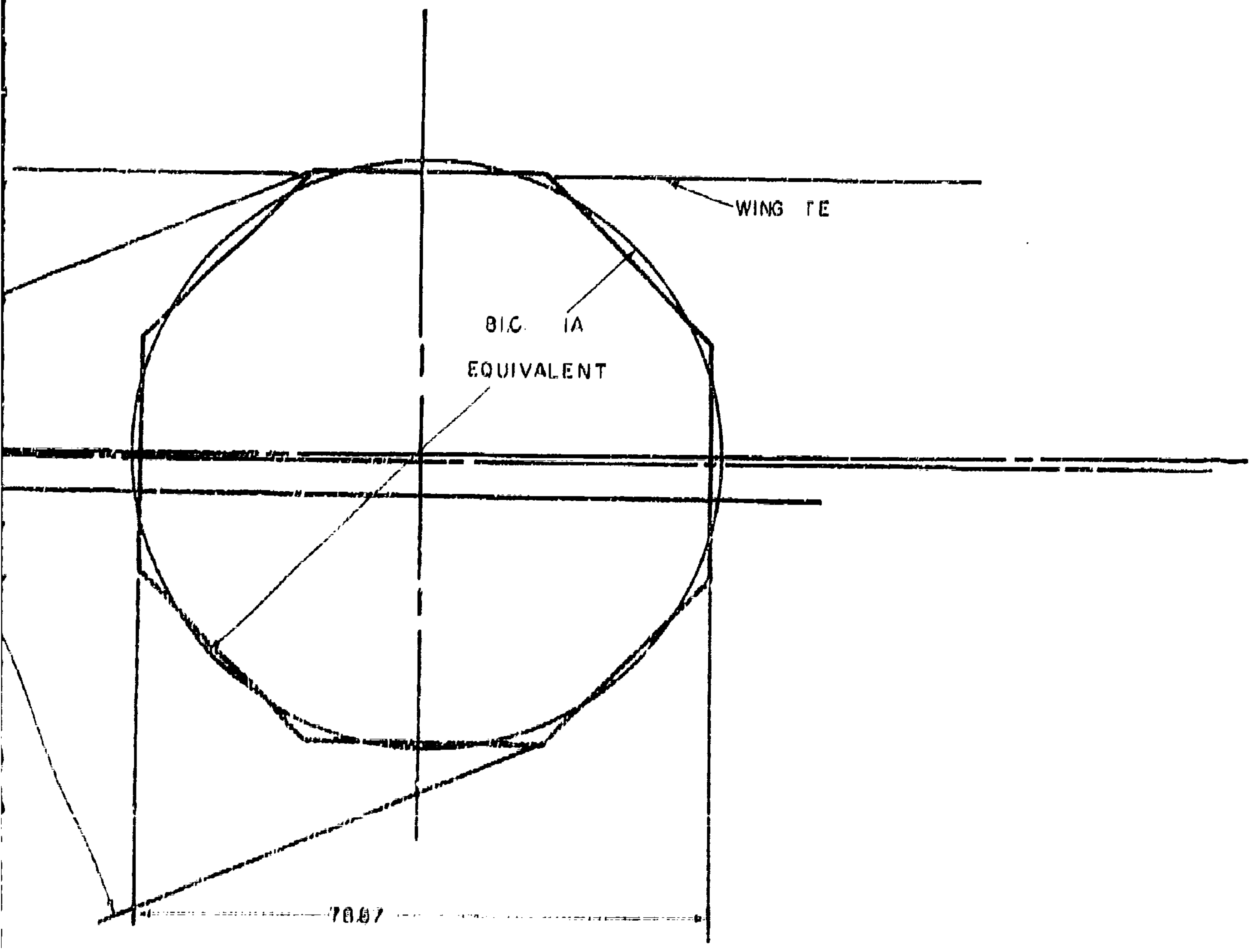
CG LOCATION DOES NOT INCLUDE INLET WEIGHT

2



2000°F MA
TEMPER
EJECTOR
MOUNT

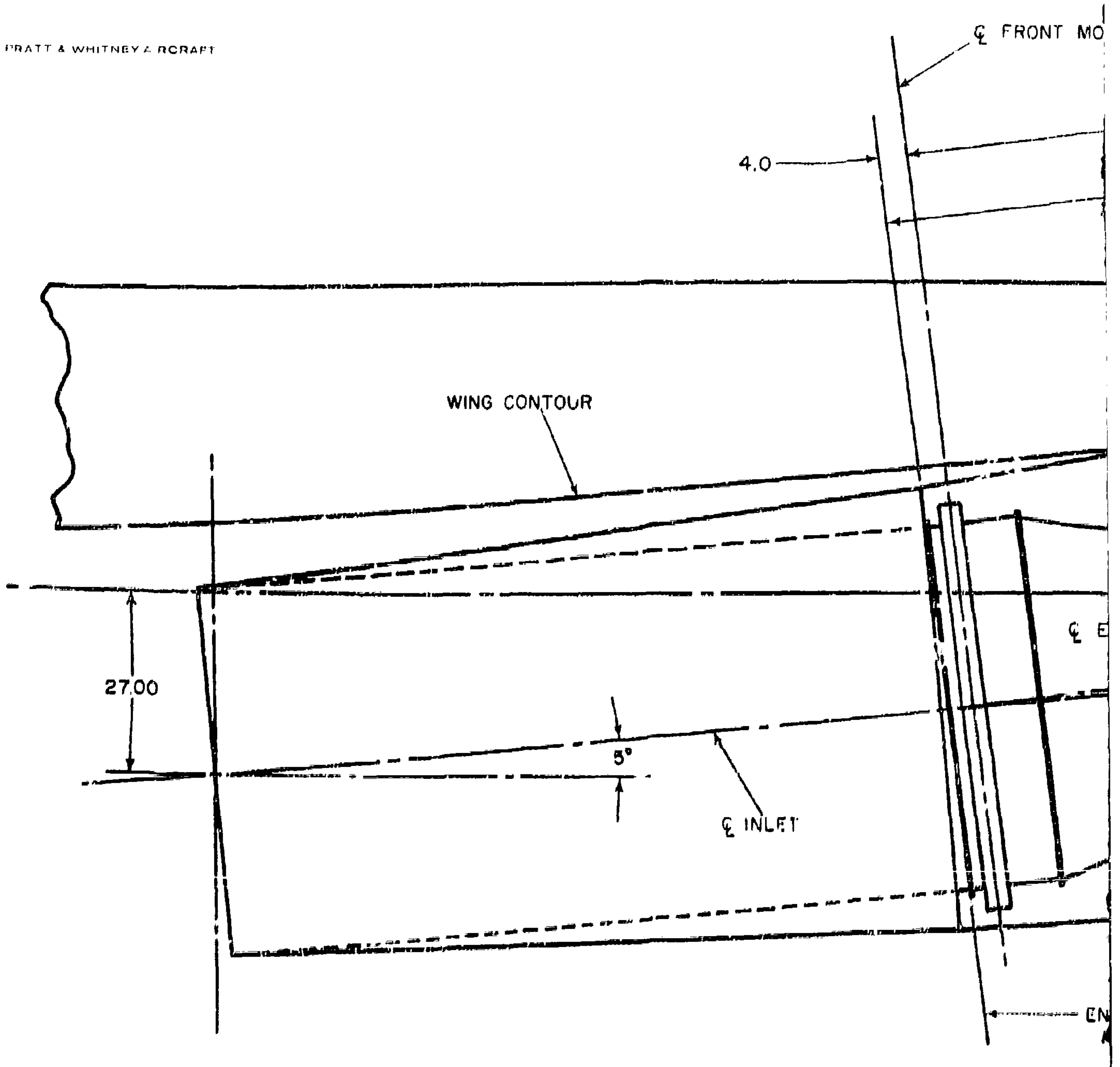
3



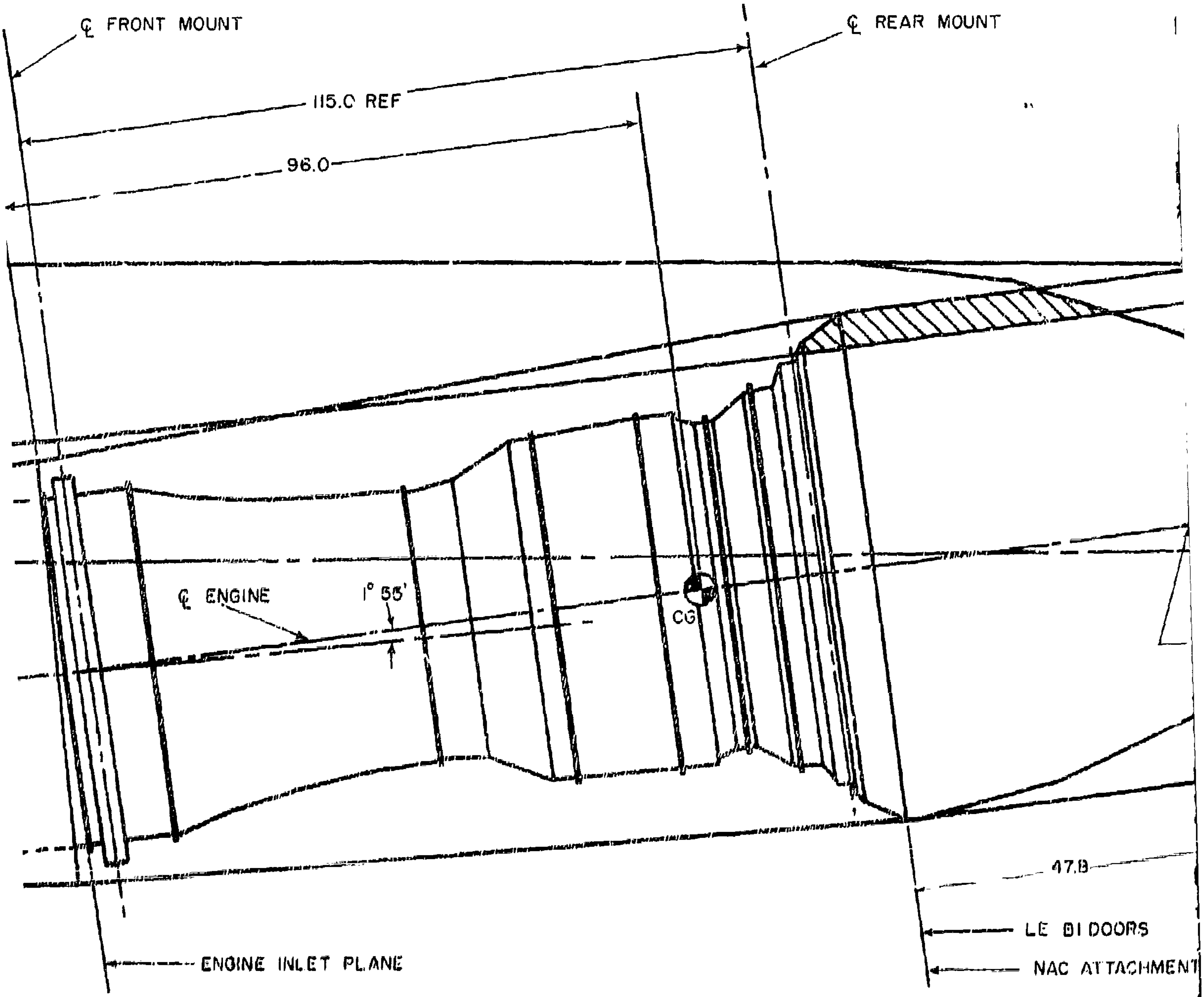
2000°F MAX. TURBINE INLET
TEMPERATURE
EJECTOR CANTED AT REAR
MOUNT PLANE

STJ227 500 LBS./SEC. TURBOJET

Figure 1-17

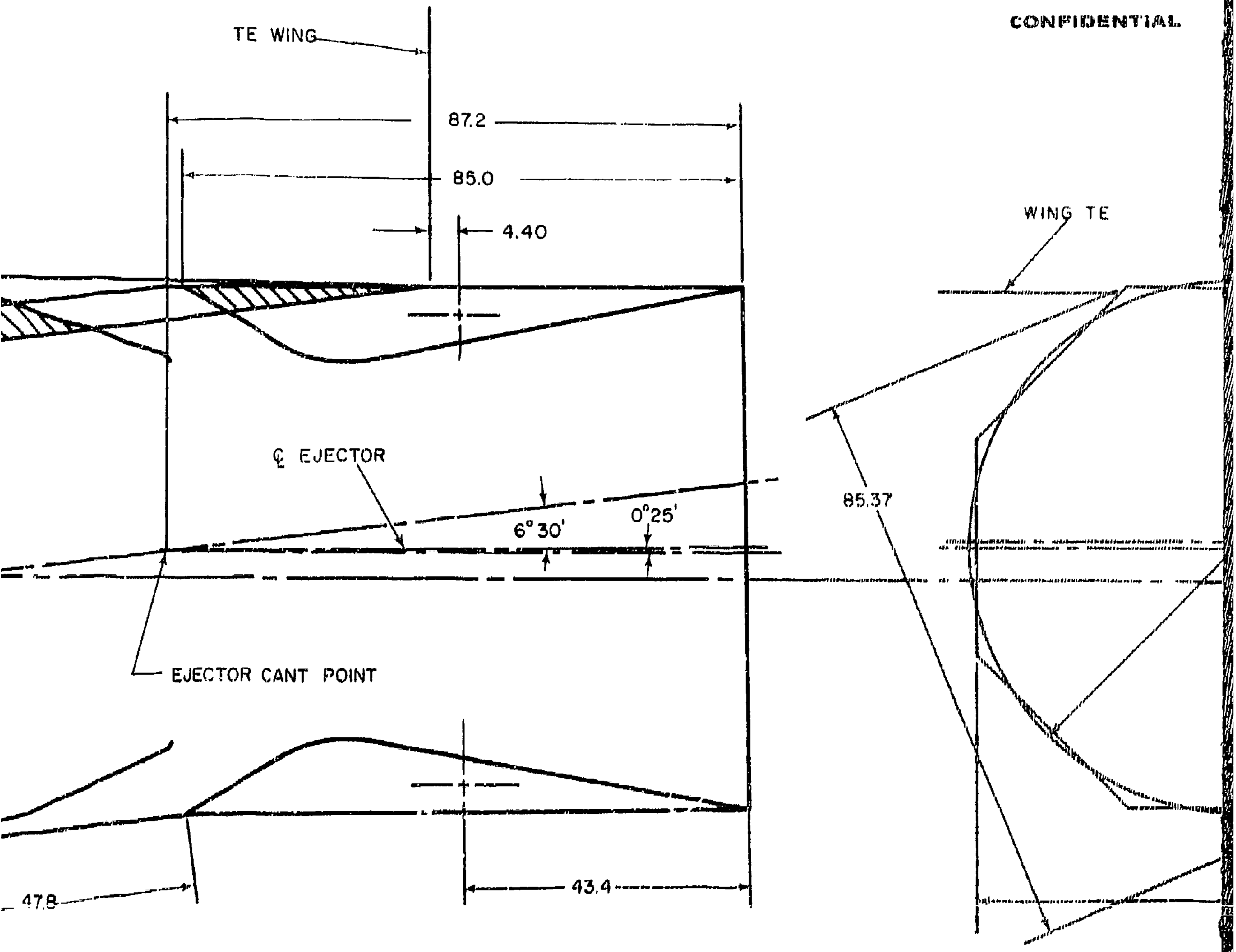


1


℄ FRONT MOUNT
℄ REAR MOUNT
115.0 REF
96.0
℄ ENGINE
1° 55'
CG
47.8
LE BI DOORS
NAC ATTACHMENT
ENGINE INLET PLANE

2

CONFIDENTIAL



.E BI DOORS

NAC ATTACHMENT PLANE

CG LOCATION DOES NOT INCLUDE INLET WEIGHT

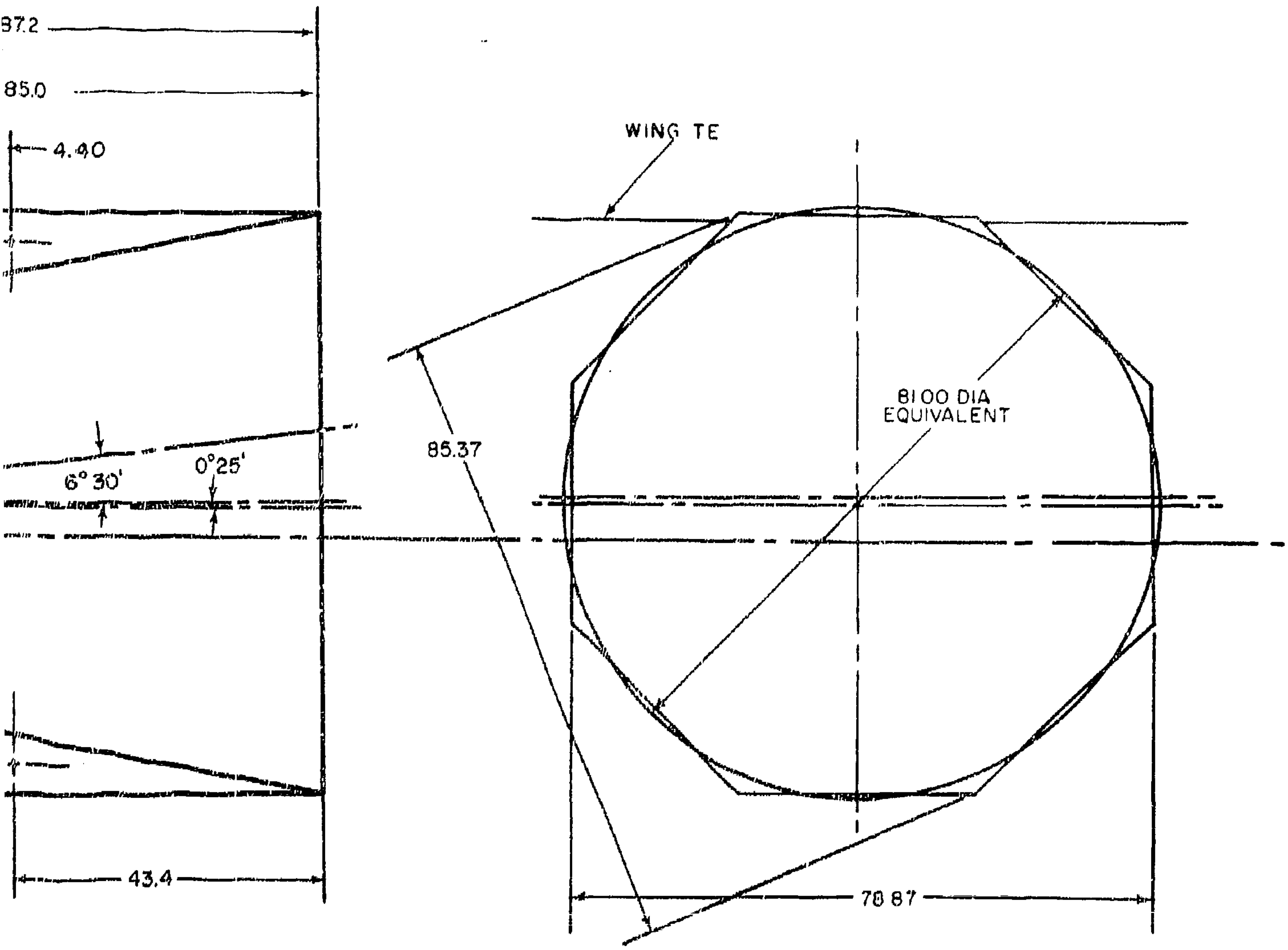
2000°F MAX TU
TEMPERATUR
EJECTOR CANT
NOZZLE PLAN

STJ227 500 LBS./SEC. TU

Figure 1-18

CONFIDENTIAL

3

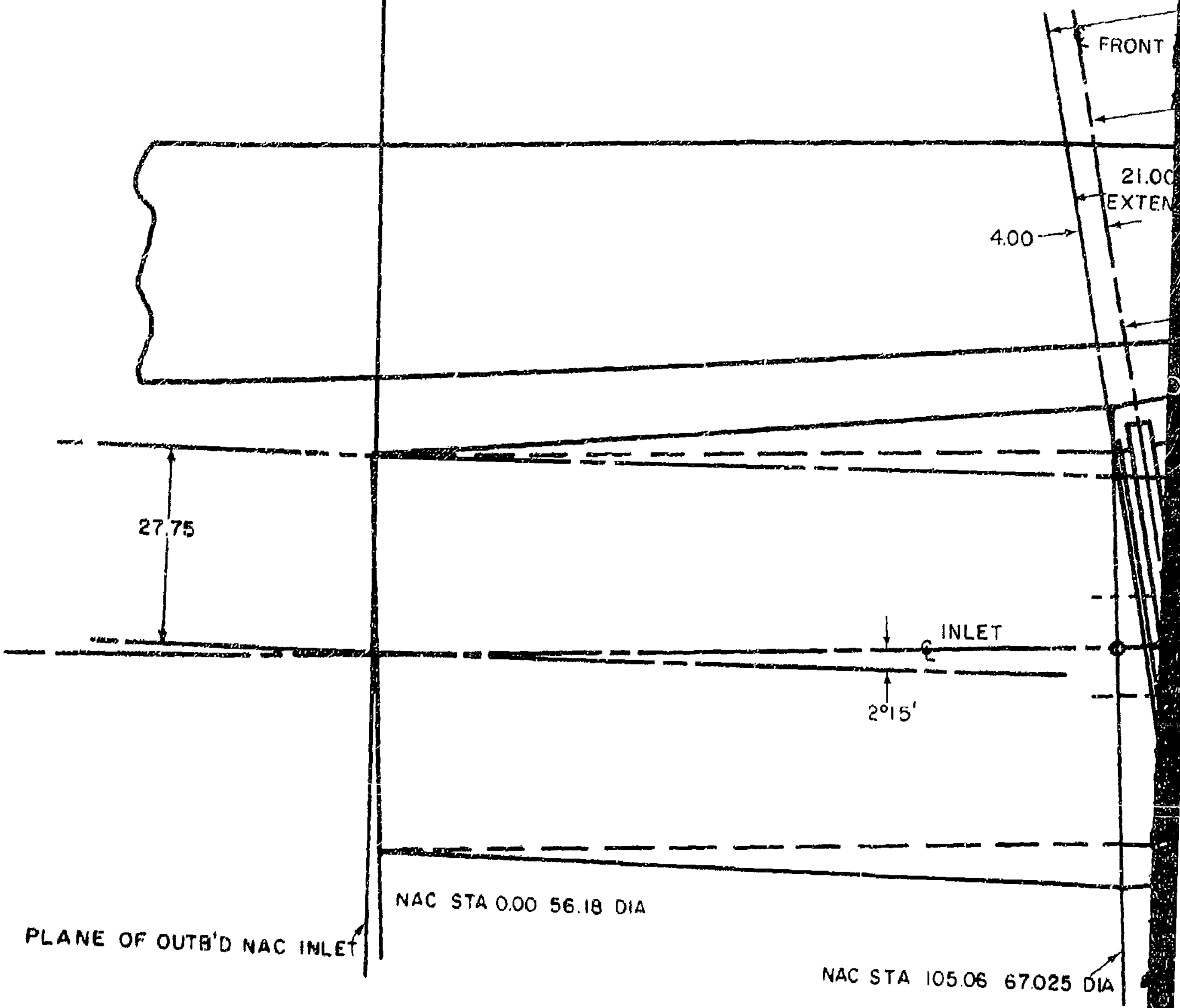


CG LOCATION DOES NOT INCLUDE INLET WEIGHT

2000°F MAX. TURBINE INLET TEMPERATURE
EJECTOR CANTED AT PRIMARY NOZZLE PLANE

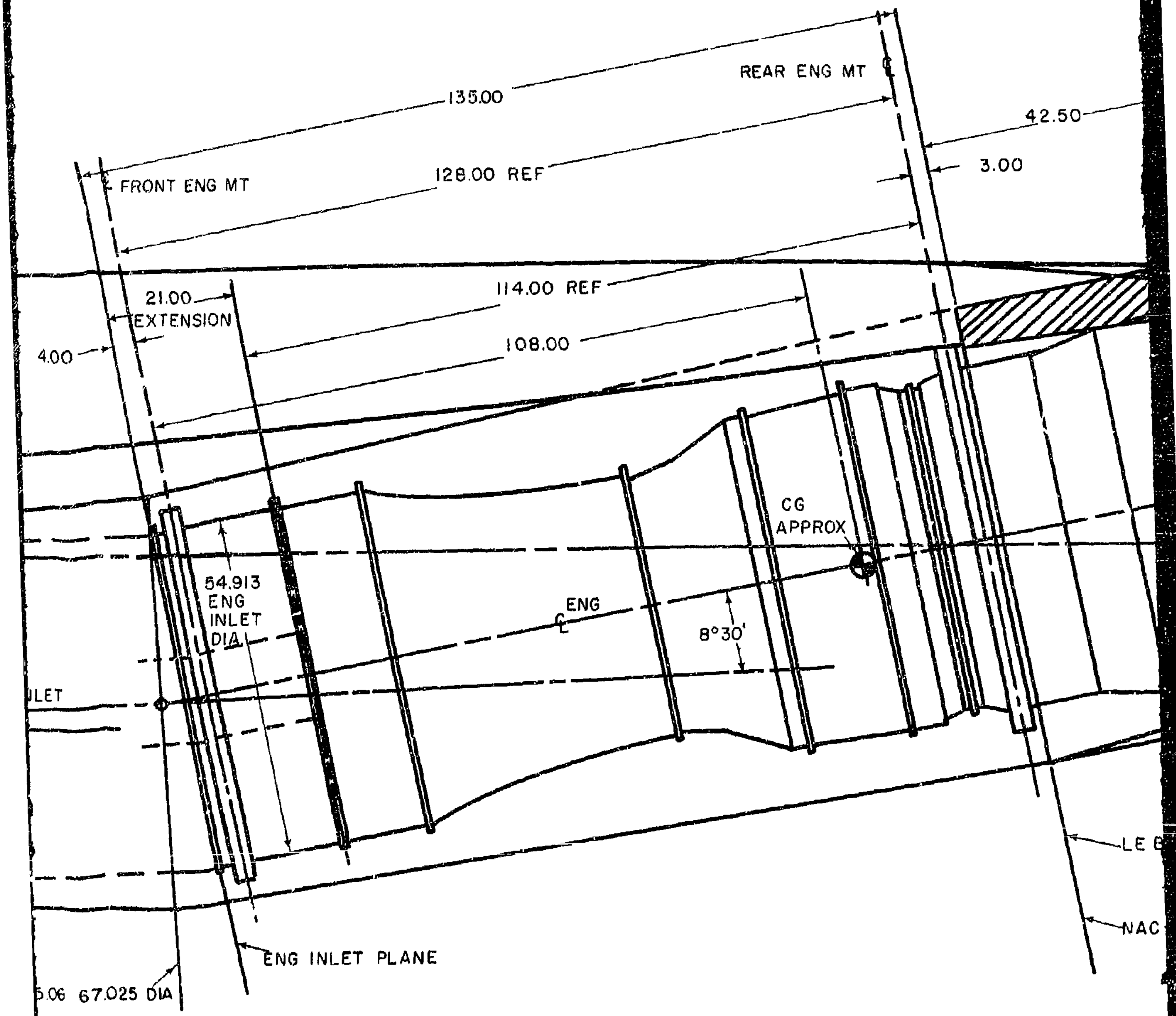
STJ227 500 LBS./SEC. TURBOJET

Figure 1-18



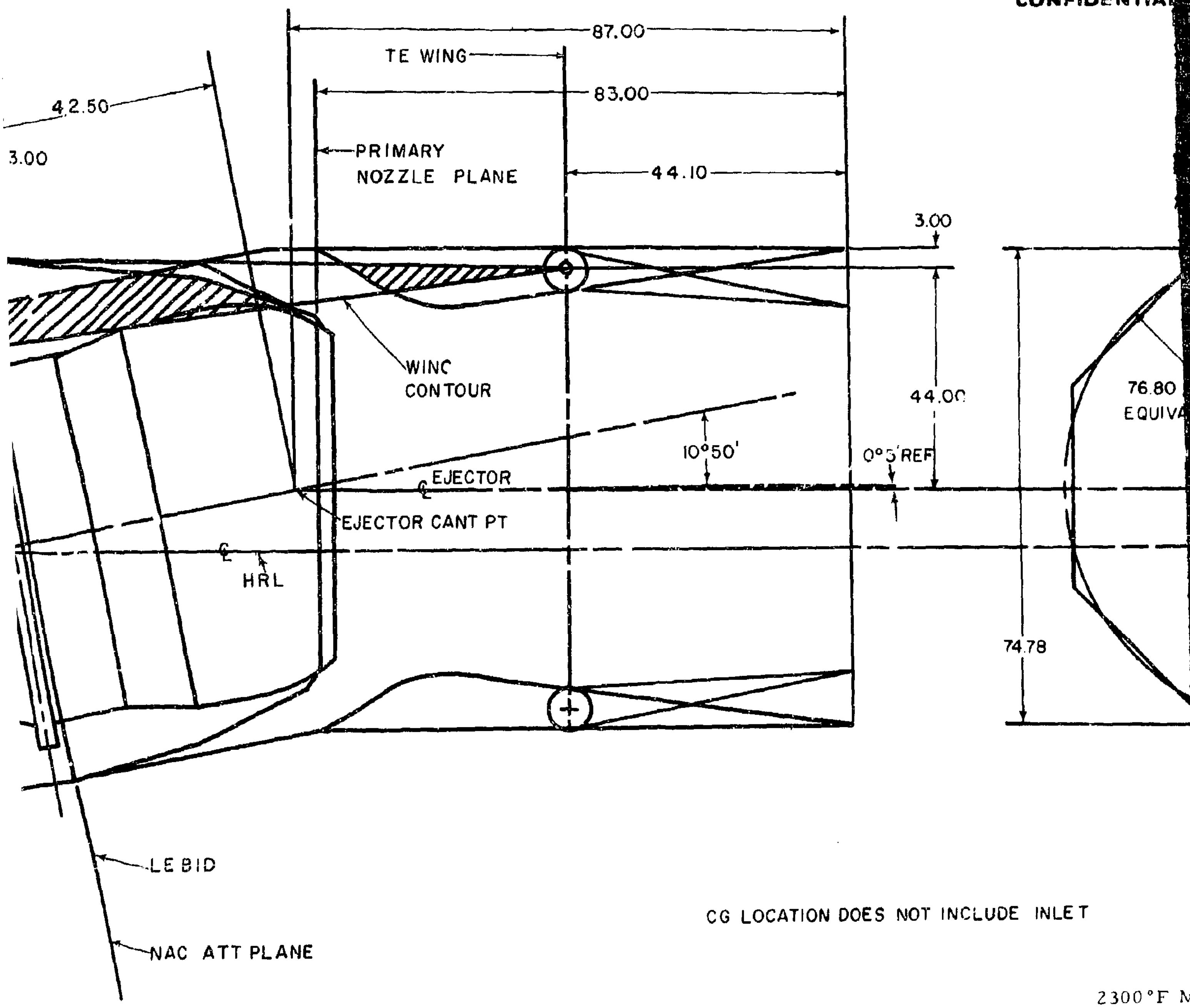
FRONT
21.00
EXTEN
4.00
27.75
INLET
2°15'
NAC STA 0.00 56.18 DIA
PLANE OF OUTB'D NAC INLET
NAC STA 105.06 67.025 DIA

1



135.00
REAR ENG MT ℄
42.50
3.00
128.00 REF
℄ FRONT ENG MT
114.00 REF
21.00
EXTENSION
4.00
108.00
CG
APPROX
54.913
ENG
INLET
DIA
ENG ℄
8°30'
LET
LE
NAC
ENG INLET PLANE
67.025 DIA

2

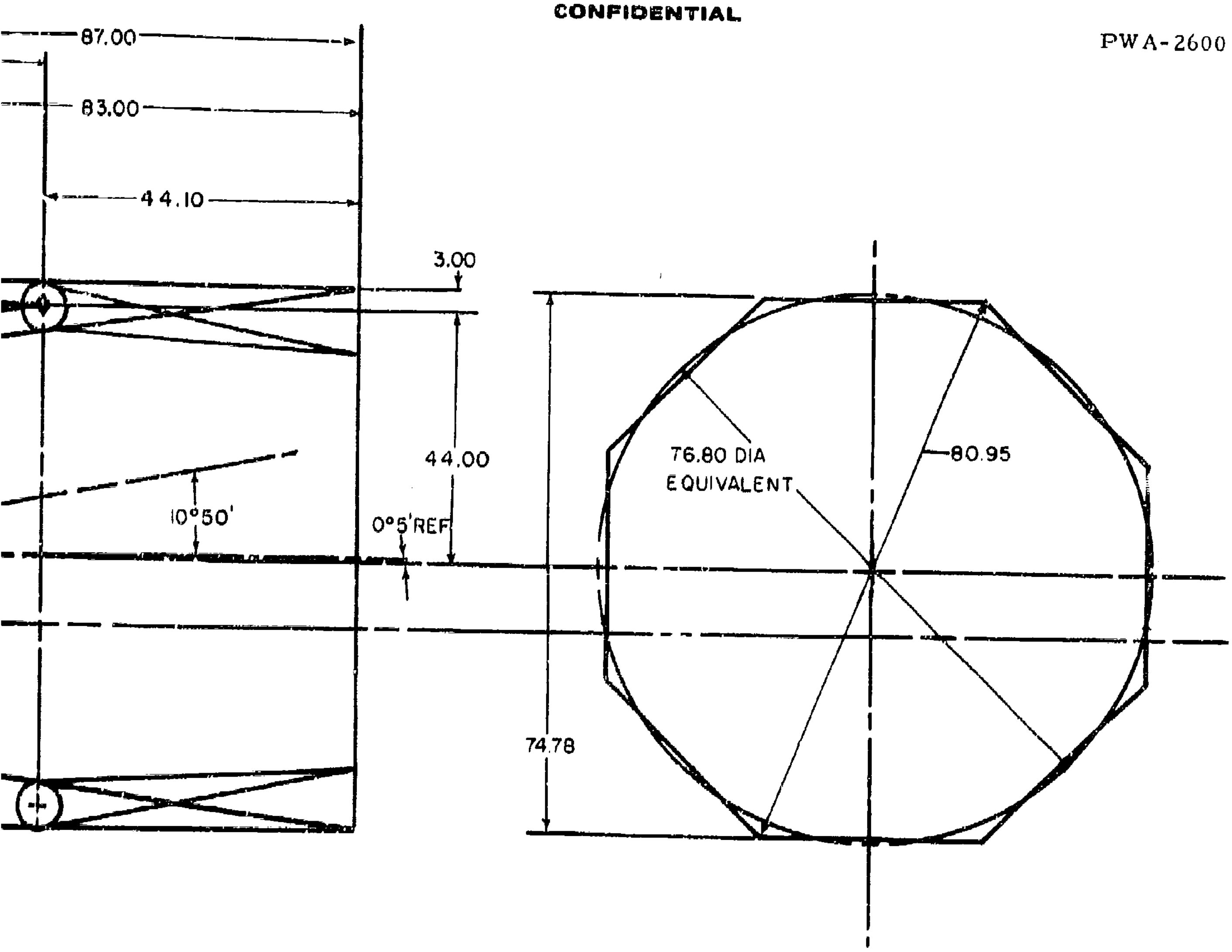


CG LOCATION DOES NOT INCLUDE INLET

2300°F M
TEMPE
EJECTOR
PRIMAR
*NOTE: C

STJ227 525 LBS./SEC. 7

Figure 1-19

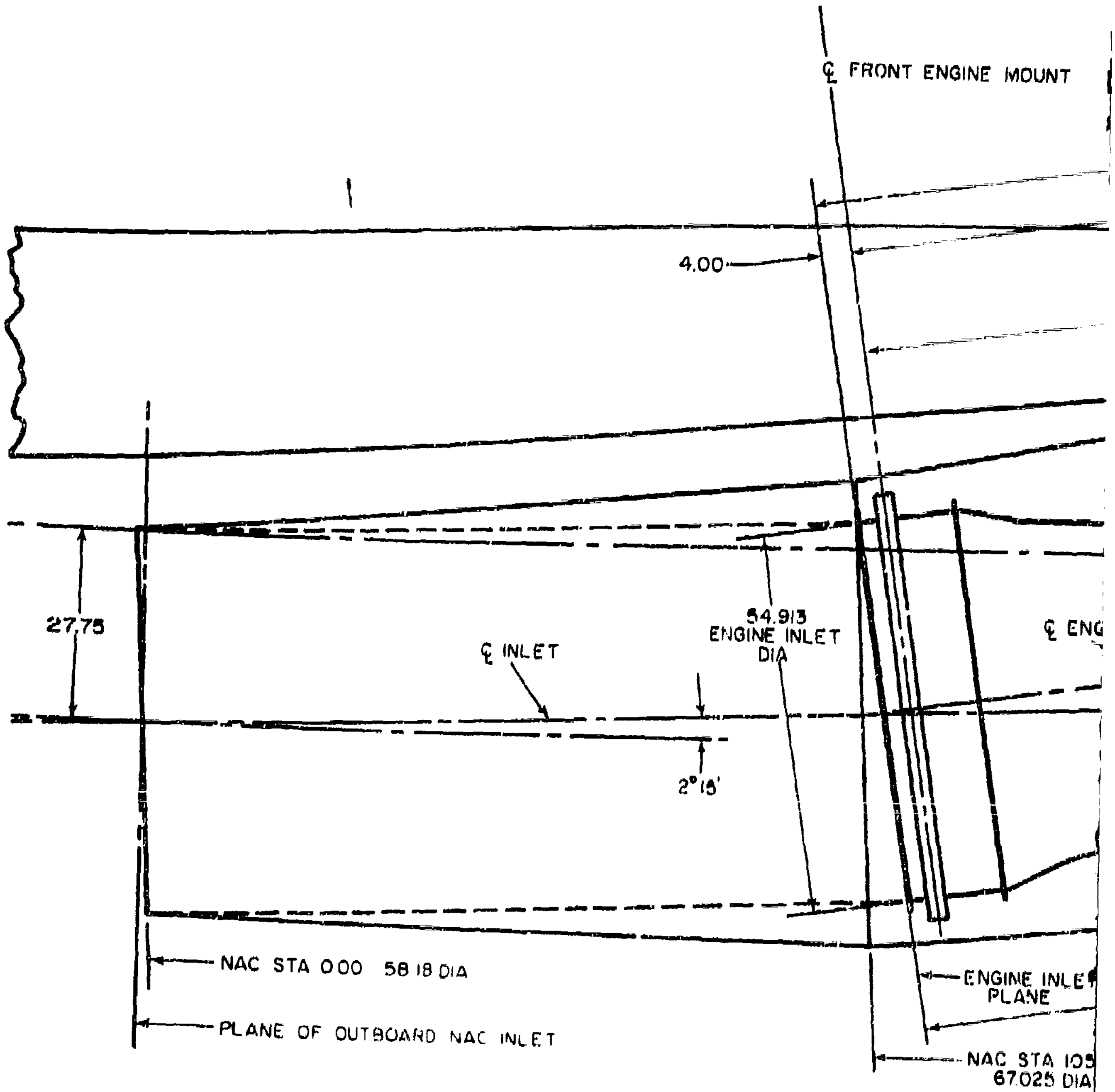


CG LOCATION DOES NOT INCLUDE INLET

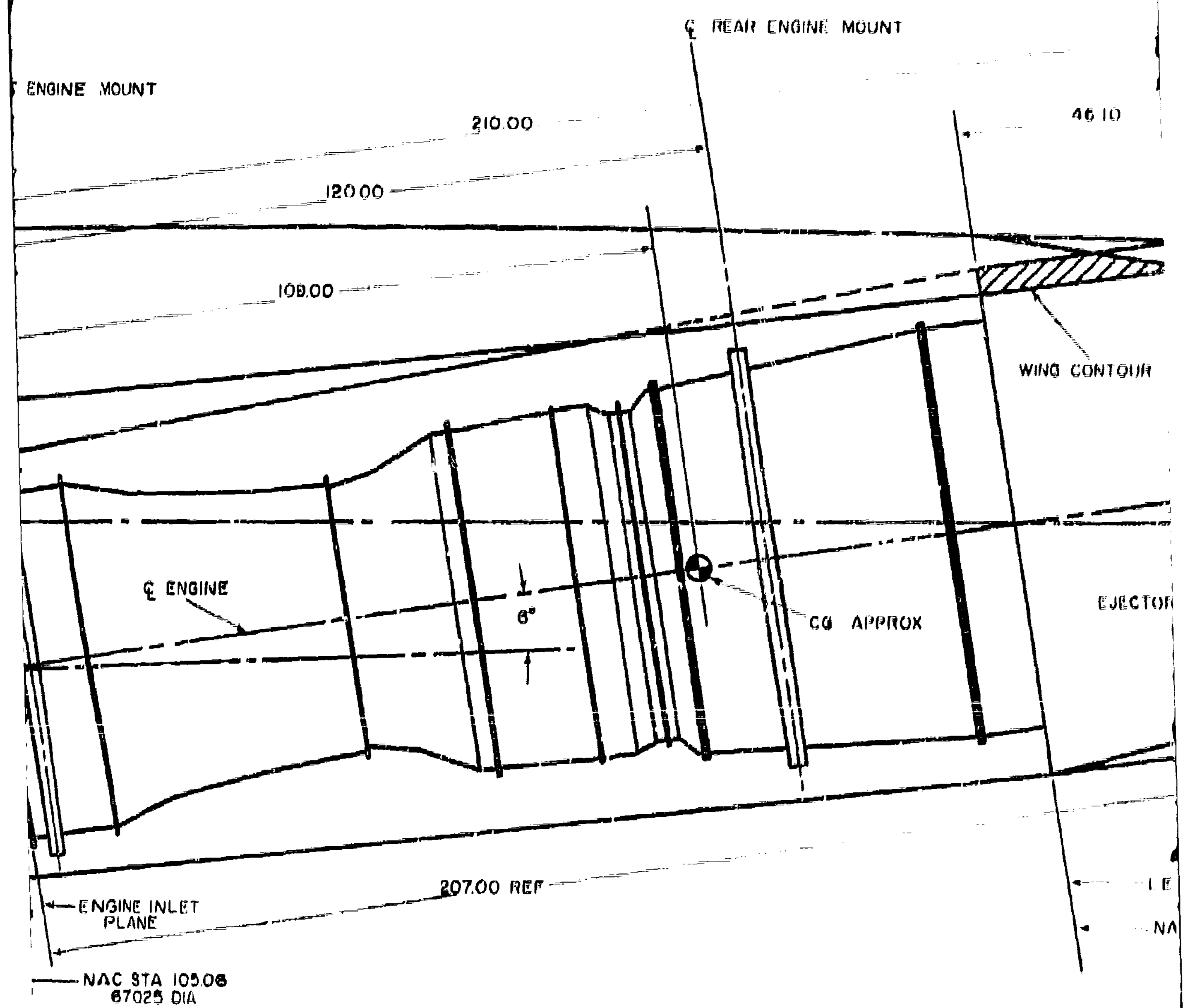
2300°F MAXIMUM TURBINE INLET TEMPERATURE
EJECTOR CANTED 4.00 FWD. OF PRIMARY NOZZLE PLANE
*NOTE: CG LOCATION DOES NOT INCLUDE INLET

STJ227 525 LBS./SEC. TURBOJET

Figure 1-19

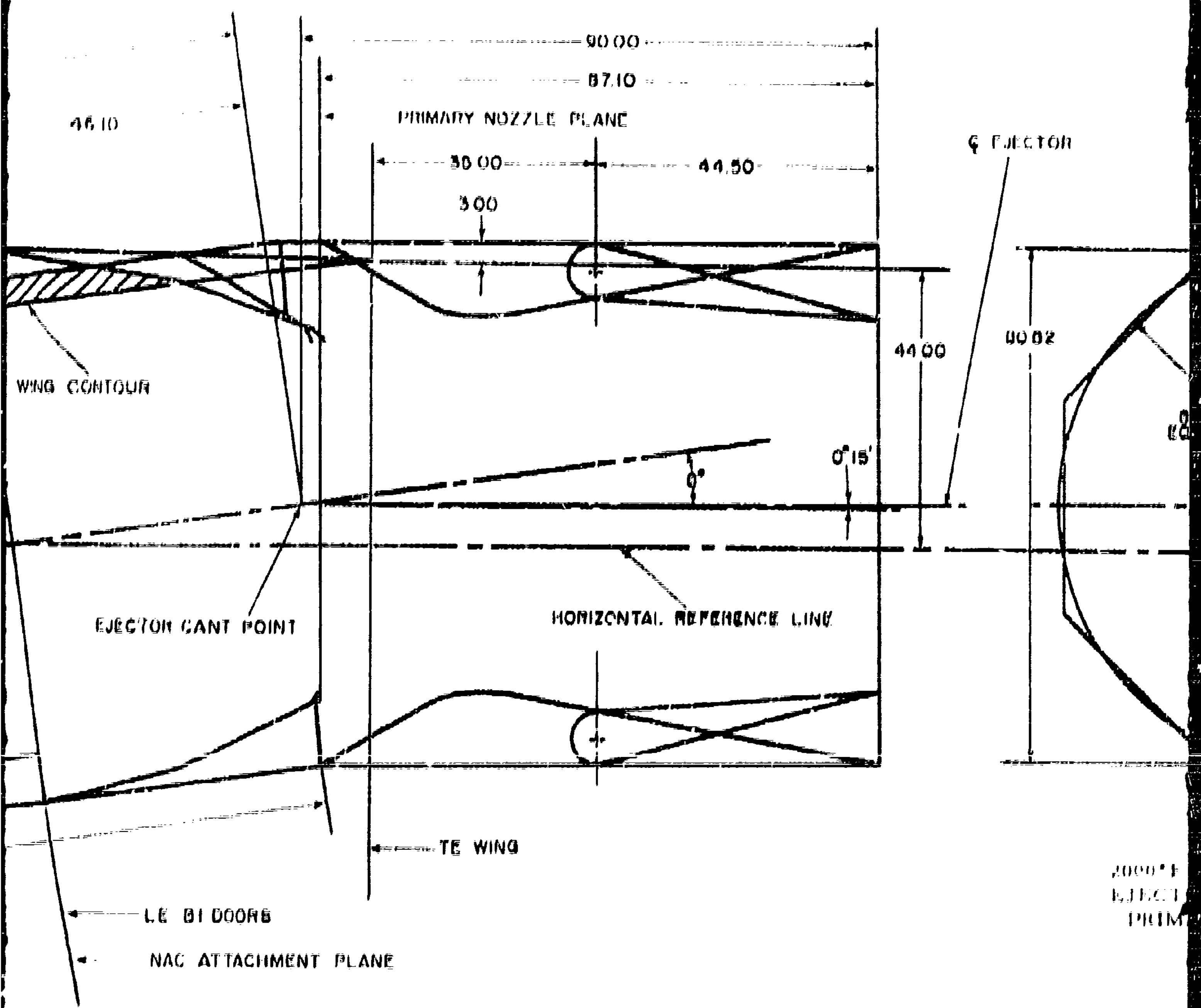

℄ FRONT ENGINE MOUNT
4.00
27.75
℄ INLET
54.913
ENGINE INLET
DIA
2°15'
℄ ENG
NAC STA 0.00 58.18 DIA
PLANE OF OUTBOARD NAC INLET
ENGINE INLET
PLANE
NAC STA 105
67.025 DIA

1



℄ REAR ENGINE MOUNT
ENGINE MOUNT
210.00
46.10
120.00
100.00
WING CONTOUR
℄ ENGINE
6°
CG APPROX
EJECTOR
207.00 REF
ENGINE INLET PLANE
NAC STA 105.06
67025 DIA

