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PAGE
MODEL
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FIG. 2

ESTIMATED PARTIAL AFTERBURNING PERFORMANCE

J-58 ENGINE

M = 3.20

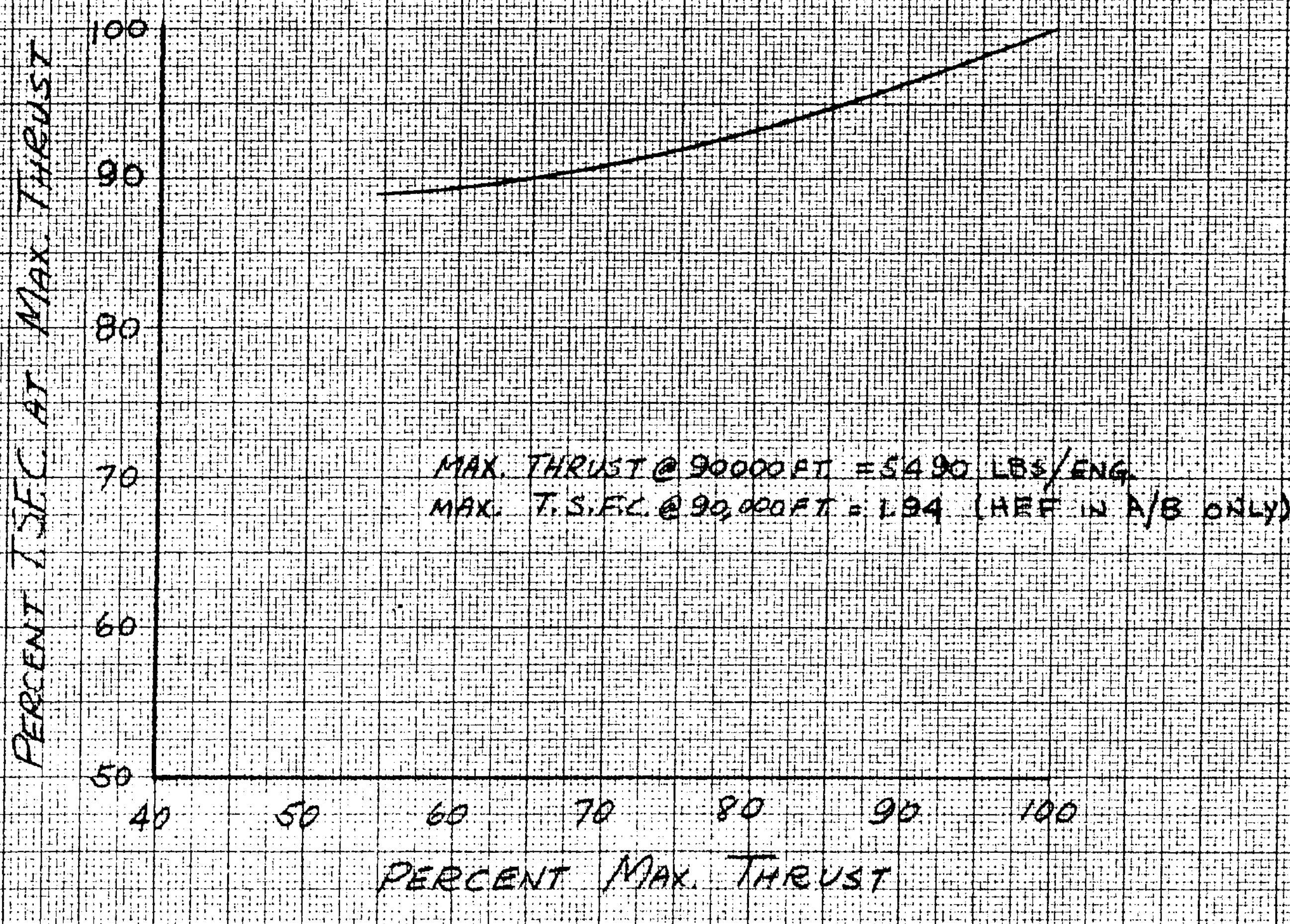

FIG. 2

FORM 5278 A

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