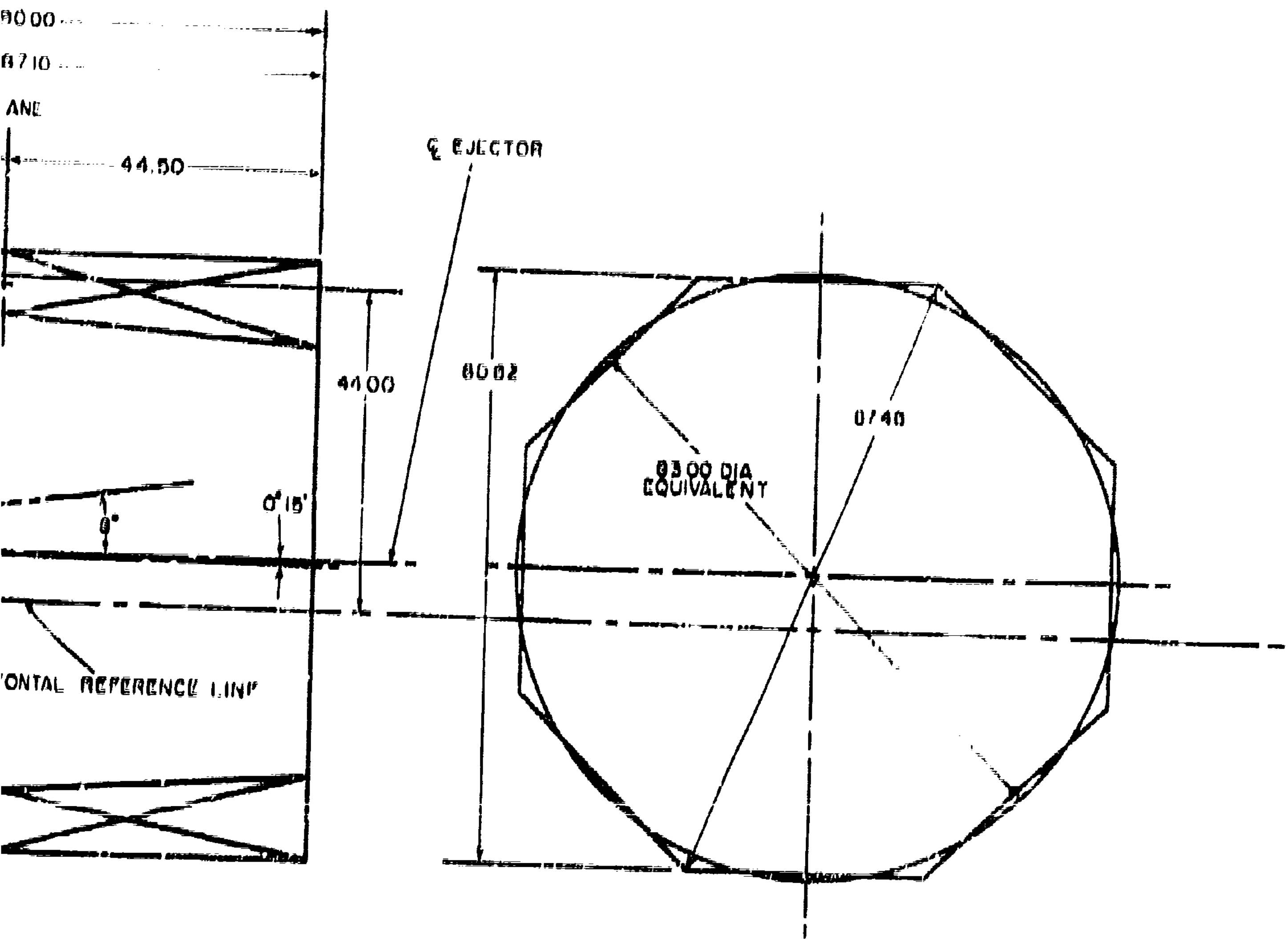
CON



2000°F
EJECT
PRIM

CON

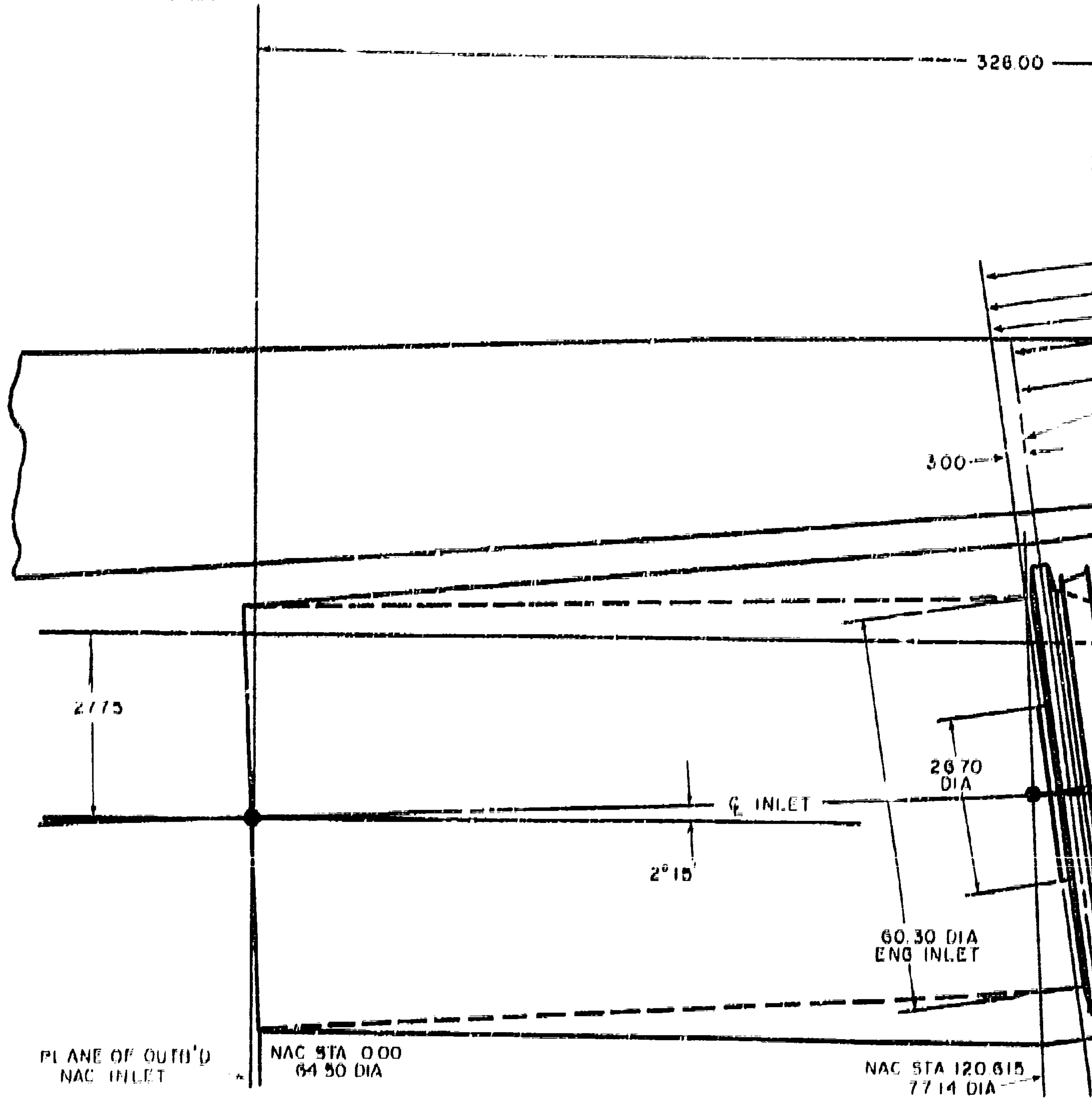
3



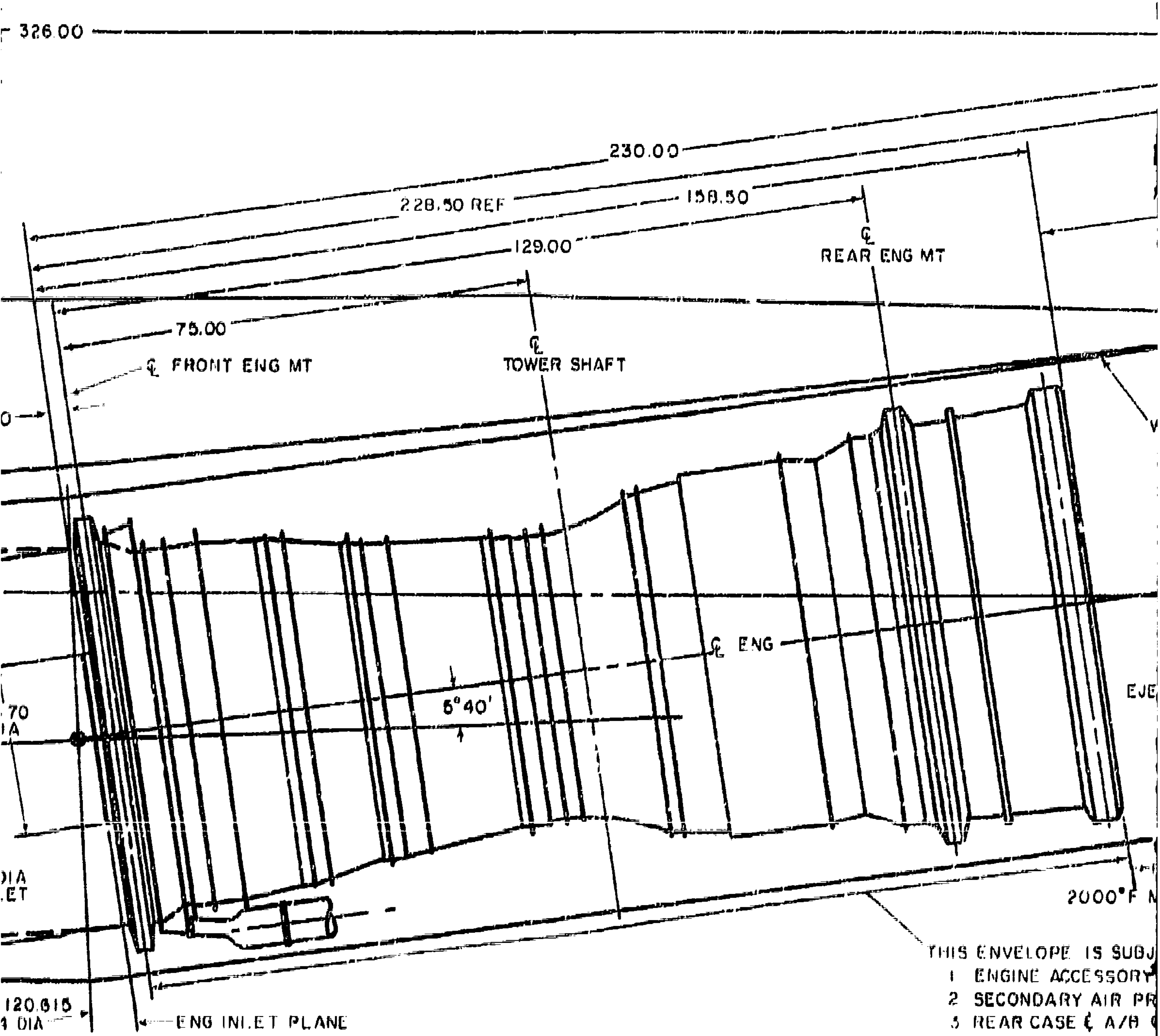
2000°F MAXIMUM TURBINE INLET
EJECTOR CANTED 2.40 FWD. OF
PRIMARY NOZZLE PLANE

STJ227 525 LBS./SEC. TURBOJET

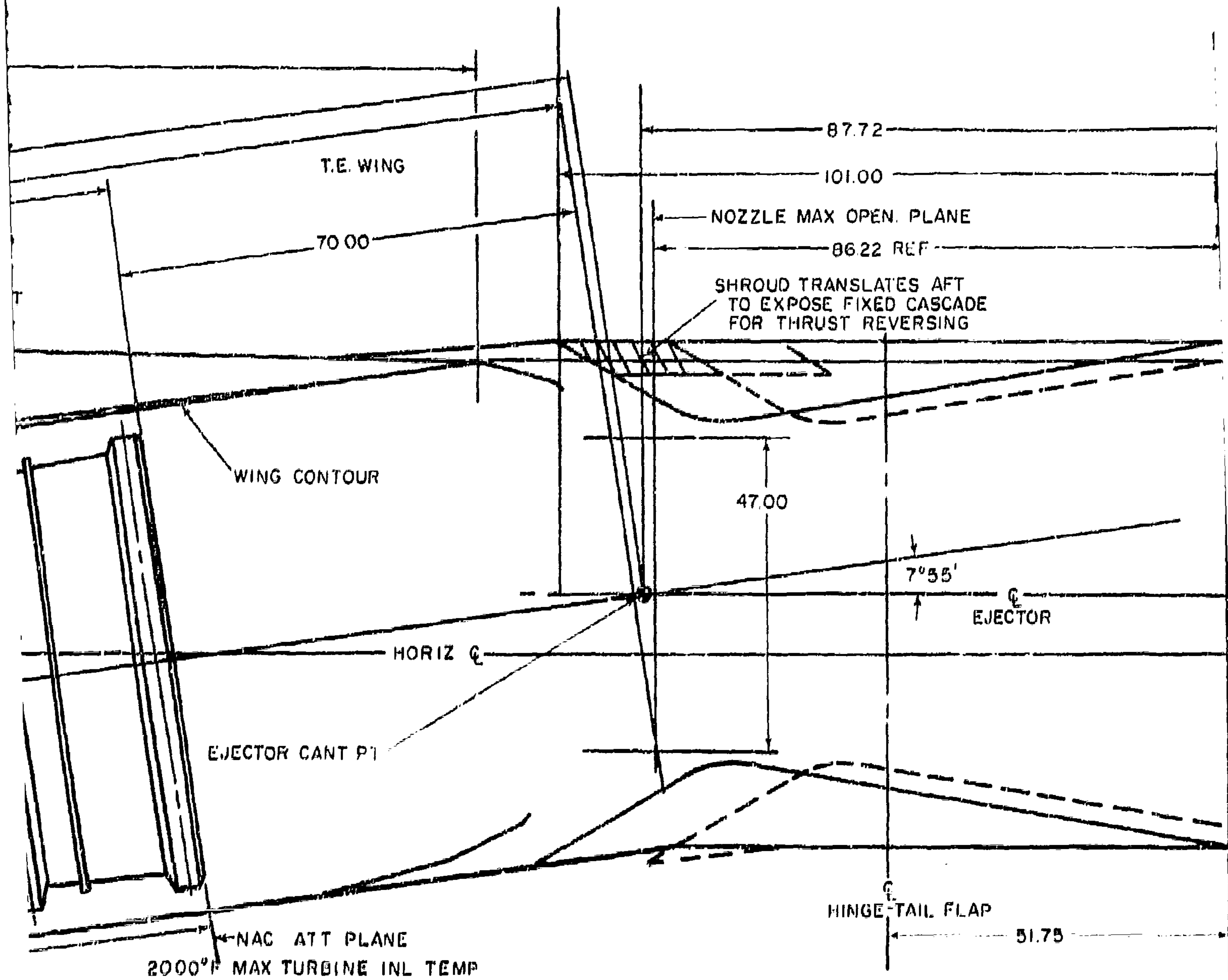
Figure 1-20



328.00
3.00
27.75
26.70
DIA
℄ INLET
2°15'
60.30 DIA
ENG INLET
PLANE OF OUTB'D
NAC INLET
NAC STA 0.00
64.50 DIA
NAC STA 120.615
77.14 DIA

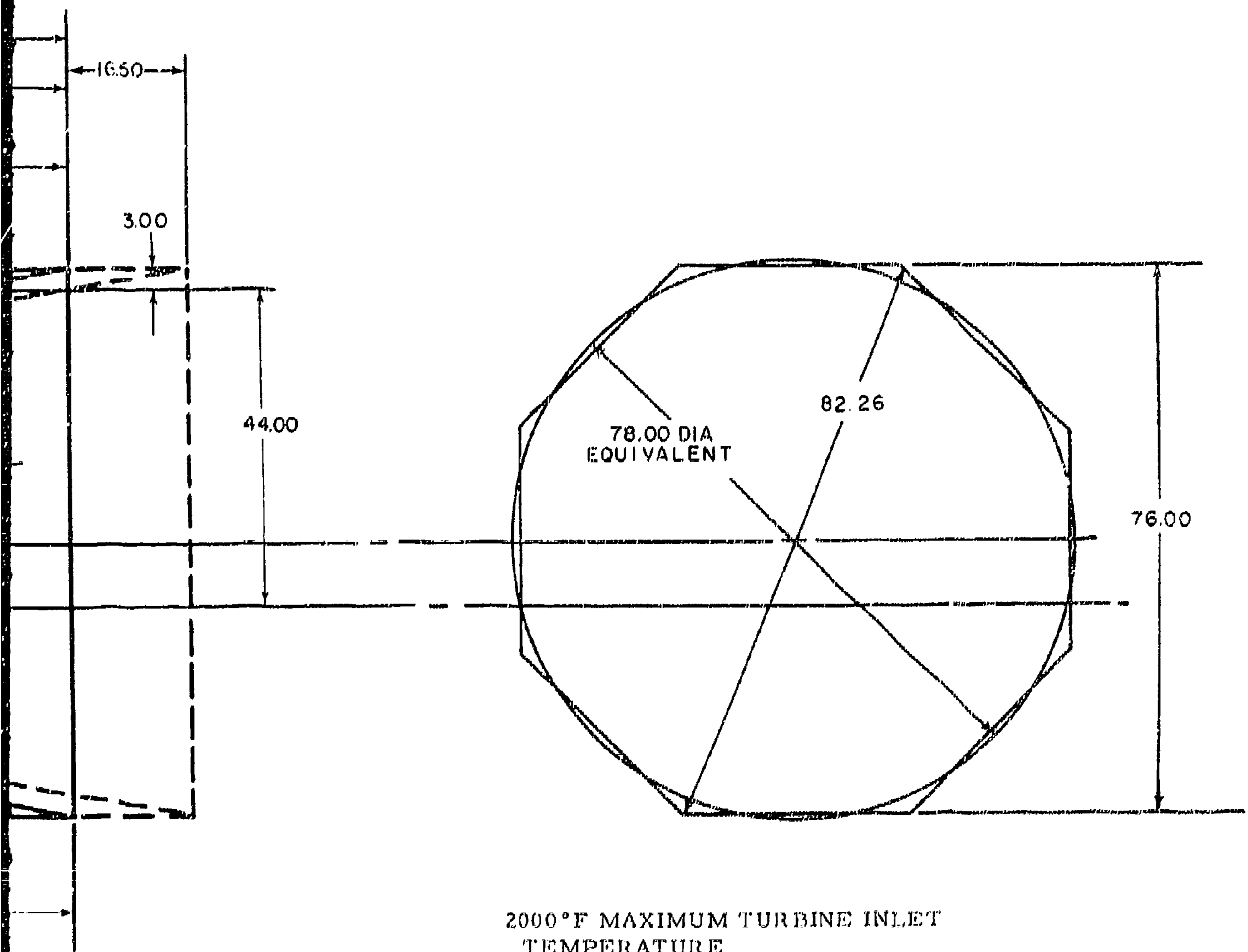


326.00
230.00
228.50 REF
158.50
129.00
℄ REAR ENG MT
75.00
℄ FRONT ENG MT
℄ TOWER SHAFT
℄ ENG
5°40'
EJE
2000°F
THIS ENVELOPE IS SUBJ
1 ENGINE ACCESSORY
2 SECONDARY AIR PR
3 REAR CASE ℄ A/B
120.615
ENG INLET PLANE



13 ENVELOPE IS SUBJECT TO CHANGE FOR:
ENGINE ACCESSORY ARRANGEMENT
2 SECONDARY AIR PROVISIONS FOR EJECTOR
3 REAR CASE & A/B COOLING PROVISIONS

3



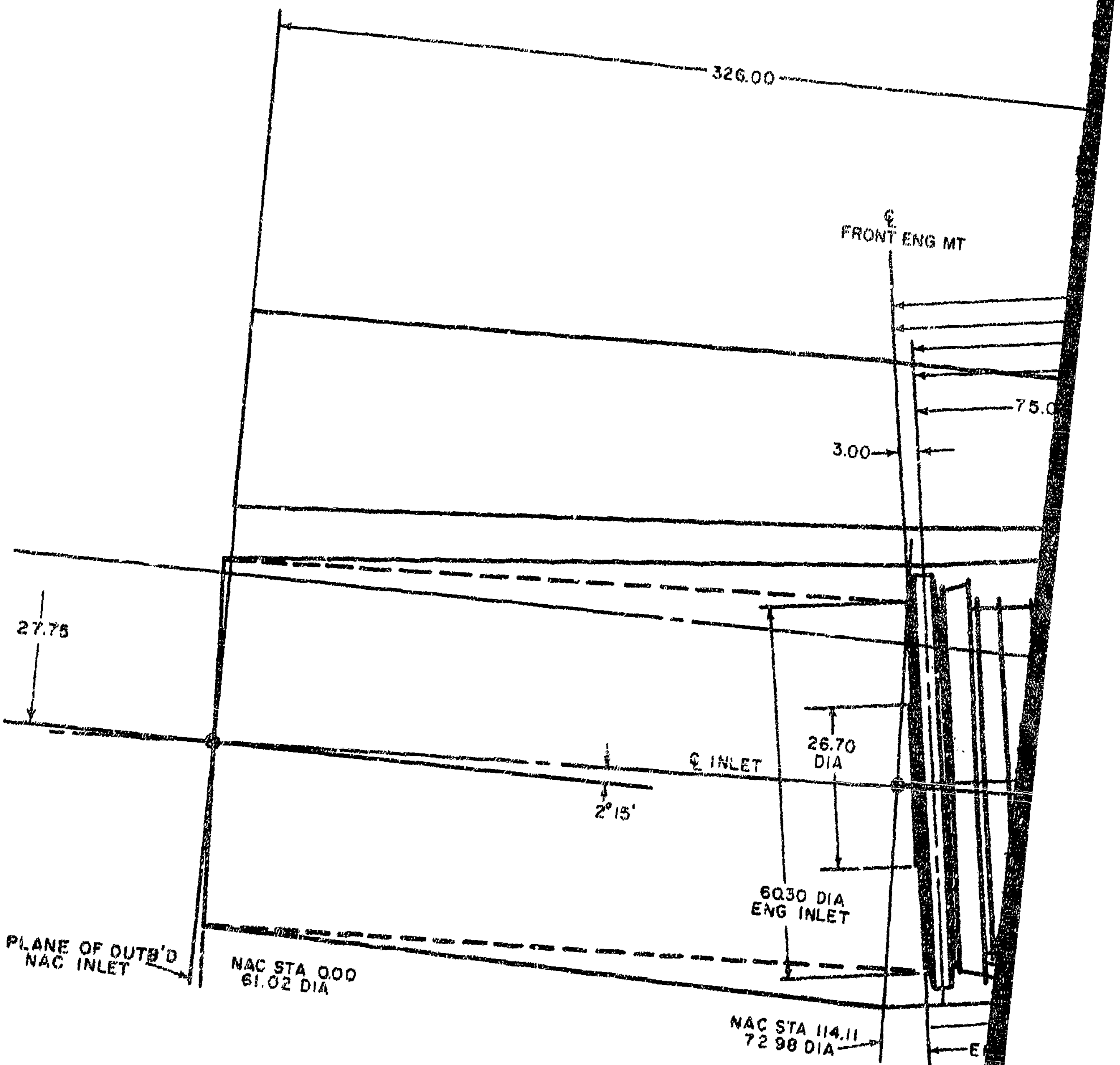
2000°F MAXIMUM TURBINE INLET
TEMPERATURE
EJECTOR CANTED 1.50 FWD. OF
PRIMARY NOZZLE PLANE

THIS ENVELOPE IS SUBJECT TO CHANGE FOR:

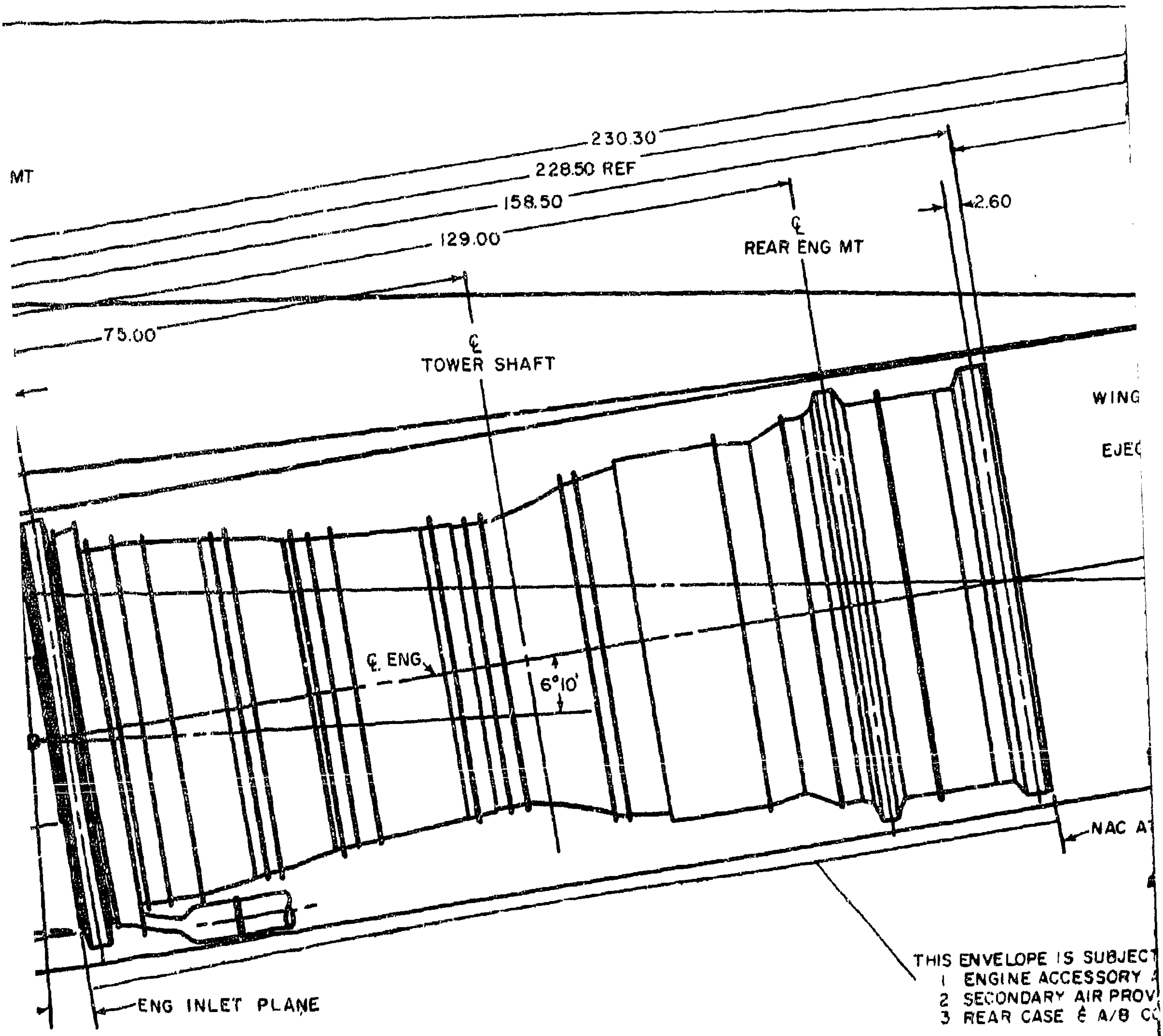
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR EJECTOR
3. REAR CASE & A/B COOLING PROVISIONS

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

Figure 1-21

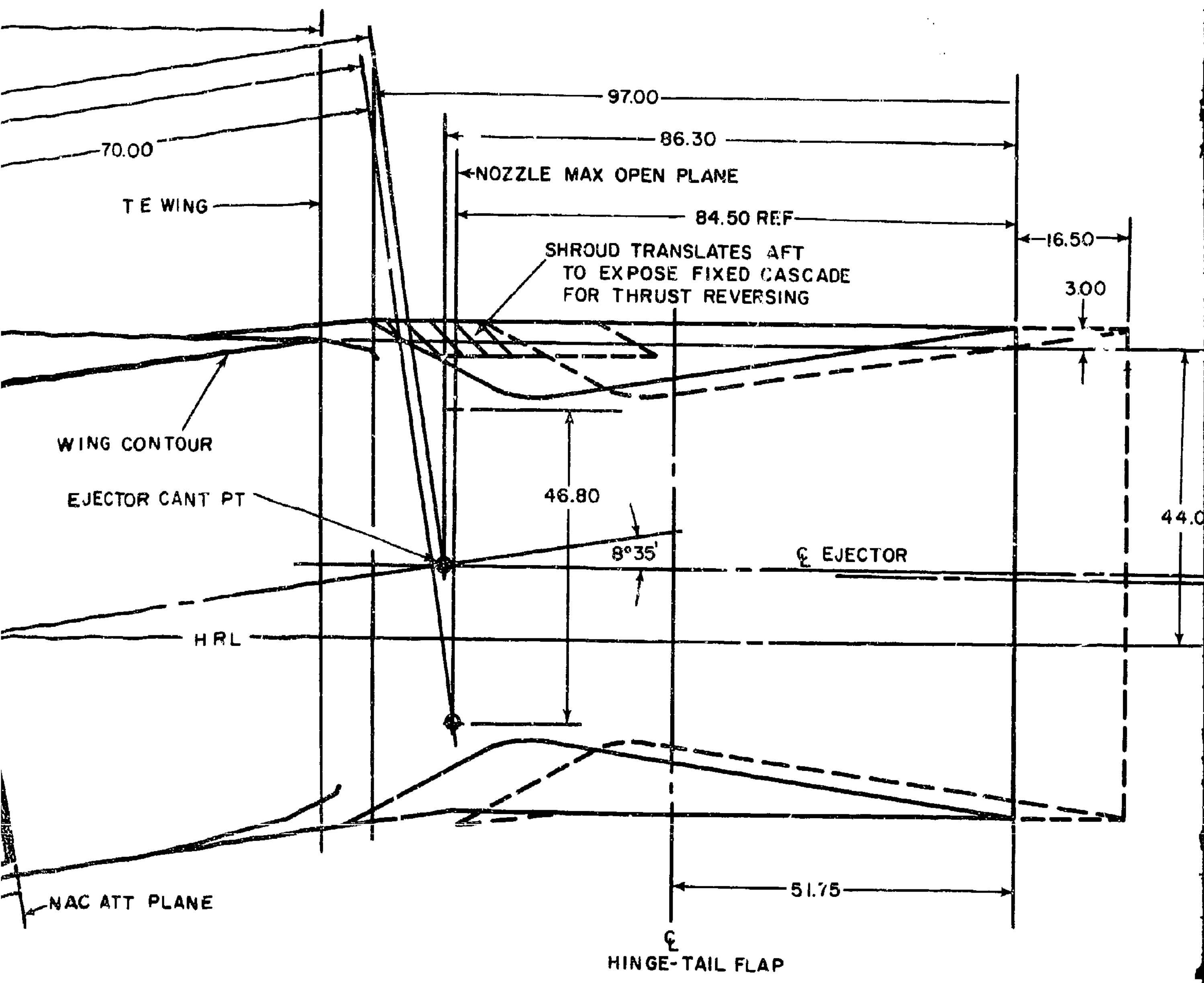

326.00
℄ FRONT ENG MT
75.0
3.00
27.75
26.70 DIA
℄ INLET
2°15'
60.30 DIA ENG INLET
PLANE OF OUTB'D NAC INLET
NAC STA 0.00 61.02 DIA
NAC STA 114.11 72.98 DIA

1



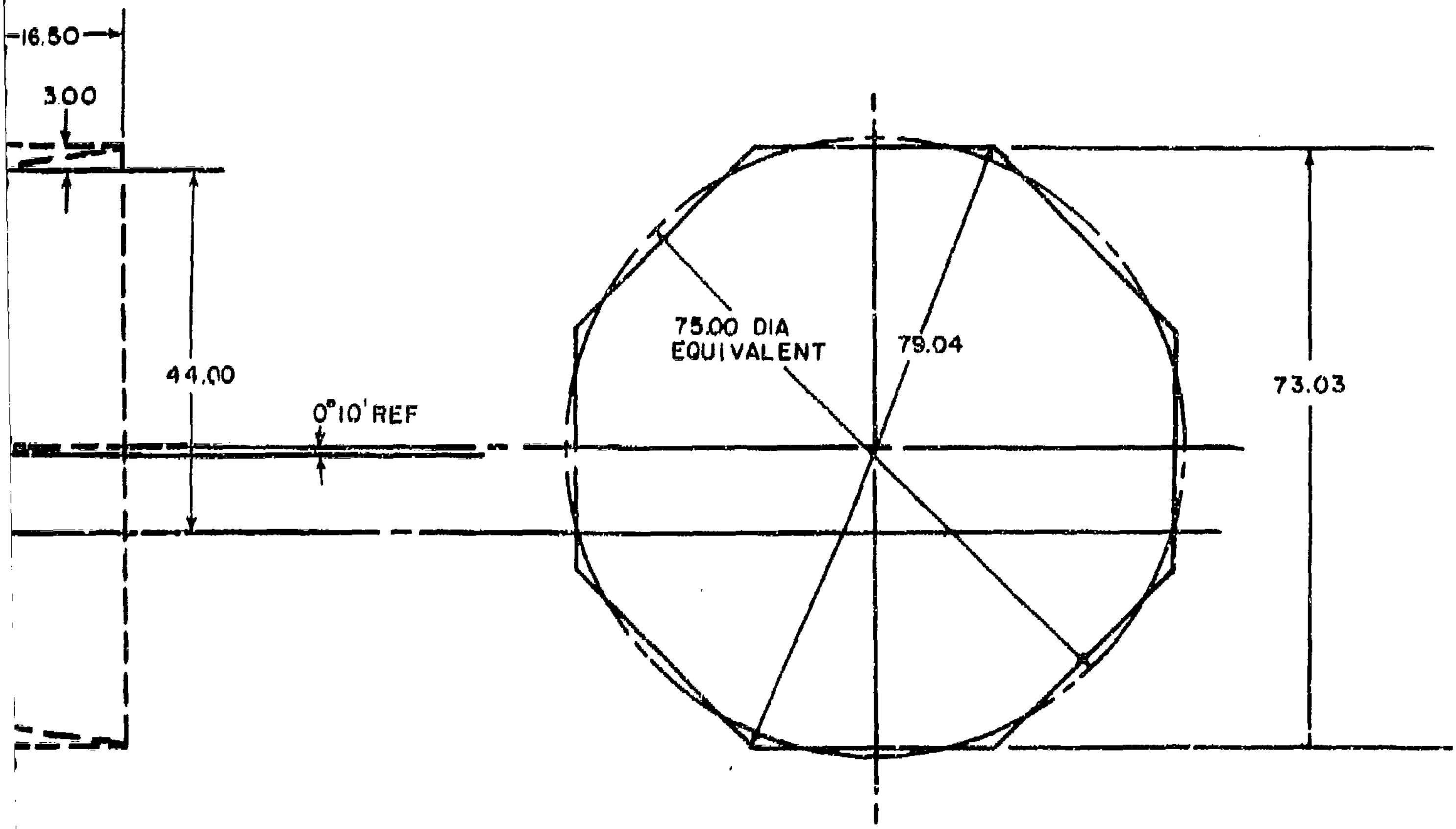
MT
230.30
228.50 REF
158.50
129.00
2.60
℄
REAR ENG MT
75.00
℄
TOWER SHAFT
WING
EJE
℄ ENG
6°10'
NAC A
THIS ENVELOPE IS SUBJECT
1 ENGINE ACCESSORY
2 SECONDARY AIR PROV
3 REAR CASE & A/B C
ENG INLET PLANE

2



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

3

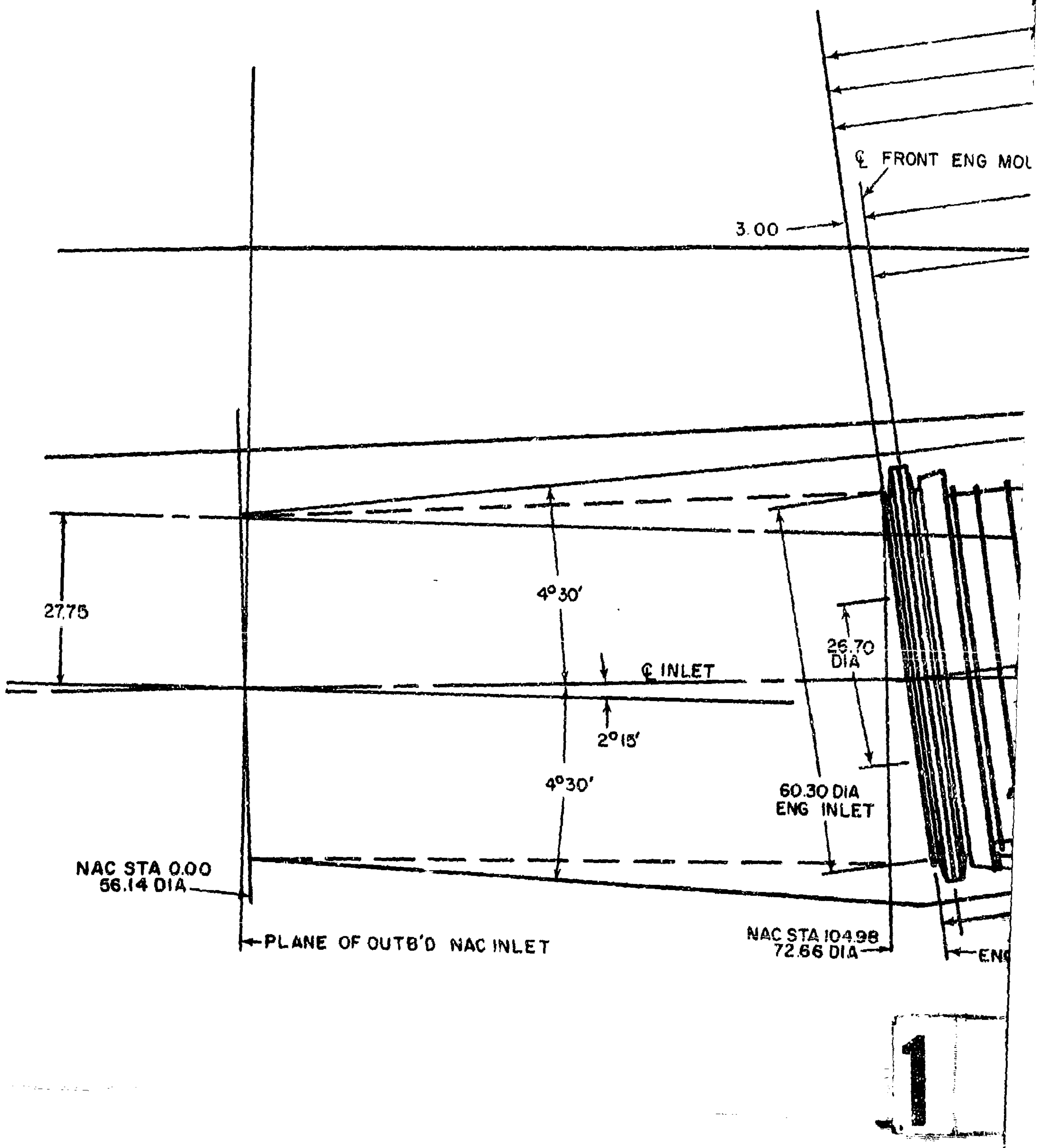


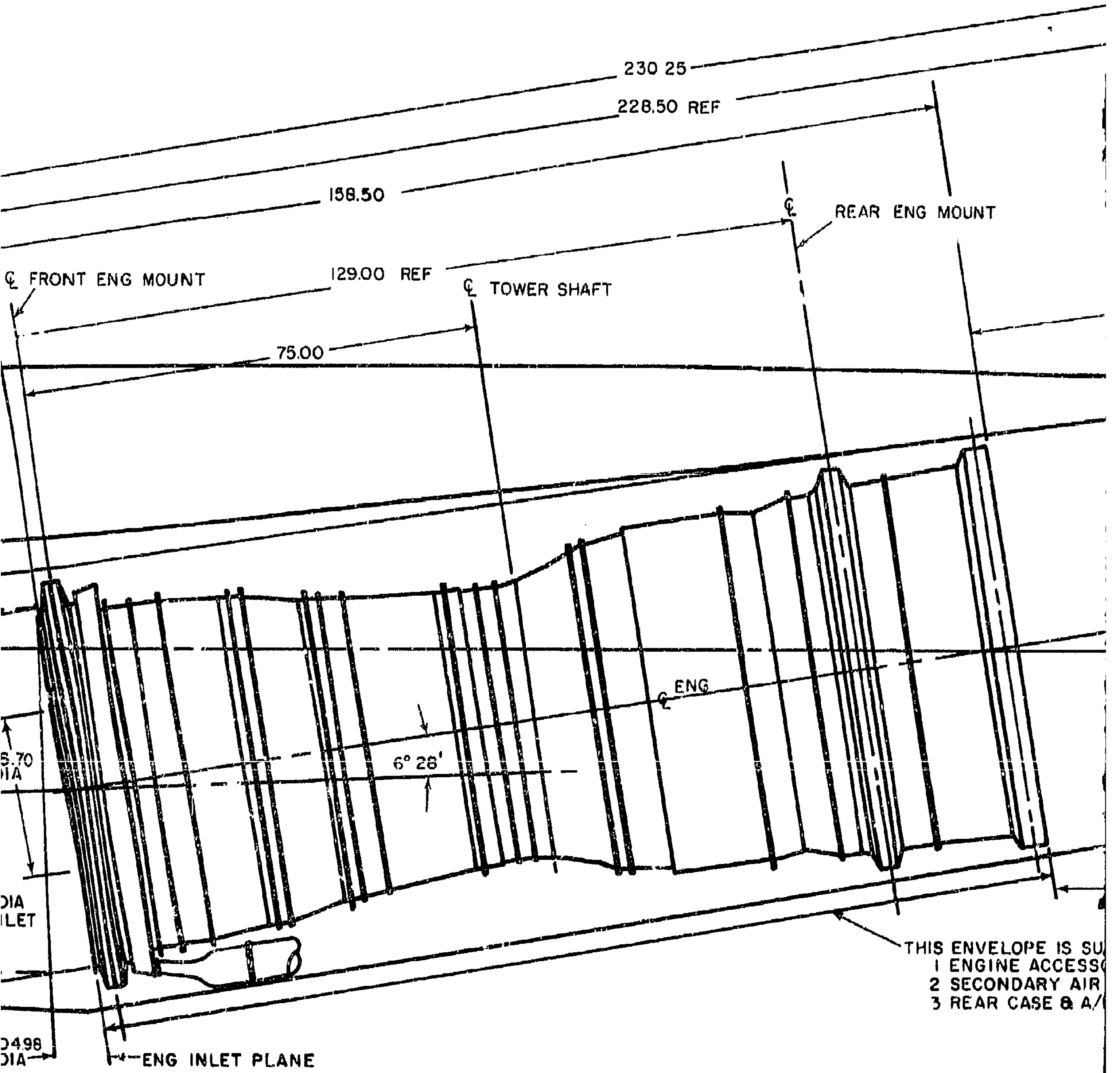
2000°F TURBINE INLET TEMPERATURE
L. H. SIDE VIEW OUTBOARD POD
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:

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

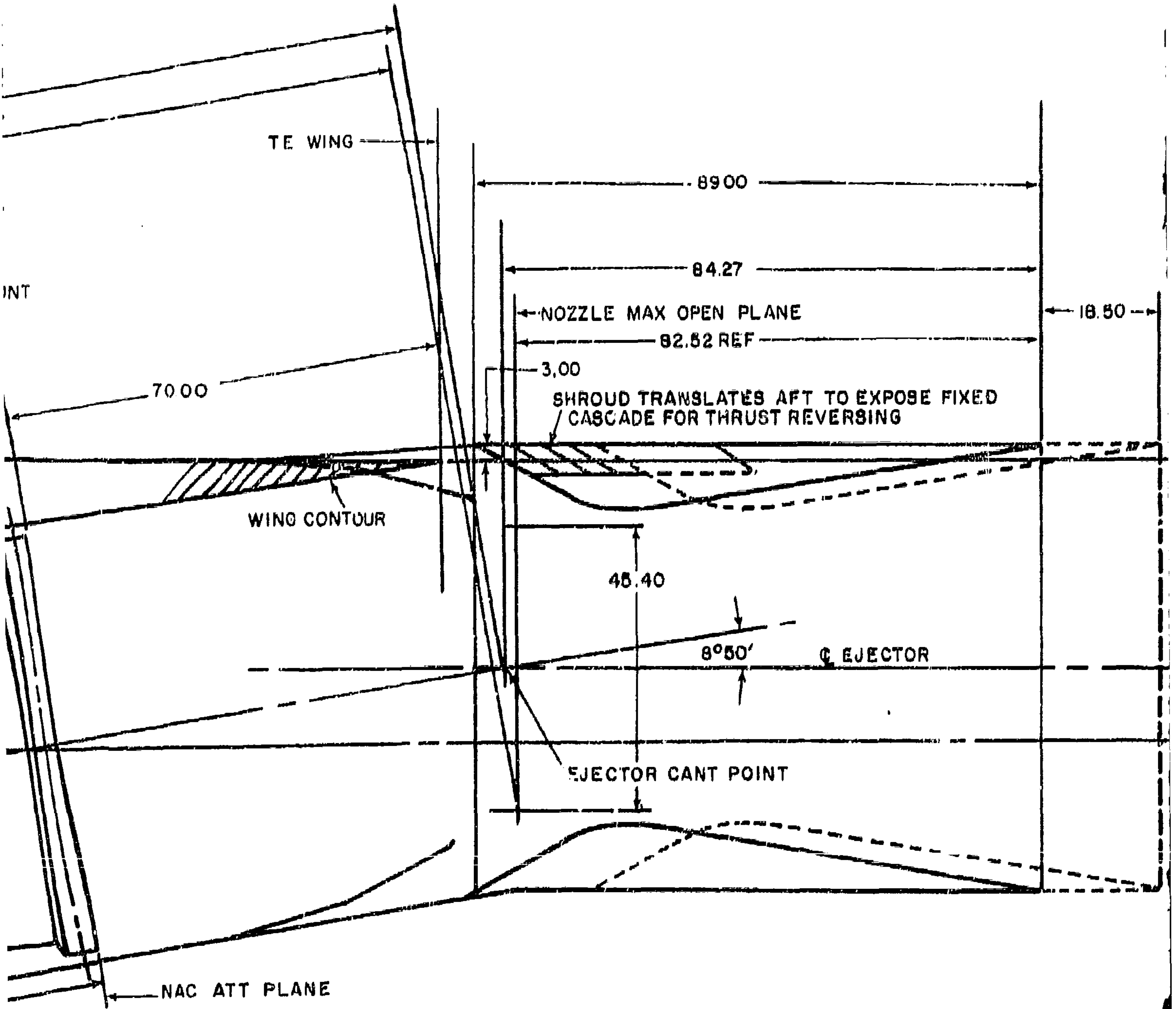
Figure 1-22


℄ FRONT ENG MOU
3.00
27.75
4°30'
℄ INLET
2°15'
4°30'
26.70 DIA
60.30 DIA ENG INLET
NAC STA 0.00
56.14 DIA
PLANE OF OUTB'D NAC INLET
NAC STA 104.98
72.66 DIA
EN
1

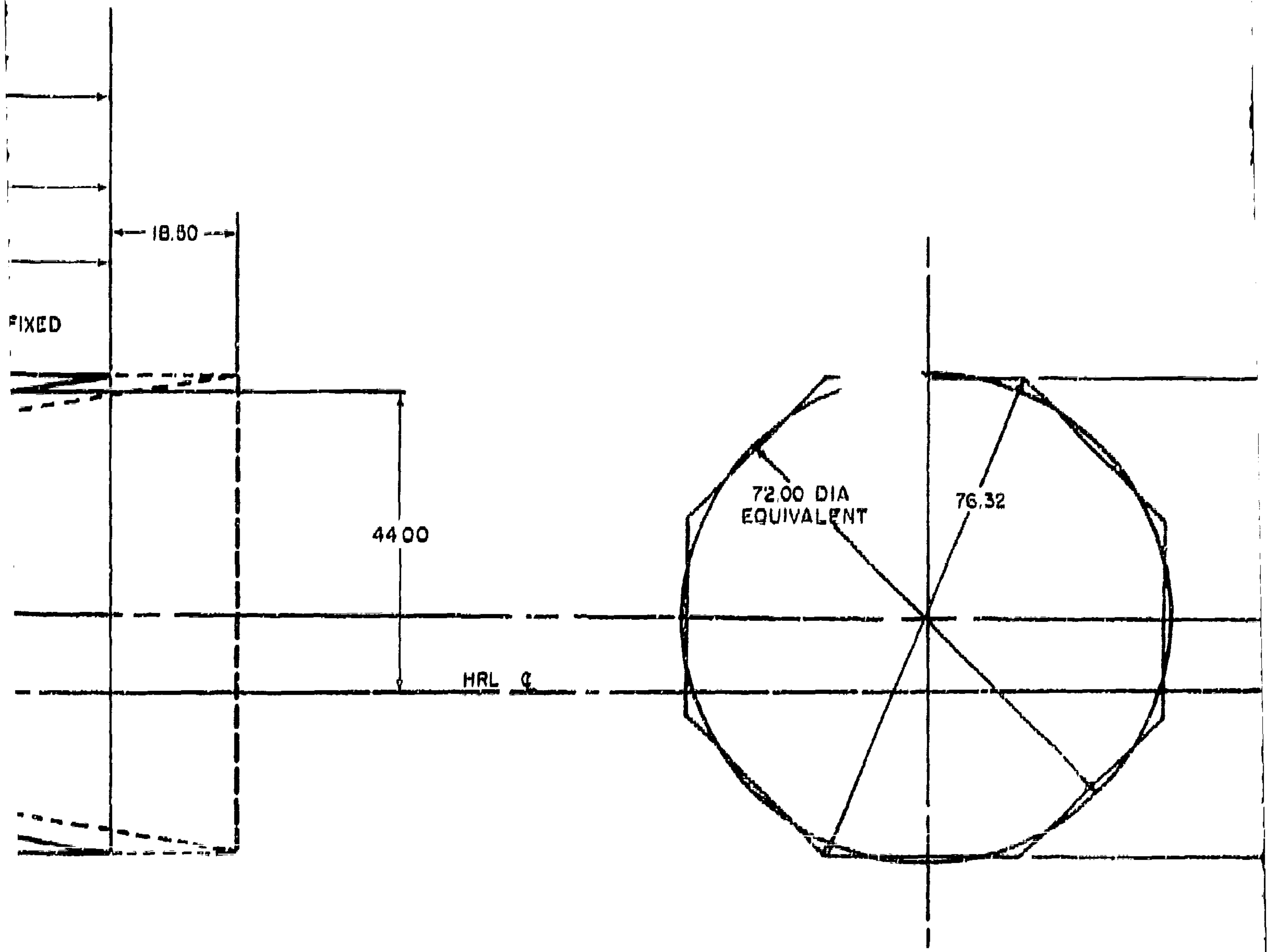


230 25
228.50 REF
158.50
℄ REAR ENG MOUNT
℄ FRONT ENG MOUNT
129.00 REF
℄ TOWER SHAFT
75.00
℄ ENG
6° 28'
6.70
DIA
DIA
ILET
THIS ENVELOPE IS SU
1 ENGINE ACCESS
2 SECONDARY AIR
3 REAR CASE & A/
04.98
DIA
ENG INLET PLANE

2



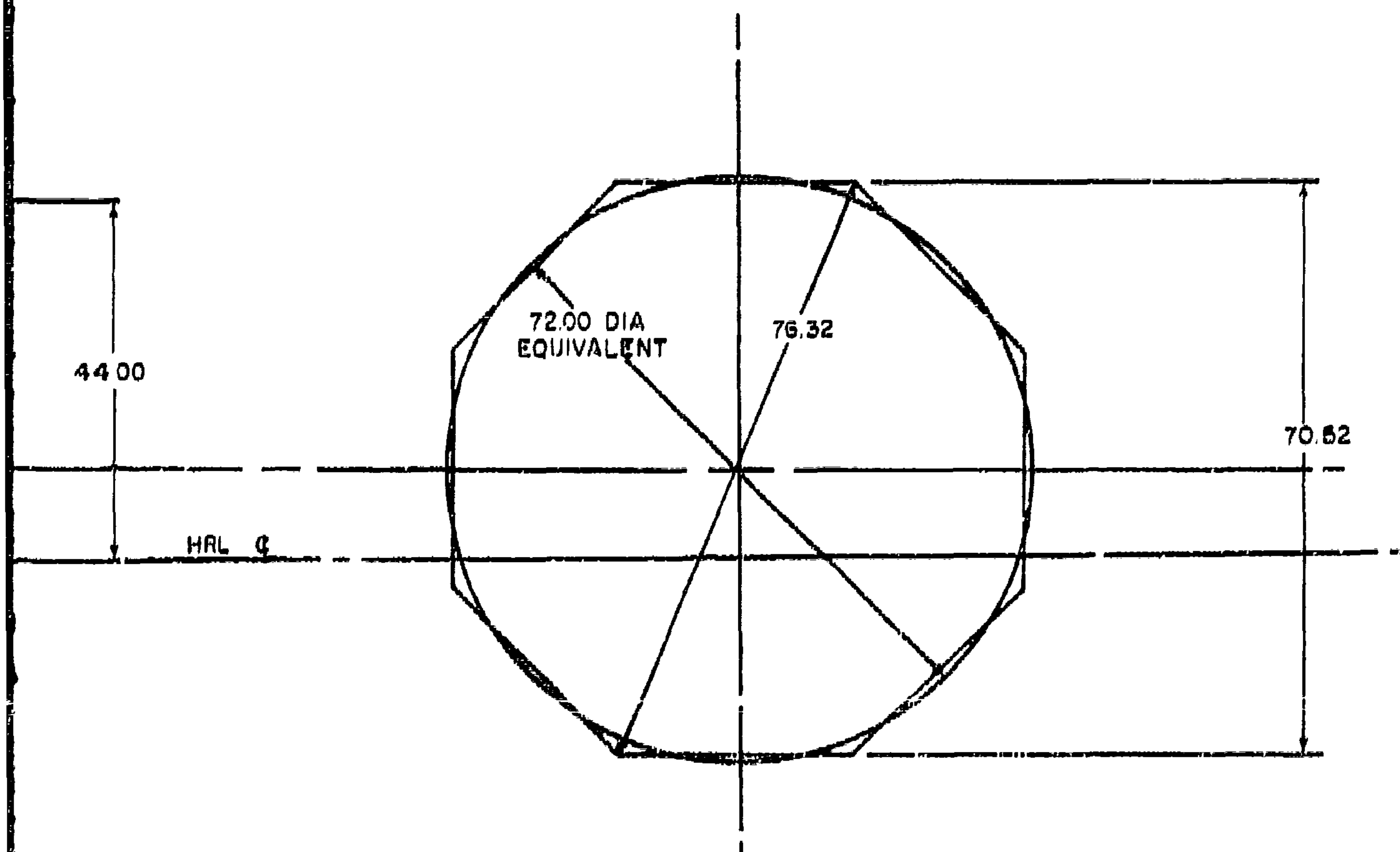
NVELOPE IS SUBJECT TO CHANGE FOR
NGINE ACCESSORY ARRANGEMENT
ECONDARY AIR PROVISIONS FOR EJECTOR
EAR CASE & A/B COOLING PROVISIONS



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

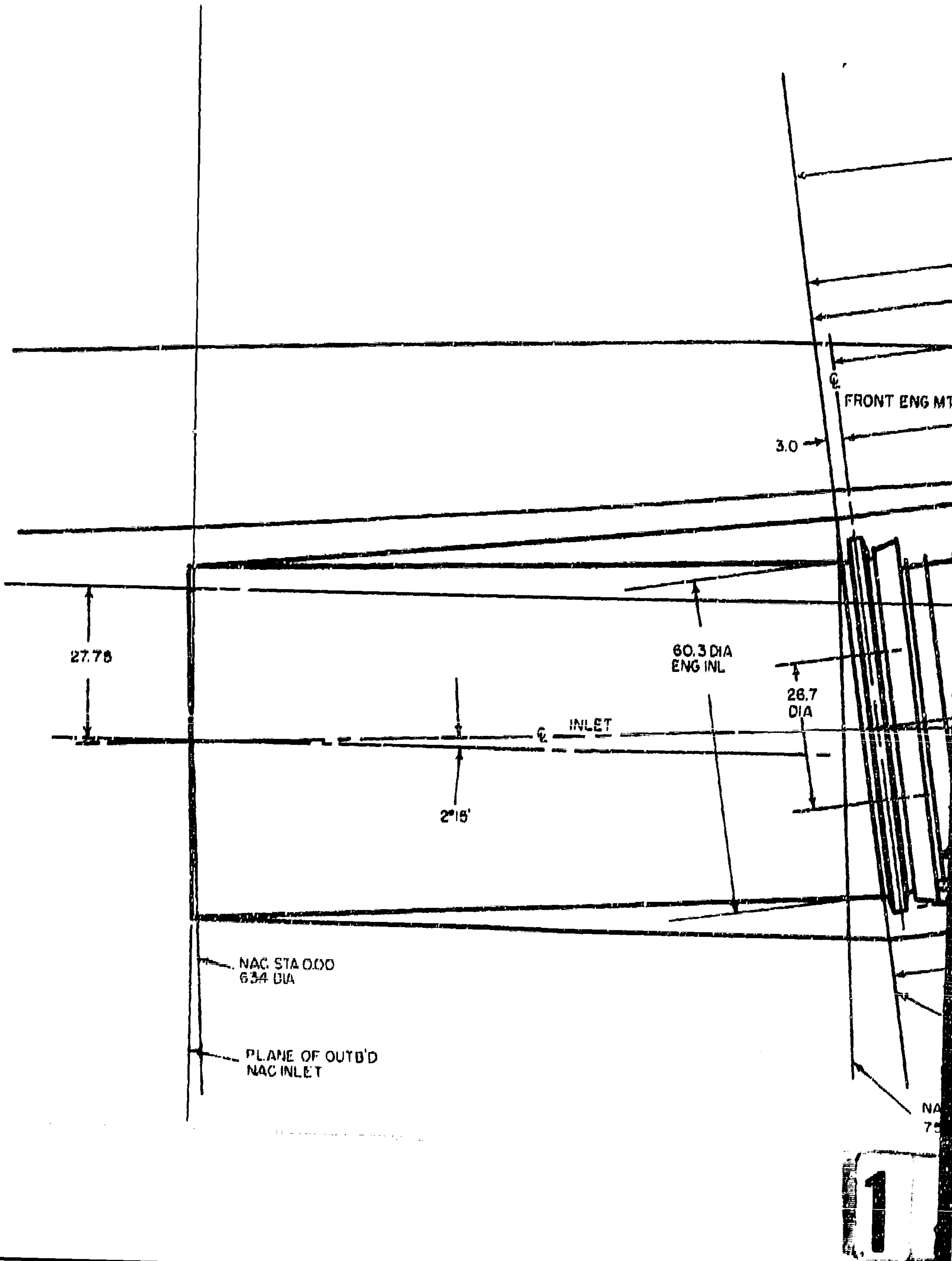
Figure 1-23

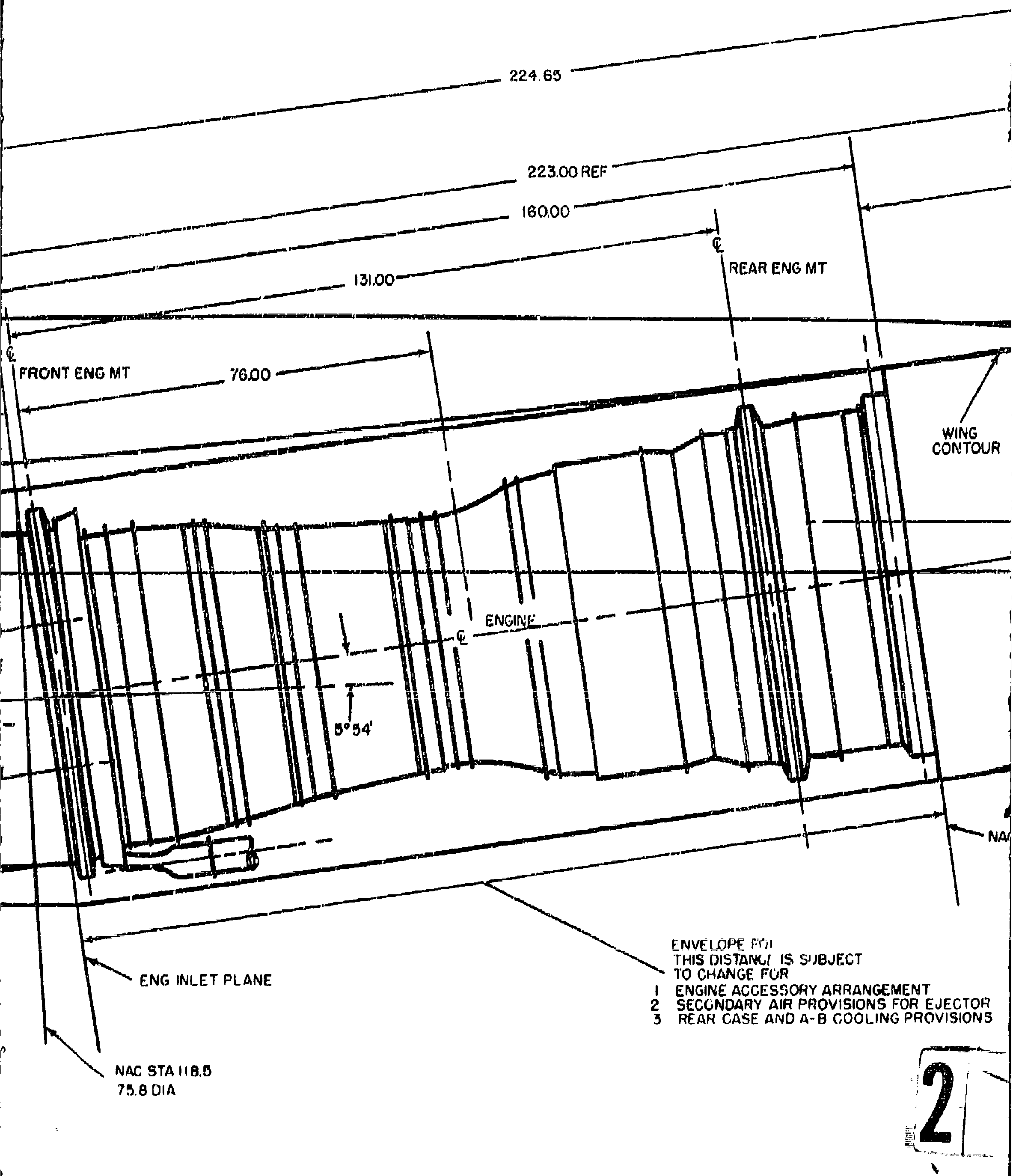


2000°F TURBINE INLET TEMPERATURE
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY A/R PROVISIONS FOR EJECTOR
3. REAR CASE & A/B COOLING PROVISIONS

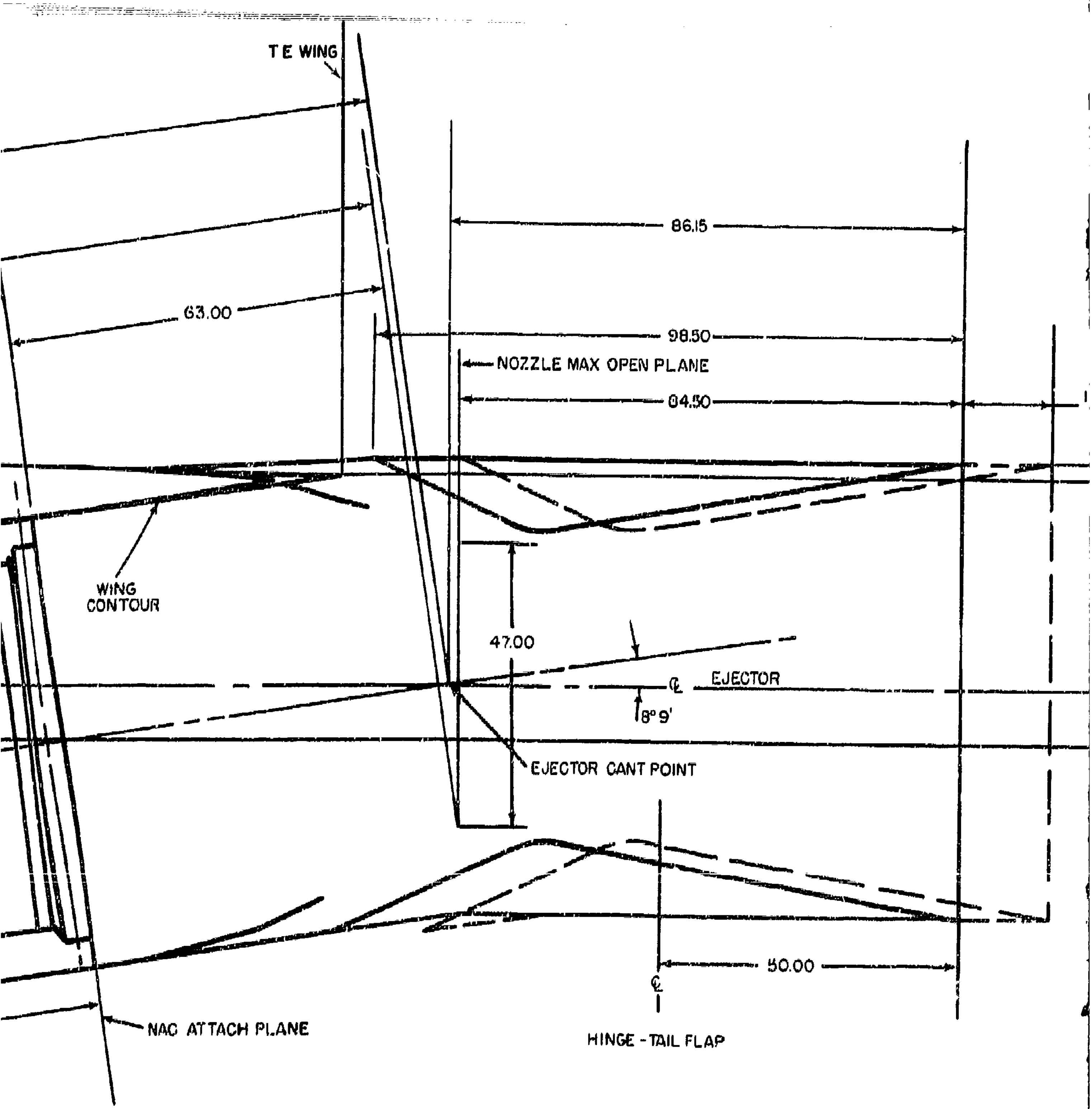
STJ227, 525 LBS./SEC. (LOW FLOW) TURBOJET

Figure 1-23


FRONT ENG MT
3.0
27.78
60.3 DIA
ENG INL
26.7
DIA
INLET
2°15'
NAC STA 0.00
63.4 DIA
PLANE OF OUTB'D
NAC INLET
NA
7



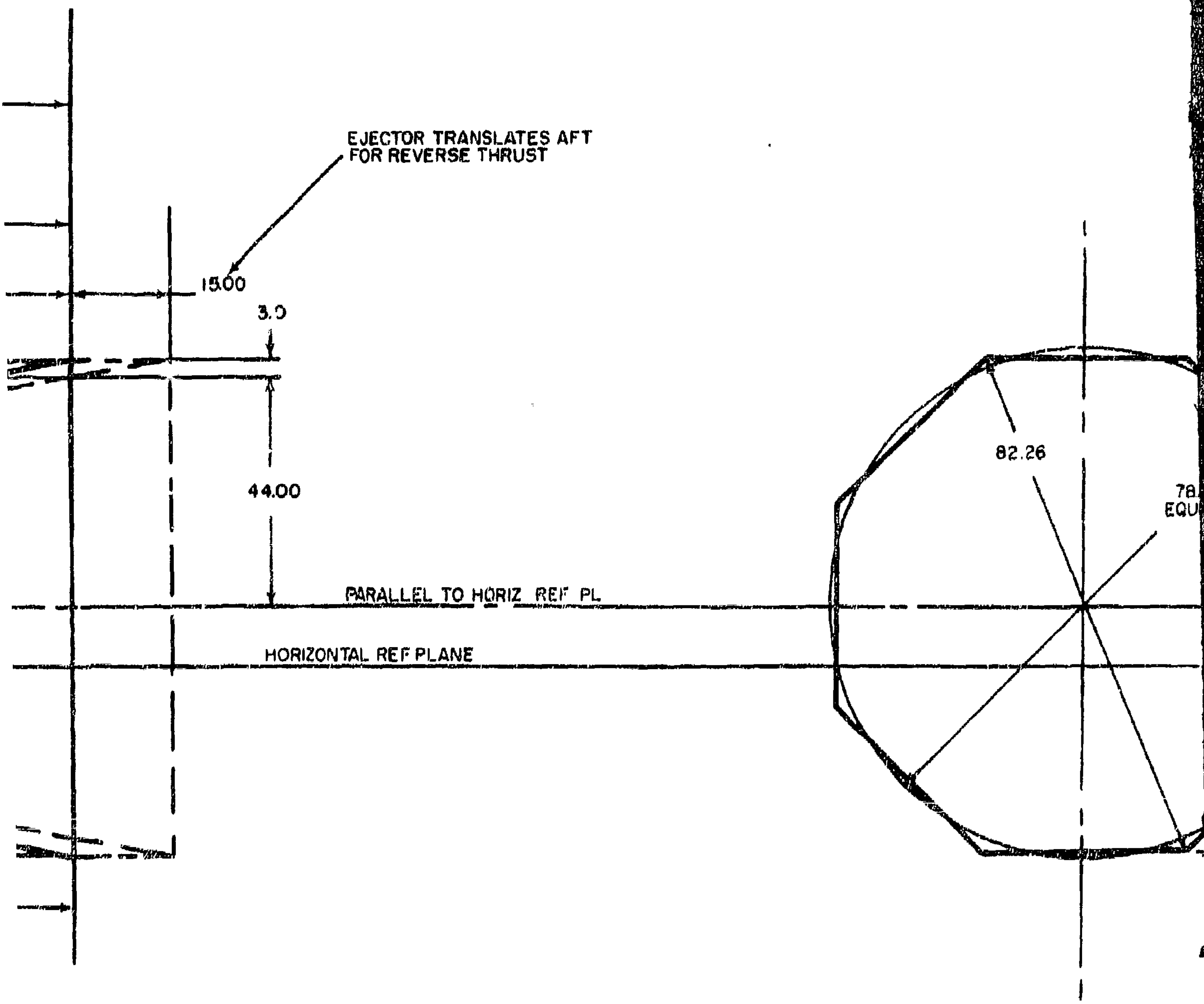
224.65
223.00 REF
160.00
131.00
REAR ENG MT
FRONT ENG MT
76.00
WING CONTOUR
ENGINE
5° 54'
ENG INLET PLANE
NAC STA 118.5
75.8 DIA
ENVELOPE FOI
THIS DISTANC/ IS SUBJECT
TO CHANGE FOR
1 ENGINE ACCESSORY ARRANGEMENT
2 SECONDARY AIR PROVISIONS FOR EJECTOR
3 REAR CASE AND A-B COOLING PROVISIONS
2



ːT

ANGEMENT
IONS FOR EJECTOR
OLING PROVISIONS

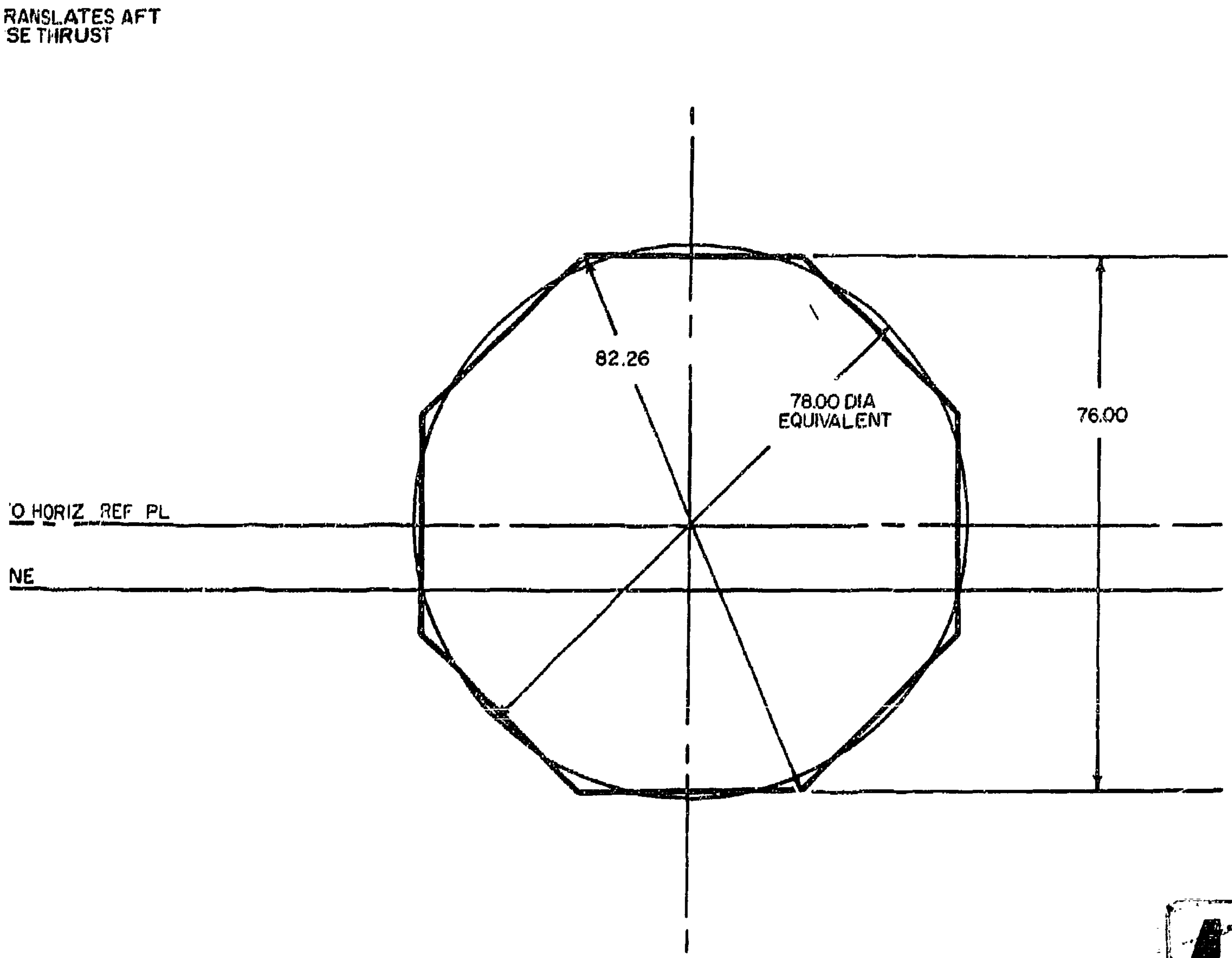
3



2300°F TURBINE INLET TEMPE

STJ227, 525 LBS./SEC. (HIGH FLO

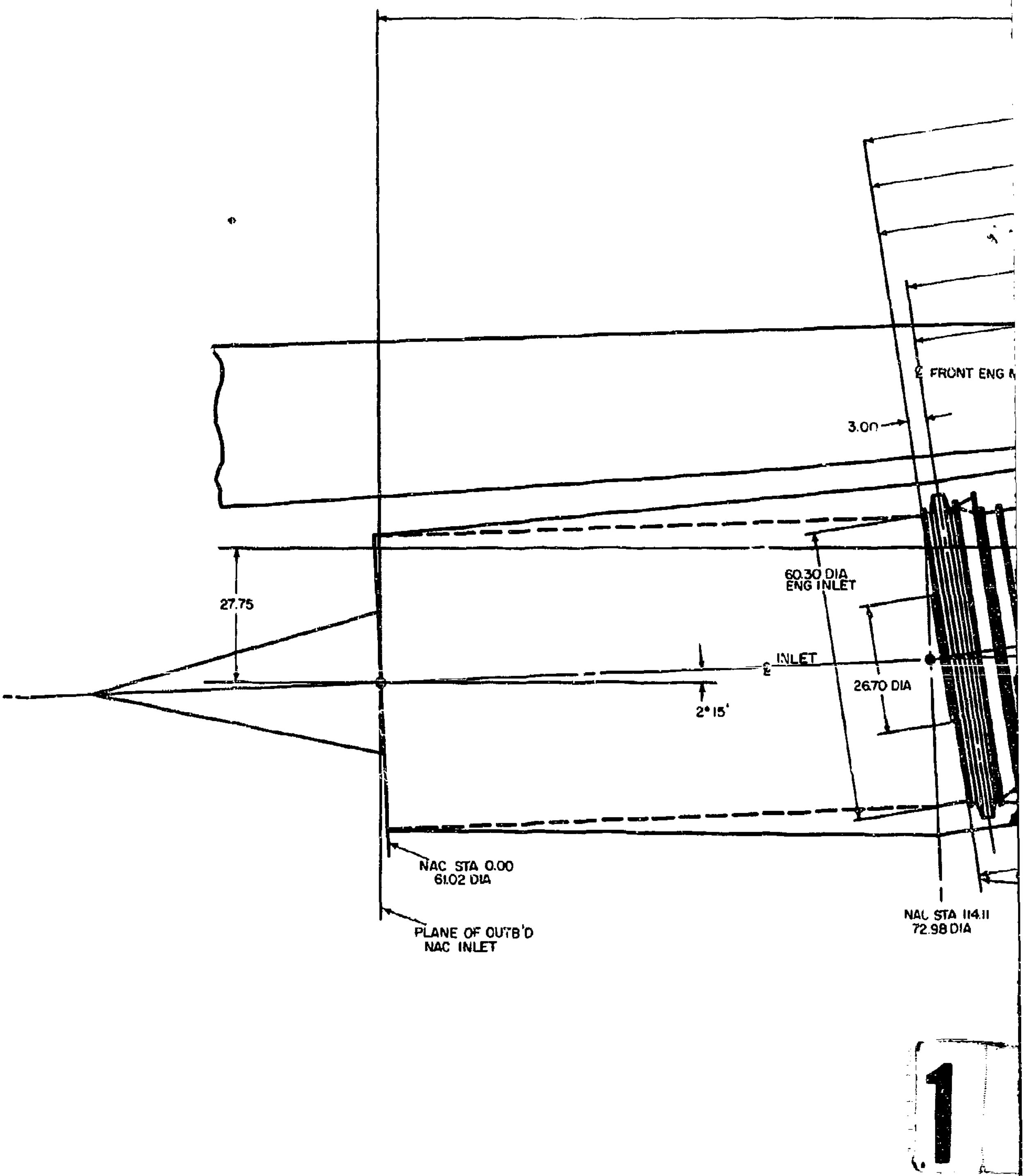
Figure 1-24



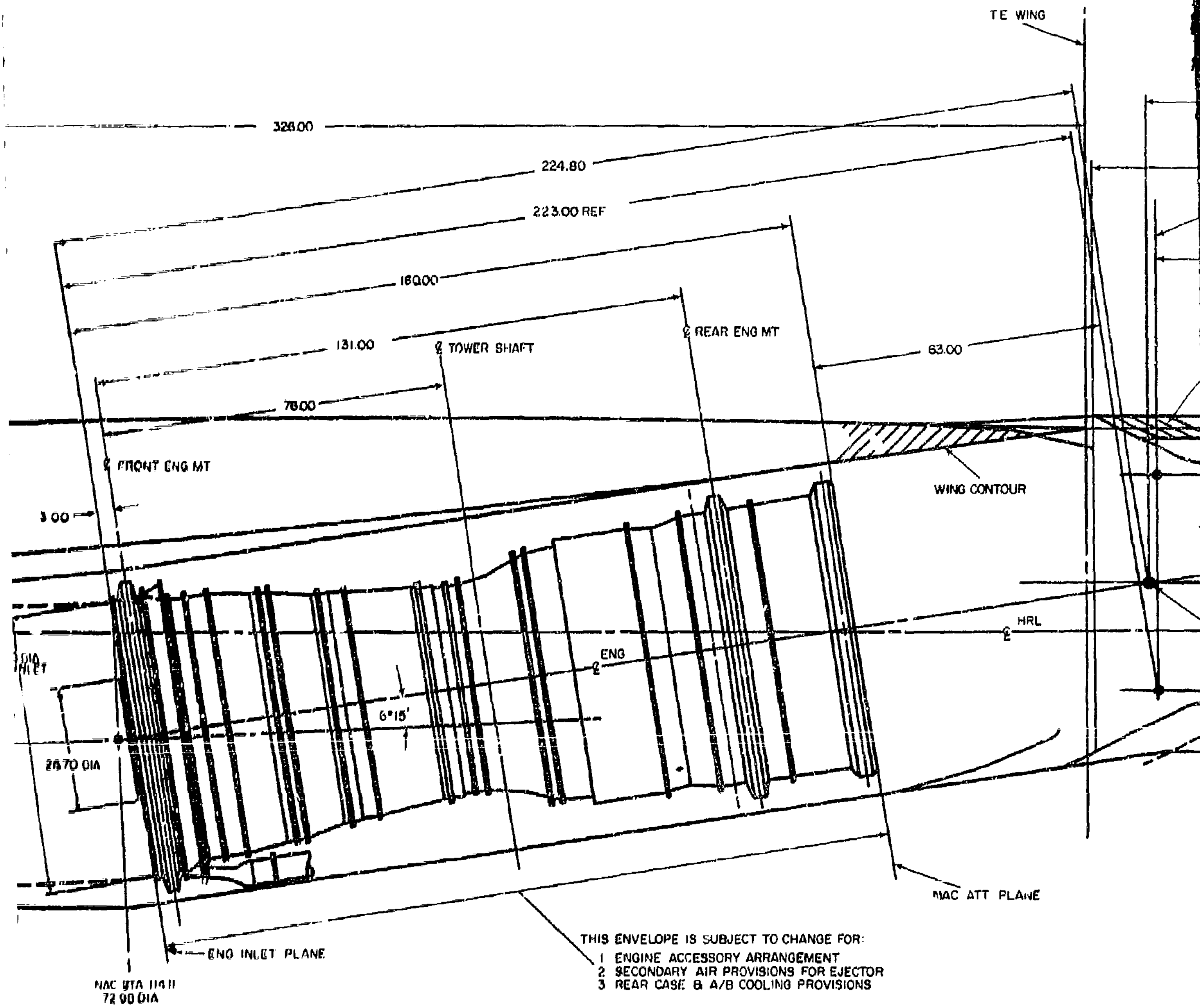
2300°F TURBINE INLET TEMPERATURE

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

Figure 1-24



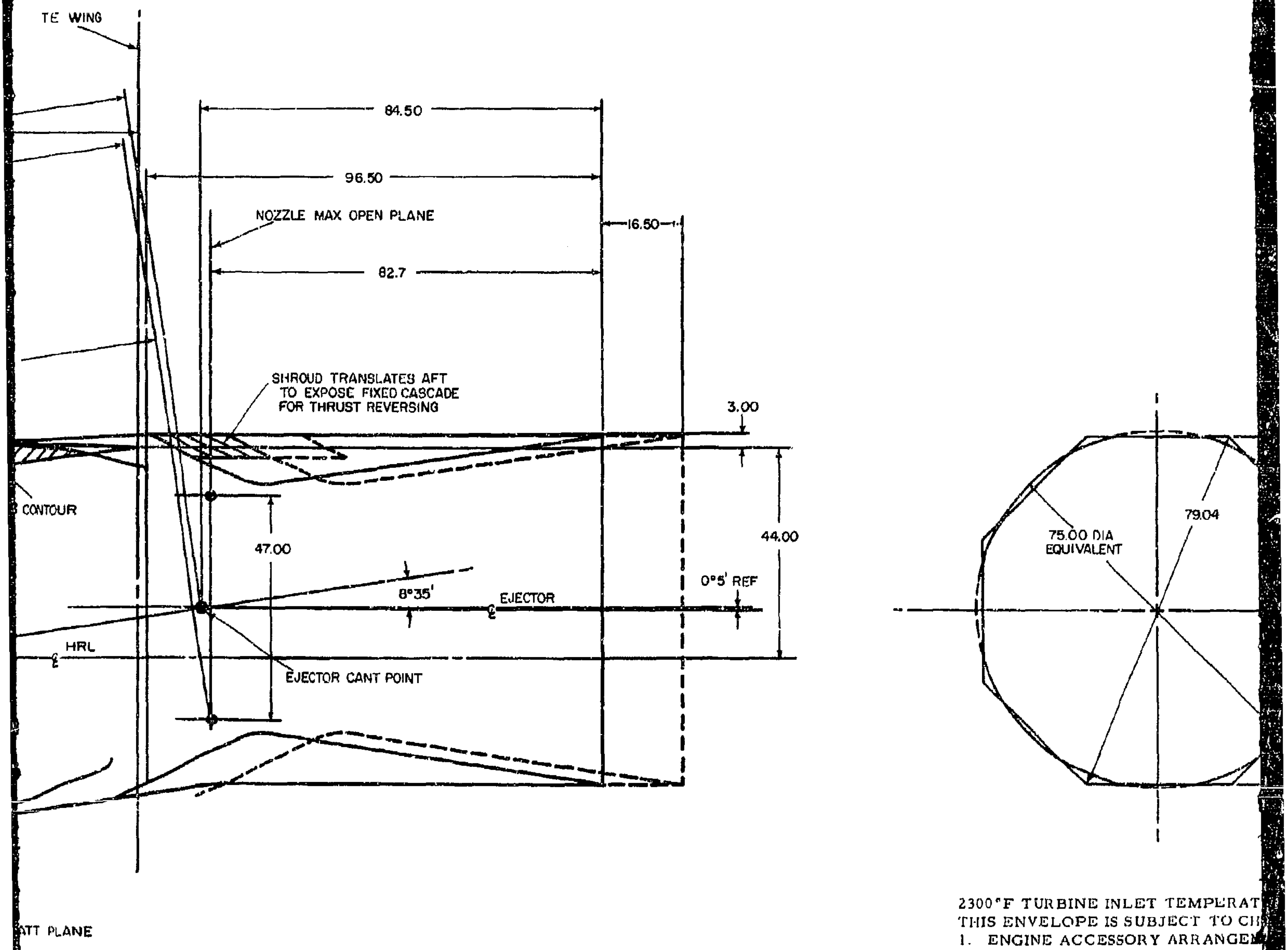
FRONT ENG M
3.00
60.30 DIA
ENG INLET
27.75
INLET
26.70 DIA
2° 15'
NAC STA 0.00
61.02 DIA
PLANE OF OUTB'D
NAC INLET
NAC STA 114.11
72.98 DIA
1



T E WING
328.00
224.80
223.00 REF
180.00
131.00
℄ TOWER SHAFT
℄ REAR ENG MT
63.00
78.00
℄ FRONT ENG MT
WING CONTOUR
3.00
HRL
ENG
6°15'
26.70 DIA
NAC ATT PLANE
ENG INLET PLANE
NAC STA 114.11
72.90 DIA
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:
1 ENGINE ACCESSORY ARRANGEMENT
2 SECONDARY AIR PROVISIONS FOR EJECTOR
3 REAR CASE & A/B COOLING PROVISIONS

2

CONFIDENTIAL

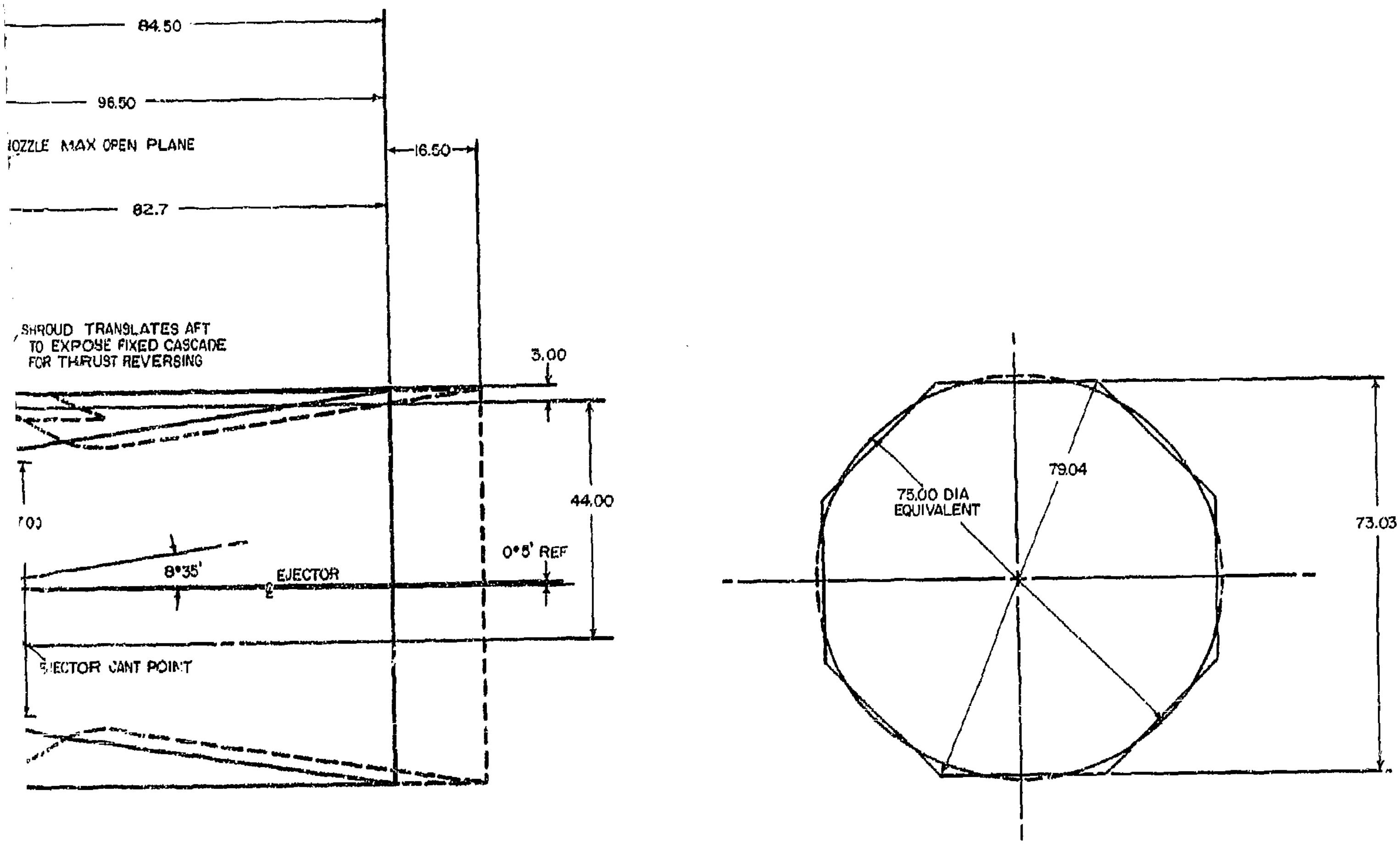


2300°F TURBINE INLET TEMPERAT
THIS ENVELOPE IS SUBJECT TO CH
1. ENGINE ACCESSORY ARRANGEM
2. SECONDARY AIR PROVISIONS FO
3. REAR CASE & A/B COOLING PRO

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

Figure 1-25

CONFIDENTIAL

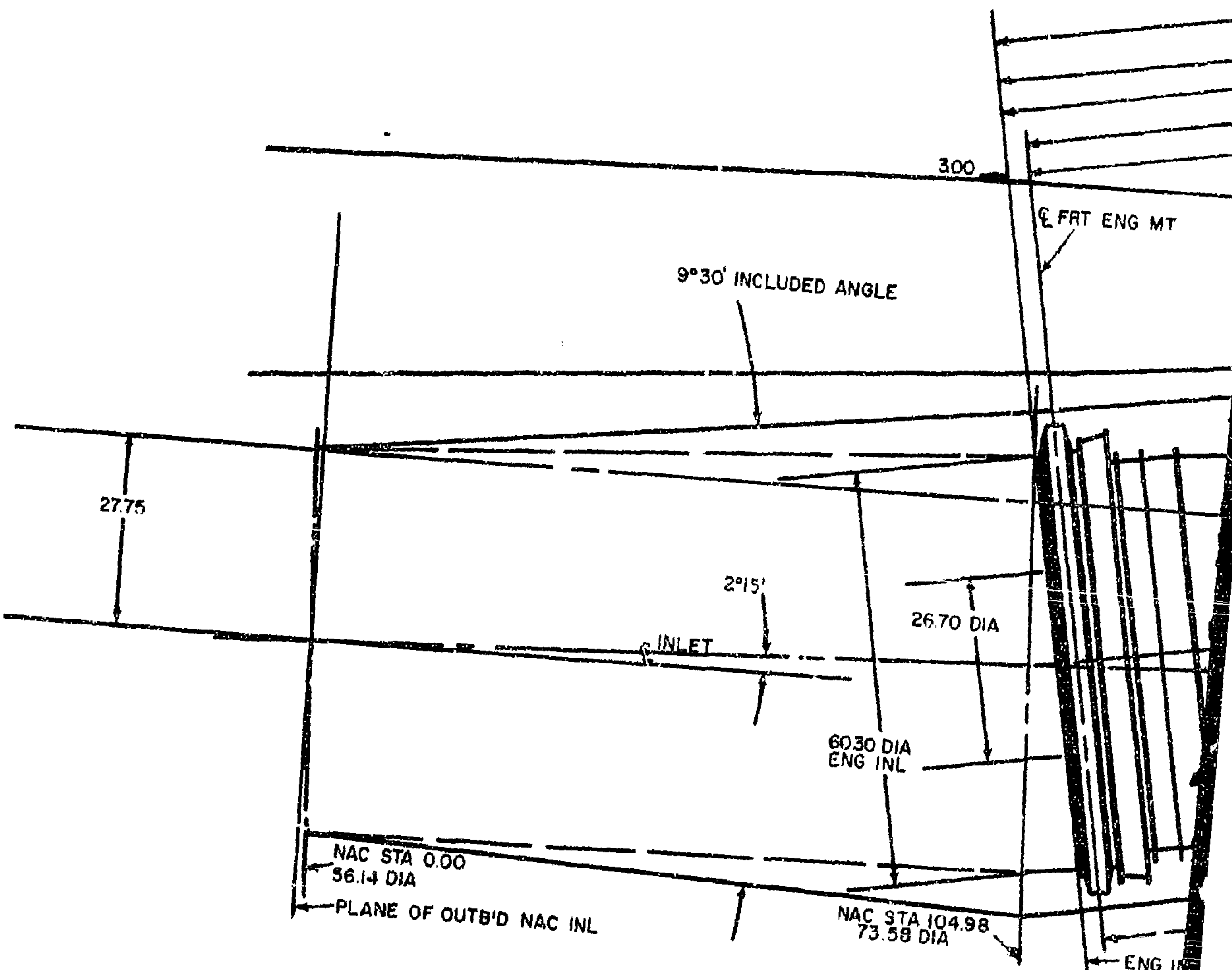


2300°F TURBINE INLET TEMPERATURE
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:

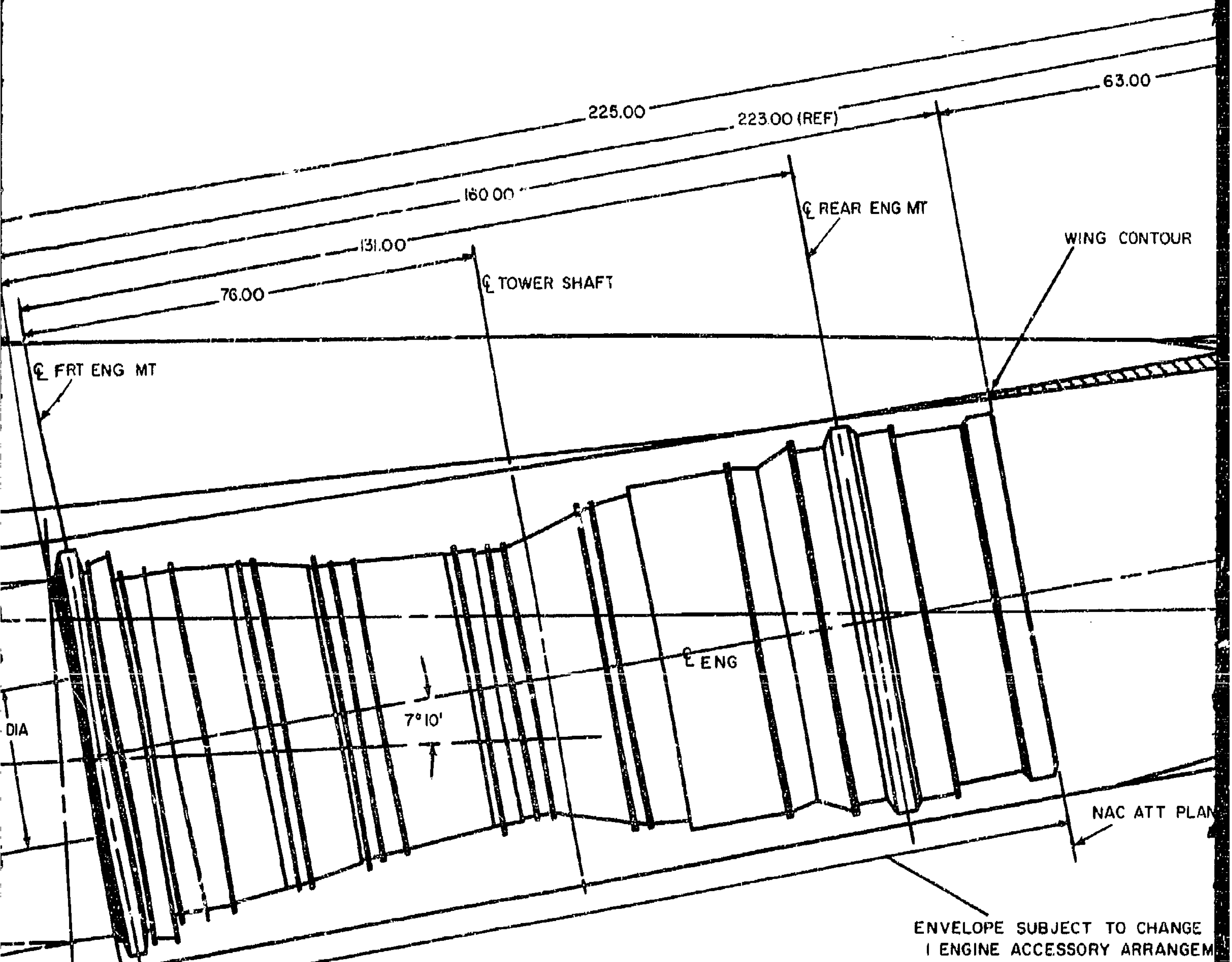
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR EJECTOR
3. REAR CASE & A/B COOLING PROVISIONS

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

Figure 1-25

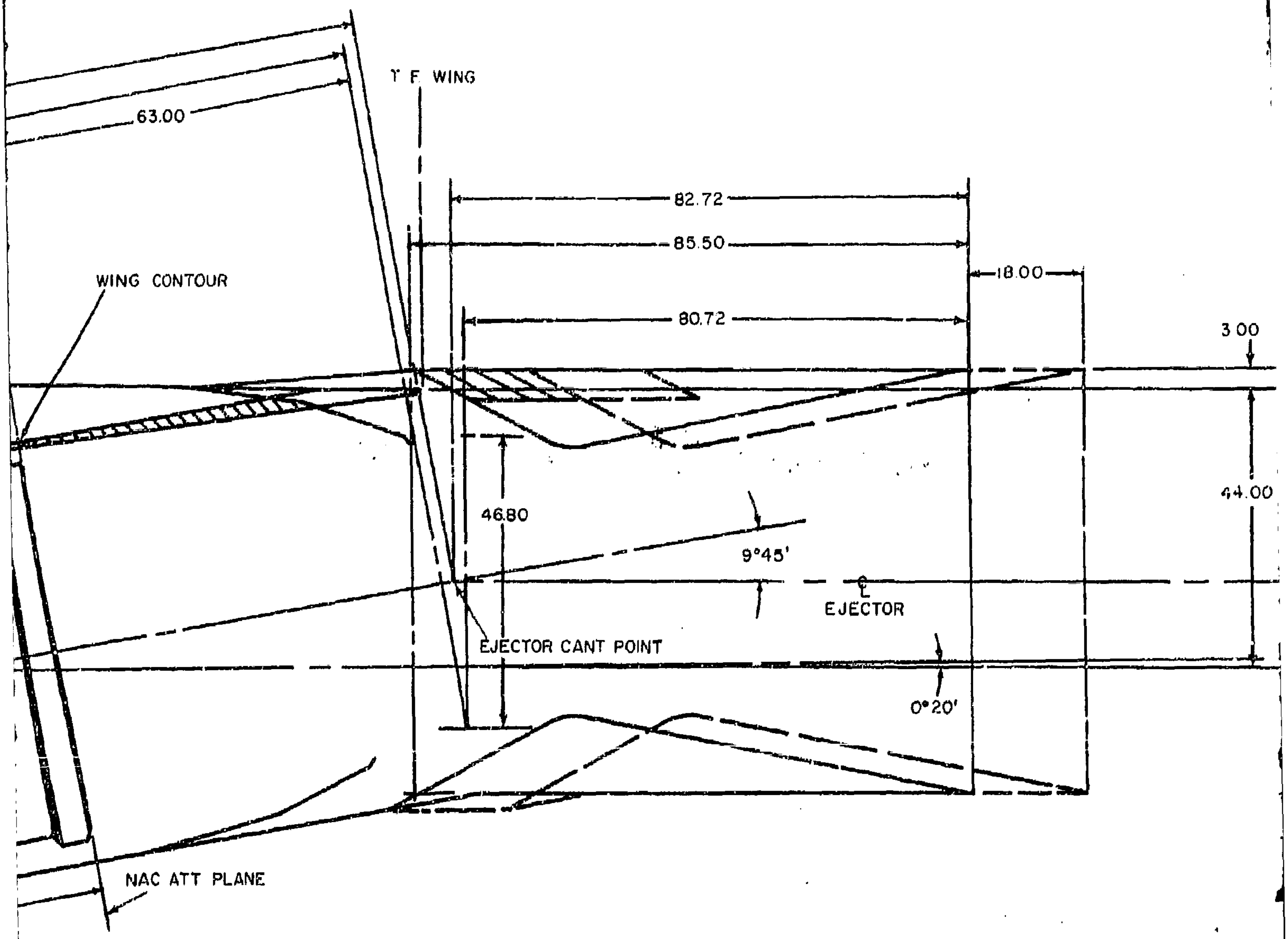

3.00
℄ FRT ENG MT
9°30' INCLUDED ANGLE
27.75
2°15'
26.70 DIA
INLET
6030 DIA
ENG INL
NAC STA 0.00
56.14 DIA
PLANE OF OUTB'D NAC INL
NAC STA 104.98
73.58 DIA
ENG I

1



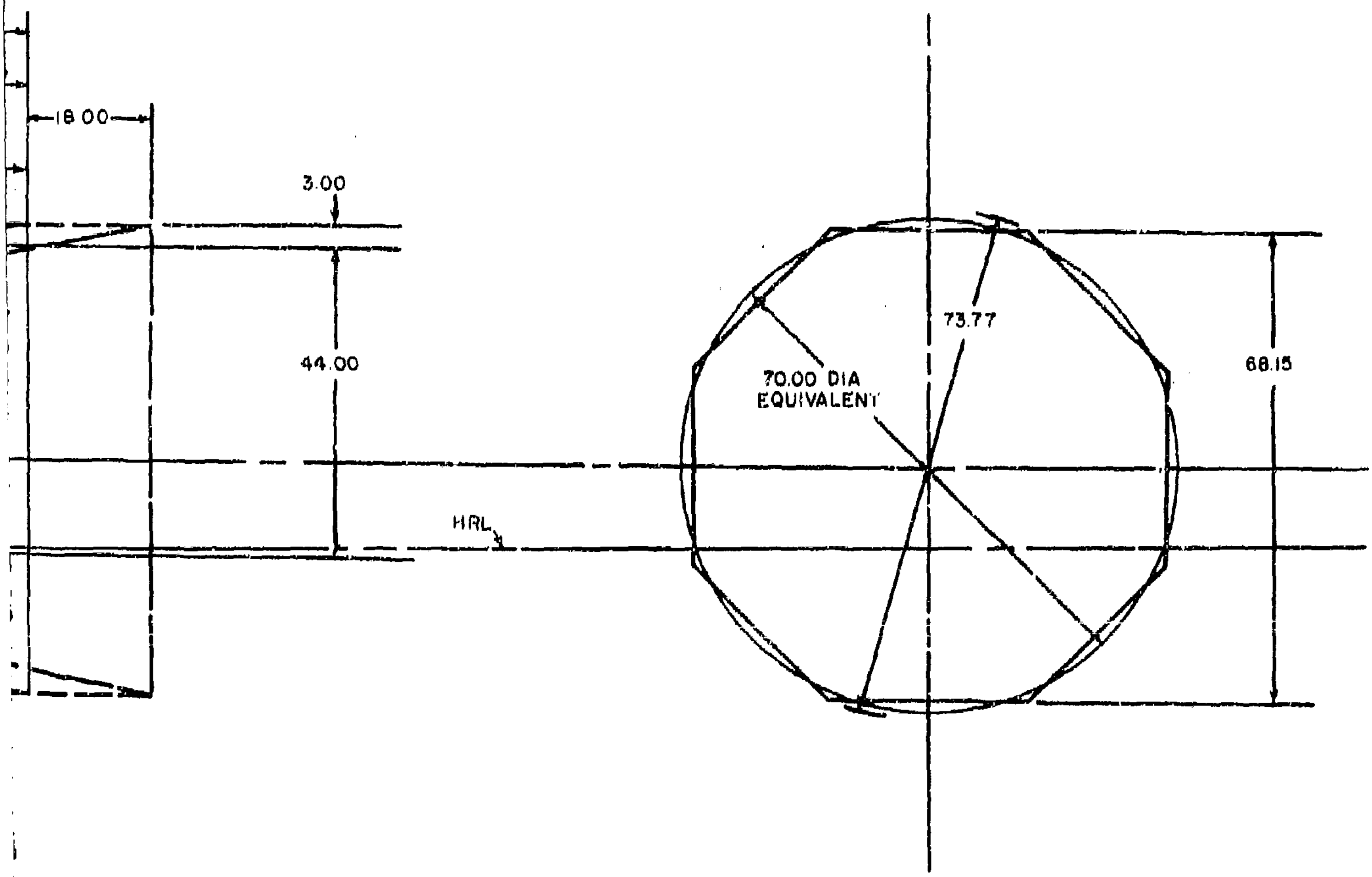
225.00
223.00 (REF)
63.00
160.00
131.00
76.00
℄ REAR ENG MT
WING CONTOUR
℄ TOWER SHAFT
℄ FRT ENG MT
℄ ENG
7°10'
DIA
NAC ATT PLAN
ENVELOPE SUBJECT TO CHANGE
1 ENGINE ACCESSORY ARRANGEM
2 SECONDARY AIR PROVISIONS, EJ
3 REAR CASE AND A/B COOLING
ENG INL PLANE

2



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

3



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

Figure 1-26

PRATT & WHITNEY AIRCRAFT

ACCESSORY DRIVE PADS

- A — POWER TAKEOFF
- B — TACHOMETER (TACH) PAD ... (AND ...)
- D — HYDRAULIC PUMP (TWO) ...

FUEL DRAIN

- FD1 — COMBUSTION CHAMBER FUEL DRAIN
- FD2 — DUMP VALVE DRAIN (MAIN)
- FD3 — DUMP VALVE DRAIN (A/B ZONE 1)
- FD4A — DUMP VALVE DRAIN (A/B ZONE 2)
- FD5 — AFTERBURNER COMBUSTION CHAMBER FUEL DRAIN

FUEL PRESSURE

- FP1 — FUEL PUMP INLET PRESSURE
- FP3 — CHECK VALVE FUEL PRESSURE (OUTLET PRESSURE)

FUEL FLOW

- F1 — FUEL PUMP & AFTERBURNER PUMP SUPPLY INLET
- F2 — MAIN FUEL FLOWMETER SUPPLY INLET
- F3 — MAIN FUEL FLOWMETER SUPPLY OUTLET
- F5 — AFTERBURNER FLOWMETER SUPPLY INLET
- F6 — AFTERBURNER FLOWMETER SUPPLY OUTLET

FUEL VENT

- FV2 — FUEL PUMP OUTLET VENT

PRESSURE SENSING

- PT5 — TURBINE EXIT PRESSURE

OIL BREATHER

- LB2 — MAIN OIL OVERBOARD BREATHER

OIL DRAIN

- LD1 — OIL TANK DRAIN
- LD2 — OIL TANK OVERFLOW DRAIN
- LD3 — OIL ... DRAIN
- LD4 — GEARBOX MAIN OIL DRAIN
- LD7 — OIL STRAINER DRAIN

OIL FLOW

- L1 — OIL TANK REMOTE FILLER
- L2 — OIL TANK MANUAL FILL

OIL PRESSURE

- LP1 — PRESSURE FOR TRANSMITTER
- LP4 — OIL FILTER INLET PRESSURE
- LP5 — OIL FILTER OUTLET PRESSURE

(LP4, LP5) PROVISIONS FOR PRESSURE DIFFERENTIAL

OIL TEMPERATURE

- LT1 — MAIN OIL TEMPERATURE

OIL VENT

- LV3 — OIL PRESSURE TRANSMITTER VENT

SEAL DRAIN

- SD1 — FUEL CONTROL SEAL DRAIN
- SD2 — FUEL PUMP SEAL DRAIN
- SD3 — HYDRAULIC PUMP SEAL DRAIN
- SD4 — AFTERBURNER FUEL PUMP SEAL DRAIN

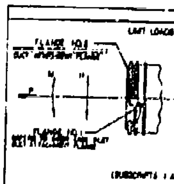
TEMPERATURE SENSING

- TT5 — TURBINE EXIT TEMPERATURE (AVG)
- TT5 — TURBINE EXIT TEMPERATURE (INDIVIDUAL)

MISCELLANEOUS

- F — MAIN FUEL PUMP
- G — FUEL PUMP FUEL FILTER (MAIN)
- H — FUEL PUMP FILTER (... SPACE FOR REMOVAL)
- J — AFTERBURNER FUEL PUMP
- K — HYDRAULIC PUMP
- L — OIL PUMP
- M — OIL FILTER
- N — FUEL OIL COOLER (MAIN)
- P — FUEL OIL COOLER (AFTERBURNER)
- R — GEARBOX
- S — AUTOMATIC RESTART SWITCH
- T — BREATHER PRESSURIZING VALVE
- U — HIGH PRESSURE BLEED PAD
- X — MAIN FUEL CONTROL
- Y — AFTERBURNER FUEL CONTROL

- Z — EXHAUST NOZZLE CONTROL
- AA — HYDRAULIC DISCHARGE FILTER
- AB — IGNITER PLUG (... SPACE FOR REMOVAL)
- AC — IGNITER PLUG (AFTERBURNER) (... SPACE FOR REMOVAL)
- AD — ENGINE FRONT MOUNTING PROVISIONS
- AE — ENGINE REAR MOUNTING PROVISIONS
- AF — FUEL RETURN TO SUPPLEMENTARY COOLING
- AG — GROUND HANDLING HOLES (FRONT MOUNT)
- AH — GROUND HANDLING AREA (REAR)
- AJ — POWER CONTROL LEVER (... ANGLE OF TRAVEL)
- AK — SHUT OFF LEVER (... ANGLE OF TRAVEL)
- AL — APPROACH VELOCITY CONTROL LEVER (... ANGLE OF TRAVEL)
- AM — MACH NO OR STOP POSITION RESET LEVER (... ANGLE OF TRAVEL)
- AN — AIR TURBINE VALVE (AFTERBURNER FUEL PUMP)
- AP — IGNITION EXCITER ELECTRICAL CONN
- AR — THERMAL ANTI ICING PAD
- AS — AERODYNAMIC BRAKE CONTROL AIR SUPPLY CONN
- AT — EXHAUST NOZZLE FEEDBACK
- AU — CHECK & DUMP VALVE (AFTERBURNER ZONE 1)
- AV — CHECK & DUMP VALVE (AFTERBURNER ZONE 2)
- AW — GAS GENERATOR FUEL FLOWMETER
- AX — MAIN FUEL FLOWMETER
- AY — AFTERBURNER FUEL FILTER (... SPACE FOR REMOVAL)
- AZ — WINDMILL BYPASS CHECK & DUMP VALVE (MAIN FUEL)
- BA — OIL TANK
- BB — NOZZLE POSITION INDICATOR MOUNTING PROVISIONS
- BC — ... POSITION INDICATOR CONN (ELECTRICAL)
- BD — THERMOSTATIC FUEL BYPASS VALVE
- BF — VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFR)
- BN — CONTROL CABLE - FUEL CONTROLS
- BP — SECONDARY AIR BYPASS (NCT)
- BS — FUEL INLET TEMPERATURE SENSOR
- BU — OIL COOLER THERMOSTATIC BYPASS VALVE AFTERBURNER FUEL
- BV — HYDRAULIC RETURN FILTER
- BW — EXHAUST NOZZLE ACTUATORS
- BY — BYPASS AND BLEED DOOR ACTUATORS
- BZ — INLET GUIDE VANE ACTUATORS
- CA — AERODYNAMIC BRAKE ACTUATORS
- CB — EXCITER (MAIN)
- CC — EXCITER (AFTERBURNER)



UNIT LOADS FOR ENGINE INLET FLANGE A

M = TOTAL MOMENT IN PLANE CONTAINING ENGINE AXIS THAT IS APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (IN LB)

H = TOTAL LOAD PERPENDICULAR TO ENGINE AXIS WHICH IS APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (LBS)

P = TOTAL AXIAL LOAD APPLIED TO FLANGE NO 1 OR NO 2 DUE TO INSTALLATION LOADS AND MANEUVER LOADS (LBS)

(SUBSCRIPTS 1 AND 2 REFER TO FLANGES 1 AND 2 RESPECTIVELY)

MAX ALLOWABLE MANEUVER LOAD EQUATIONS FOR FLANGE NO

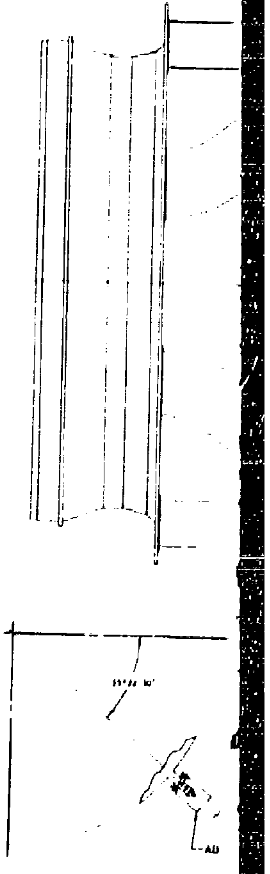
1 [3.43 P1 + 75 H1] 10^{-3} ...

2 [60 P2 + 28 H2 + ...] 10^{-3} ...

MAX ALLOWABLE 1 G (GROUND) LOAD EQUATIONS FOR FLANGE NO 2

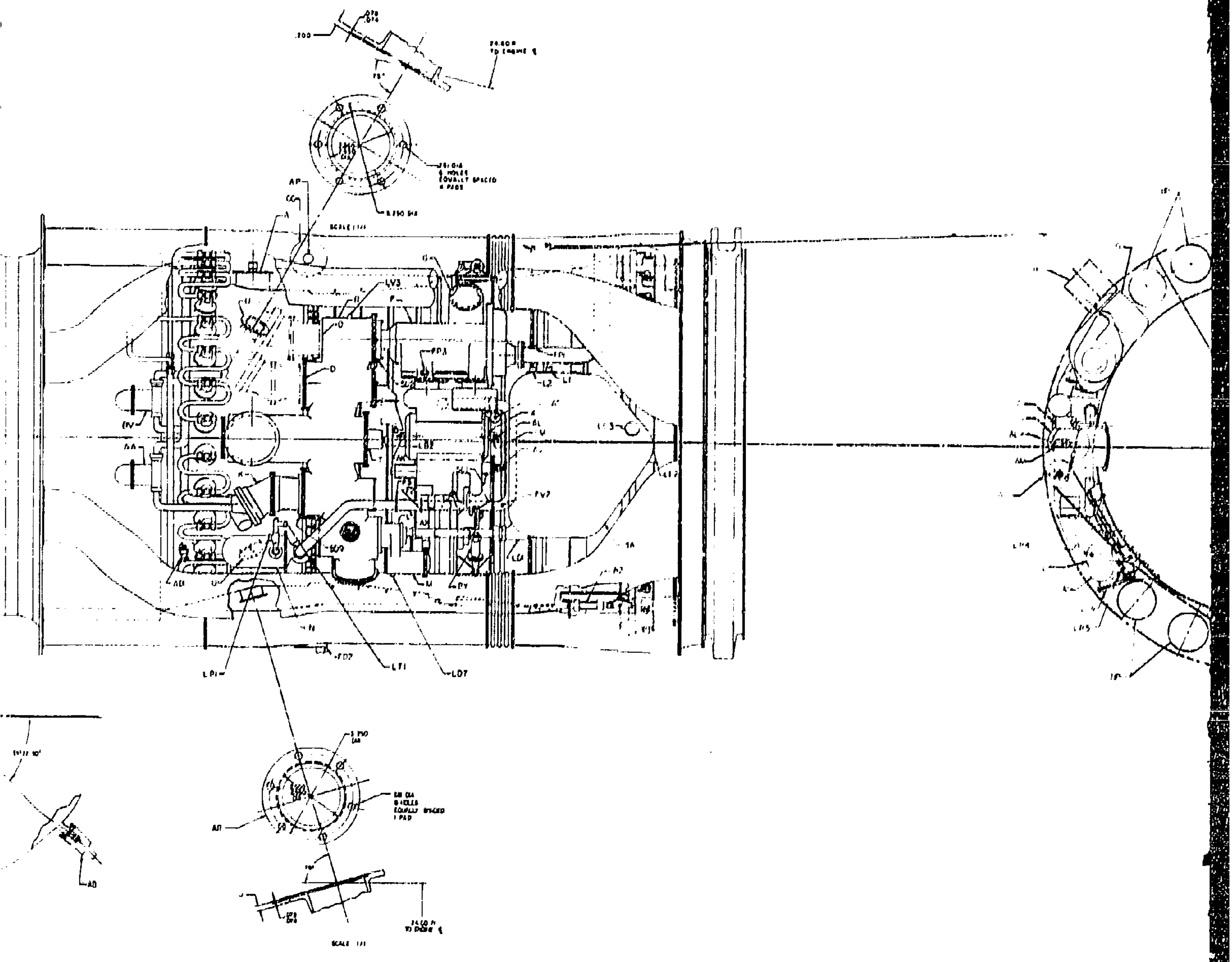
1 [M2 = 60,000 IN LBS H2 = 5000 LBS P2 = 3000 LBS

ALL LOADS AND MOMENTS TO BE USED AS ABSOLUTE VALUES

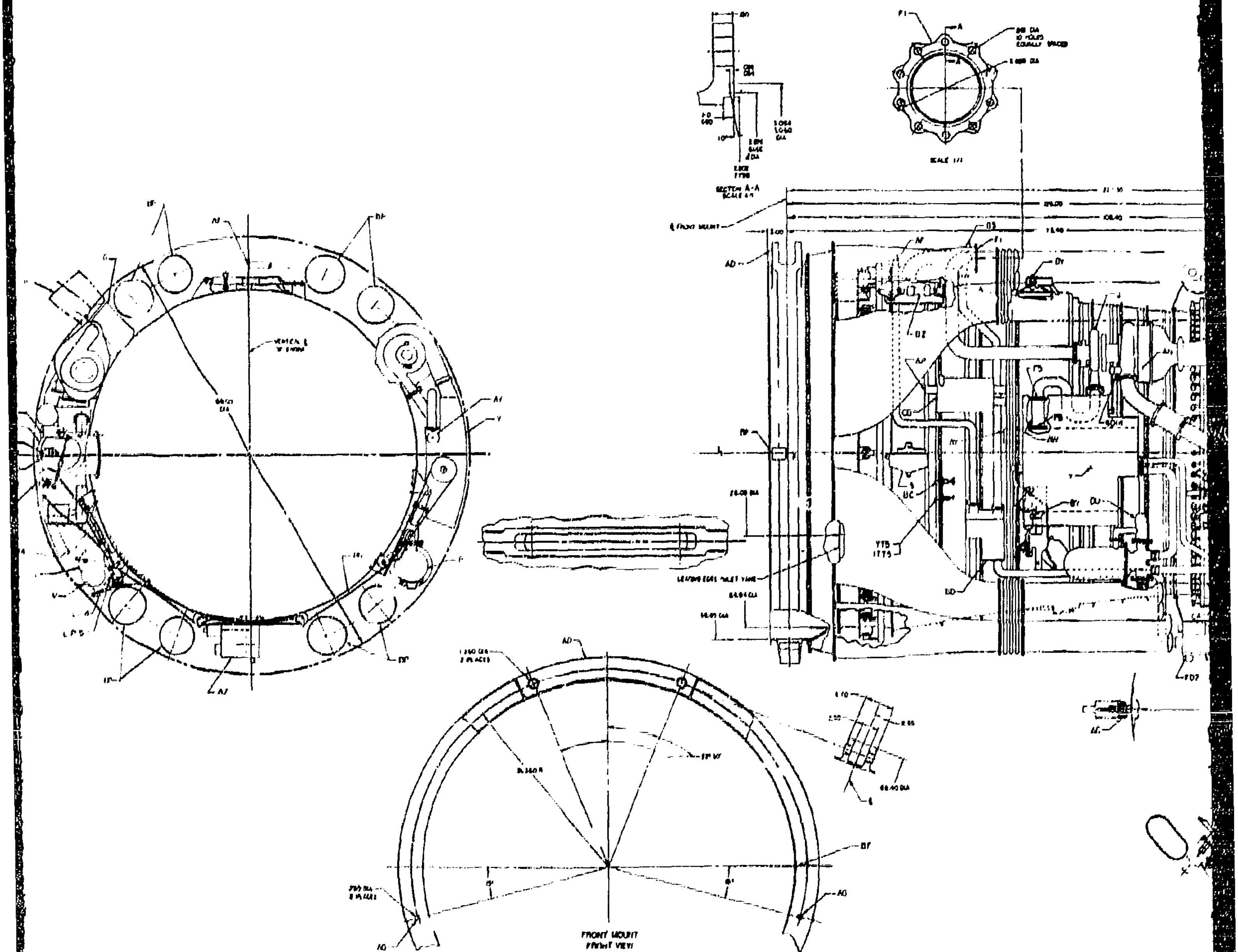


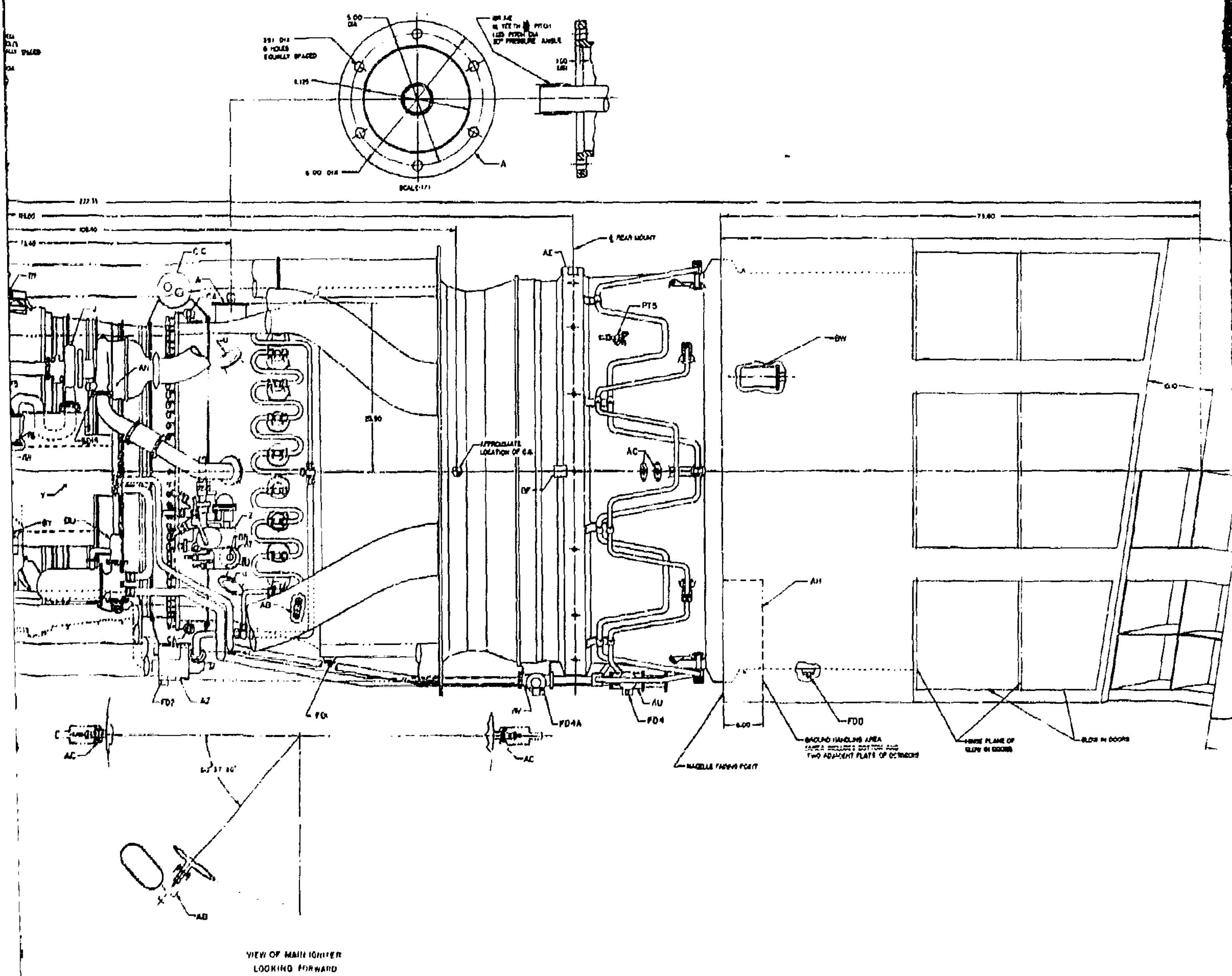
VIEW ... [illegible]

1



2

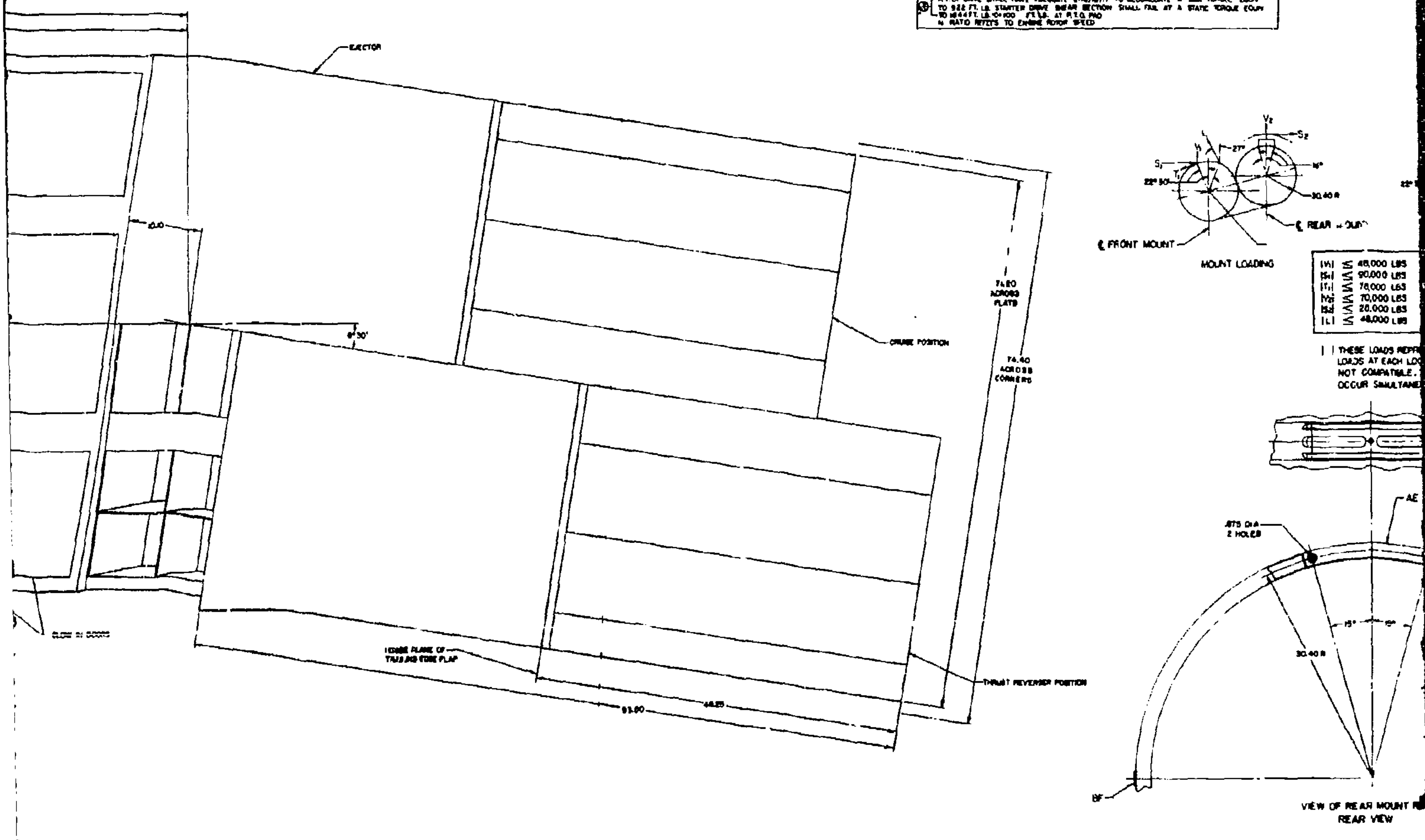

SECTION A-A
FRONT MOUNT
FRONT VIEW



5.00 DIA
6 HOLES
EQUALLY SPACED
6.00 DIA
SCALE 1/1
A
C C
PT5
BW
AC
AE
AH
AB
AU
AV
AZ
FD2
FD4
FD4A
FD8
℄ REAR MOUNT
APPROXIMATE LOCATION OF C.G.
73.60
25.90
GROUND HANDLING AREA
NACELLE FAIRING POINT
HINGE PLANE OF BLOW IN DOORS
BLOW IN DOORS
VIEW OF MAIN IGNITER
LOOKING FORWARD

TORQUE & MOMENT LIMITATION TABLE-ACCESS DRIVES							
DRIVE PAD	NOMINAL USE	TORQUE (LB. IN.)			SPEED RATIO	ROTATION (FACING ENGINE PAD)	1 G OVERHUNG MOMENT (LB IN.)
		CONTINUOUS	OVERLOAD	STATIC			
A	[illegible]	2334	3500 □	24200	2.94 N	CCW	———
B	TACH.	7	—	50	.762 N	CCW	———
C	HYD PUMP	946	1420	4730	.980 N	CCW	———

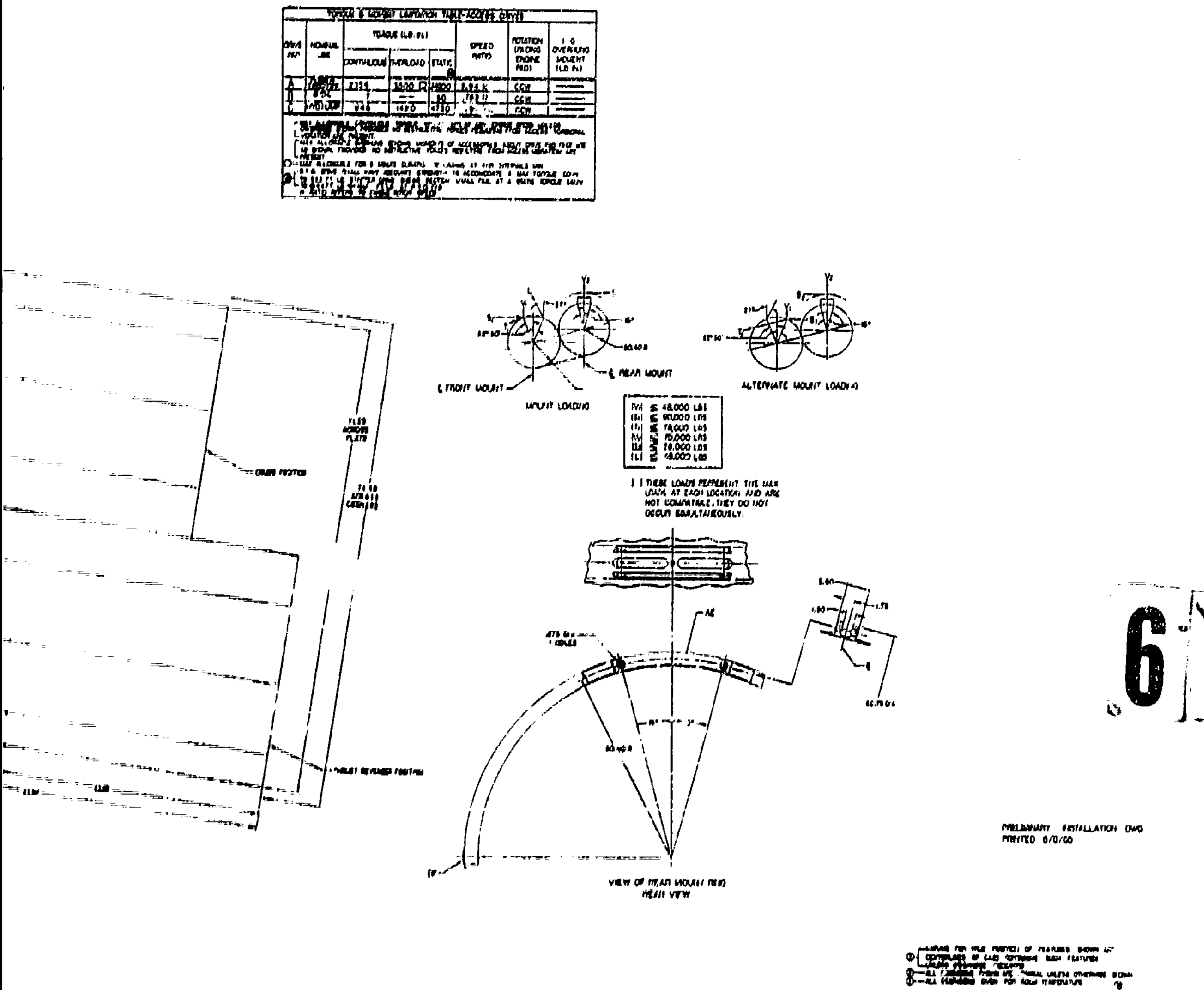
— MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT ANY ENGINE SPEED UNLESS OTHERWISE SHOWN, PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS. TORSIONAL VIBRATION ARE PRESENT.
— MAX ALLOWABLE OVERHUNG BENDING MOMENTS OF ACCESSORIES ABOUT DRIVE PAD FACE ARE AS SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION ARE PRESENT.
□ — MAX ALLOWABLE FOR 5 MINUTE DURATION RECURRING AT [illegible] INTERVALS [illegible]
— P.T.O. DRIVE SHALL HAVE ADEQUATE STRENGTH TO ACCOMMODATE A MAX TORQUE EQUAL TO 922 FT. LB. STARTER DRIVE SHEAR SECTION SHALL FAIL AT A STATIC TORQUE EQUAL TO [illegible] FT. LB. AT P.T.O. PAD
N RATIO REFERS TO ENGINE ROTOR SPEED



PROPOSED TURBOJET ACCESSO

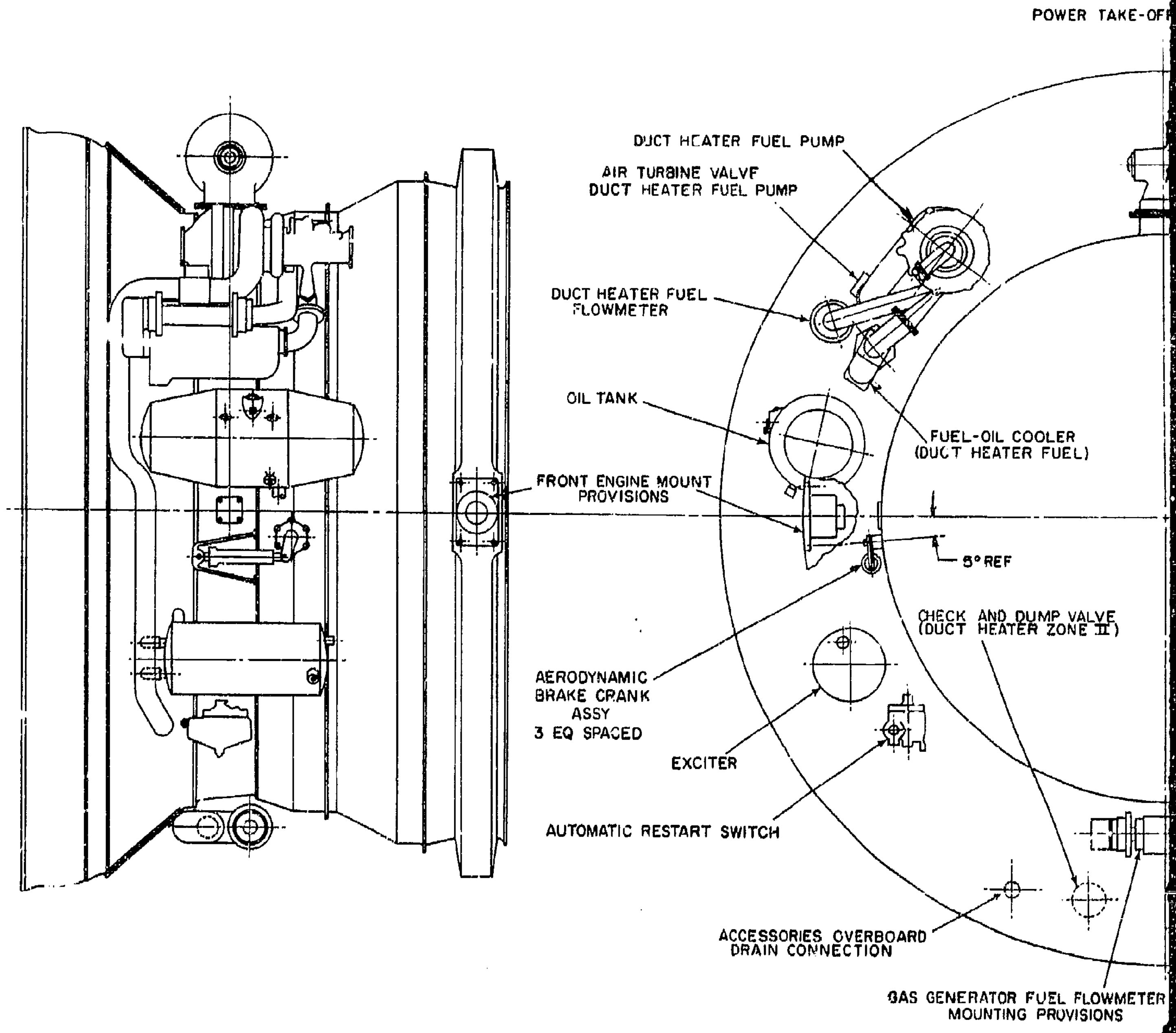
Figure 1-27

5



PROPOSED TURBOJET ACCESSORY ARRANGEMENT

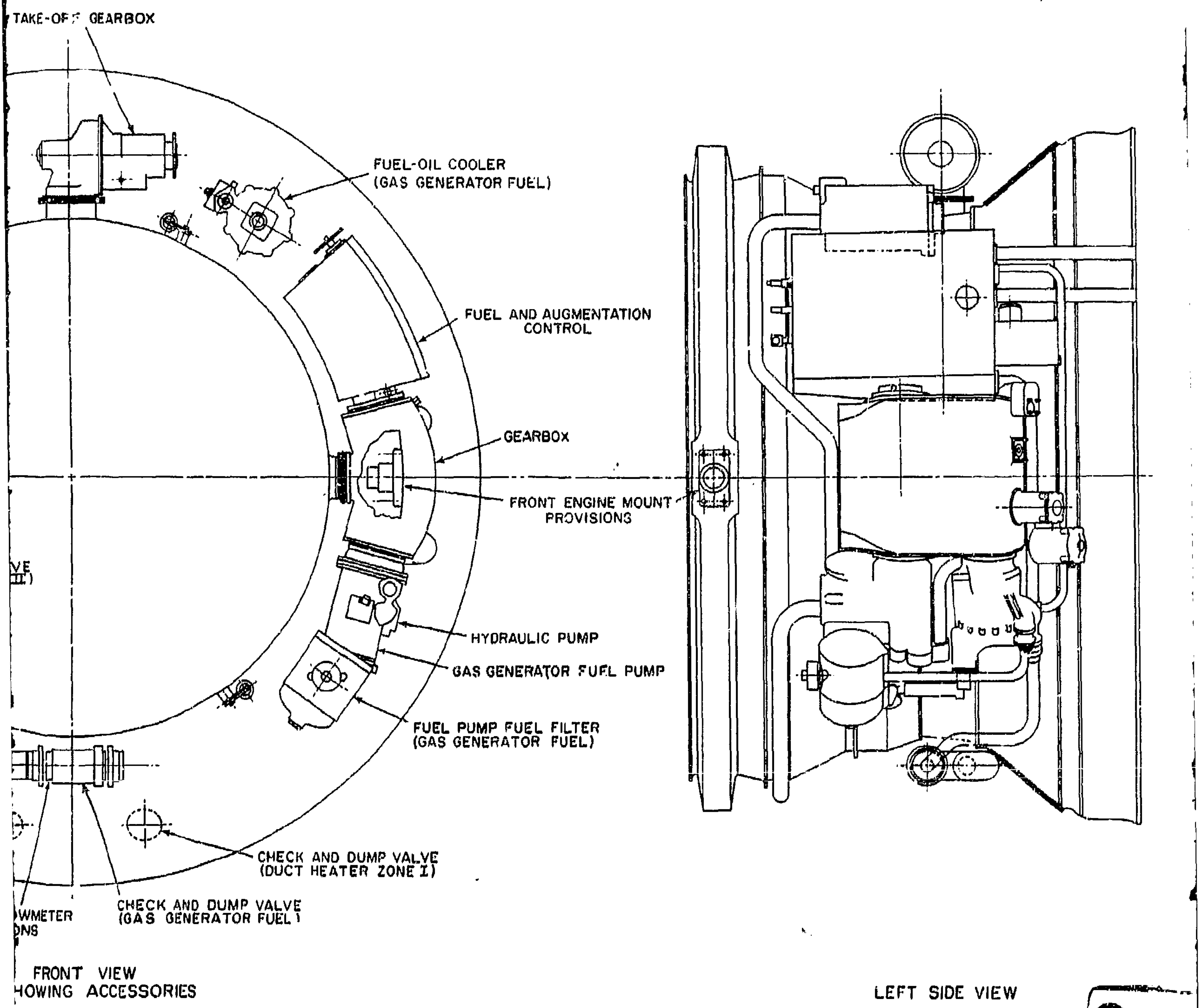
Figure 1-27



RIGHT SIDE VIEW

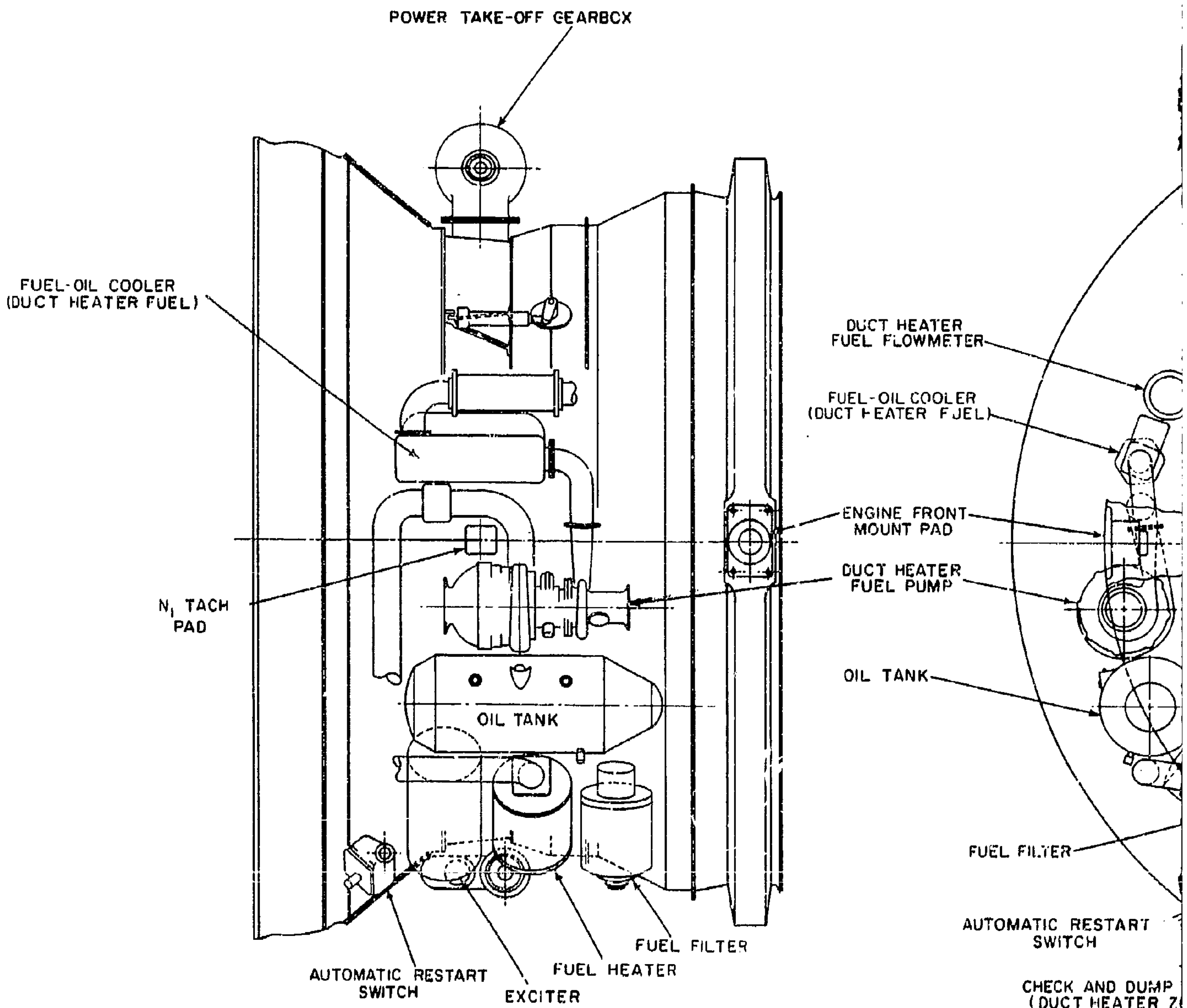
FRON
SHOWING

1

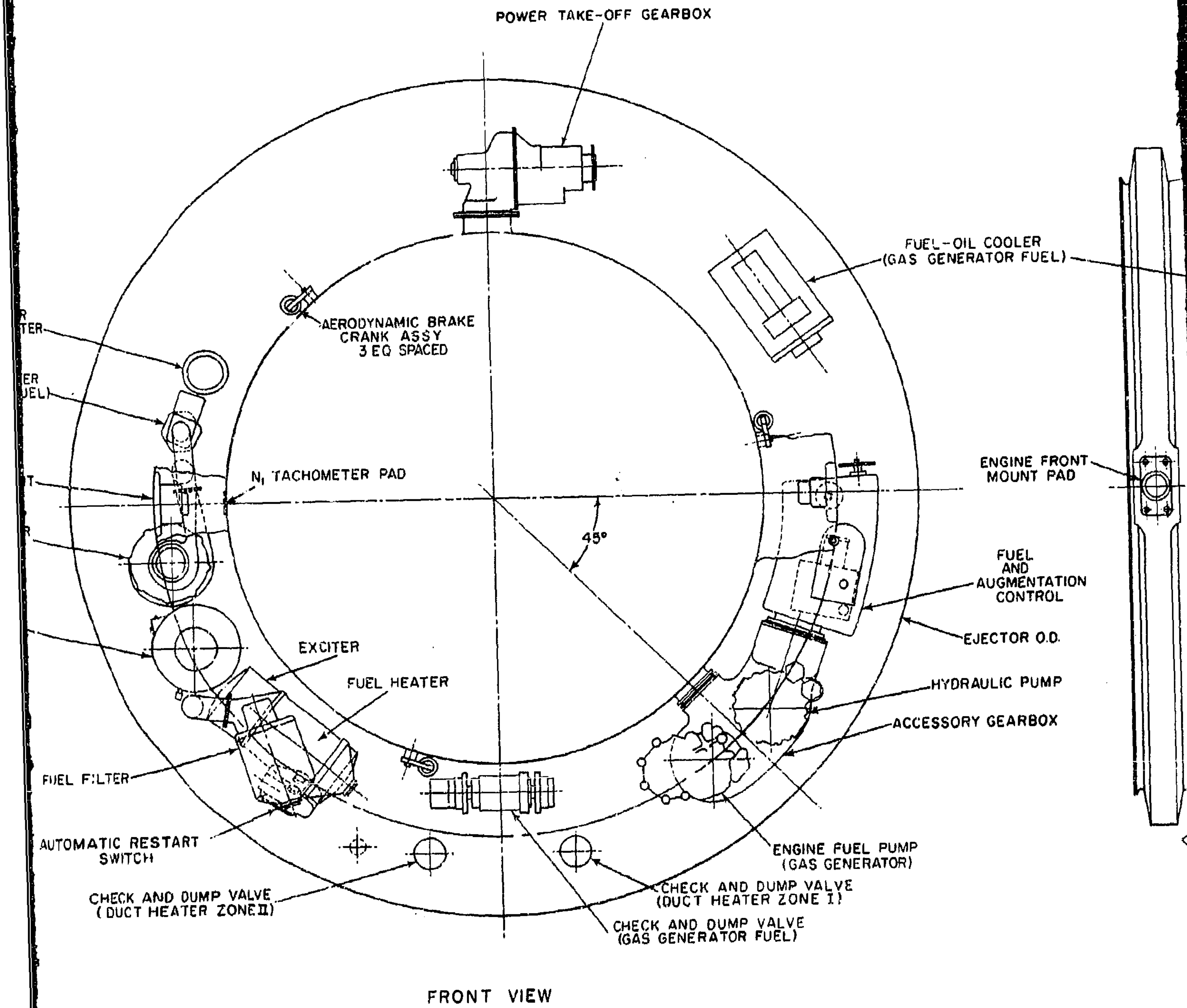


REVISED ACCESSORY ARRANGEMENT SHOWING FUEL COMPONENTS REMOVED FROM BOTTOM OF ENGINE

Figure 1-28

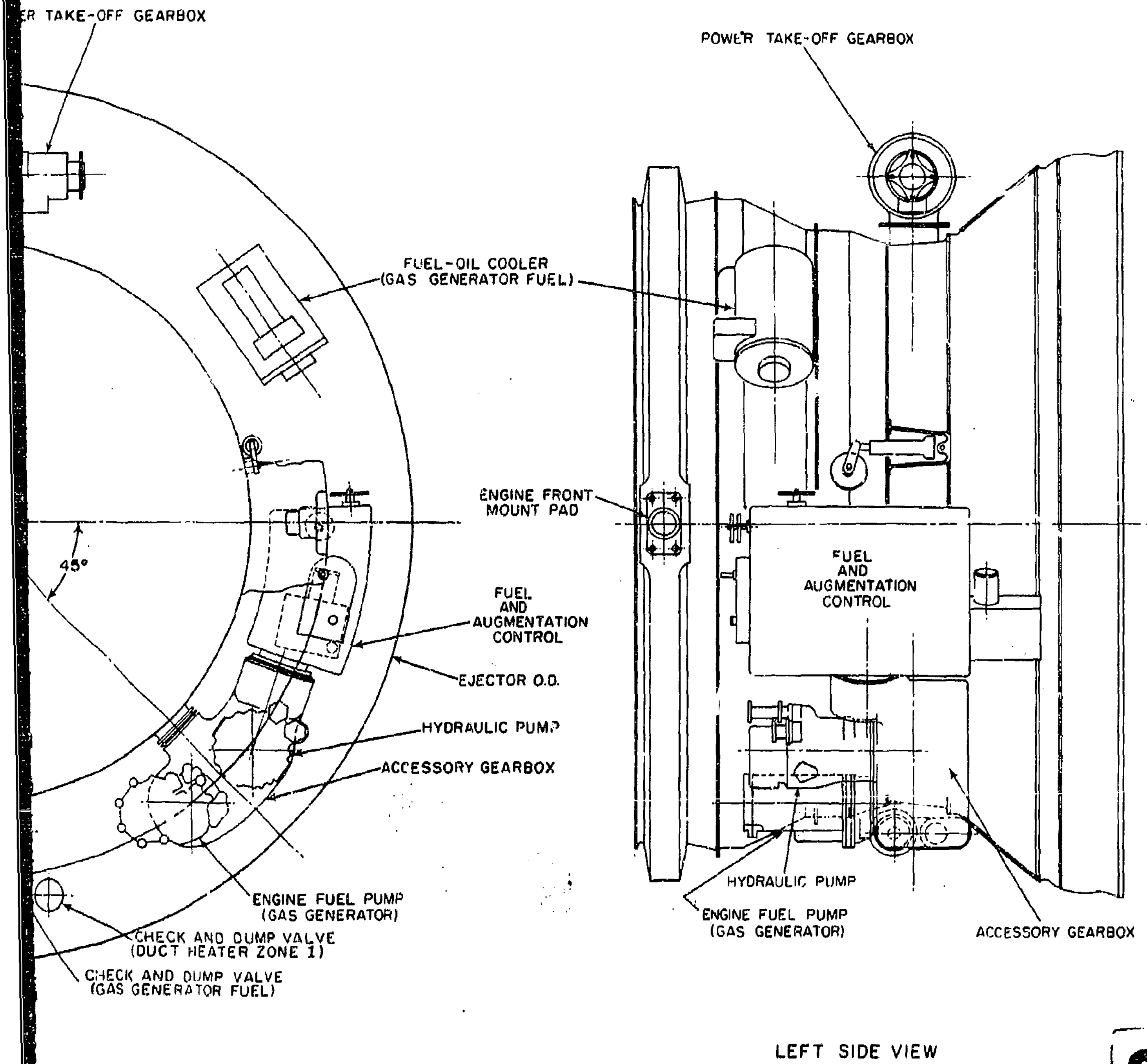


RIGHT SIDE VIEW



REVISED ACCESSORY ARRANGEM
MOUNT STRUCTURE AT

Figure 1-2



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

Figure 1-29

ACCESSORY DRIVE PADS

[illegible]	G	[illegible]
[illegible]	H	TACHOMETER MTG PAD (N2) (AND 20005 XVDI)
[illegible]	[illegible]	TACHOMETER MTG PAD (N1) (AND 20005 XVBI)

FUEL DRAIN

[illegible]	FD 1	COMBUSTION CHAMBER FUEL DRAIN
[illegible]	FD 2	DUMP VALVE DRAIN (GAS GEN)
[illegible]	FD 4	DUMP VALVE DRAIN (DUCT HTR) ZONE I
[illegible]	FD 4A	DUMP VALVE DRAIN (DUCT HTR) ZONE II

FUEL PRESSURE

[illegible]	FP 1	FUEL PUMP INLET PRESSURE
[illegible]	FP 3	CHECK VALVE FUEL PRESSURE

FUEL FLOW

[illegible]	F 1	FUEL PUMP SUPPLY INLET
[illegible]	F 2	MAIN FUEL FLOWMETER SUPPLY INLET
[illegible]	F 3	MAIN FUEL FLOWMETER SUPPLY OUTLET
[illegible]	F 4	DUCT HEATER FUEL PUMP INLET
[illegible]	F [illegible]	DUCT HEATER FLOWMETER SUPPLY INLET
[illegible]	F [illegible]	DUCT HEATER FLOWMETER SUPPLY OUTLET

FUEL VENT

[illegible]	[illegible]	FUEL PUMP OUTLET VENT

FUEL TEMPERATURE

[illegible]	[illegible]	HEATER OUTLET FUEL TEMP

OIL BREATHER

[illegible]	[illegible]	MAIN OIL OVERBOARD BREATHER

OIL DRAIN

[illegible]	LD 1	OIL TANK DRAIN
[illegible]	LD 2	OIL TANK OVERFLOW DRAIN
[illegible]	LD 3	[illegible] OVERFLOW DRAIN
[illegible]	LD 4	GEARBOX MAIN OIL DRAIN
[illegible]	LD 7	OIL STRAINER DRAIN

OIL FLOW

[illegible]	L1	OIL TANK REMOTE FILLER
[illegible]	L2	OIL TANK MANUAL FILL

OIL PRESSURE

[illegible]	LP 1	PRESSURE FOR TRANSMITTER
[illegible]	LP 4	OIL FILTER INLET PRESSURE } PROV FOR ΔP
[illegible]	LP 5	OIL FILTER OUTLET PRESSURE } PROV FOR ΔP

OIL TEMPERATURE

[illegible]	LT 1	MAIN OIL TEMP

OIL VENT

[illegible]	LV 3	OIL PRESSURE TRANSMITTER VENT

SEAL DRAIN

[illegible]	SD1	FUEL CONTROL SEAL DRAIN
[illegible]	SD2	FUEL PUMP SEAL DRAIN
[illegible]	SD9	HYDRAULIC PUMP SEAL DRAIN
[illegible]	SD14	DUCT HEATER FUEL PUMP SEAL DRAIN
[illegible]	SD3	ACCESSORY DRIVE OVERBOARD DRAIN (TACHOMETER)
[illegible]	SD4	ACCESSORY DRIVE OVERBOARD DRAIN (HYD PUMP)
[illegible]	SD5	ACCESSORY DRIVE OVERBOARD DRAIN (FUEL PUMP)

TEMPERATURE SENSING

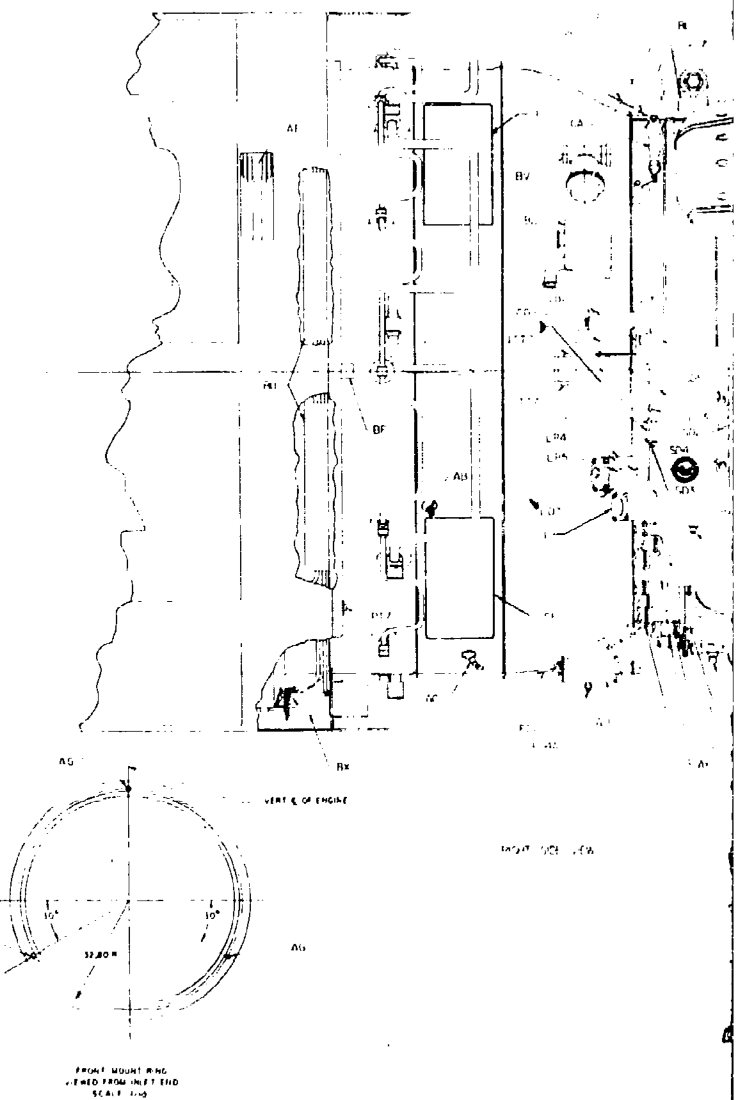
[illegible]	TT 7	TURBINE EXIT TEMP (AVG)
[illegible]	TT 7	TURBINE EXIT TEMP (INDIVIDUAL)

PRESSURE SENSING

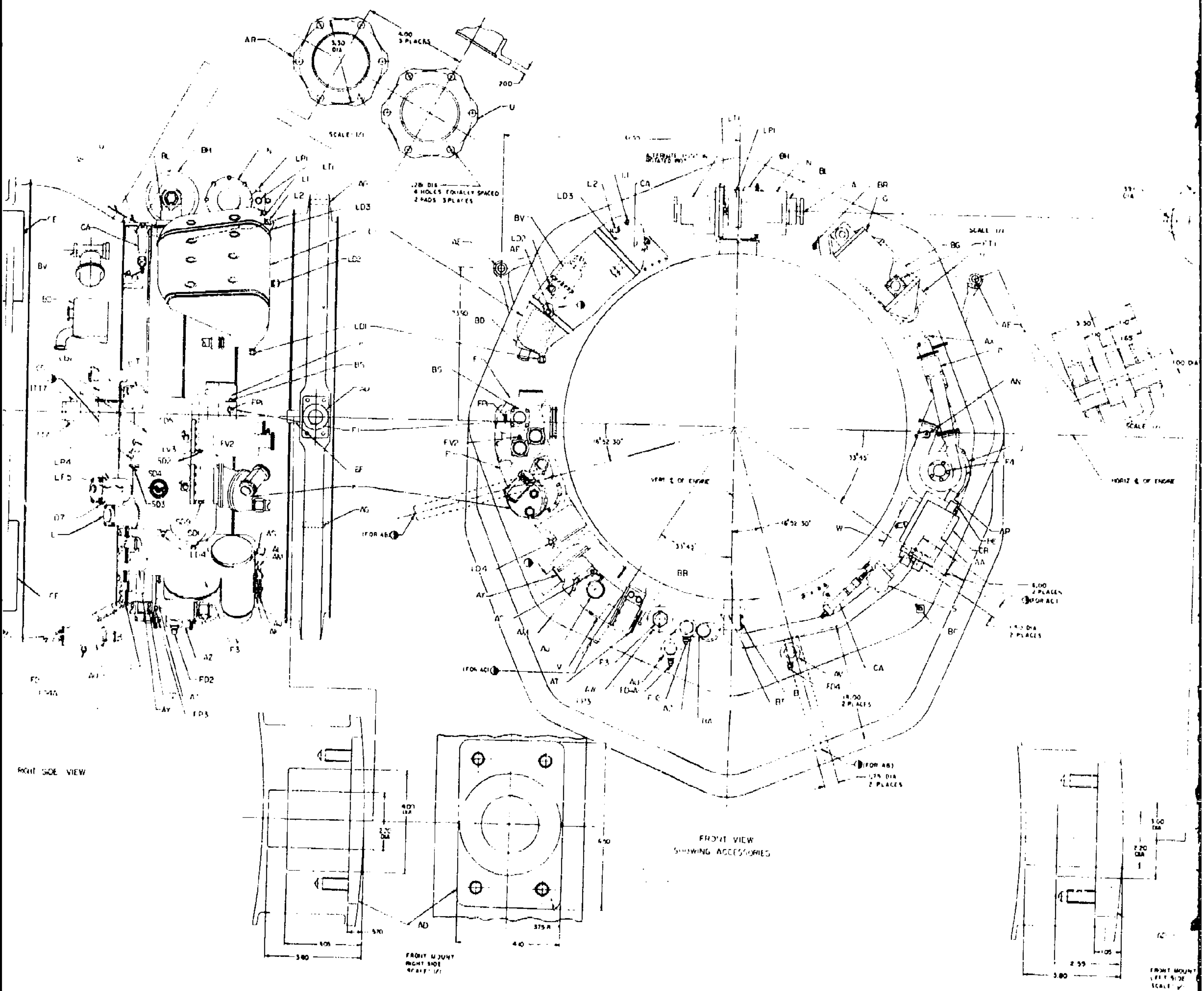
[illegible]	PT 7	TURBINE EXIT PRESSURE

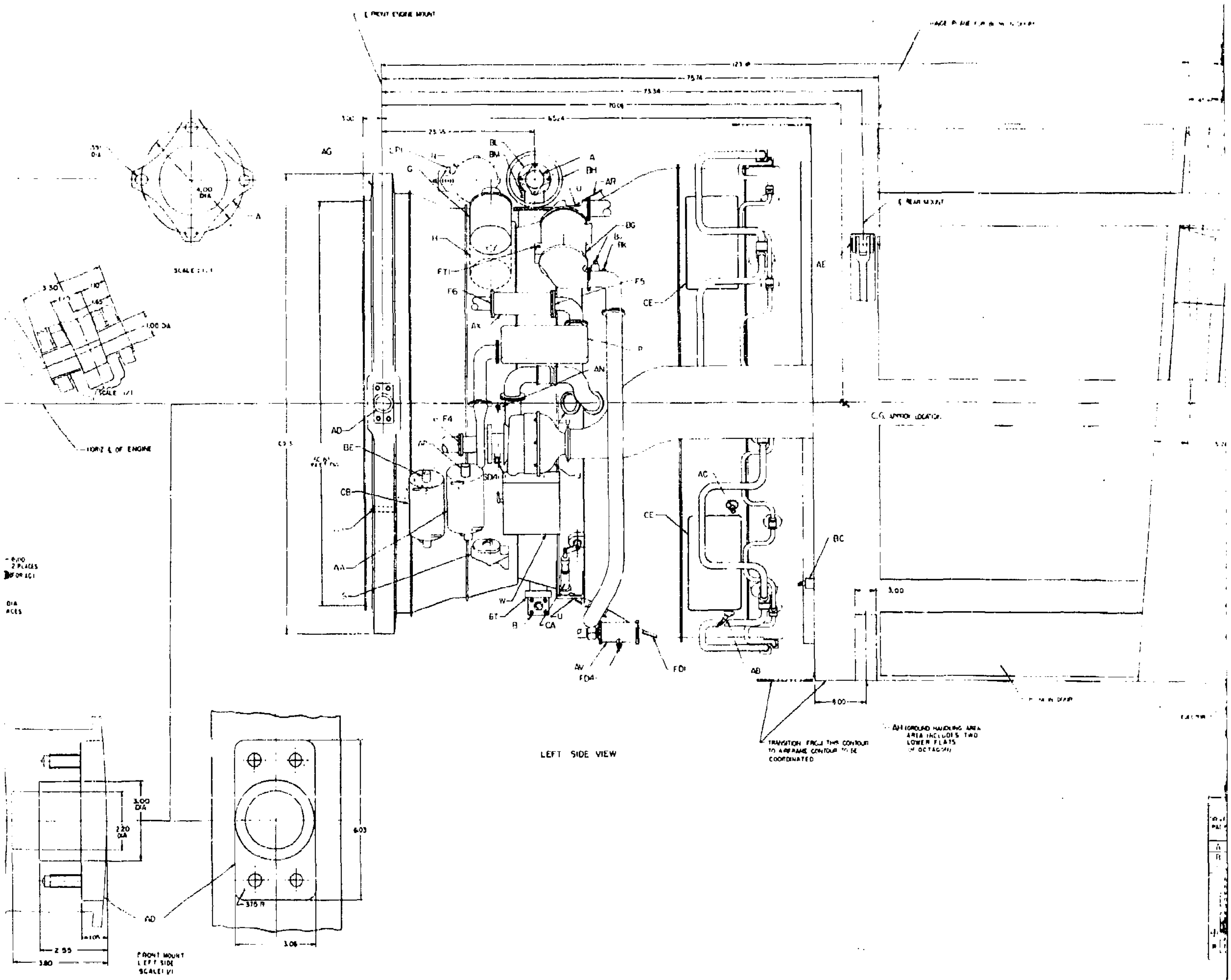
MISCELLANEOUS

[illegible]	E	OIL TANK
[illegible]	F	GAS GEN FUEL PUMP
[illegible]	G	FUEL PUMP FILTER (GAS GEN)
[illegible]	H	FUEL PUMP FILTER (MIN SPACE FOR REMOVAL)
[illegible]	J	DUCT HEATER FUEL PUMP
[illegible]	K	HYDRAULIC PUMP
[illegible]	L	OIL PUMP
[illegible]	M	OIL FILTER
[illegible]	N	FUEL OIL COOLER (GAS GEN)
[illegible]	P	FUEL OIL COOLER (DUCT HEATER)
[illegible]	R	GEARBOX
[illegible]	S	AUTOMATIC RESTART SWITCH
[illegible]	T	BREATHER PRESSURIZING VALVE
[illegible]	U	HIGH PRESSURE BLEED PAD
[illegible]	V	UNITIZED CONTROL (FUEL, AUGMENTATION, & NOZZLE)
[illegible]	W	DUCT TURBOPUMP CONTROLLER
[illegible]	AA	EXCITER (GAS GEN IGNITION)
[illegible]	AB	IGNITER PLUG (GAS GEN)
[illegible]	AC	IGNITER PLUG (DUCT HEATER)
[illegible]	AD	ENGINE FRONT MOUNTING PROVISIONS
[illegible]	AE	ENGINE REAR MOUNTING PROVISIONS
[illegible]	AF	FUEL RETURN TO SUPPLEMENTARY COOLING
[illegible]	AG	GROUND HANDLING HOLES (FRONT MOUNT)
[illegible]	AH	GROUND HANDLING AREA (REAR)
[illegible]	AJ	POWER CONTROL SHAFT (120° ANGLE OF TRAVEL)
[illegible]	AK	SHUT OFF SHEAVE (90° ANGLE OF TRAVEL)
[illegible]	AL	APPROACH VELOCITY CONTROL LEVER (60° ANGLE OF TRAVEL)
[illegible]	AM	MACH NO OR SHOCK POSITION RESET LEVER (60° ANGLE OF TRAVEL)
[illegible]	AN	AIR TURBINE VALVE DUCT HEATER FUEL PUMP
[illegible]	AP	IGNITION EXCITER ELECTRICAL CONN GAS GEN
[illegible]	AR	THERMAL ANTI ICING PAD
[illegible]	AS	AERODYNAMIC BRAKE CONTROL AIR SUPPLY CONN
[illegible]	AT	DUCT NOZZLE FEEDBACK
[illegible]	AU	CHECK & DUMP VALVE (DUCT HTR ZONE I)
[illegible]	AV	CHECK & DUMP VALVE (DUCT HTR ZONE II)
[illegible]	AW	GAS GEN FUEL FLOWMETER MTG PROVISIONS
[illegible]	AX	DUCT HEATER FUEL FLOWMETER MTG PROVISIONS
[illegible]	AY	FUEL CONTROL FUEL FILTER
[illegible]	AZ	CHECK & DUMP AND LOW INLET VALVE (GAS GEN FUEL)
[illegible]	BA	CHECK & DUMP VALVE FILTER
[illegible]	BB	NOZZLE POSITION INDICATOR MOUNTING PROVISIONS
[illegible]	BC	REVERSER POSITION INDICATOR CONN (ELECTRICAL)
[illegible]	BD	THERMOSTATIC FUEL BYPASS VALVE
[illegible]	BE	IGNITION EXCITER ELECTRICAL CONN (DUCT HEATER)
[illegible]	BF	VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFR)
[illegible]	BG	FUEL HEATER
[illegible]	BH	POWER TAKEOFF GEARBOX
[illegible]	BJ	FUEL HEATER VALVE CONN (ELECTRICAL)
[illegible]	BK	FUEL HEATER VALVE POSITION INDICATOR (ELECTRICAL)
[illegible]	BL	POWER TAKEOFF DECOUPLER
[illegible]	BM	POWER TAKEOFF DECOUPLER ACTIVATING SHAFT
[illegible]	BR	FUEL FILTER PRESSURE CONN (ELECTRICAL)
[illegible]	BS	FUEL INLET TEMPERATURE SENSOR
[illegible]	BT	OIL SCAVENGE PUMP - BEARING COMPARTMENT
[illegible]	BU	BACK PRESSURE FLAPPER DOORS (4 REQD)
[illegible]	BV	HYDRAULIC RETURN FILTER
[illegible]	BX	DUCT NOZZLE ACTUATORS (4 REQD)
[illegible]	CA	AERODYNAMIC BRAKE ACTUATORS
[illegible]	CB	EXCITER (DUCT HEATER IGNITION)
[illegible]	CC	THRUST REVERSING ACTUATORS (2 REQD)
[illegible]	CD	HYDRAULIC DISCHARGE FILTER
[illegible]	CF	GAS GENERATOR ACCESS DOORS

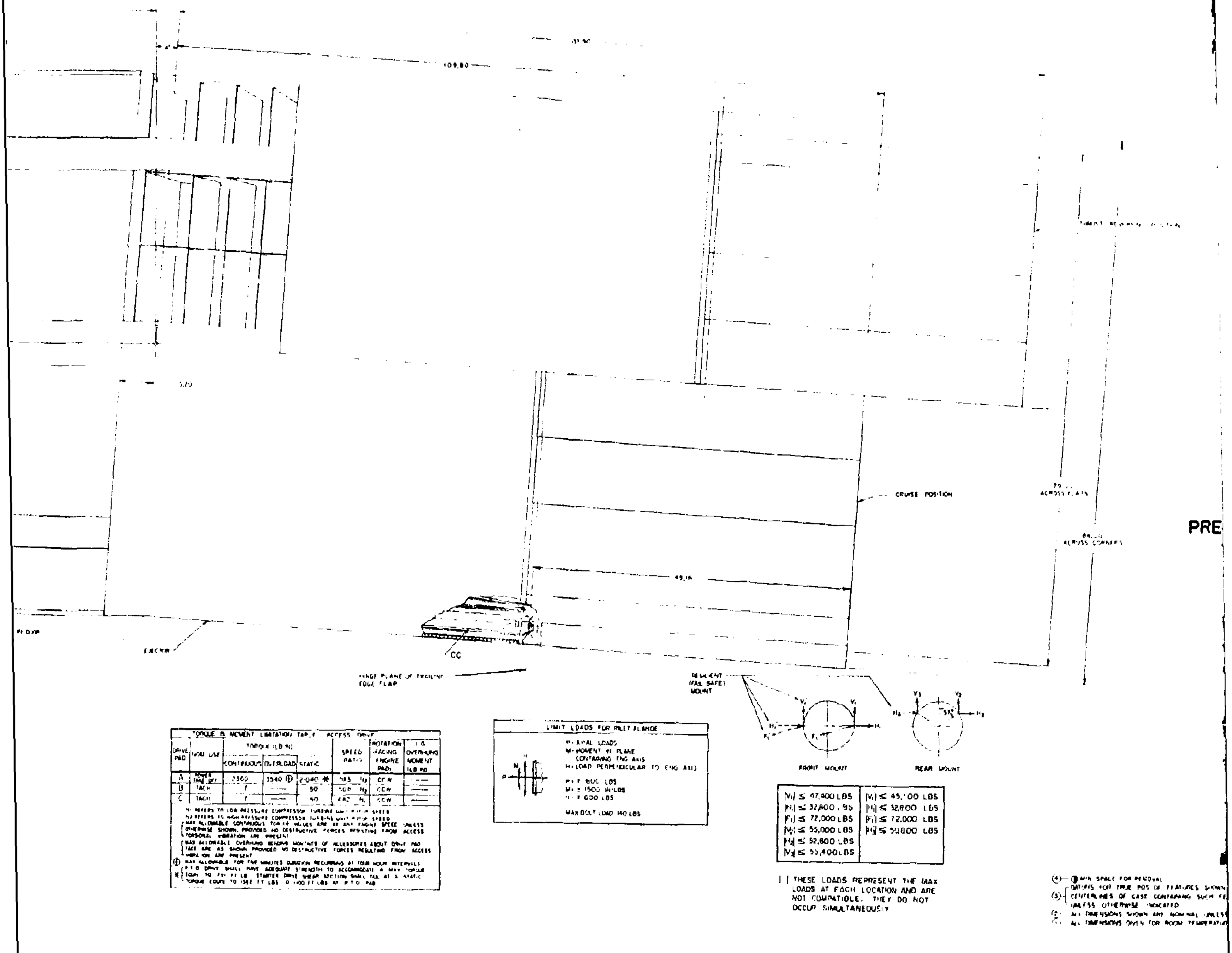


1


RIGHT SIDE VIEW
FRONT VIEW
SHOWING ACCESSORIES
VERT ℄ OF ENGINE
HORIZ ℄ OF ENGINE
FRONT MOUNT
RIGHT SIDE
SCALE 1/1


LEFT SIDE VIEW
C.G. APPROX LOCATION
HORIZ ℄ OF ENGINE
FRONT MOUNT
LEFT SIDE
SCALE 1/1
TRANSITION FROM THIS CONTOUR
TO AIRFRAME CONTOUR TO BE
COORDINATED
AH GROUND HANDLING AREA
AREA INCLUDES TWO
LOWER FLATS
OF OCTAGON
3.00
DIA
2.20
DIA
6.03
3.06
2.95
3.80
4.00
DIA
1.00 DIA
3.00
6.00

3



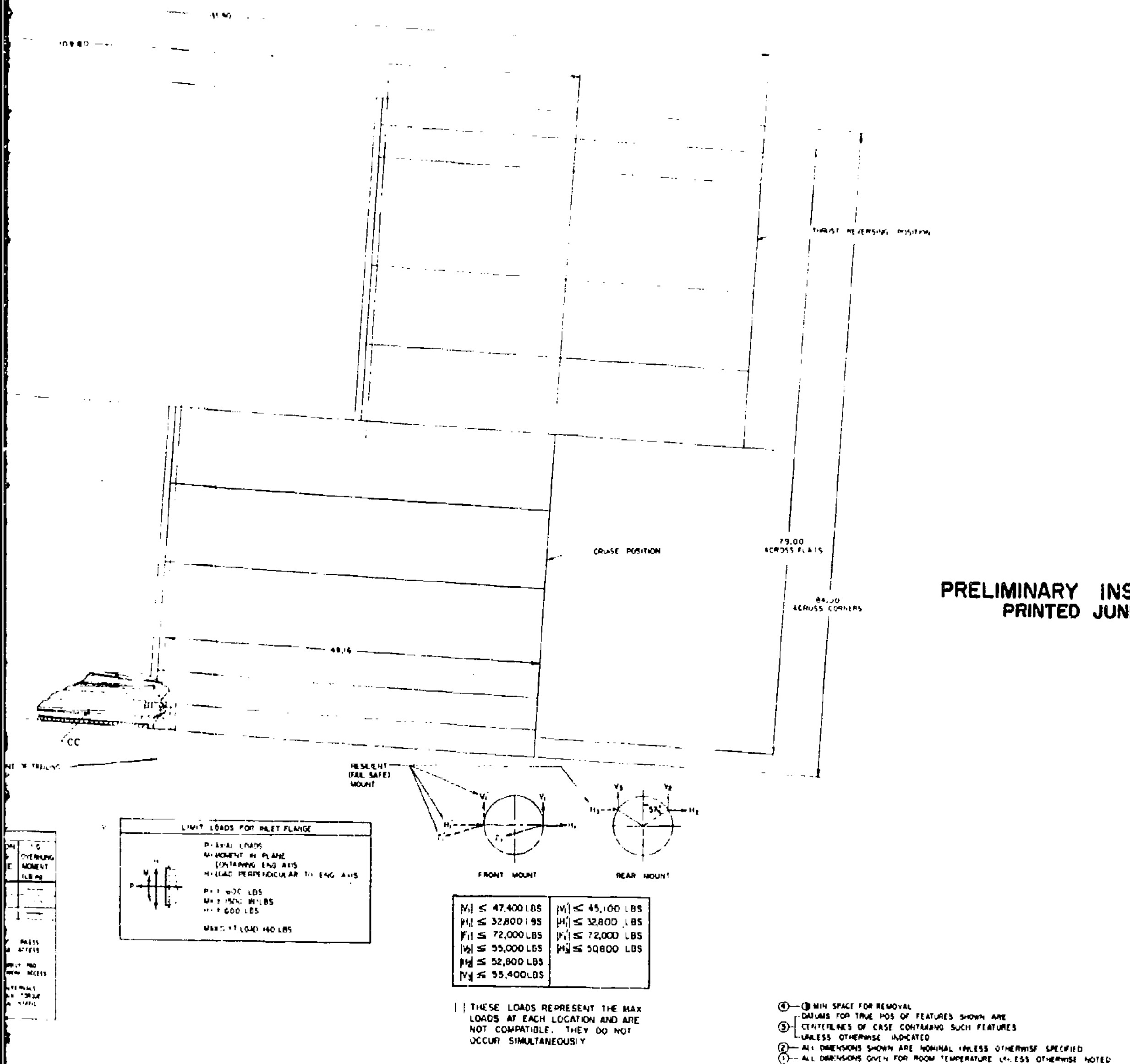
TORQUE & MOMENT LIMITATION TABLE ACCESS DRIVE

DRIVE PAD	USE	TORQUE (LB IN) CONTINUOUS	OVERLOAD	STATIC	SPEED RATIO	ROTATION (FACING ENGINE PAD)	OVERHUNG MOMENT (LB IN)
A	POWER TAKE OFF	2300	3540 (Ⓣ)	2,040 *	[illegible]	CCW	—
B	TACH	7	—	50	500	CCW	—
C	TACH	7	—	50	[illegible]	CCW	—

FRONT MOUNT	REAR MOUNT
$\lvert V_1 \rvert \le$ 47,400 LBS	$\lvert V_1 \rvert \le$ 45,100 LBS
$\lvert H_1 \rvert \le$ 32,800 LBS	$\lvert H_1 \rvert \le$ 32,800 LBS
$\lvert F_1 \rvert \le$ 72,000 LBS	$\lvert F_1 \rvert \le$ 72,000 LBS
$\lvert V_2 \rvert \le$ 55,000 LBS	$\lvert H_3 \rvert \le$ 50,000 LBS
$\lvert H_2 \rvert \le$ 52,600 LBS	
$\lvert V_3 \rvert \le$ 55,400 LBS	

PROPOSED TURBOFAN ACCESSORY ARRANG

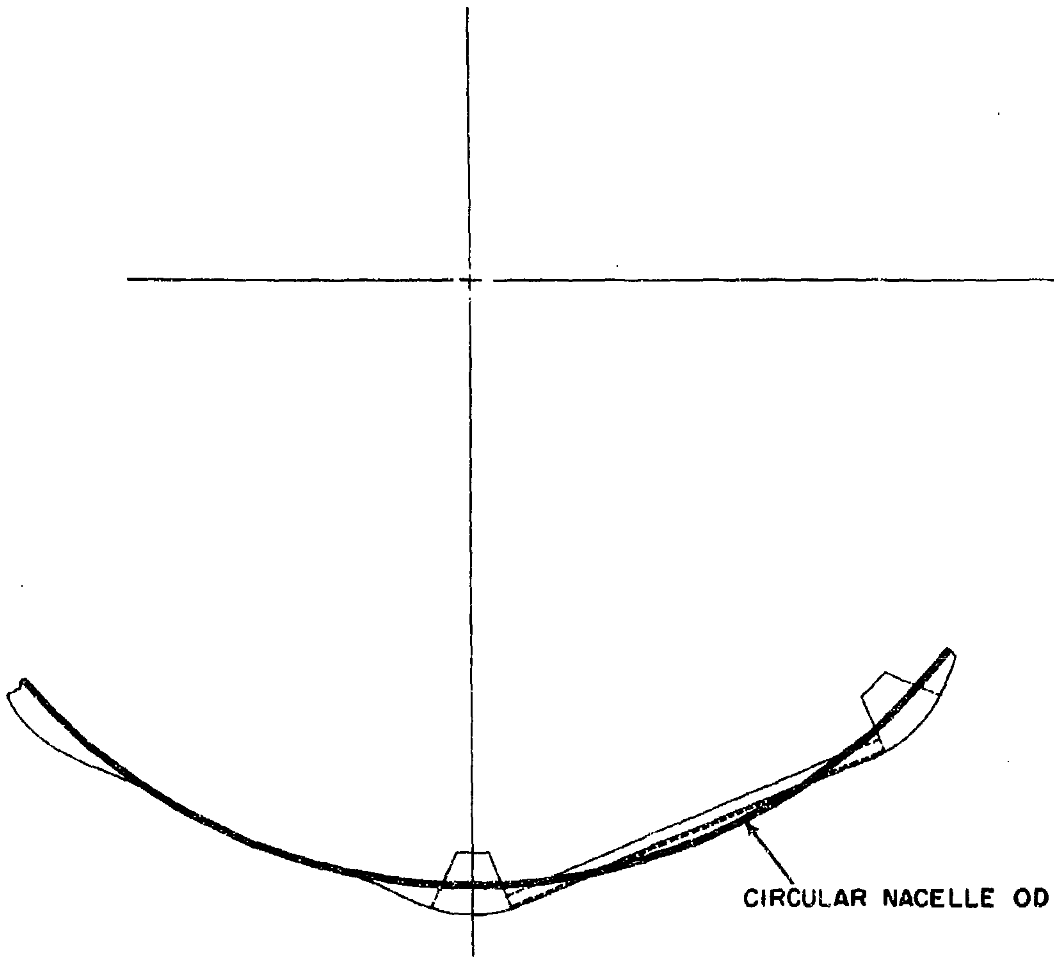
Figure 1-30

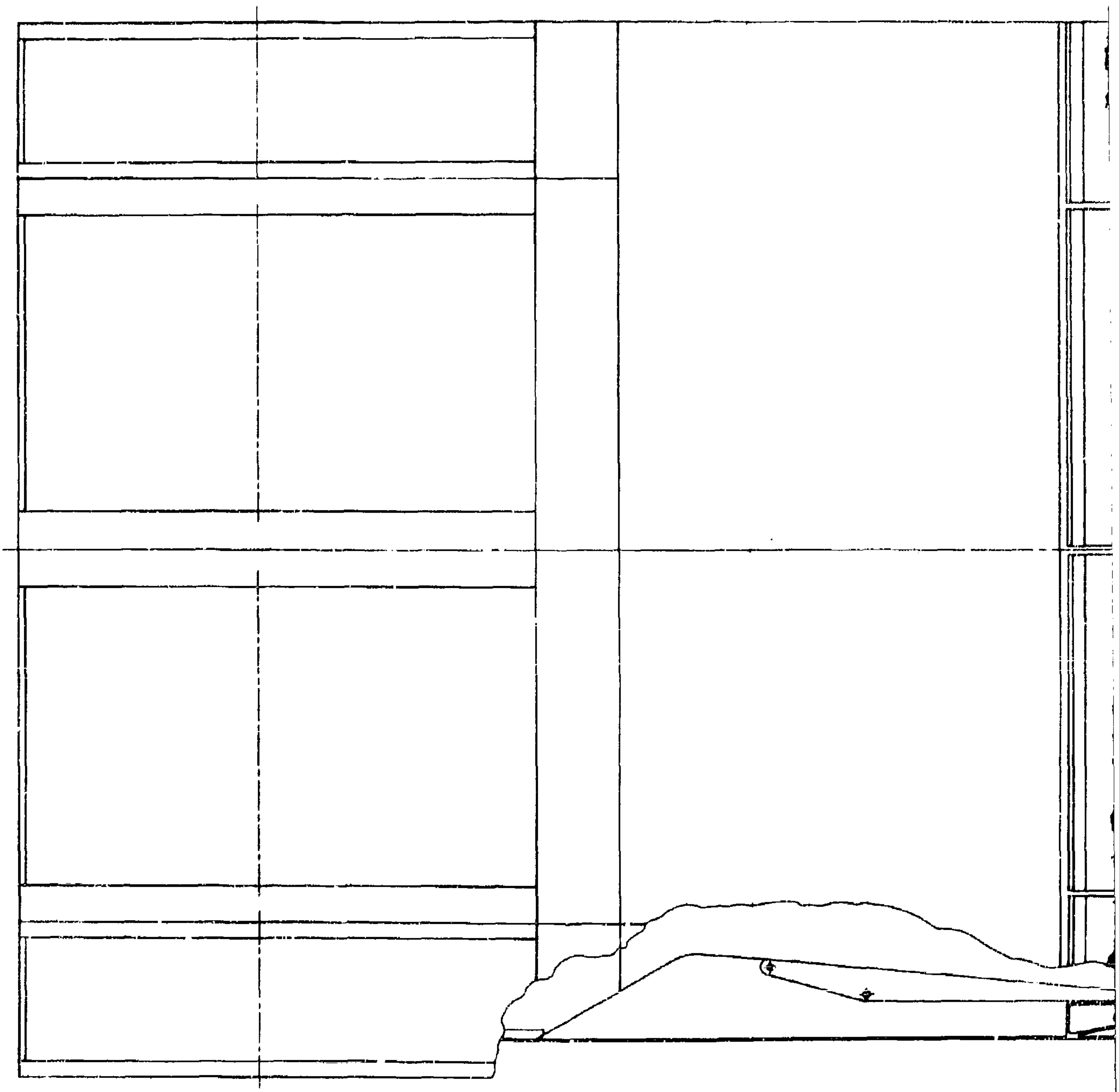


PRELIMINARY INSTALLATION DWG
PRINTED JUNE 23, 1965

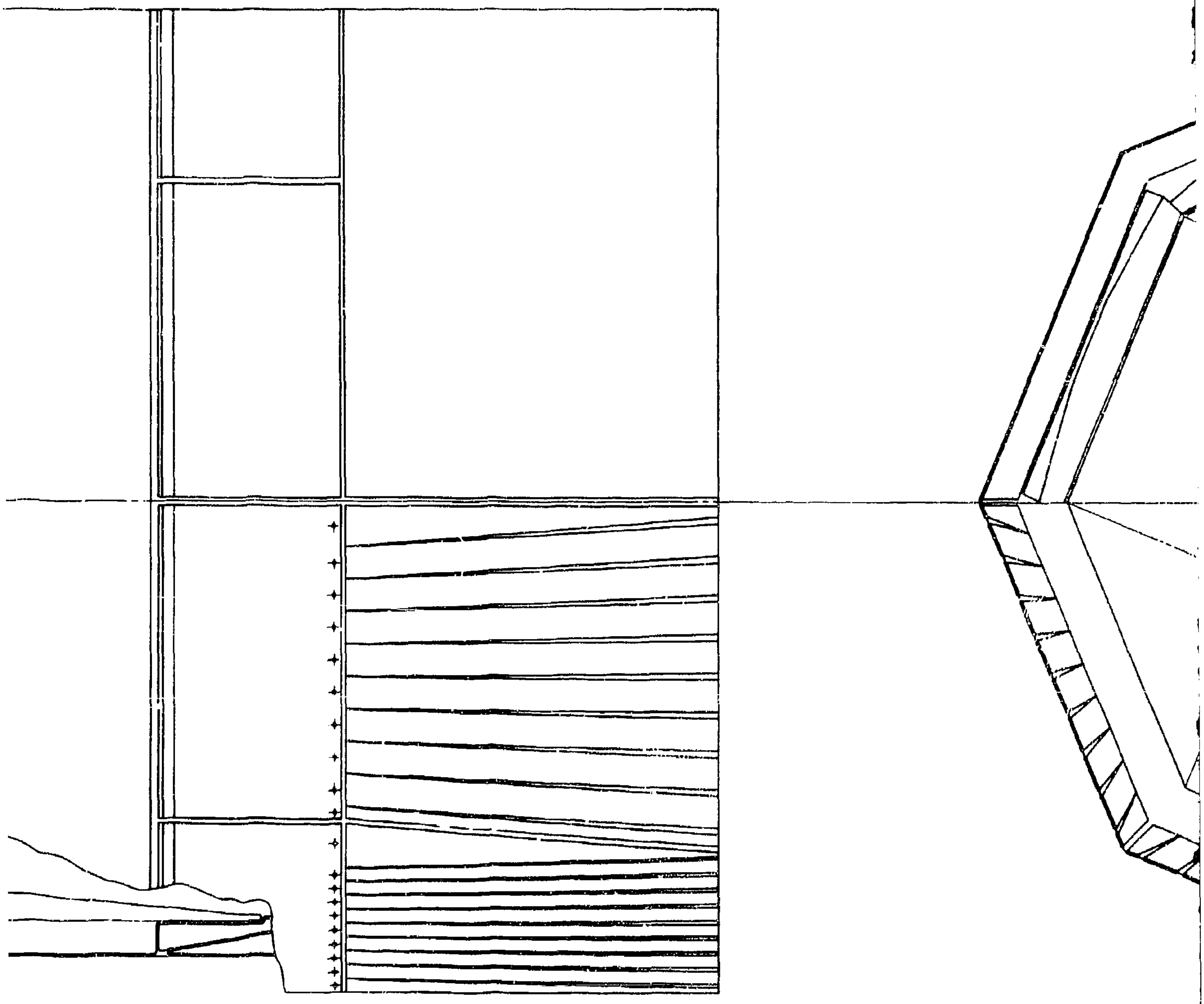
PROPOSED TURBOFAN ACCESSORY ARRANGEMENT

Figure 1-30

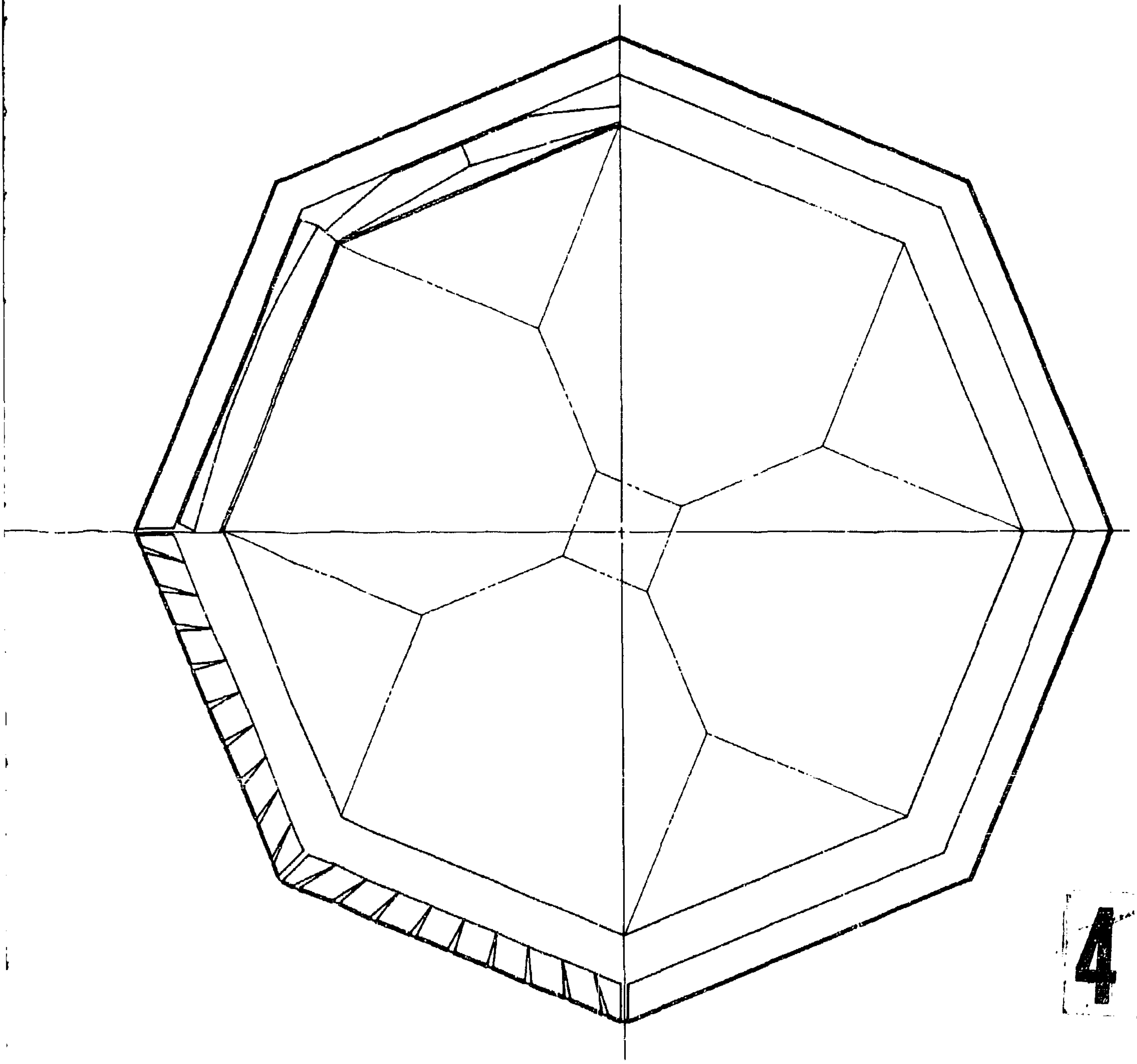

CIRCULAR NACELLE OD



2

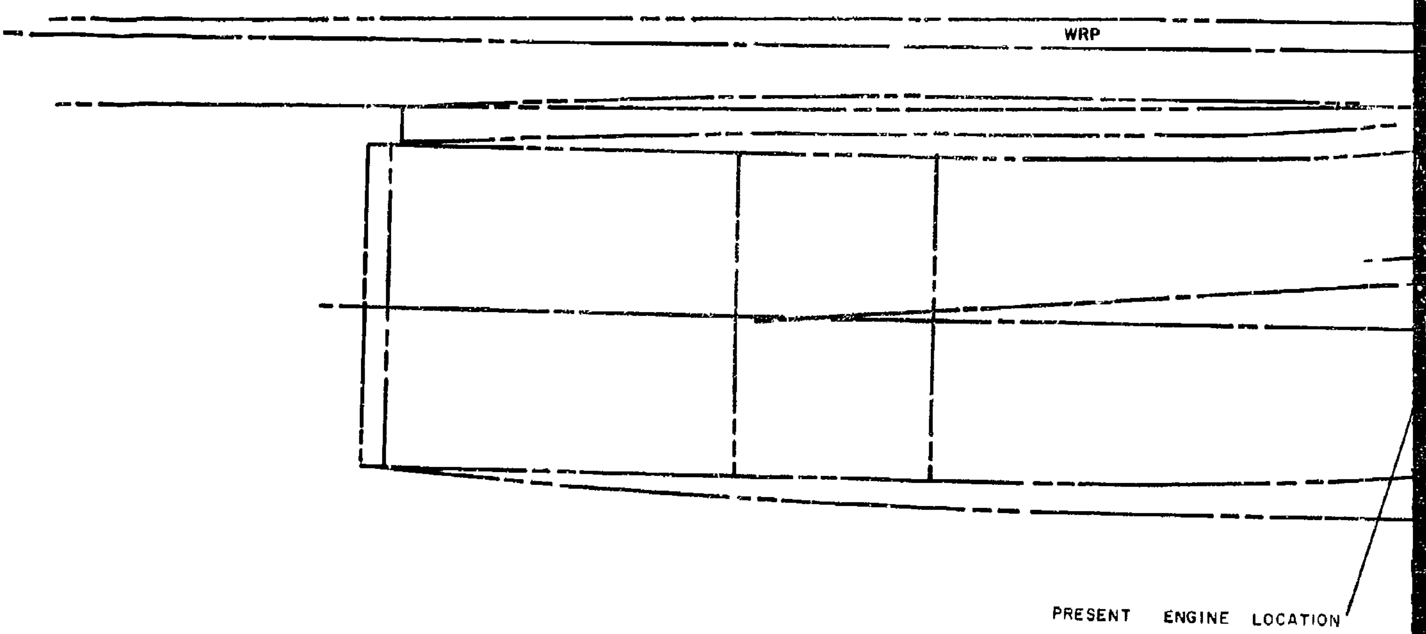


3

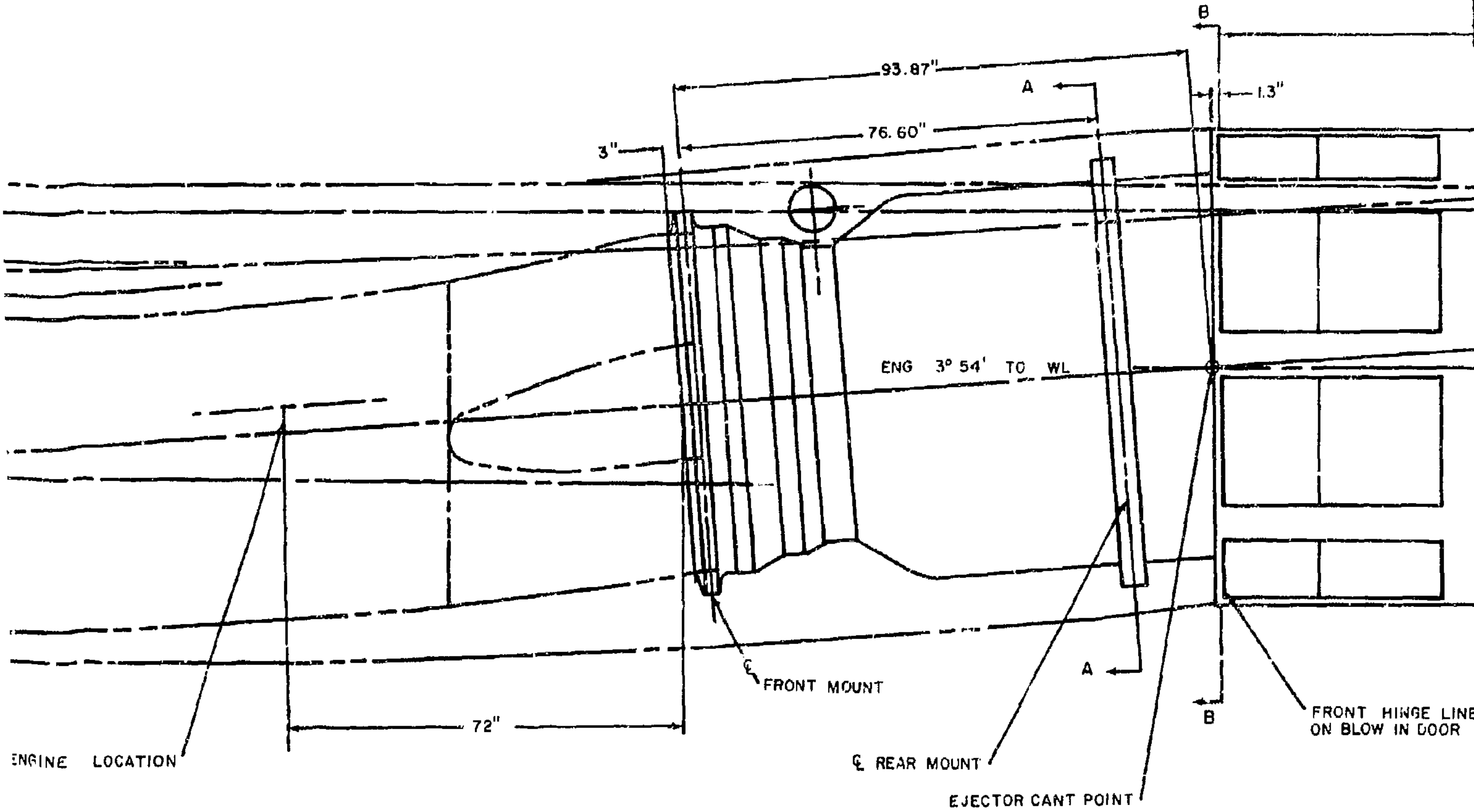


SCHEMATIC OF OCTAGONAL EJECTOR

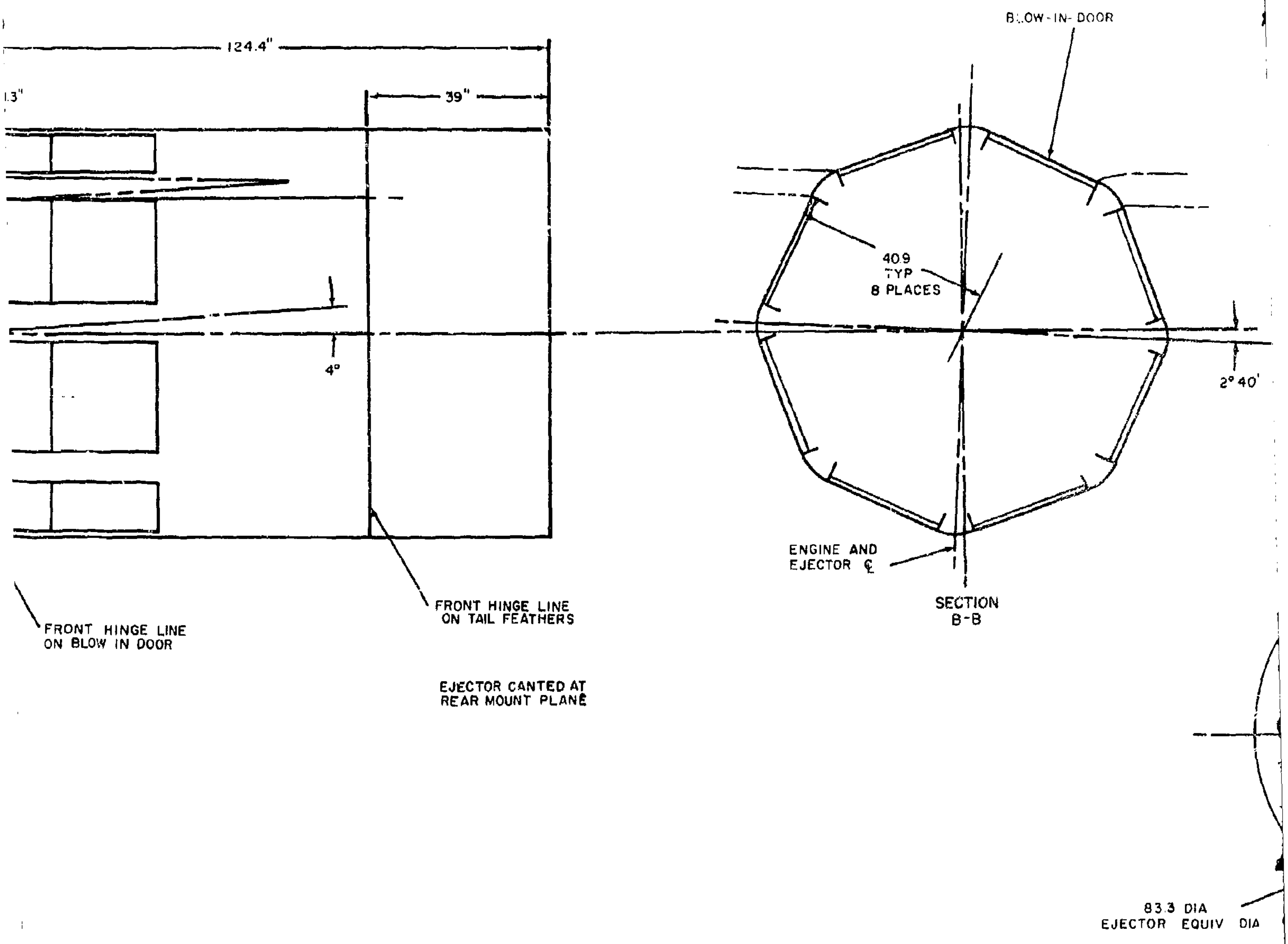
Figure 1-31



1

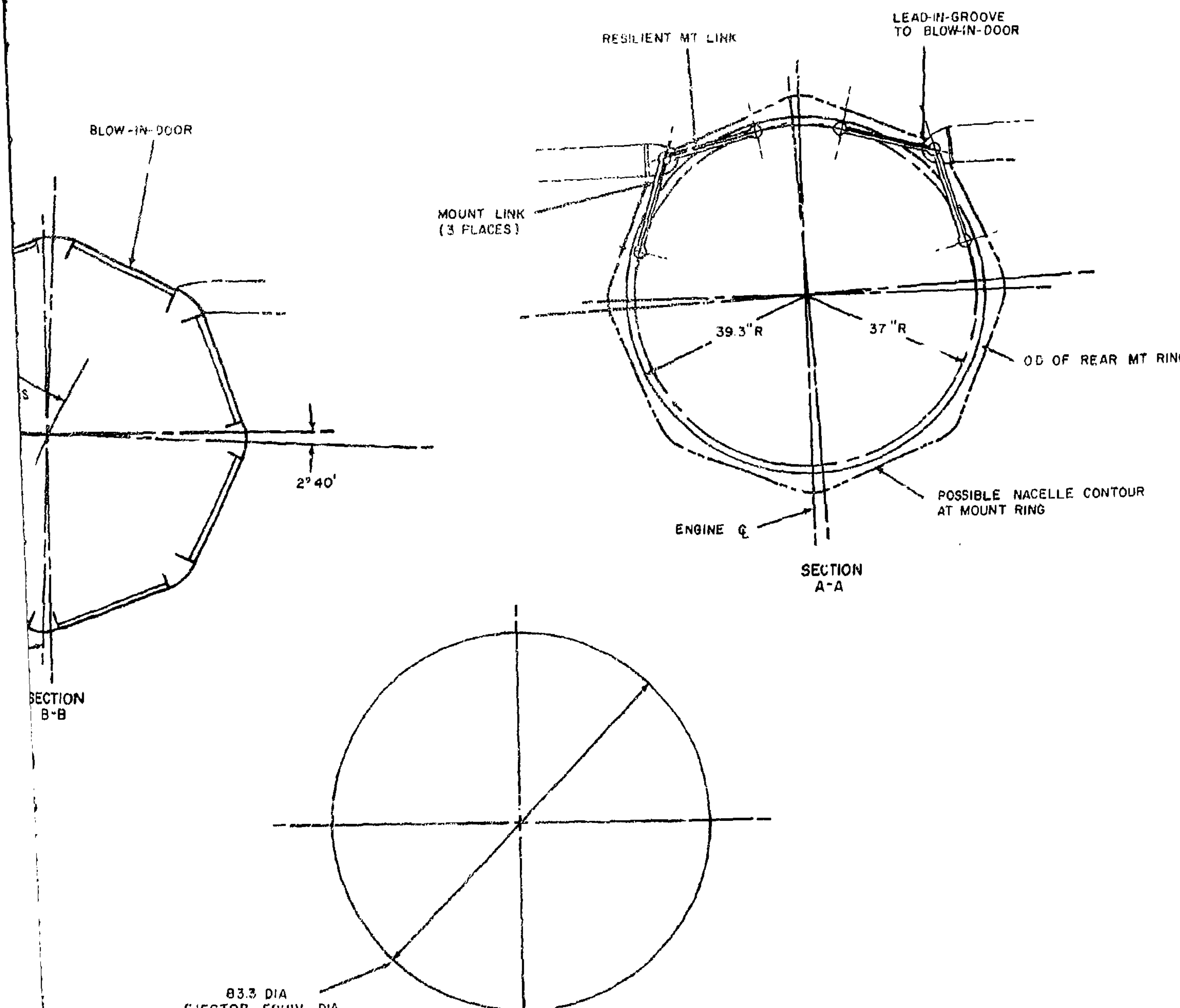


B
93.87"
A
1.3"
76.60"
3"
ENG 3° 54' TO WL
A
B
℄ FRONT MOUNT
72"
ENGINE LOCATION
℄ REAR MOUNT
FRONT HINGE LINE
ON BLOW IN DOOR
EJECTOR CANT POINT



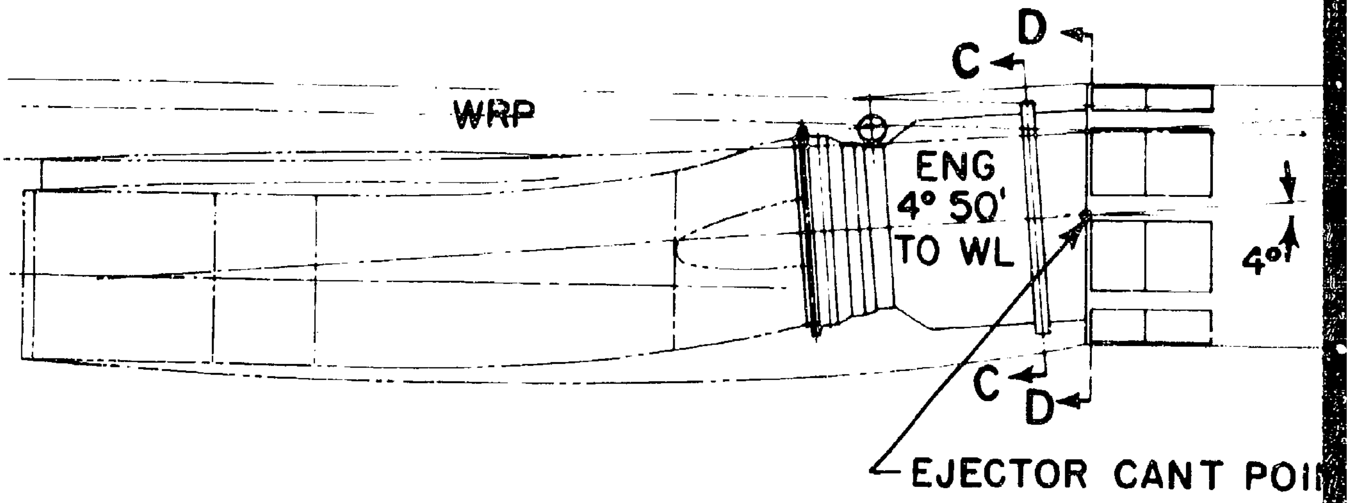
124.4"
1.3"
39"
4°
FRONT HINGE LINE ON BLOW IN DOOR
FRONT HINGE LINE ON TAIL FEATHERS
EJECTOR CANTED AT REAR MOUNT PLANE
BLOW-IN-DOOR
40.9 TYP 8 PLACES
2°40'
ENGINE AND EJECTOR ℄
SECTION B-B
83.3 DIA EJECTOR EQUIV DIA

3

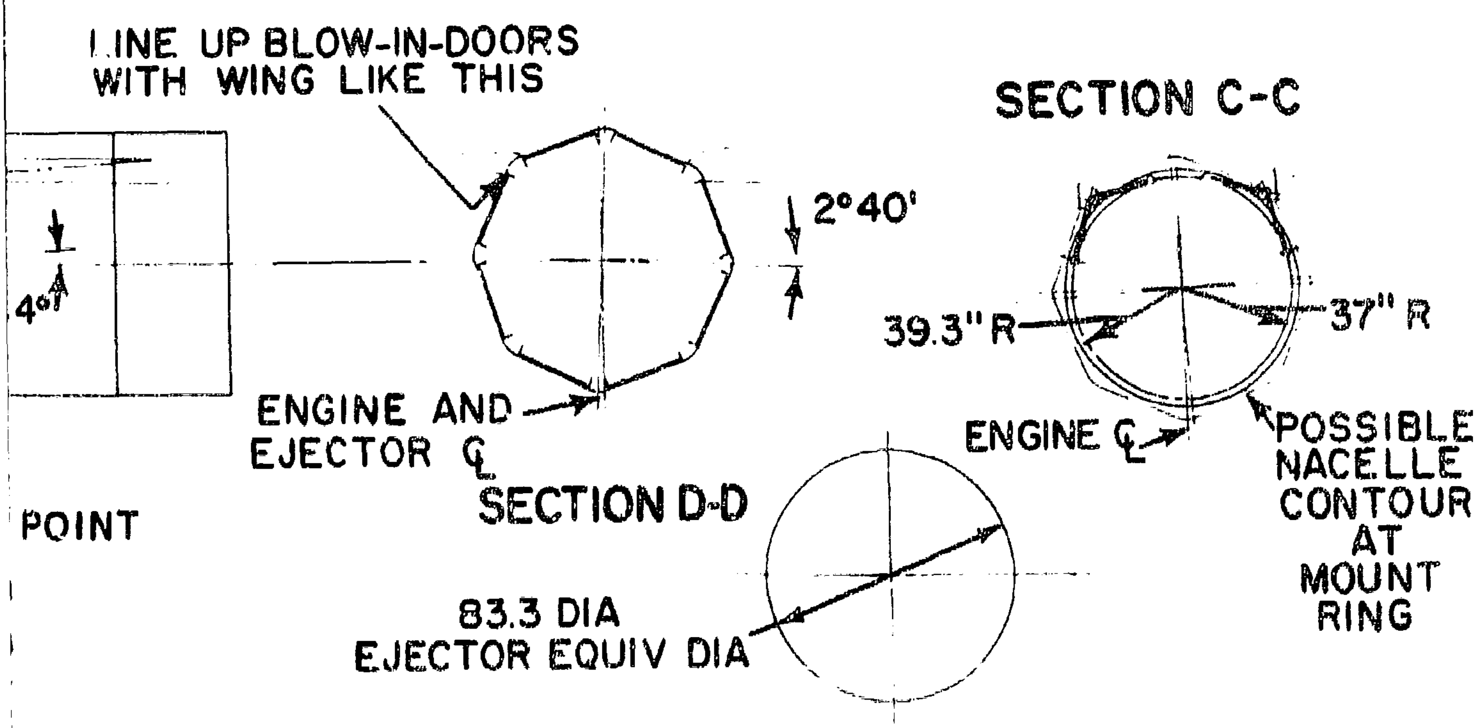


STF219L 700 LB/SEC TURBOFAN OUTBOARD ENGINE

Figure 1-32



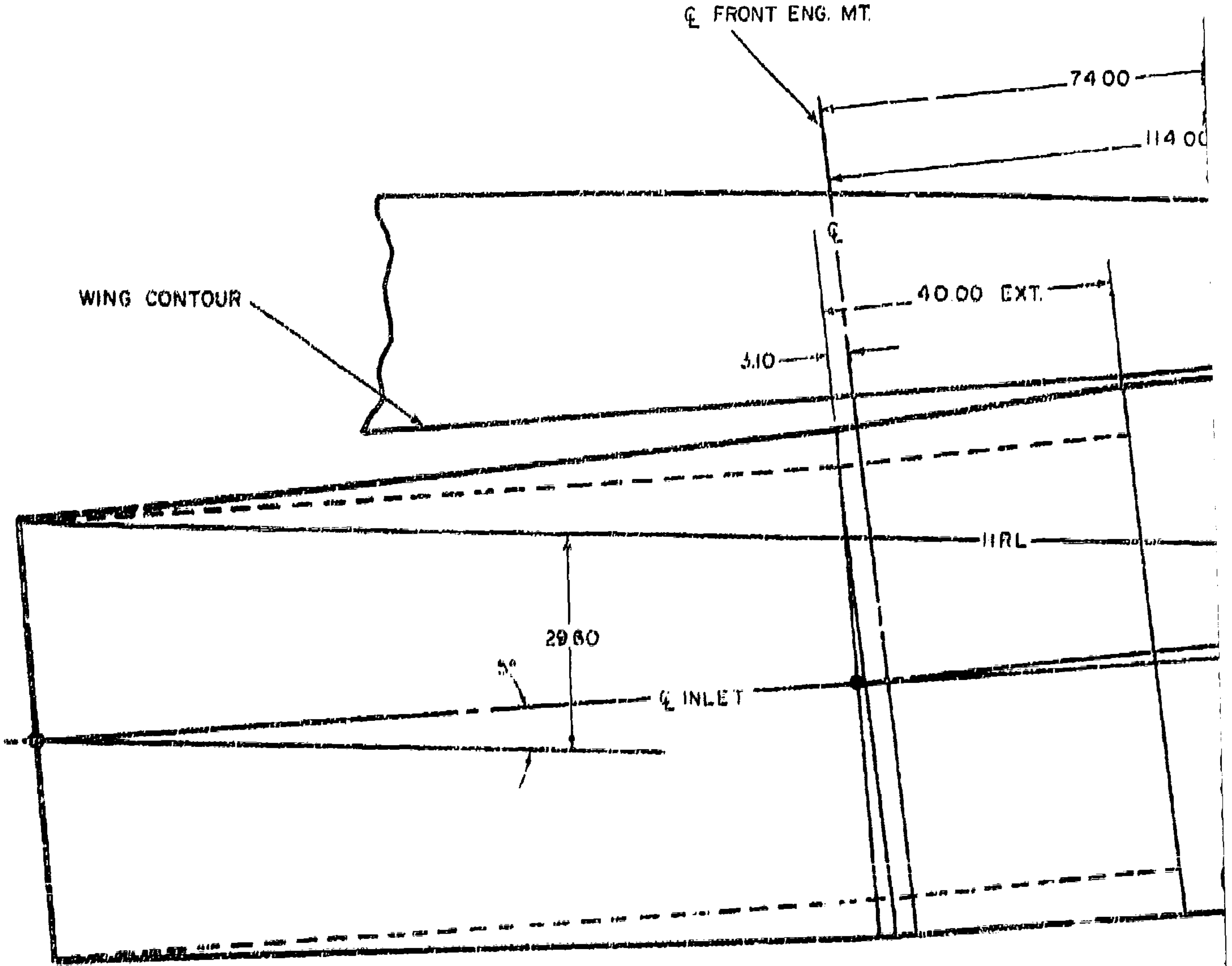
1



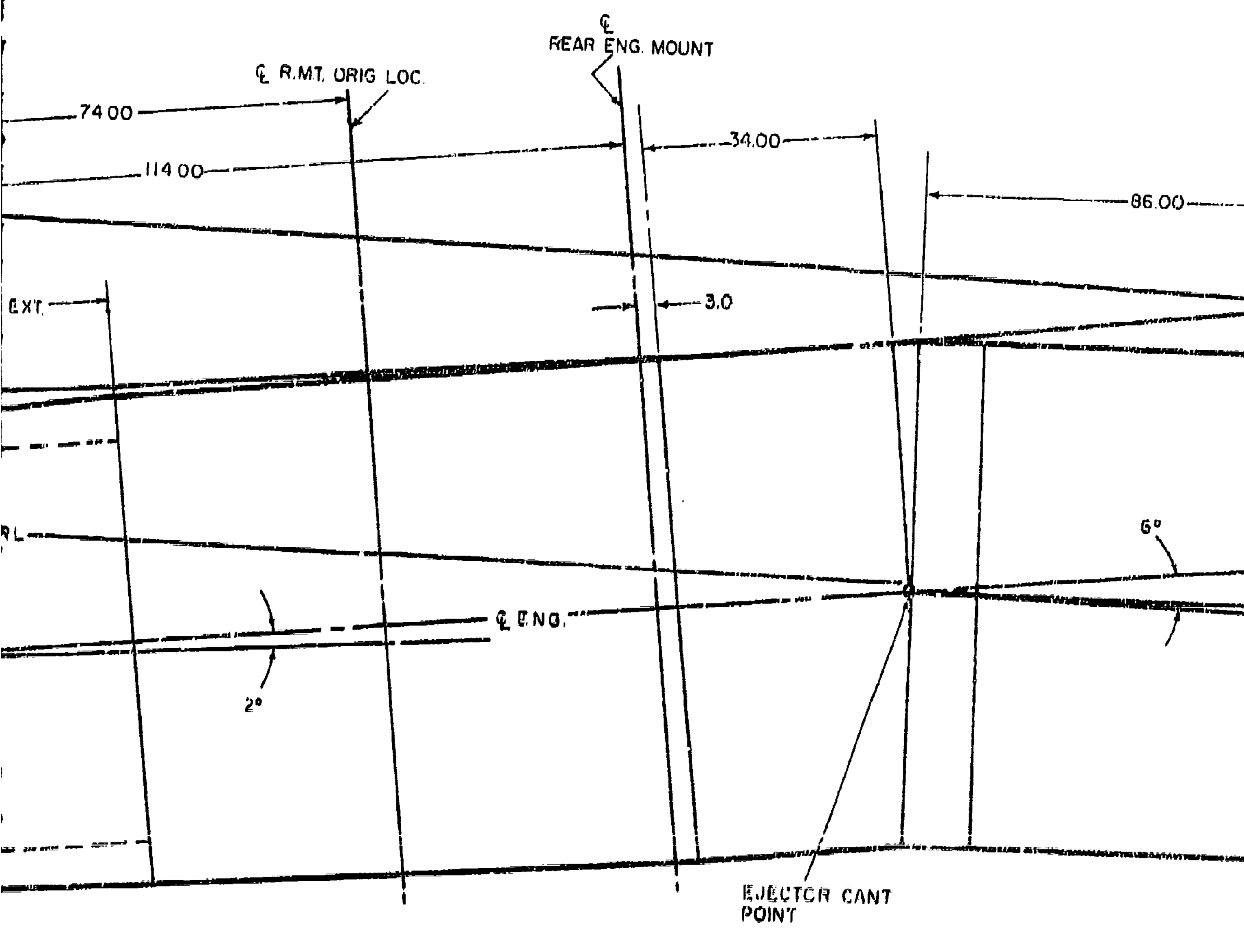
2

STF219L 700 LB/SEC TURBOFAN INBOARD ENGINE

Figure 1-33

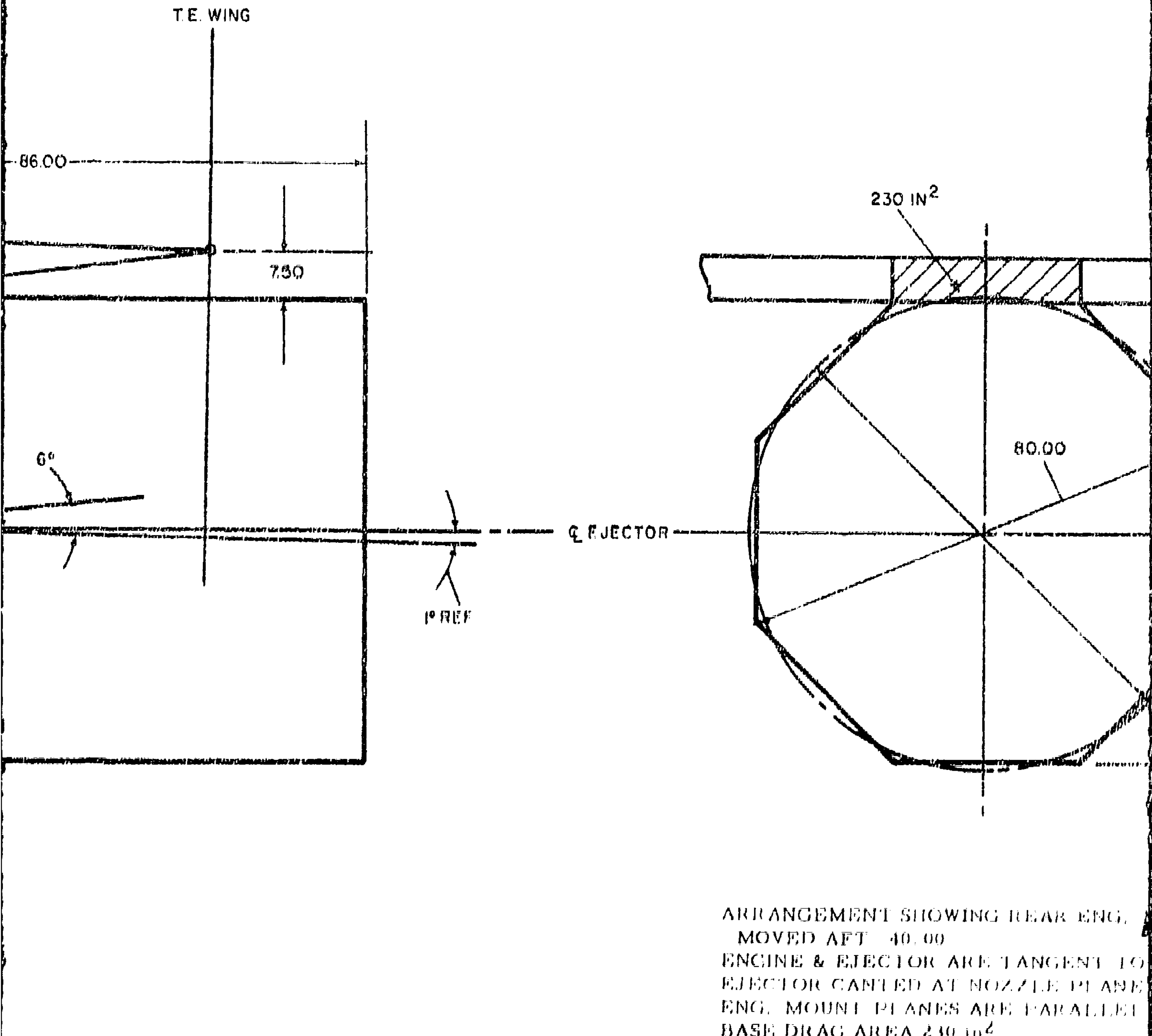


℄ FRONT ENG. MT.
74.00
114.00
℄
WING CONTOUR
40.00 EXT.
3.10
HRL
29.60
5°
℄ INLET



℄ REAR ENG. MOUNT
℄ R.MT. ORIG LOC.
74 00
114 00
34.00
86.00
EXT.
3.0
RL
6°
℄ ENG.
2°
EJECTOR CANT POINT

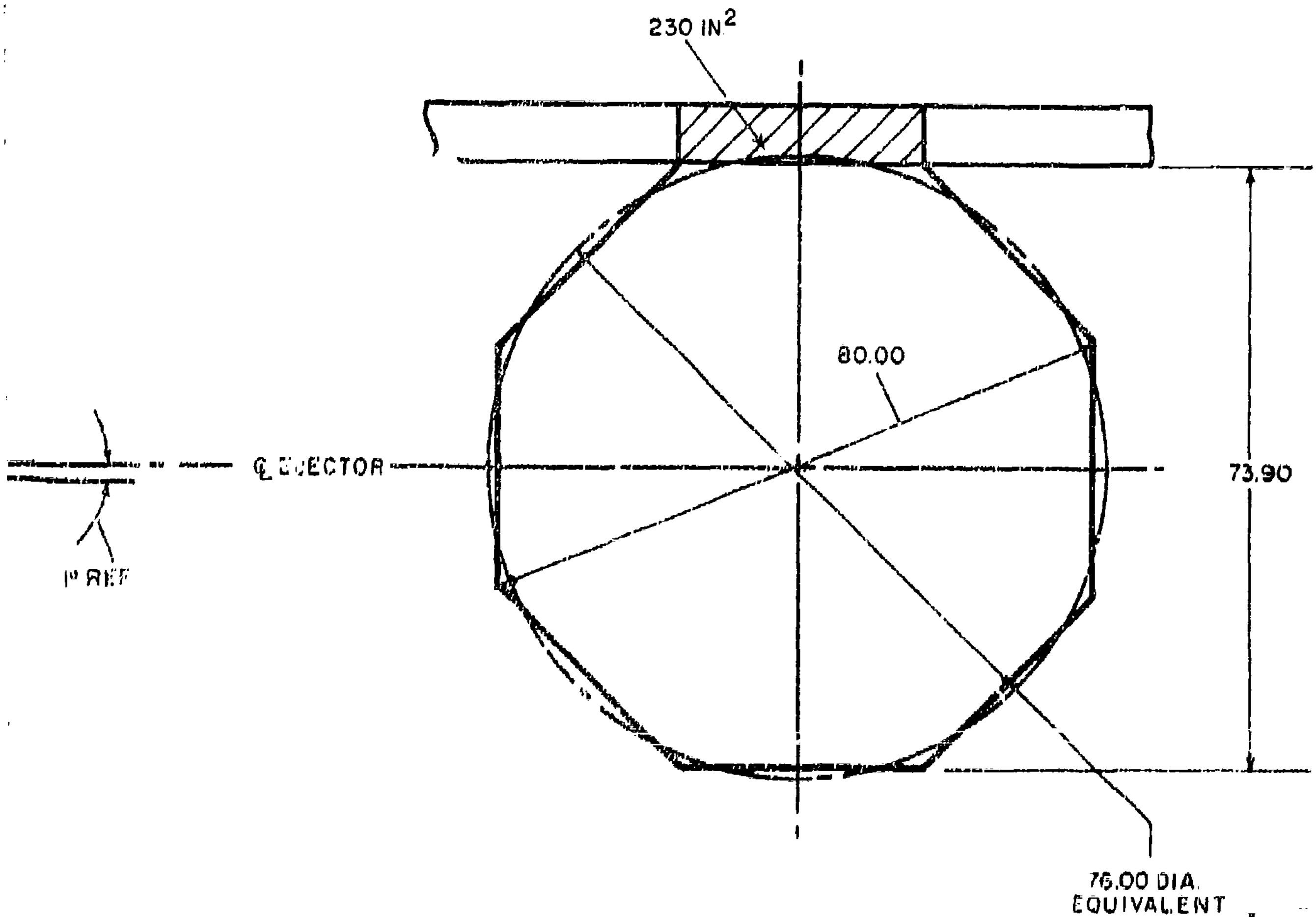
2



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

STF219L 650 LB/SEC TURBOFAN

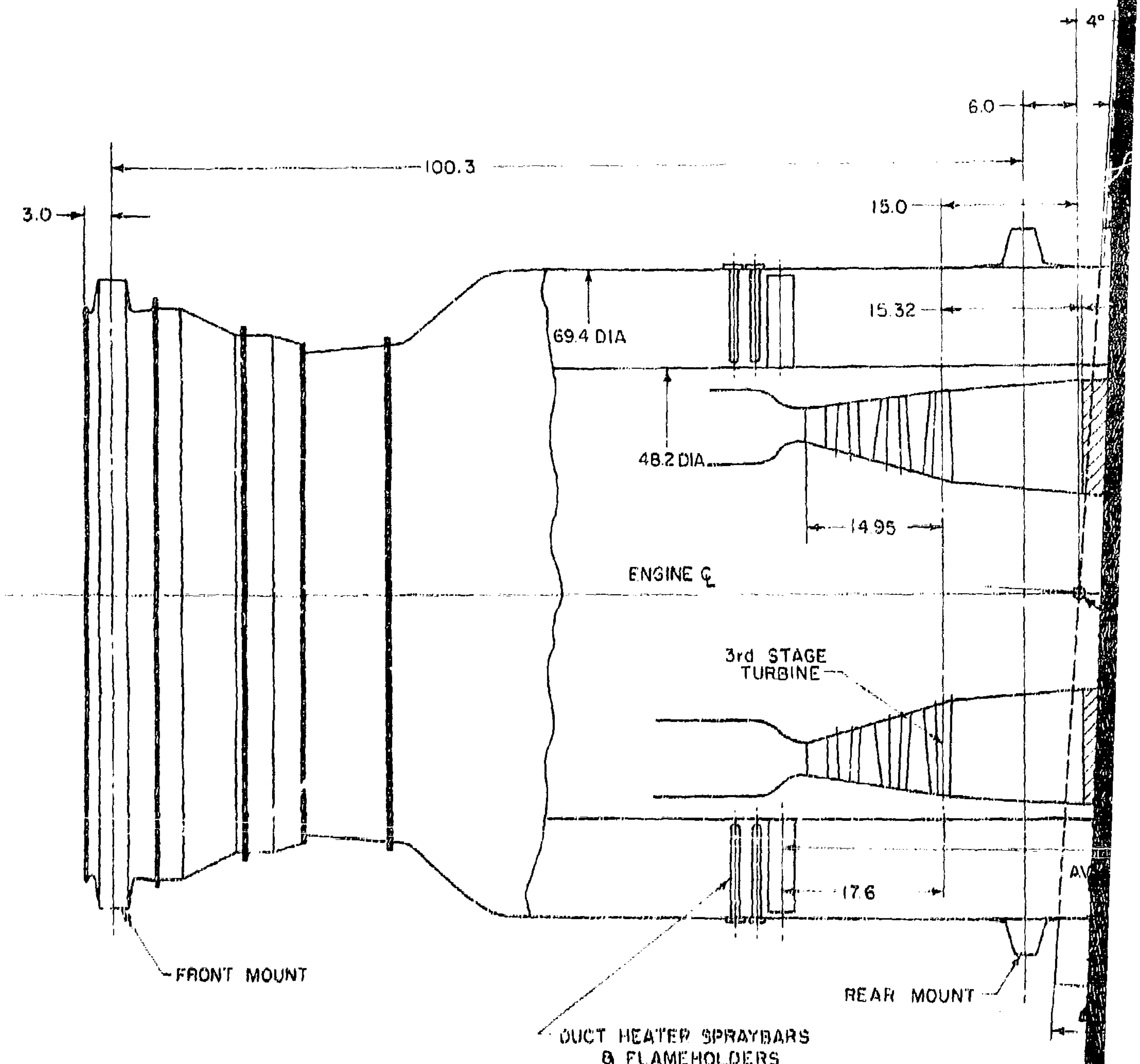
Figure 1-74



ARRANGEMENT SHOWING REAR ENG. 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

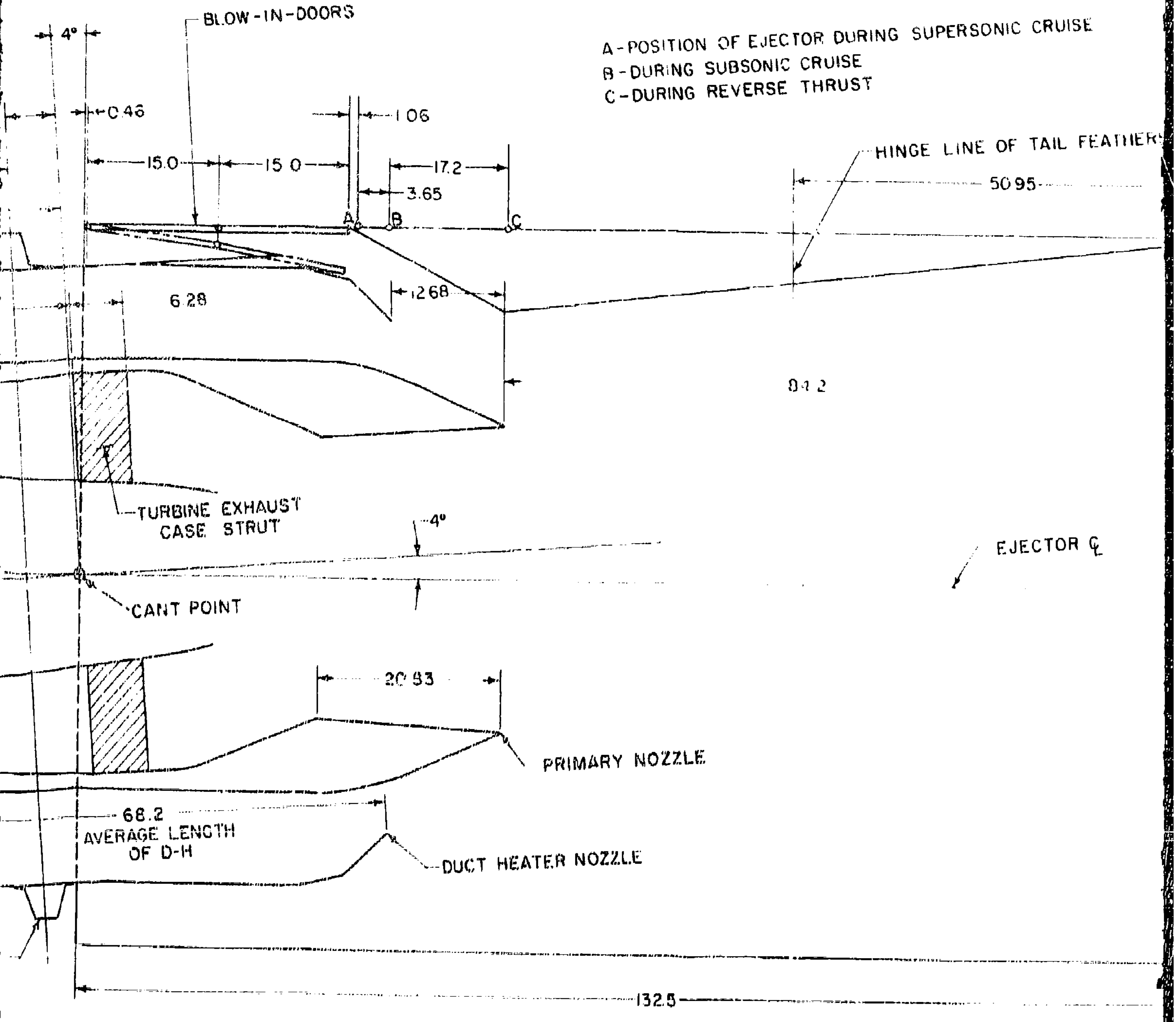
STF219L 650 LB/SEC TURBOFAN

Figure 1-34


4°
6.0
100.3
3.0
15.0
15.32
69.4 DIA
48.2 DIA
14.95
ENGINE ℄
3rd STAGE
TURBINE
17.6
FRONT MOUNT
REAR MOUNT
DUCT HEATER SPRAYBARS
& FLAMEHOLDERS

SECTION A-

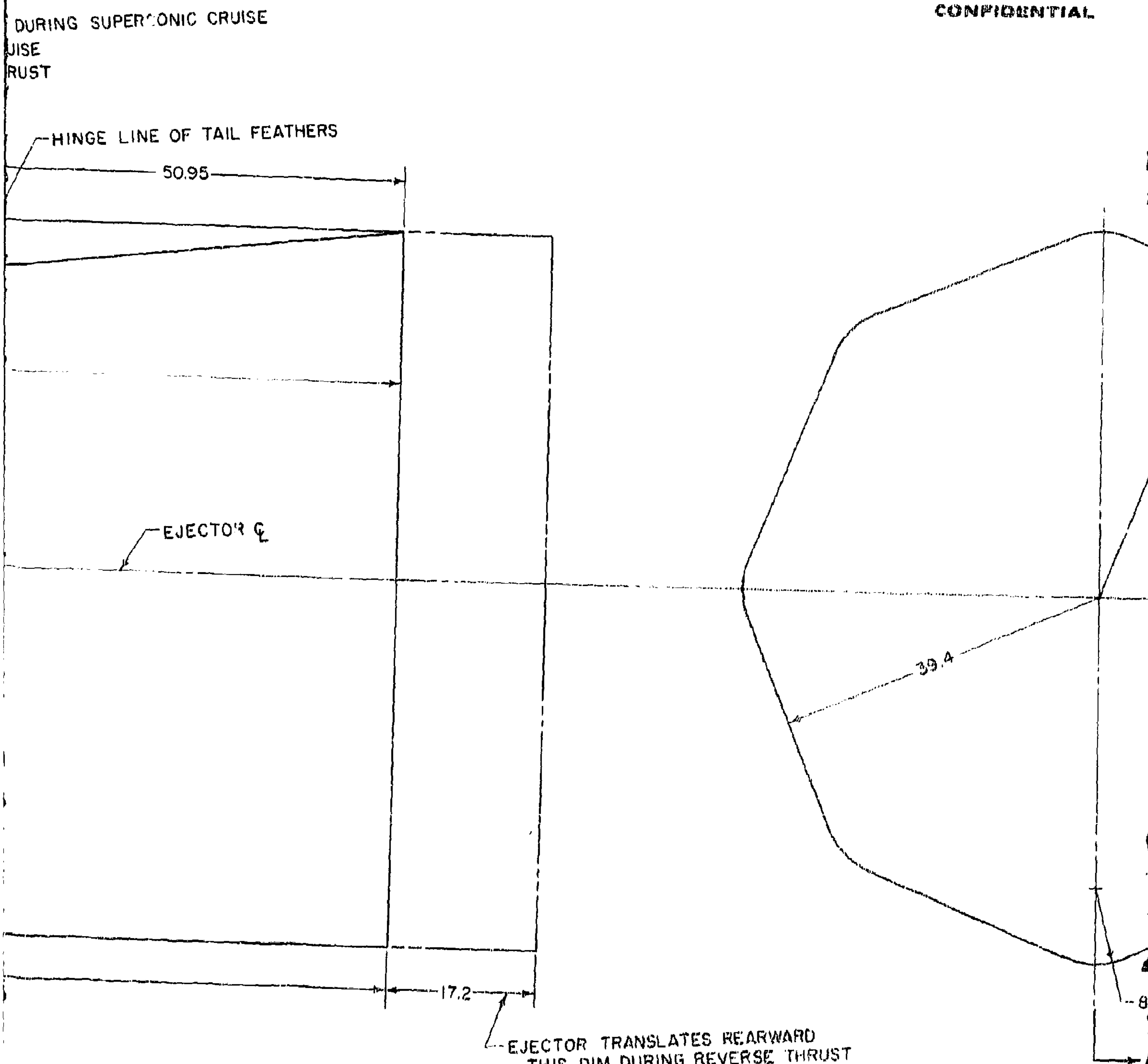
1



A-POSITION OF EJECTOR DURING SUPERSONIC CRUISE
B-DURING SUBSONIC CRUISE
C-DURING REVERSE THRUST
BLOW-IN-DOORS
4°
0.46
1.06
15.0
15.0
17.2
3.65
HINGE LINE OF TAIL FEATHERS
50.95
A
B
C
6.28
12.68
TURBINE EXHAUST CASE STRUT
4°
EJECTOR ℄
CANT POINT
20.83
PRIMARY NOZZLE
68.2
AVERAGE LENGTH OF D-H
DUCT HEATER NOZZLE
132.5

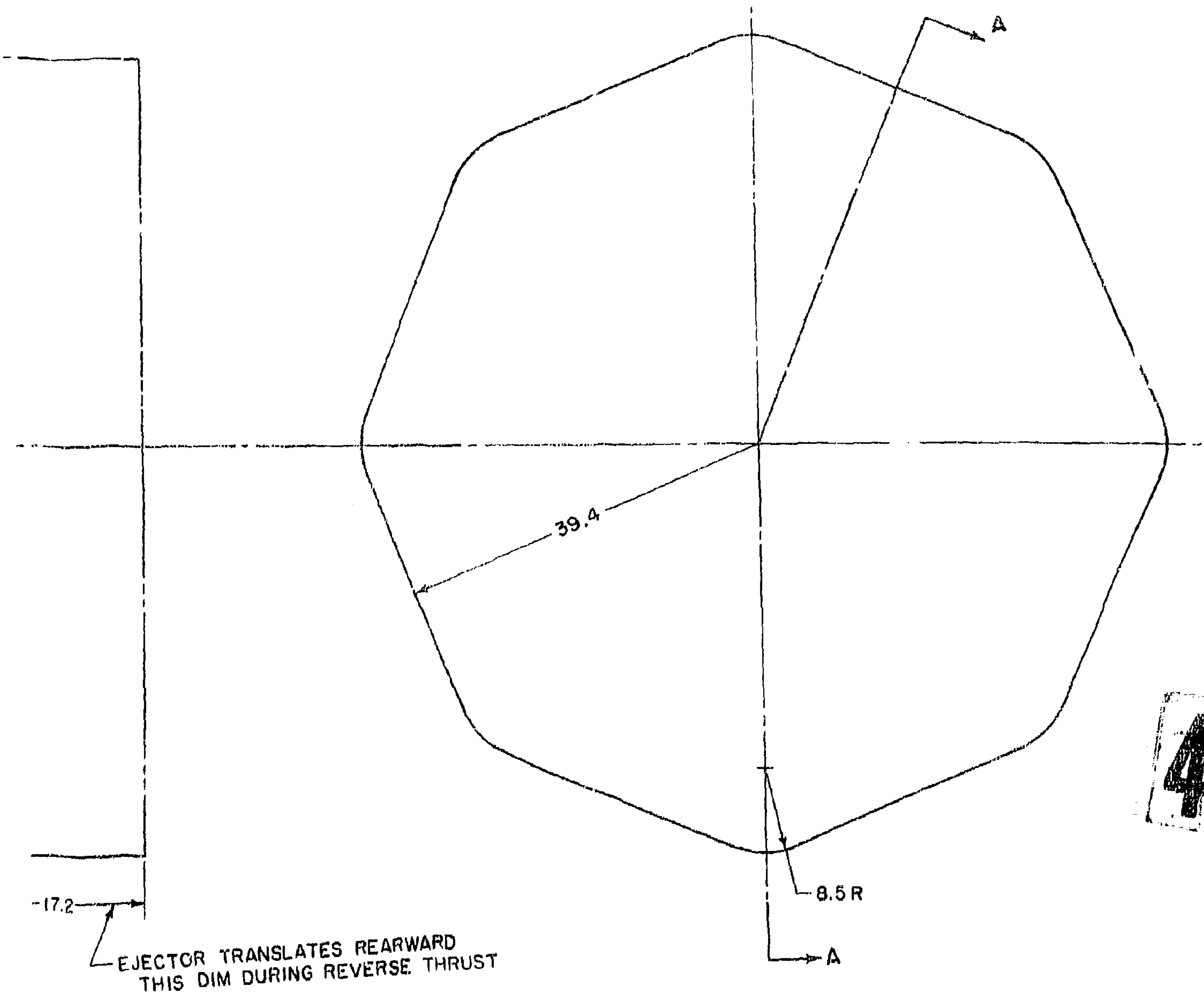
CTION A-A

2



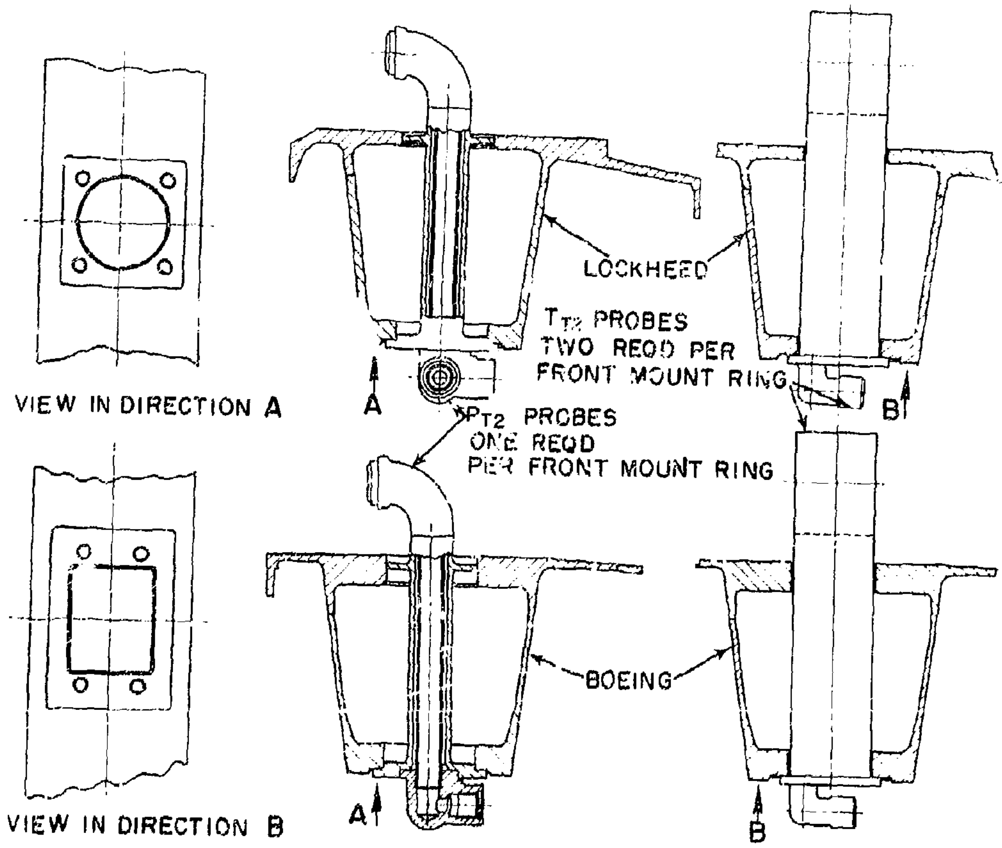
STF219L 650 LB/SEC TURBOFAN
ENGINE/EJECTOR RELATION

Figure 1-35



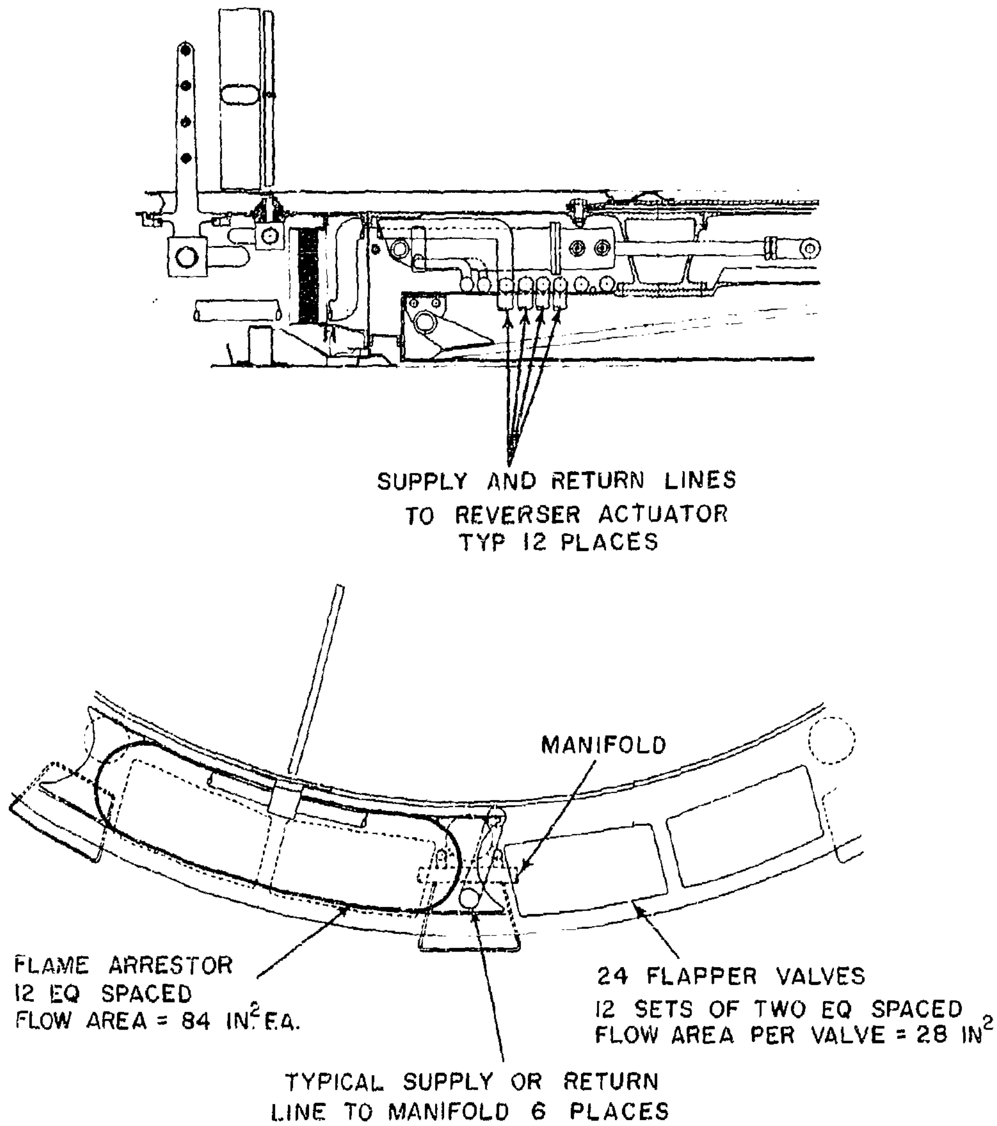
STF219L 650 LB/SEC TURBOFAN SHOWING
ENGINE/EJECTOR RELATIONSHIP

Figure 1-35



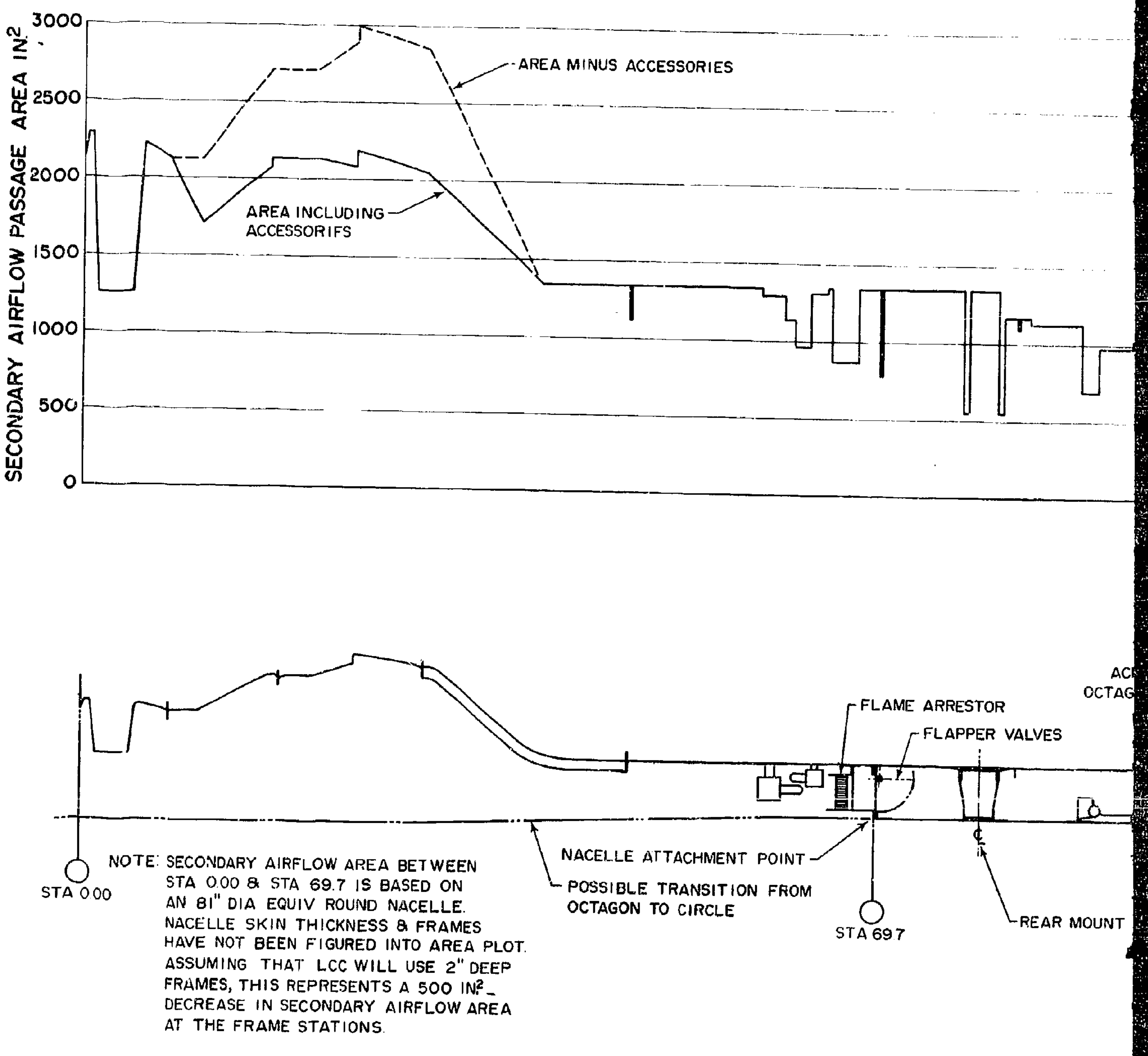
INLET PRESSURE AND TEMPERATURE PROBES FOR BIASING ENGINE FUEL CONTROL

Figure 1-36

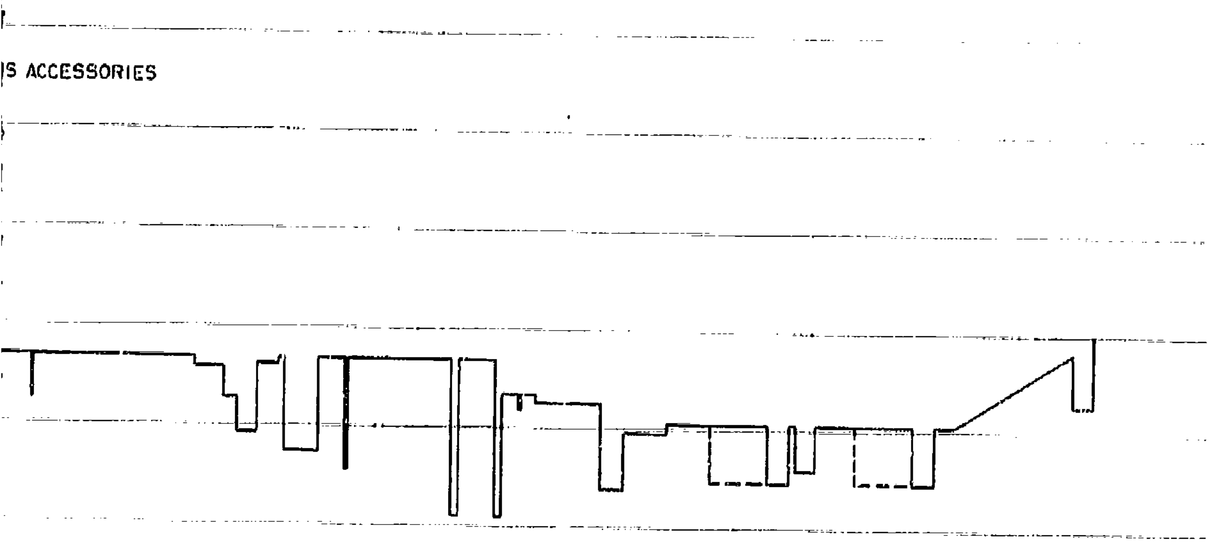


ARRANGEMENT OF FLAME ARRESTORS
IN SECONDARY AIR PASSAGE

Figure 1-37

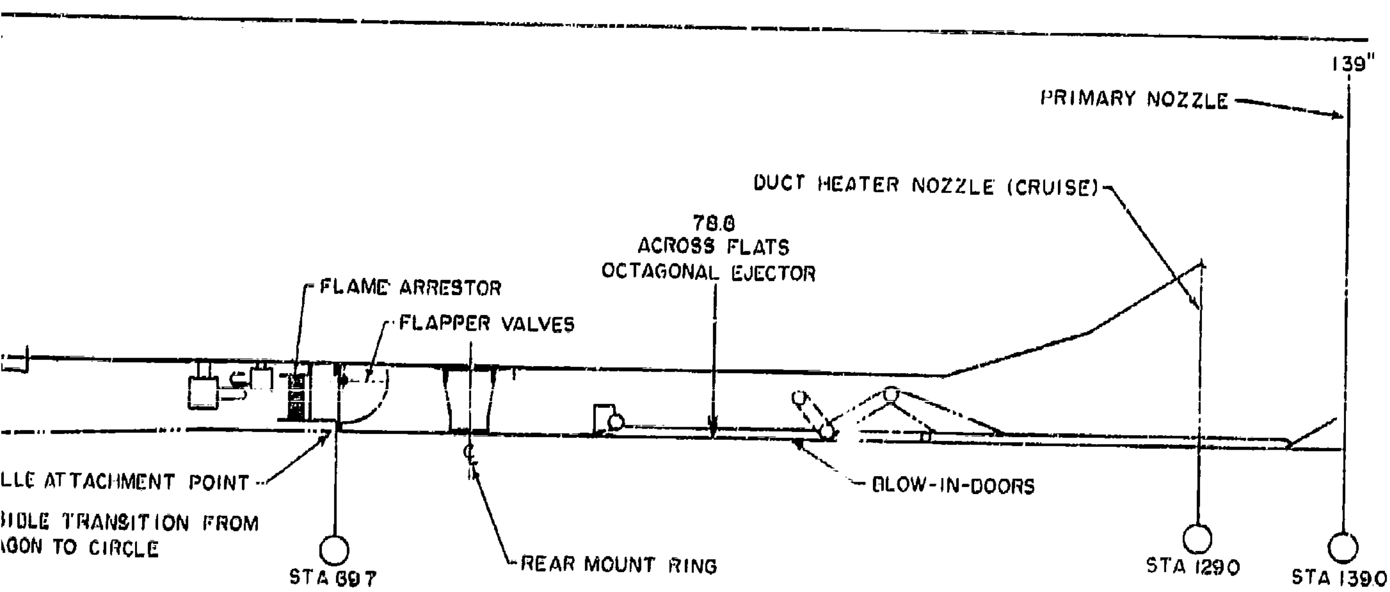


SECONDARY AIRFL



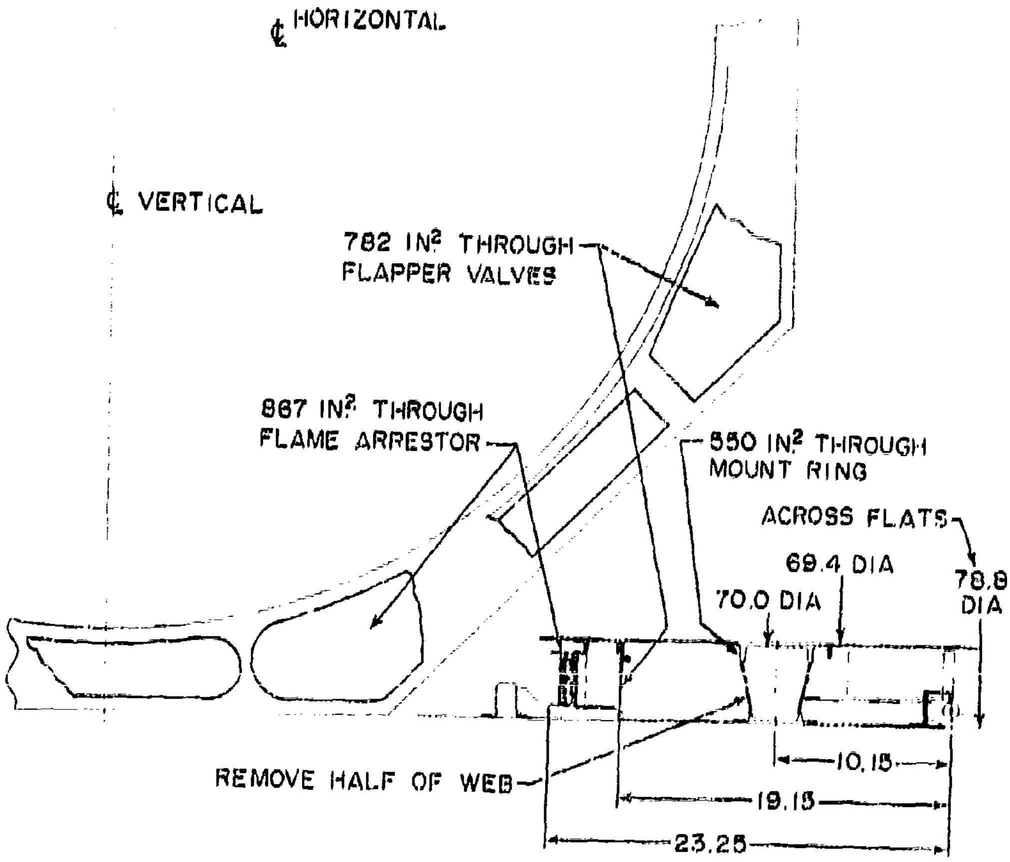
Mo	AREA	% PRESS LOSS
0.3	920	5 %
0.5	605	15 %
0.6	537	20 %
0.7	494	25 %

BASED ON MAX SECONDARY AIRFLOW FROM COORD SHEET 547



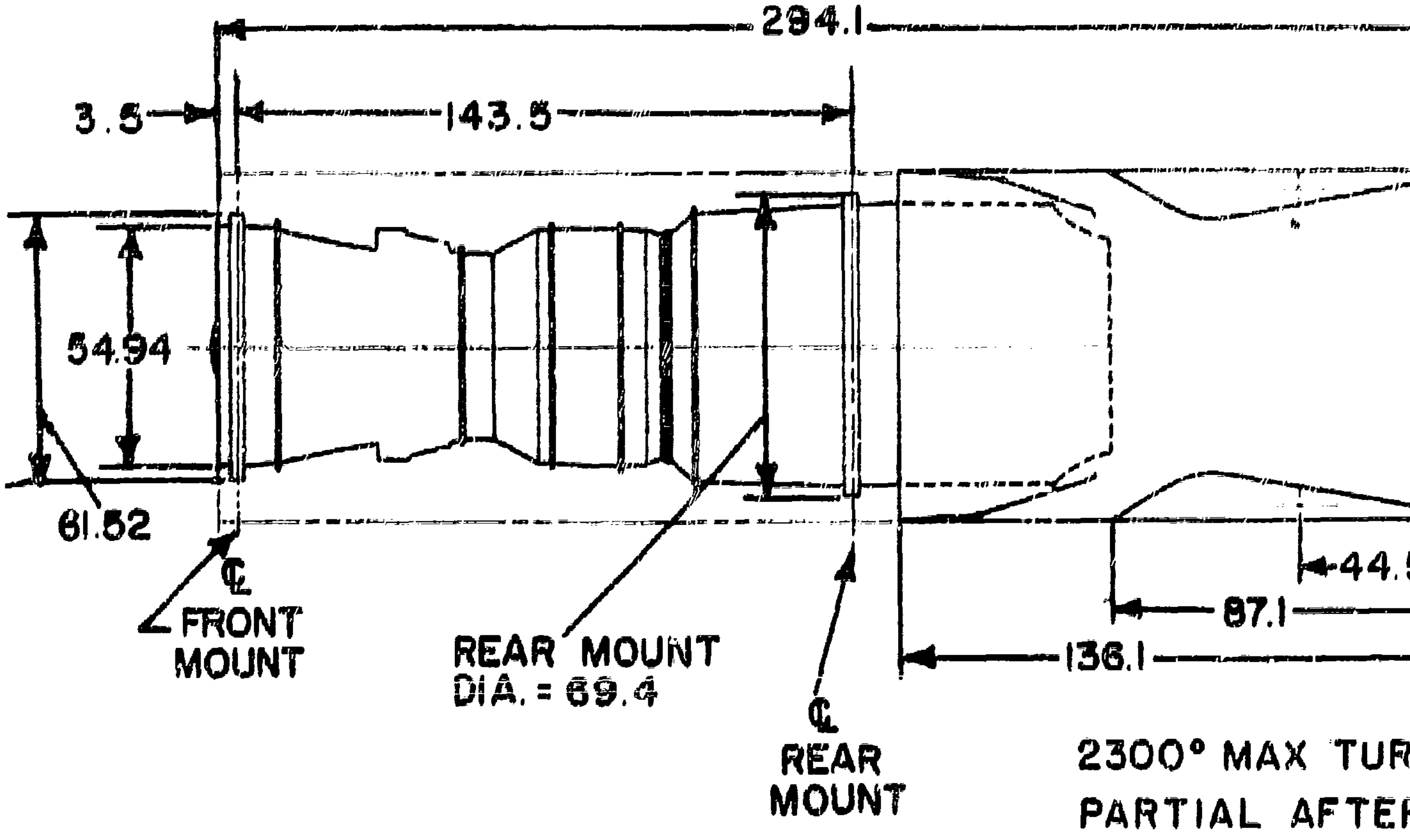
SECONDARY AIRFLOW PASSAGE GENERAL ARRANGEMENT

Figure 1-38



DETAILS OF SECONDARY AIRFLOW PASSAGE

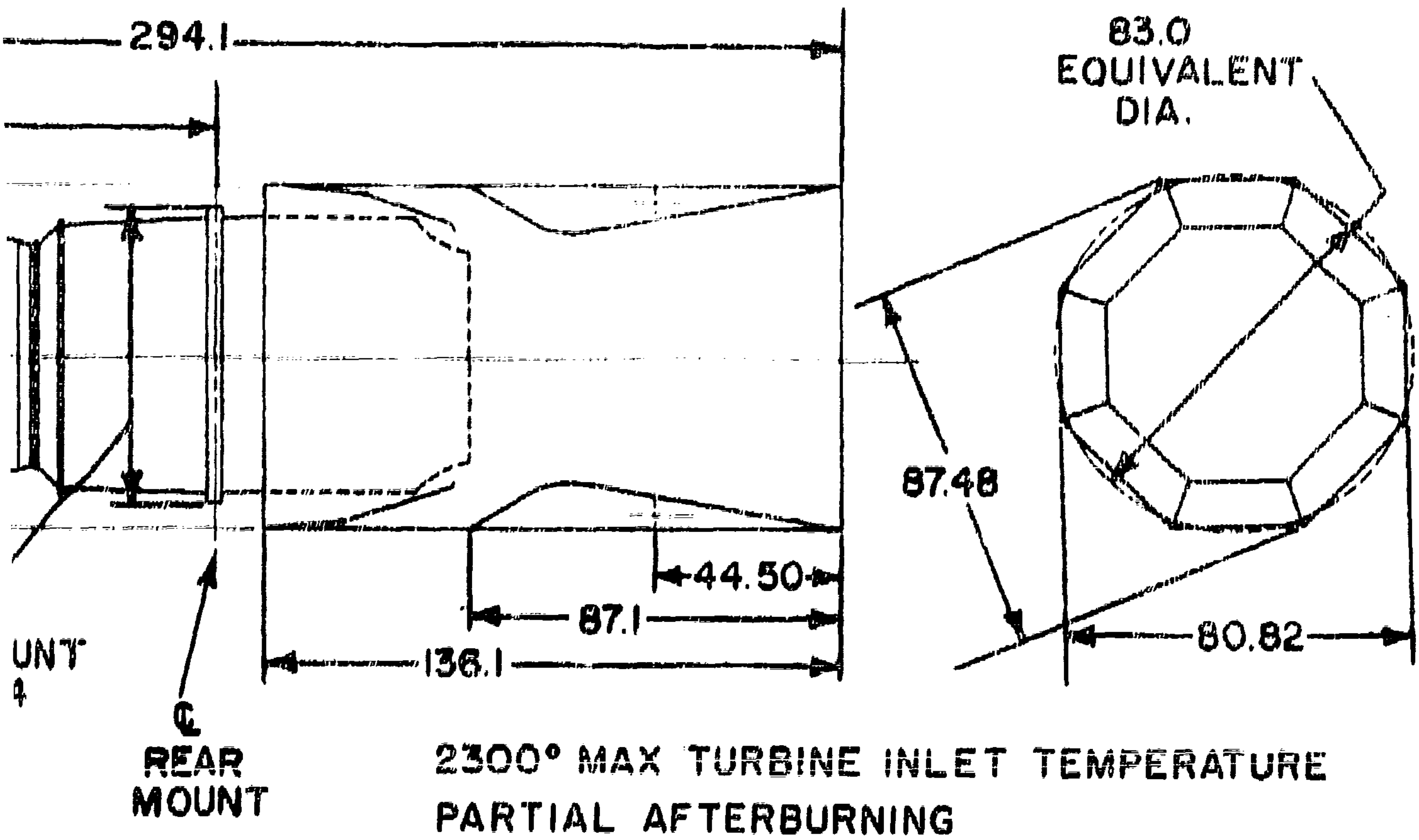
Figure 1-39



2300° MAX TUR
PARTIAL AFTER

STJ227 525 LB/

Figu

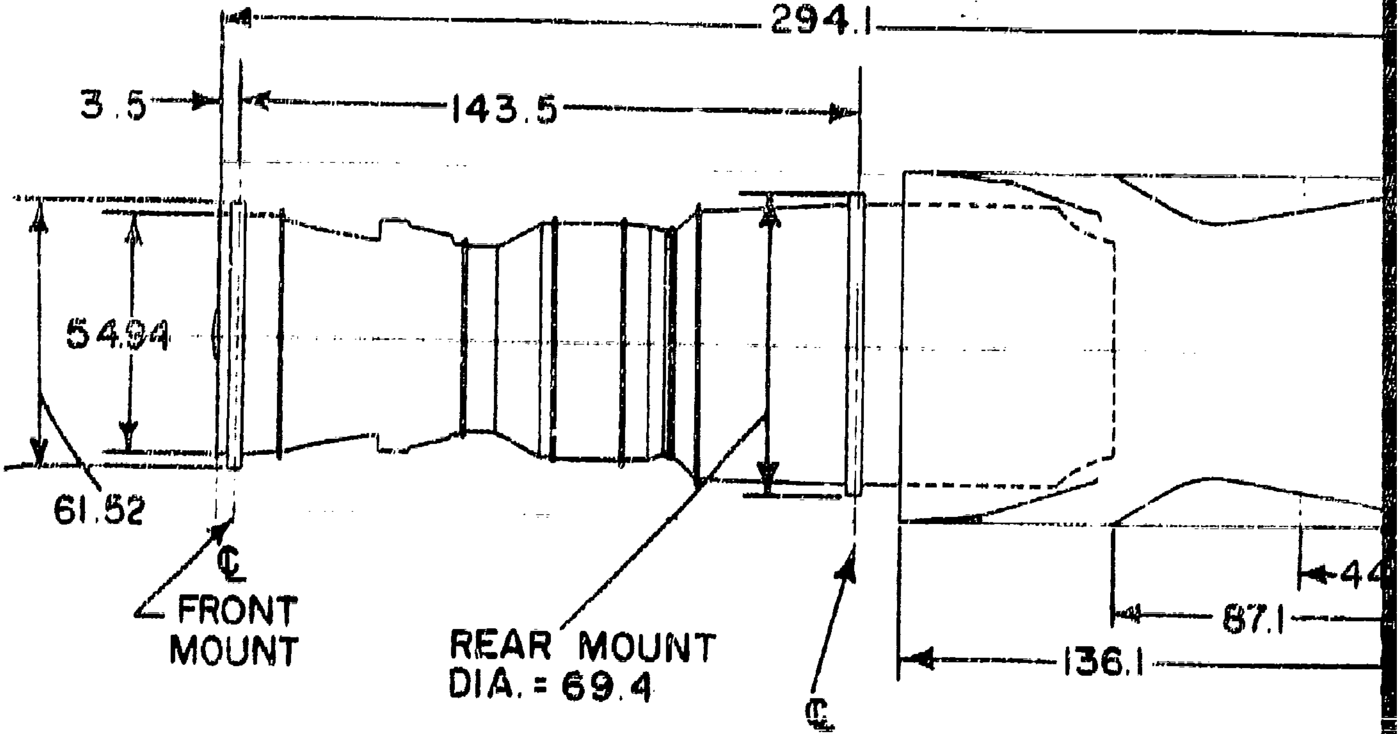


2300° MAX TURBINE INLET TEMPERATURE
PARTIAL AFTERBURNING

2

STJ227 525 LB/SEC TURBOJET

Figure 1-40

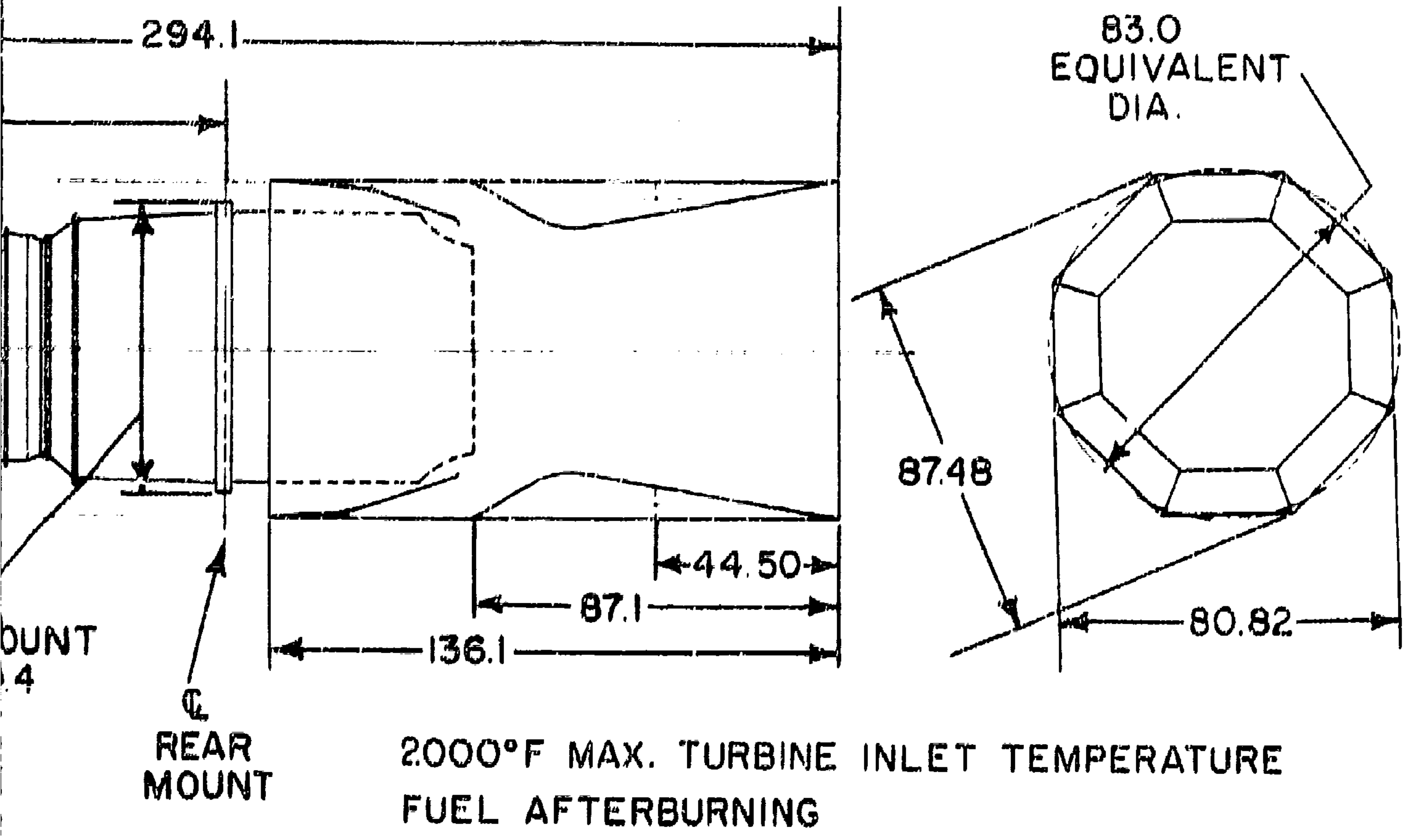


2000°F MAX. TU
FUEL AFTERBUR

STJ227 S25 I

FI

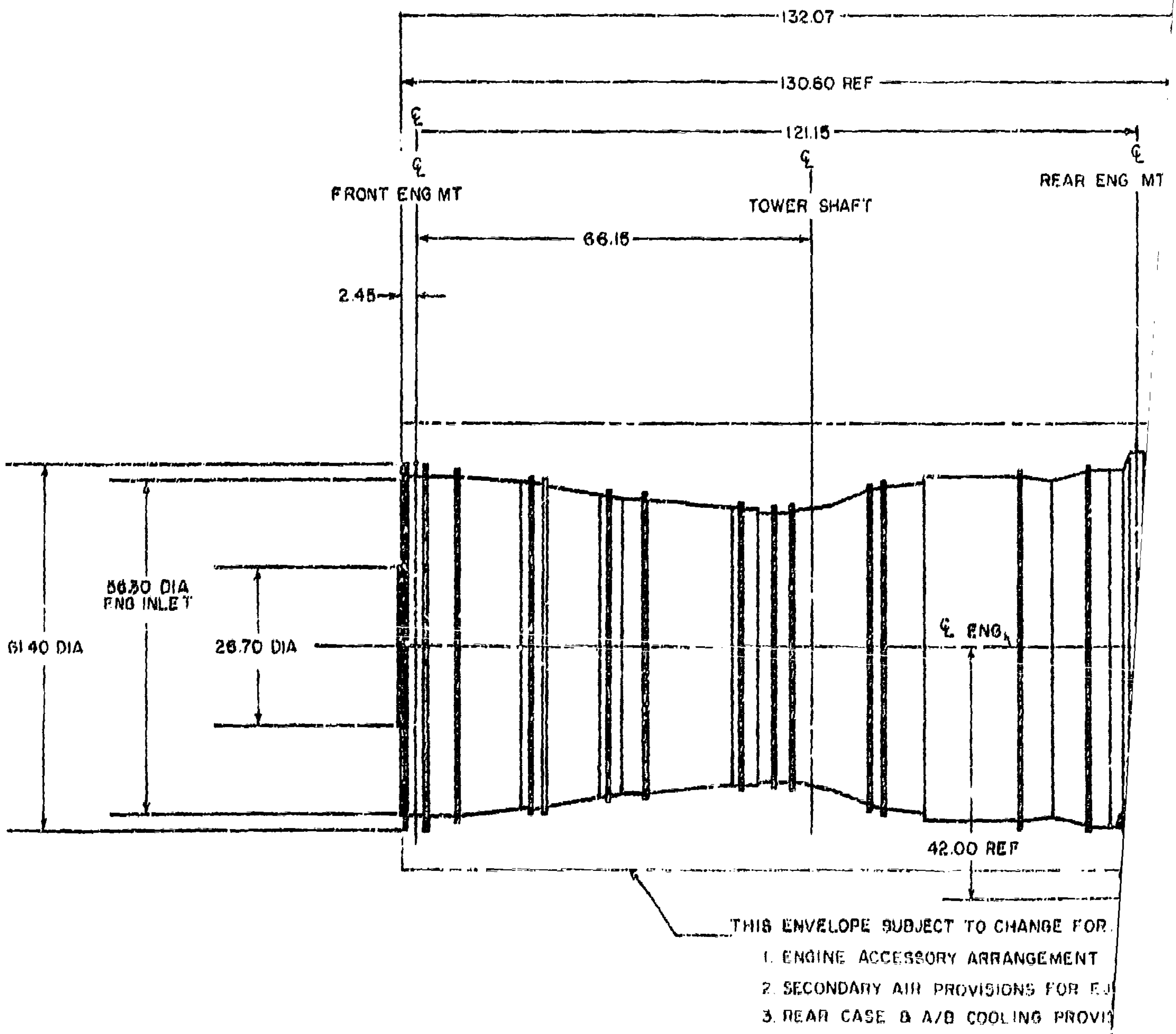
CO



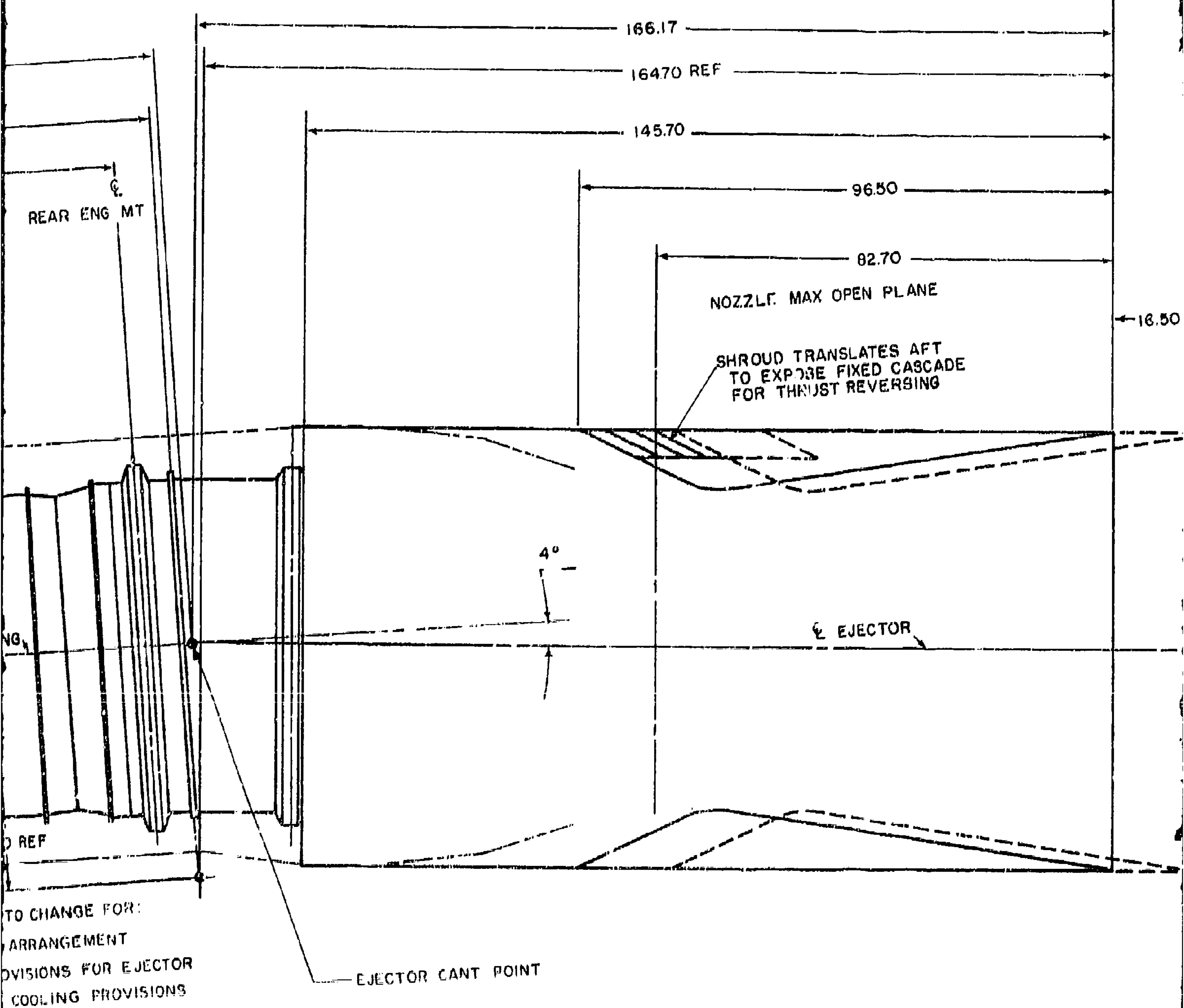
2

STJ227 625 LB/SEC TURBOJET

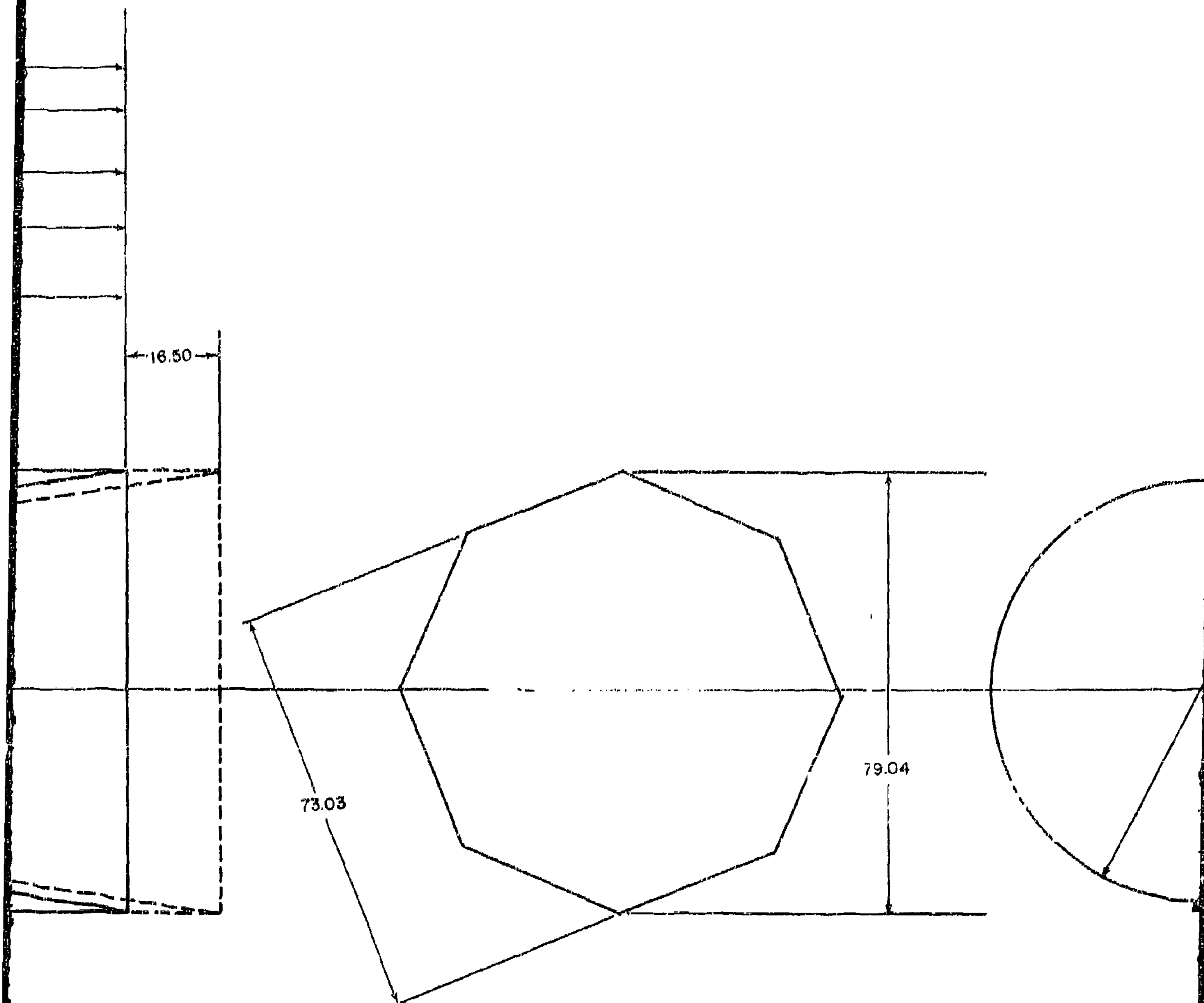
Figure 1-41


132.07
130.60 REF
121.15
℄
FRONT ENG MT
℄
TOWER SHAFT
℄
REAR ENG MT
66.15
2.45
56.30 DIA
ENG INLET
61.40 DIA
26.70 DIA
℄ ENG
42.00 REF
THIS ENVELOPE SUBJECT TO CHANGE FOR
1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR EJ
3. REAR CASE & A/B COOLING PROVI

1

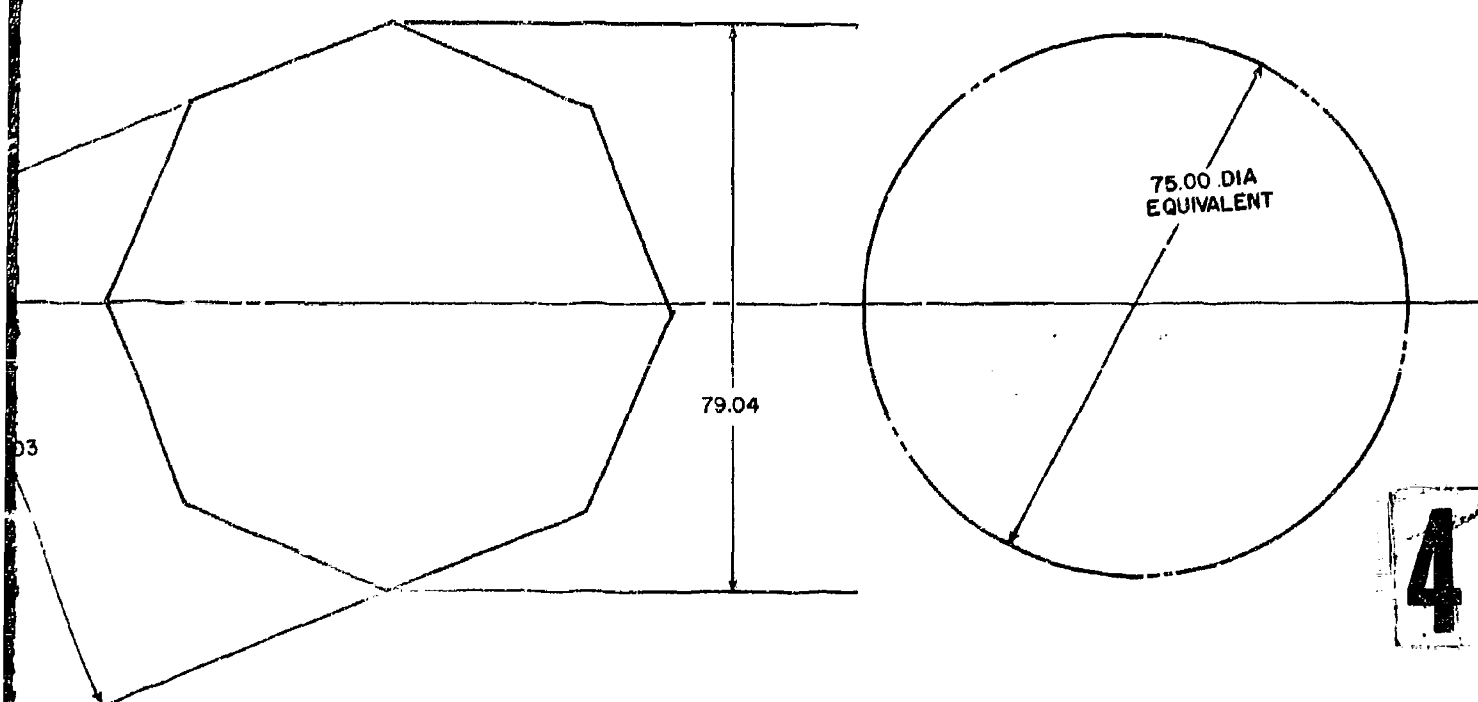

166.17
164.70 REF
145.70
96.50
82.70
REAR ENG MT
NOZZLE MAX OPEN PLANE
16.50
SHROUD TRANSLATES AFT TO EXPOSE FIXED CASCADE FOR THRUST REVERSING
4°
℄ EJECTOR
REF
TO CHANGE FOR:
ARRANGEMENT
OVISIONS FOR EJECTOR
COOLING PROVISIONS
EJECTOR CANT POINT

2



TYPICAL TURBOJET ENGINE. FULL AFTERBU

Figure 1-42

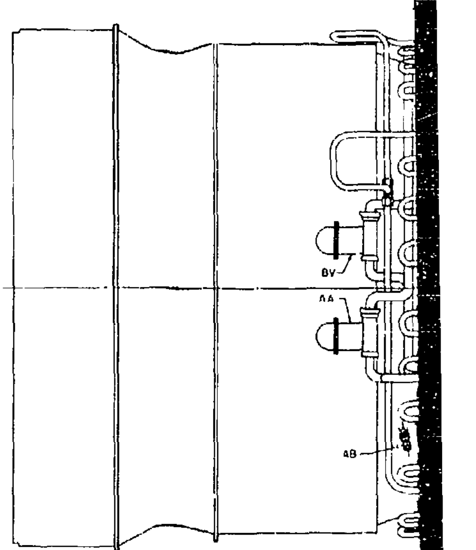


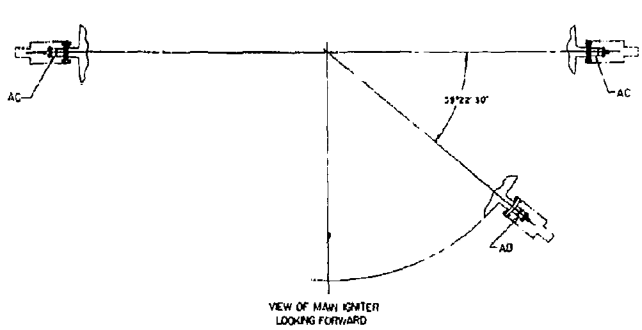
TYPICAL TURBOJET ENGINE. FULL AFTERBURNING.

Figure 1-42

ZONE		
		ACCESSORY DRIVE PADS
25F,17F,	A	POWER TAKEOFF
30E	B	TACHOMETER MOUNTING PAD WITH AND 20005XVB)
		FUEL DRAIN
17C	FD1	COMBUSTION CHAMBER FUEL DRAIN
18C	FD2	DUMP VALVE DRAIN (MAIN)
15C	FD4	DUMP VALVE DRAIN (AFTERBURNER) ZONE 1
14C	FD4A	DUMP VALVE DRAIN (AFTERBURNER) ZONE 2
13C	FD8	AFTERBURNER COMBUSTION CHAMBER FUEL DRAIN
		FUEL PRESSURE
29E	FP1	FUEL PUMP INLET PRESSURE
30E	FP3	CHECK VALVE FUEL PRESSURE (OUTLET PRESSURE)
		FUEL FLOW
19F, 30F	F1	FUEL PUMP & AFTERBURNER PUMP SUPPLY INLET
300	F2	MAIN FUEL FLOWMETER SUPPLY INLET
300	F3	MAIN FUEL FLOWMETER SUPPLY OUTLET
19E	F5	AFTERBURNER FLOWMETER SUPPLY INLET
19D	F6	AFTERBURNER FLOWMETER SUPPLY OUTLET
		FUEL VENT
29D	FV2	FUEL PUMP OUTLET VENT
		FUEL TEMPERATURE
30E	FT1	HEATER OUTLET FUEL TEMPERATURE
		OIL BREATHER
30D	LB2	MAIN OIL OVERBOARD BREATHER
		OIL DRAIN
28C	LD1	OIL TANK DRAIN
28D	LD2	OIL TANK OVERFLOW DRAIN
29D	LD3	OIL CUP OVERFLOW DRAIN
25C	LD4	GEARBOX MAIN OIL DRAIN
30C	LD7	OIL STRAINER DRAIN
		OIL FLOW
28E	L1	OIL TANK REMOTE FILLER
29E	L2	OIL TANK MANUAL FILL
		OIL PRESSURE
31D	LP1	PRESSURE FOR TRANSMITTER
25C	LP4	OIL FILTER INLET PRESSURE — PROVISIONS FOR PRESSURE DIFFERENTIAL
25C	LP5	OIL FILTER OUTLET PRESSURE — PROVISIONS FOR PRESSURE DIFFERENTIAL
		OIL TEMPERATURE
31D	LT1	MAIN OIL TEMPERATURE
		OIL VENT
30F	LV3	OIL PRESSURE TRANSMITTER VENT
		SEAL DRAIN
29D	SD1	FUEL CONTROL SEAL DRAIN
30E	SD2	FUEL PUMP SEAL DRAIN
30D	SD9	HYDRAULIC PUMP SEAL DRAIN
18D	SD14	AFTERBURNER FUEL PUMP SEAL DRAIN
		TEMPERATURE SENSING
20D	TT5	TURBINE EXIT TEMPERATURE (AVG)
20D	ITT5	TURBINE EXIT TEMPERATURE (INDIVIDUAL)
		PRESSURE SENSING
15F	PT5	TURBINE EXIT PRESSURE
		MISCELLANEOUS
24E, 30E	F	MAIN FUEL PUMP
29E 30E	G	FUEL PUMP FUEL FILTER (MAIN)
24E	H	FUEL PUMP FILTER (MIN SPACE FOR REMOVAL)
19E, 23E	J	AFTERBURNER FUEL PUMP
31D	K	HYDRAULIC PUMP
28C, 30D	L	OIL PUMP
28C, 30C	M	OIL FILTER
31C	N	FUEL-OIL COOLER (MAIN)
19C, 23C	P	FUEL-OIL COOLER (AFTERBURNER)
30E	R	GEARBOX
20D	S	AUTOMATIC RESTART SWITCH
30D	T	BREATHER PRESSURIZING VALVE
18E, 19F, 31C	U	HIGH PRESSURE BLEED PAD
28D, 29D	X	MAIN FUEL CONTROL
18D, 23D	Y	AFTERBURNER FUEL CONTROL
17D	Z	EXHAUST NOZZLE CONTROL
32D	AA	HYDRAULIC DISCHARGE FILTER
32D, 34D, 17C, 22C	AB	IGNITER PLUG (MAIN)
8D, 33B, 34B	AC	IGNITER PLUG (AFTERBURNER)
21B, 21E, 20A, 20E	AD	ENGINE FRONT MOUNTING PROVISIONS
13B, 18B, 18E	AE	ENGINE REAR MOUNTING PROVISIONS
20F	AF	FUEL RETURN TO SUPPLEMENTARY COOLING

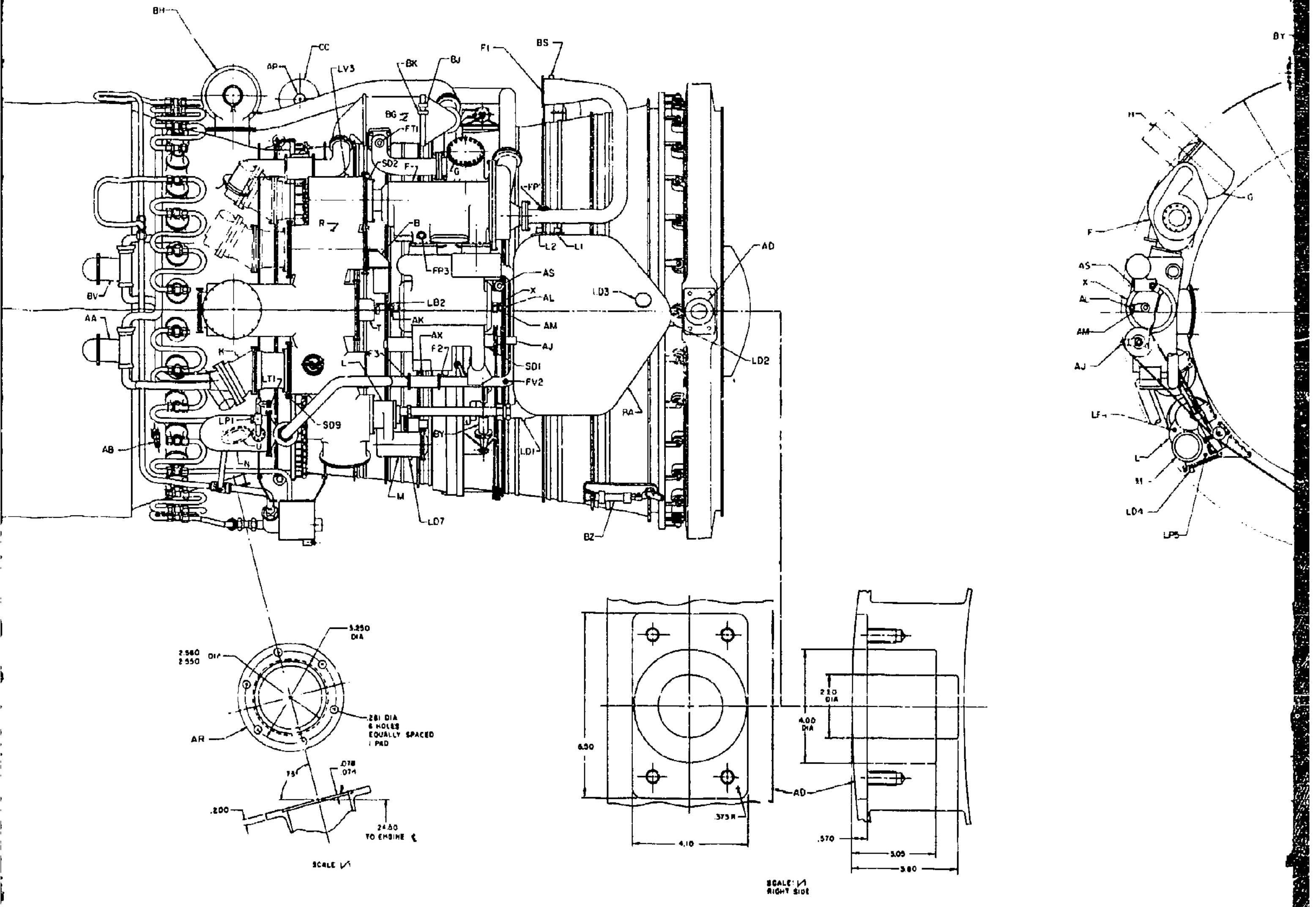
ZONE		
23A	AG	GROUND HANDLING HOLES (FRONT MOUNT)
13C	AH	GROUND HANDLING AREA (REAR)
28D	AJ	POWER CONTROL LEVER (120° ANGLE OF TRAVEL)
30D	AK	SHUT OFF LEVER (90° ANGLE OF TRAVEL)
26D	AL	APPROACH VELOCITY CONTROL LEVER (60° ANGLE OF TRAVEL)
28D, 29D	AM	MACH NO. OR SHOCK POSITION RESET LEVER (60° ANGLE OF TRAVEL)
18E	AN	AIR TURBINE VALVE AFTERBURNER FUEL PUMP
31F	AP	IGNITION EXCITER ELECTRICAL CONN.
31B	AR	THERMAL ANTI ICING PAD
24E, 28D	AS	AERODYNAMIC BRAKE CONTROL AIR SUPPLY CONN
17D	AT	EXHAUST NOZZLE FEEDBACK
14C	AU	CHECK & DUMP VALVE (AFTERBURNER ZONE I)
18C	AV	CHECK & DUMP VALVE (AFTERBURNER ZONE II)
19D	AW	MAIN FUEL FLOWMETER MOUNTING PROVISIONS
30D	AX	AFTERBURNER FUEL FLOWMETER MOUNTING PROVISIONS
25E, 19E	AY	FUEL CONTROL FUEL FILTER (MIN SPACE FOR REMOVAL)
18C, 25C	AZ	WINDMILL BYPASS CHECK & DUMP LOW IDLE VALVE (MAIN)
29D	BA	OIL TANK
17D	BB	NOZZLE POSITION INDICATOR MOUNTING PROVISIONS
20D	BC	REVERSER POSITION INDICATOR CONN. (ELECTRICAL)
19C	BD	THERMOSTATIC FUEL BYPASS VALVE
14A, 22B, 16D, 20D	BF	VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFG)
20E	BG	FUEL HEATER
17F, 24F, 31F	BH	POWER TAKEOFF GEARBOX
30F	BJ	FUEL HEATER VALVE (ELECTRICAL CONN)
30F	BK	FUEL HEATER VALVE POSITION INDICATOR (ELECTRICAL)
17F	BL	POWER TAKEOFF DECOUPLER
18F, 24F	BM	POWER TAKEOFF DECOUPLER ACTIVATING SHAFT
24D	BN	CONTROL CABLE - FUEL CONTROLS
19F, 29F	BS	FUEL INLET TEMPERATURE SENSOR
18D	BU	AFTERBURNER FUEL OIL COOLER THERMOSTATIC BYPASS VALVE
32D	BV	HYDRAULIC RETURN FILTERS
14E	BW	EXHAUST NOZZLE ACTUATORS
13D, 18D, 19F, 25F, 30C	BY	STARTING BLEED DOOR ACTUATORS
20E, 29C	BZ	INLET GUIDE VANE ACTUATORS
18C, 18F	CA	AERODYNAMIC BRAKE ACTUATORS
20D	CB	EXCITER (MAIN)
18F, 30F	CC	EXCITER AFTERBURNER



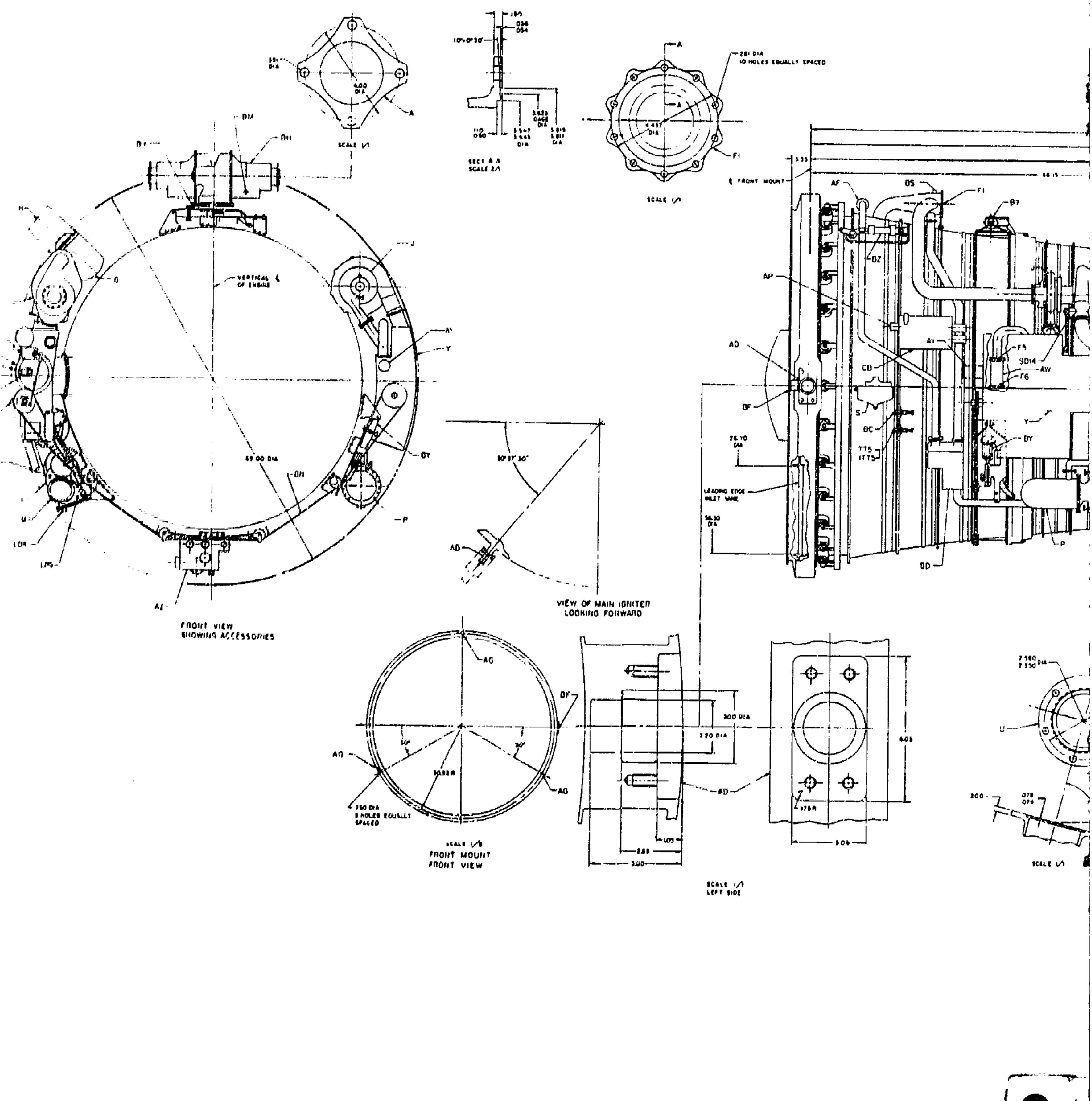


VIEW OF MAIN IGNITER LOOKING FORWARD

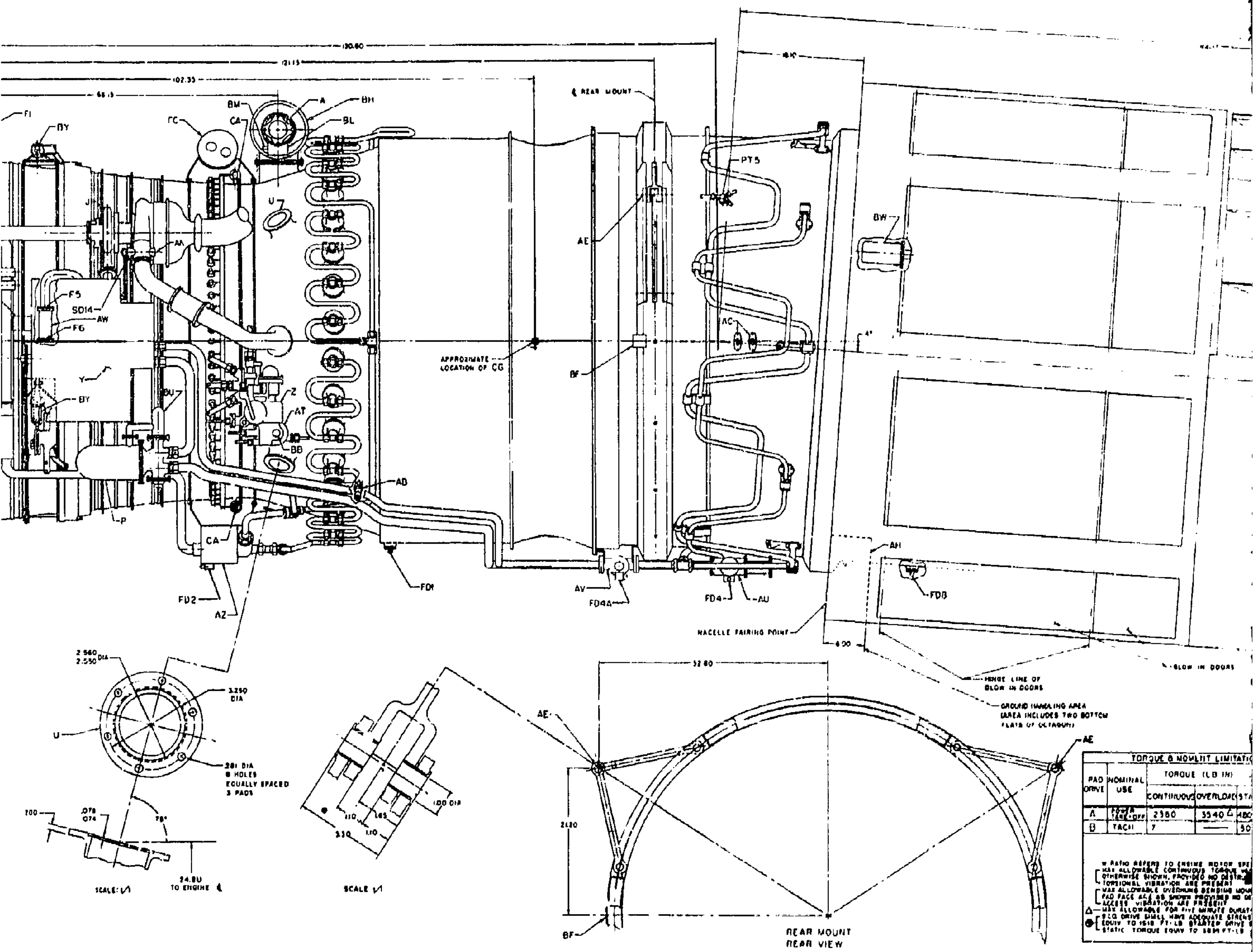
1


BH
AP
CC
LV3
BK
BJ
F1
BS
BG
FT1
SD2
F
G
FP
R
B
L2
L1
AD
FP3
AS
X
AL
LD3
AM
BV
AA
LB2
AK
AJ
AX
F2
LD2
K
F3
SD1
LT1
FV2
RA
LP1
SD9
BY
AB
LD1
N
M
LD7
BZ
3.250 DIA
2.560 2.550 DIA
.281 DIA 6 HOLES EQUALLY SPACED 1 PAD
AR
.078 .074
.200
24.80 TO ENGINE ℄
SCALE 1/1
2.20 DIA
4.00 DIA
6.50
.375 R
4.10
.570
3.05
3.80
SCALE 1/1 RIGHT SIDE
LF
LD4
LP5

2


SCALE 1/1
SECT A-A
SCALE 2/1
281 DIA
10 HOLES EQUALLY SPACED
SCALE 1/1
FRONT MOUNT
VERTICAL ℄ OF ENGINE
69.00 DIA
FRONT VIEW
SHOWING ACCESSORIES
VIEW OF MAIN IGNITER
LOOKING FORWARD
LEADING EDGE INLET VANE
SCALE 1/8
FRONT MOUNT
FRONT VIEW
SCALE 1/1
LEFT SIDE
SCALE 1/1

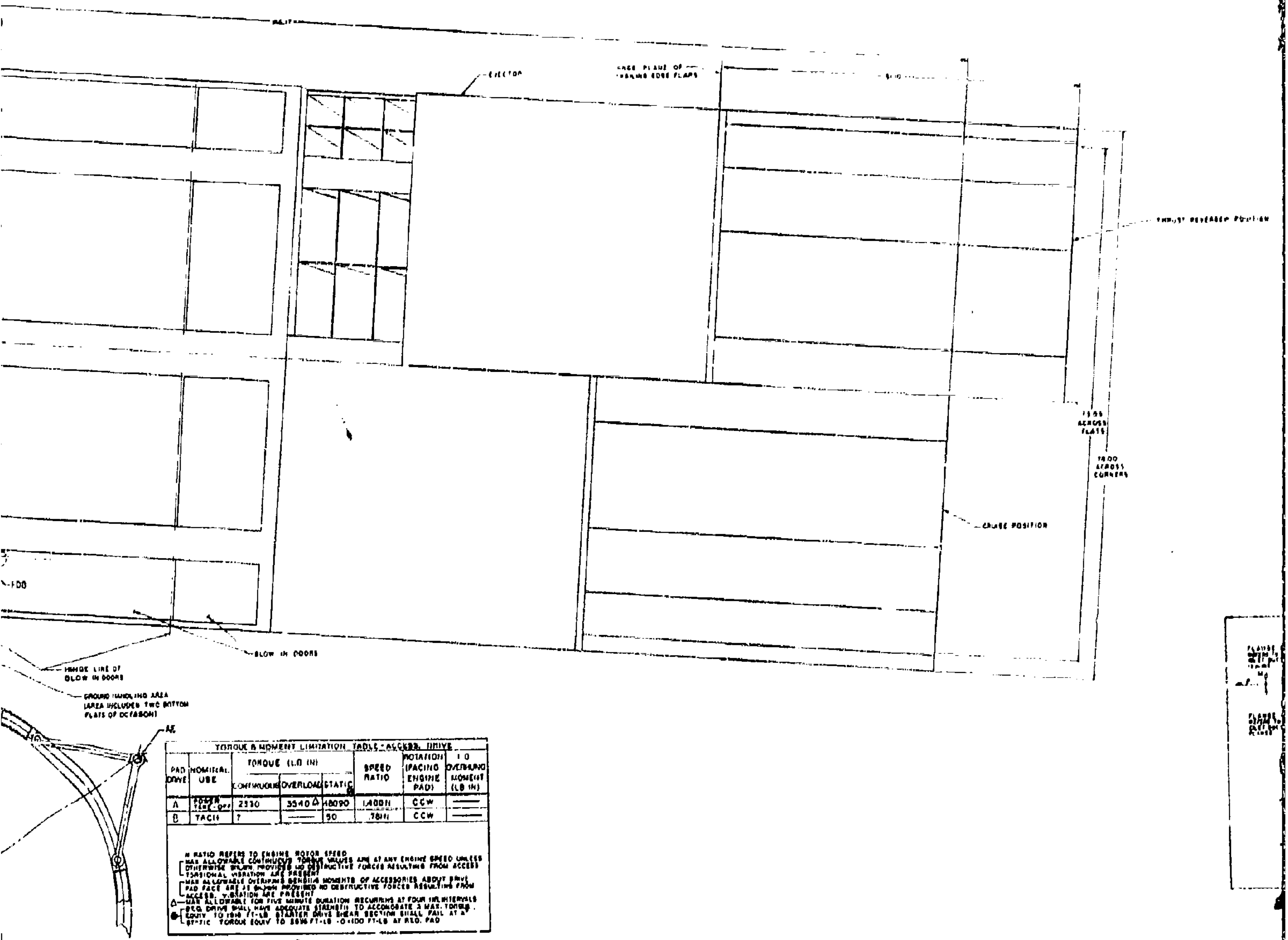
3



TORQUE & MOMENT LIMITATI[ONS]

PAD DRIVE	NOMINAL USE	TORQUE (LB IN) CONTINUOUS	OVERLOAD	STA[TIC]
A	POWER TAKE-OFF	2380	3540 △	400[...]
B	TACH	7	———	50[...]

* RATIO REFERS TO ENGINE ROTOR SPE[ED]
MAX ALLOWABLE CONTINUOUS TORQUE UN[LESS] OTHERWISE SHOWN, PROVIDED NO DESTR[...] TORSIONAL VIBRATION ARE PRESENT
MAX ALLOWABLE OVERHUNG BENDING MOM[ENT] PAD FACE ACE AS SHOWN PROVIDED NO DE[...] ACCESS VIBRATION ARE PRESENT
△ — MAX ALLOWABLE FOR FIVE MINUTE DURAT[ION]
● P.T.O. DRIVE SHALL HAVE ADEQUATE STRENG[TH] EQUIV TO 1510 FT-LB STARTER DRIVE [...] STATIC TORQUE EQUIV TO 5891 FT-LB [...]



TORQUE & MOMENT LIMITATION TABLE - ACCESS. DRIVE

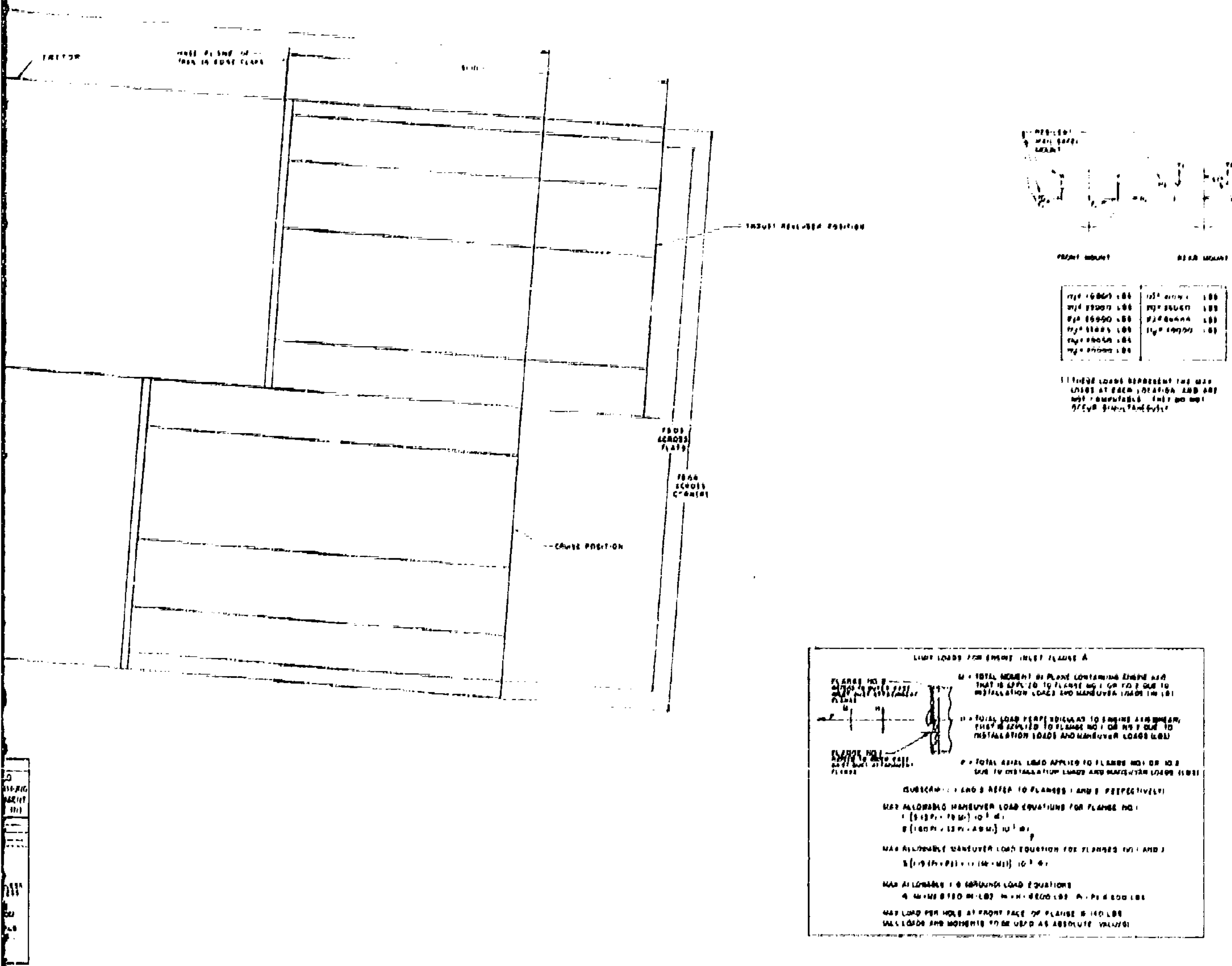
PAD DRIVE	NOMINAL USE	TORQUE (LB IN)			SPEED RATIO	ROTATION (FACING ENGINE PAD)	I O OVERHUNG MOMENT (LB IN)
		CONTINUOUS	OVERLOAD	STATIC			
A	POWER TAKE-OFF	2330	3540 Δ	48090	1.400N	CCW	——
B	TACH	7	——	50	.781N	CCW	——

N RATIO REFERS TO ENGINE ROTOR SPEED
MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT ANY ENGINE SPEED UNLESS OTHERWISE SHOWN, PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS. TORSIONAL VIBRATION ARE PRESENT
MAX ALLOWABLE OVERHUNG BENDING MOMENTS OF ACCESSORIES ABOUT DRIVE PAD FACE ARE AS SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS. VIBRATION ARE PRESENT
Δ—MAX ALLOWABLE FOR FIVE MINUTE DURATION RECURRING AT FOUR [illegible] INTERVALS
● PTO DRIVE SHALL HAVE ADEQUATE STRENGTH TO ACCOMODATE A MAX. TORQUE EQUIV TO [illegible] FT-LB STARTER DRIVE SHEAR SECTION SHALL FAIL AT A STATIC TORQUE EQUIV TO [illegible] FT-LB -0 +100 FT-LB AT RED. PAD

PROPOSED TURBOJET ACCESSORY AR

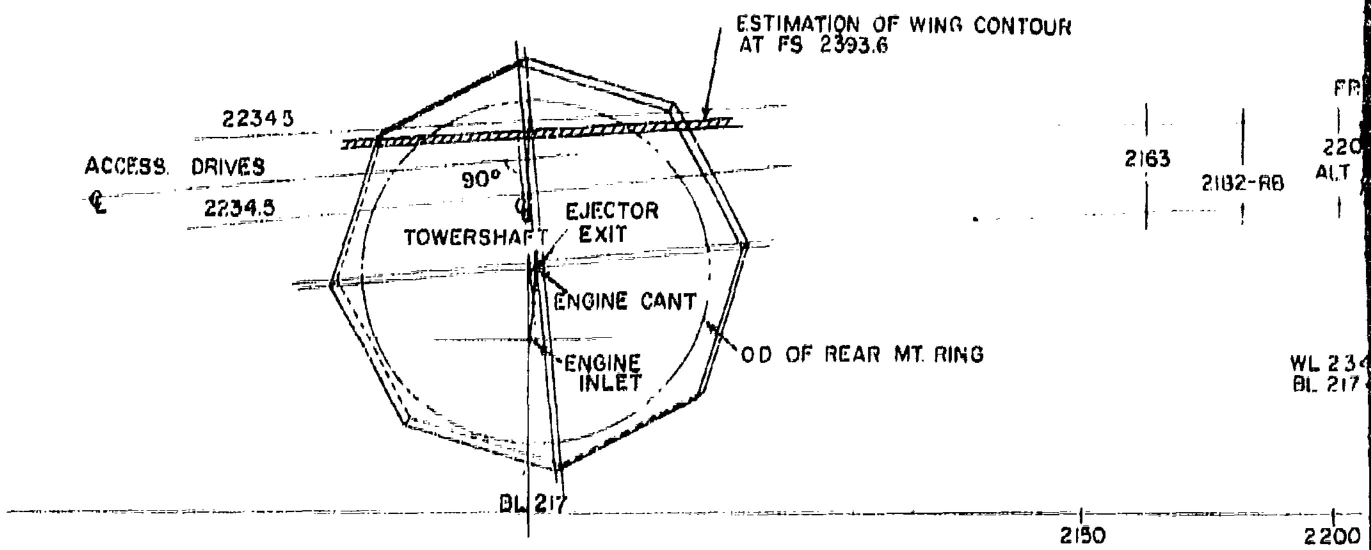
Figure 1-43

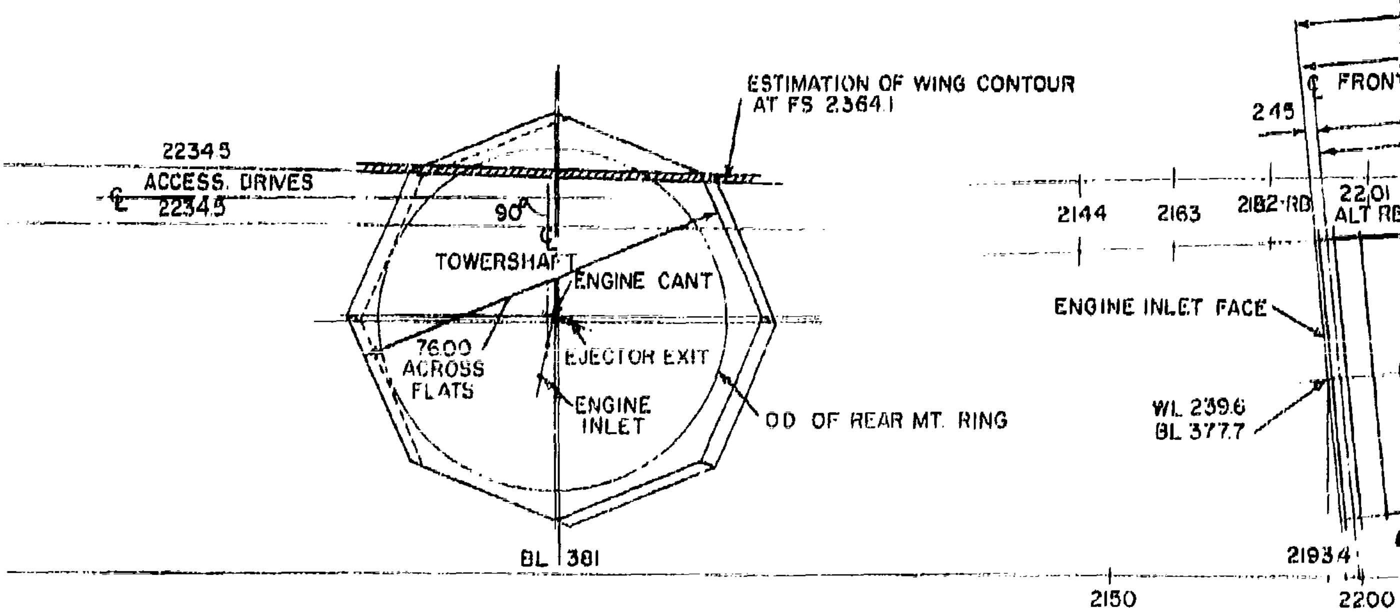
5

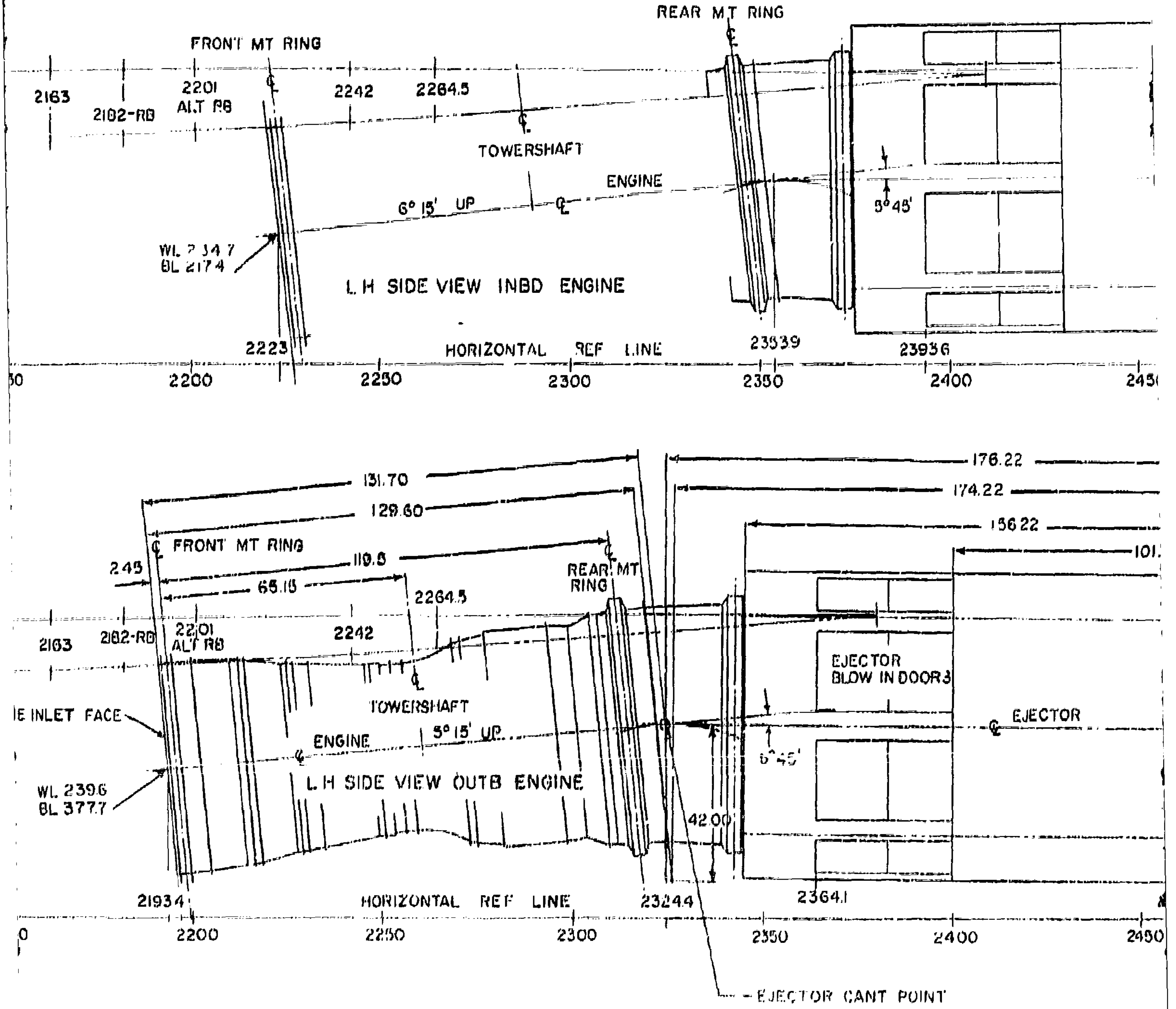


PROPOSED TURBOJET ACCESSORY ARRANGEMENT

Figure 1-43

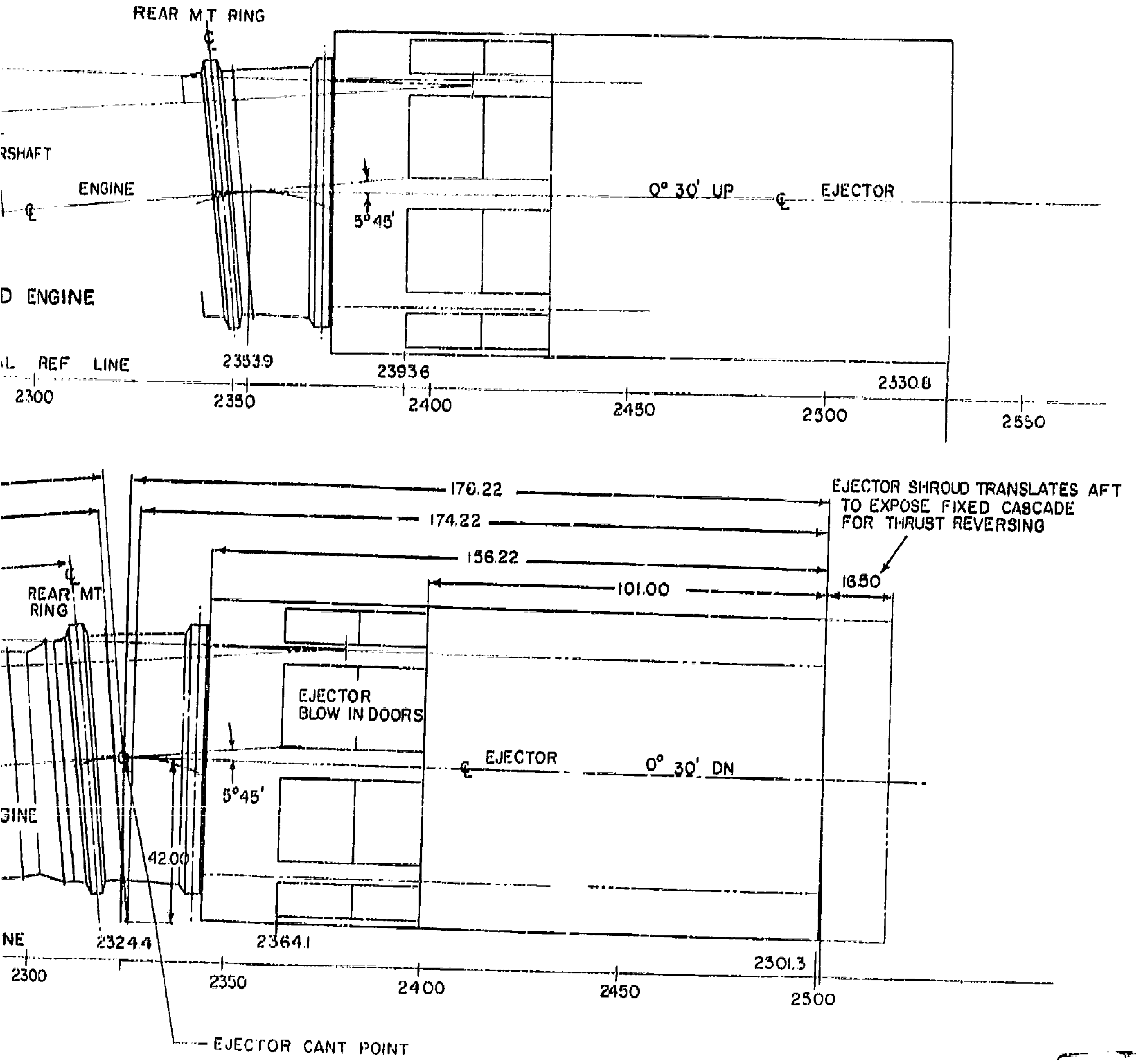

ESTIMATION OF WING CONTOUR
AT FS 2393.6
2234.5
ACCESS. DRIVES
℄
2234.5
90°
℄
TOWERSHAFT
EJECTOR
EXIT
ENGINE CANT
ENGINE
INLET
OD OF REAR MT. RING
BL 217
2163
2182-RB
2150
2200
WL 234
BL 217


ESTIMATION OF WING CONTOUR
AT FS 2364.1
2234.5
ACCESS. DRIVES
℄
2234.5
90°
℄
TOWERSHAFT
ENGINE CANT
76.00
ACROSS
FLATS
EJECTOR EXIT
ENGINE
INLET
OD OF REAR MT. RING
BL 381
℄ FRONT
245
2144
2163
2182-RB
2201
ALT RB
ENGINE INLET FACE
WL 239.6
BL 377.7
2193.4
2150
2200



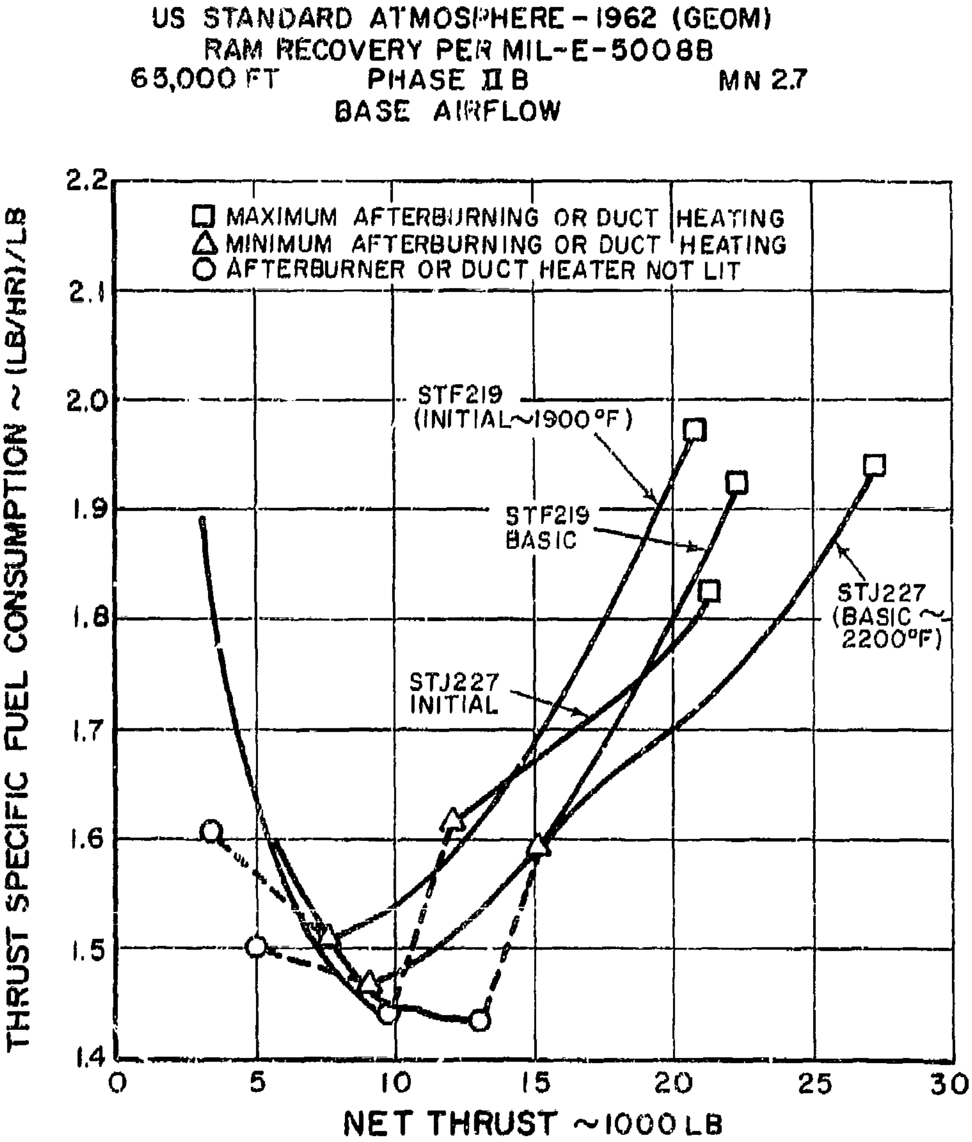
TYPICAL TURBOJET INST
EFFECT OF INCREAS

Figure 1



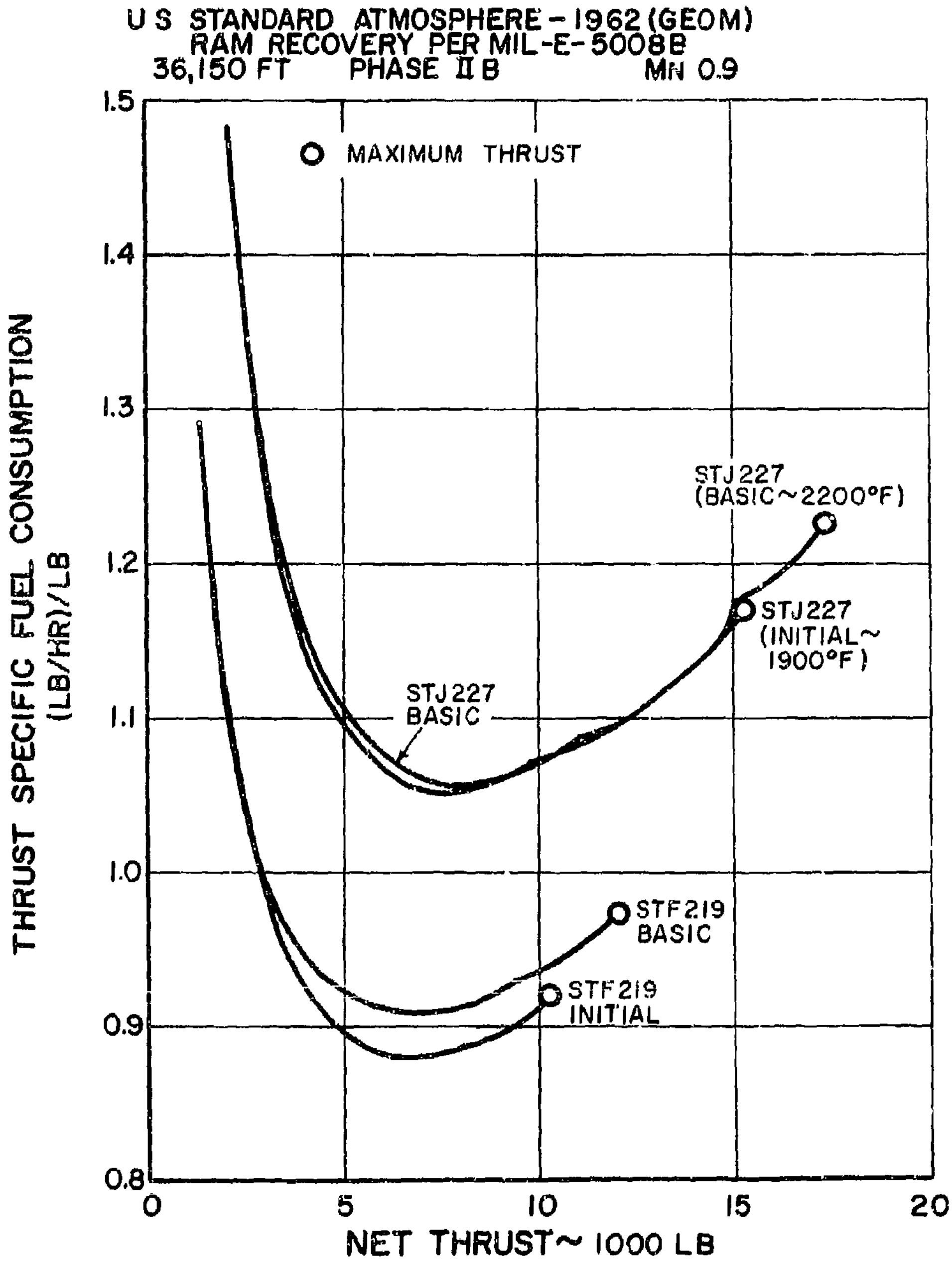
TYPICAL TURBOJET INSTALLATION SHOWING EFFECT OF INCREASED CANT ANGLE

Figure 1-44



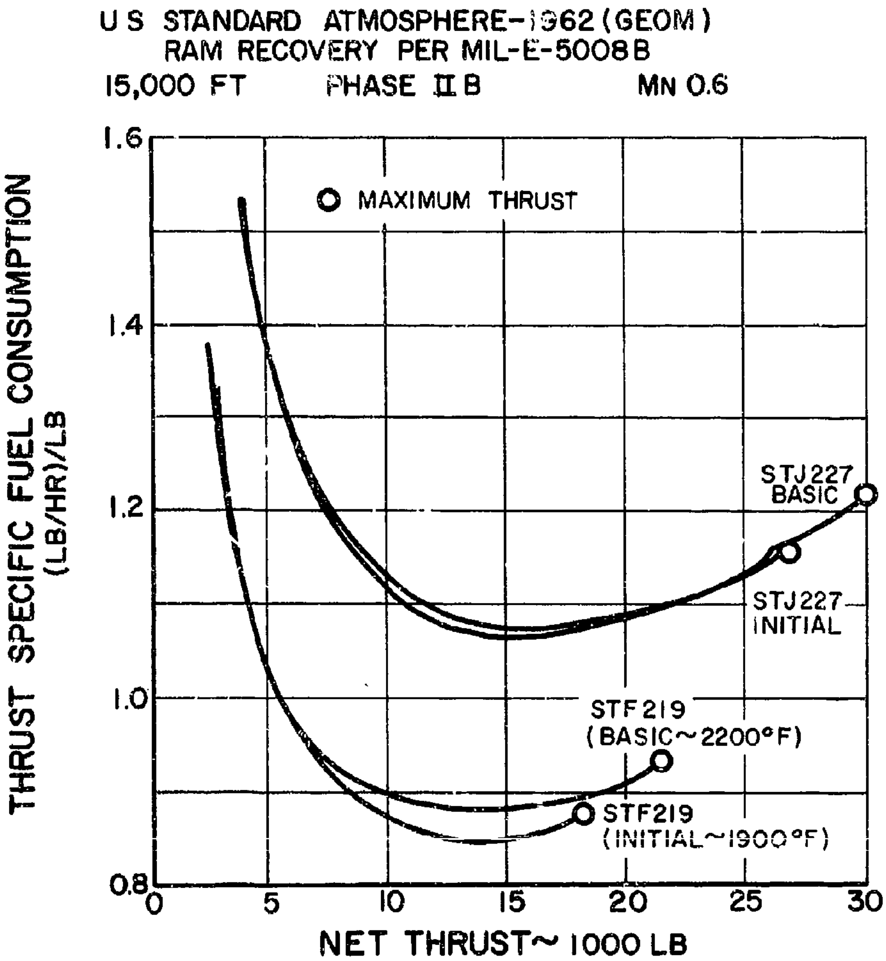
ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES AT MACH 2.7 AT 65,000 FEET

Figure 1-45



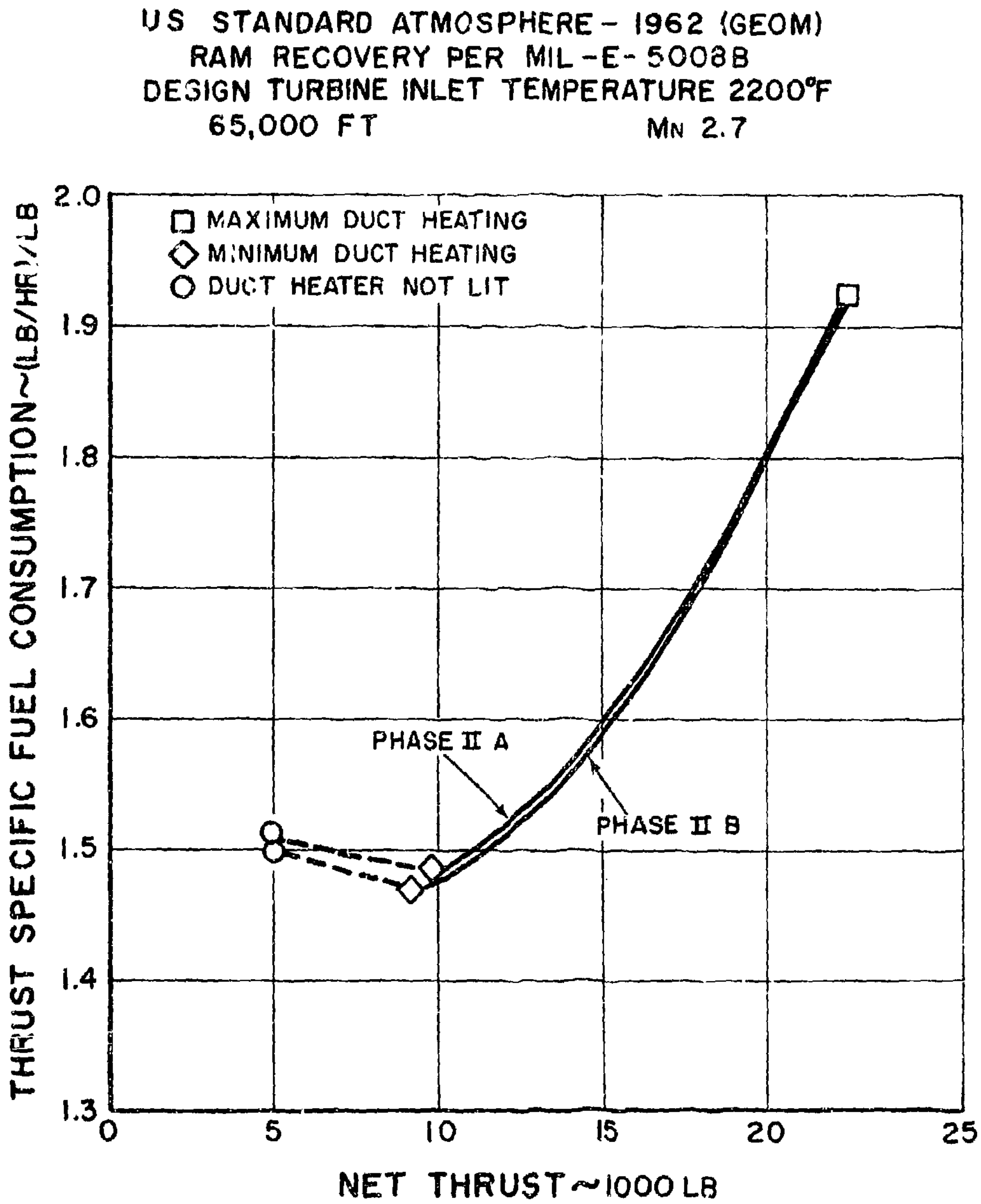
ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES
AT MACH 0.9 AT 36,150 FEET

Figure 1-46



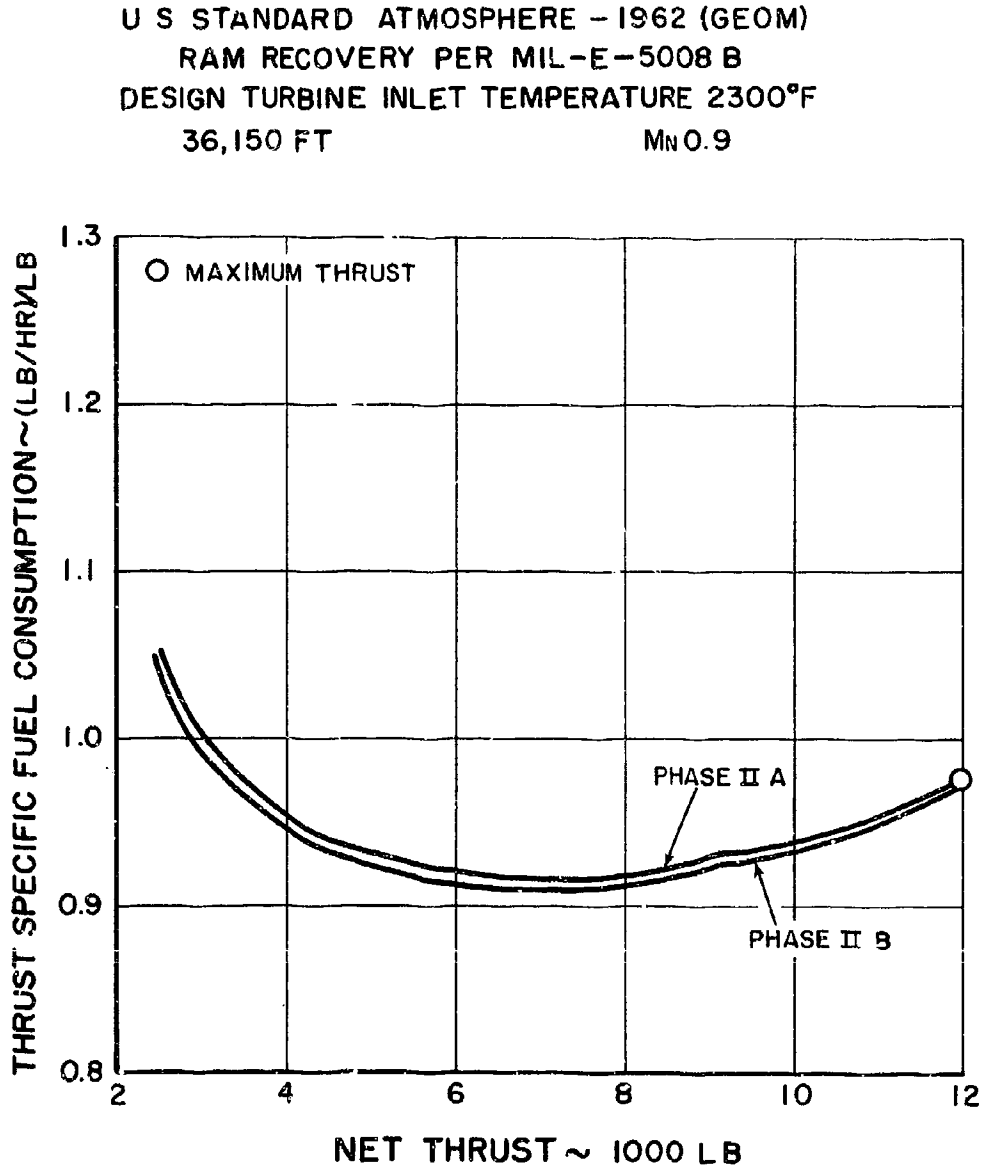
ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES
AT MACH 0.6 AT 15,000 FEET

Figure 1-47



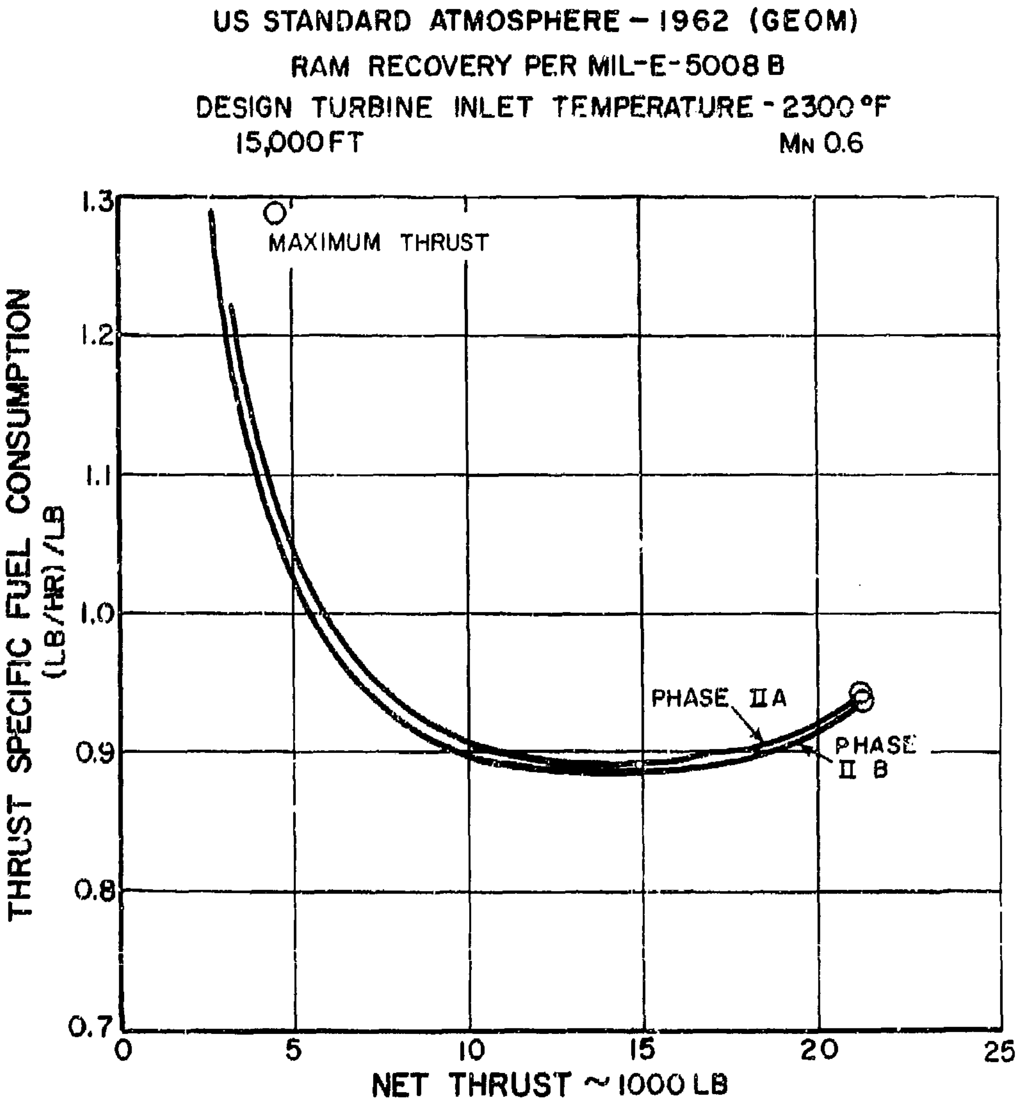
ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA AND AFTER PHASE IIB AT MACH 2.7 AT 65,000 FEET

Figure 1-48



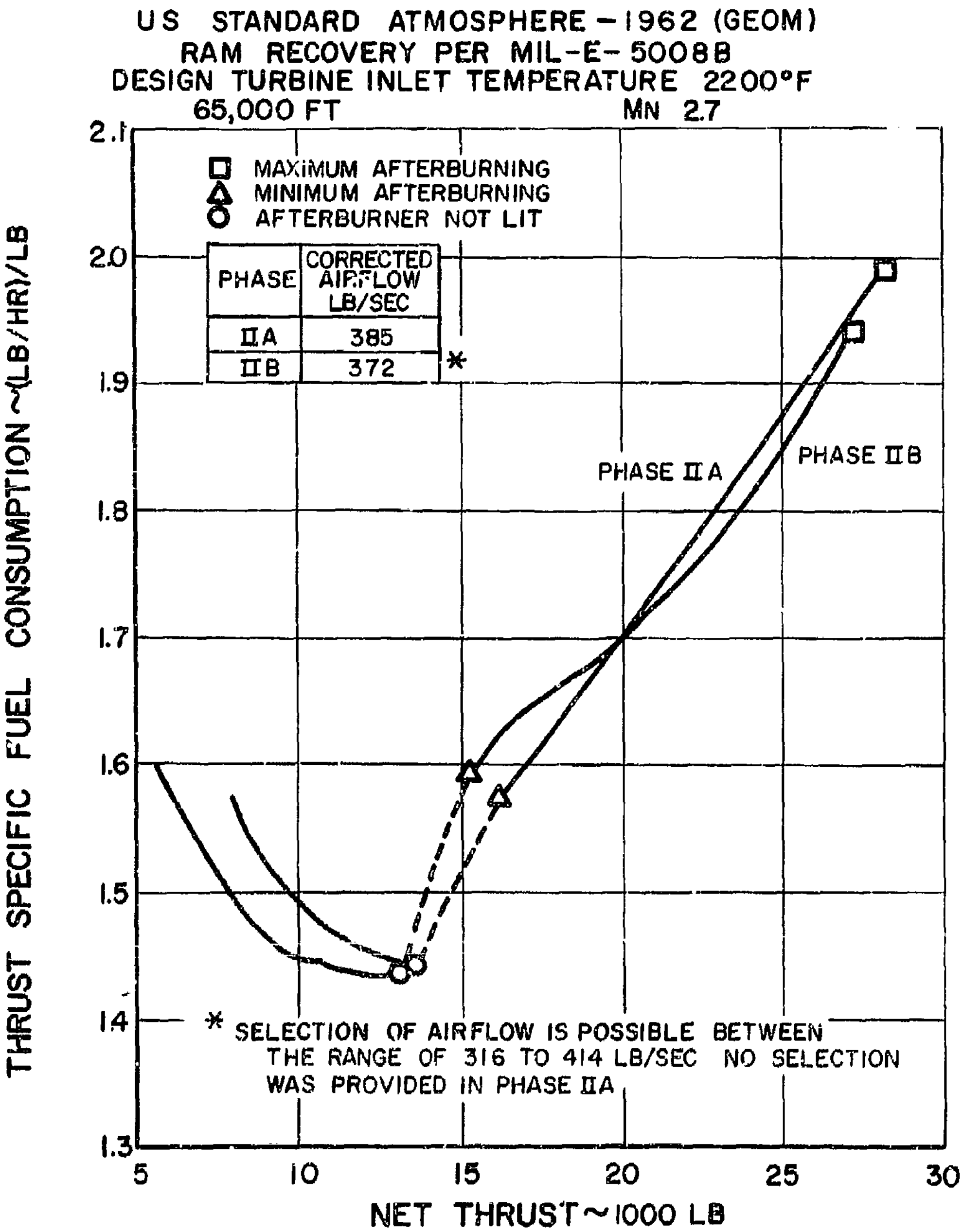
ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA AND AFTER PHASE IIB AT MACH 0.9 AT 36,150 FEET

Figure 1-49



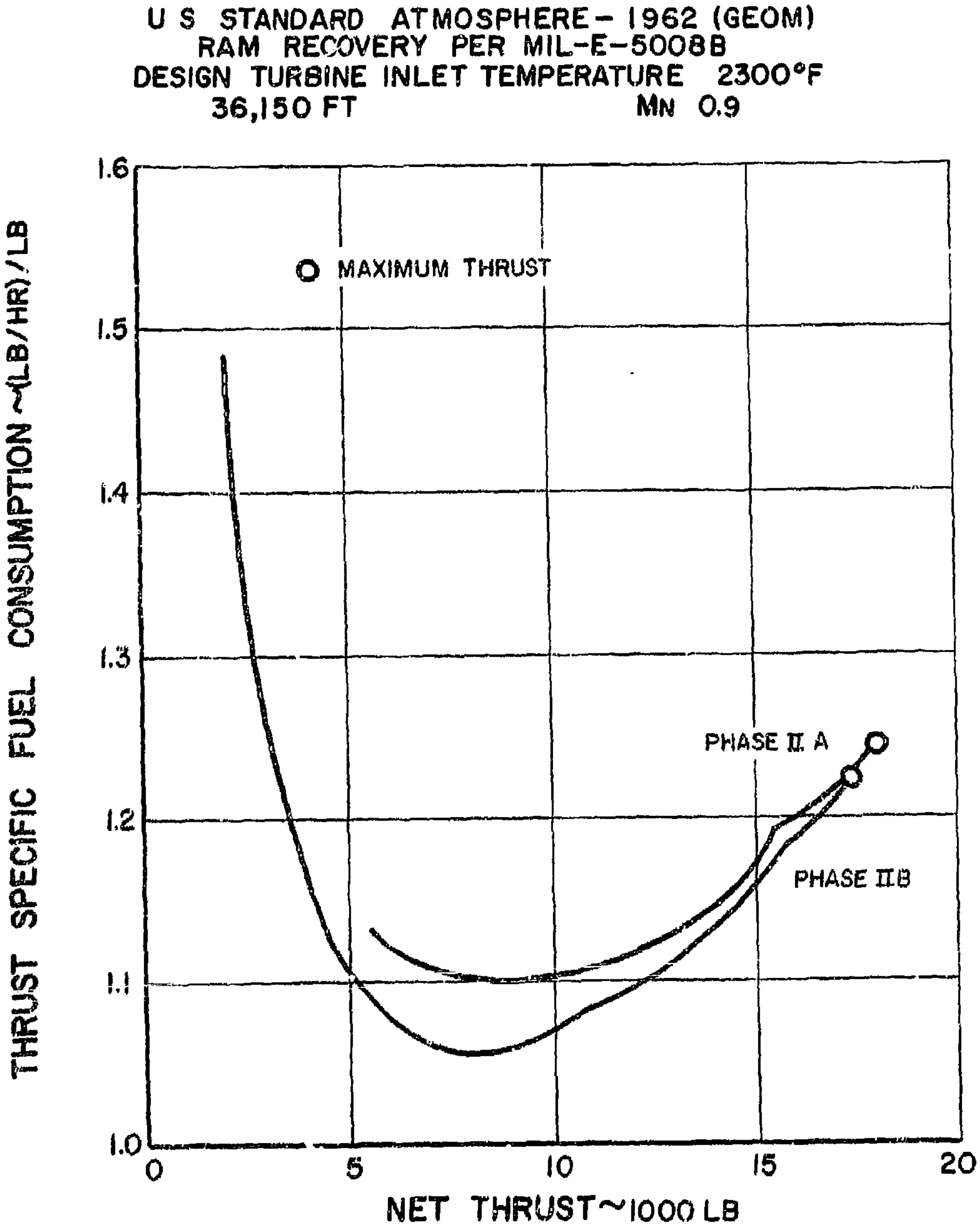
ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA AND AFTER PHASE IIB AT MACH 0.6 AT 15,000 FEET

Figure 1-50



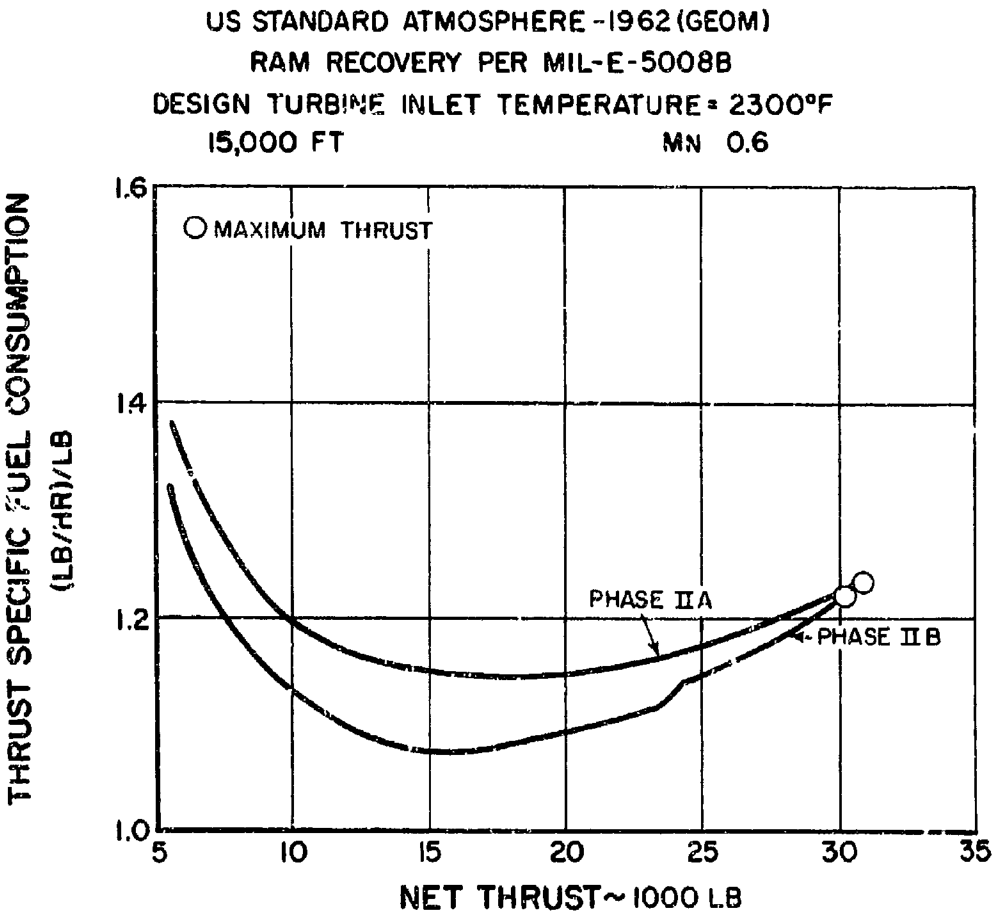
ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA AND AFTER PHASE IIB AT MACH 2.7 AT 65,000 FEET

Figure 1-51



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

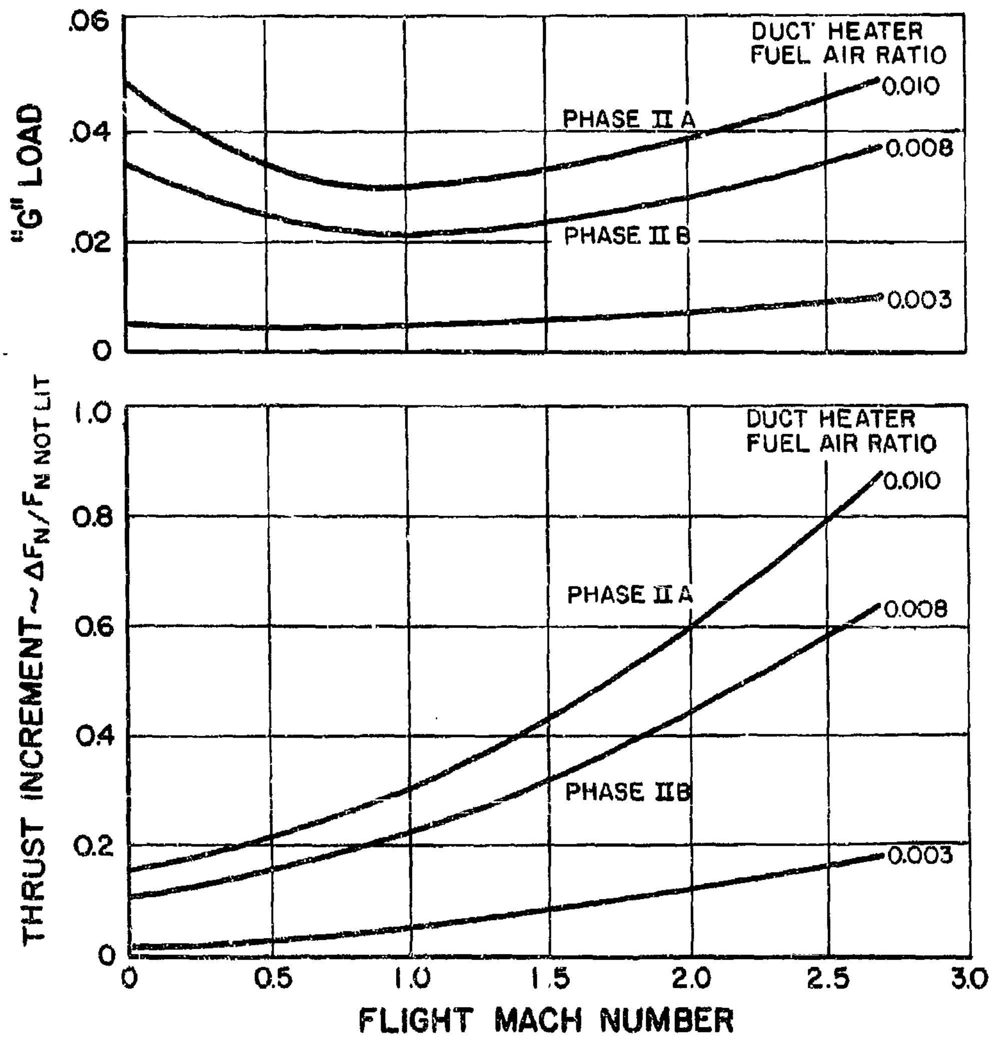
Figure 1-52



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

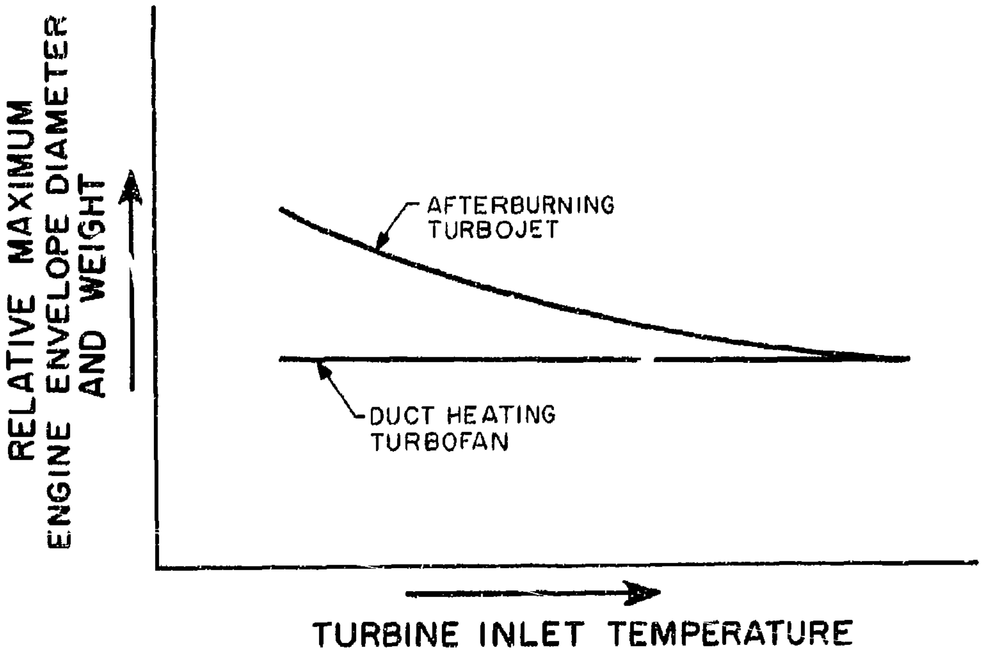
Figure 1-53

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



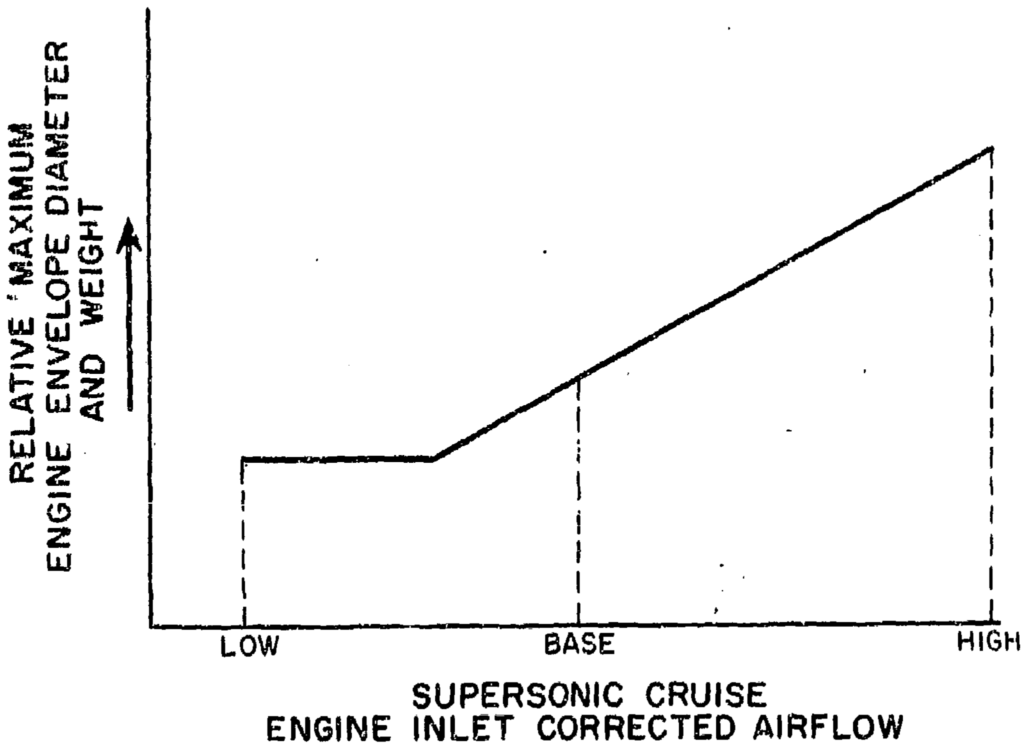
LOADING AND THRUST INCREMENT PRODUCED BY
LIGHTING DUCT HEATER OF STF219 ENGINE

Figure 1-54



EFFECT OF TURBINE INLET TEMPERATURE ON ENGINE
WEIGHT AND MAXIMUM DIAMETER

Figure 1-55



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

Figure 1-56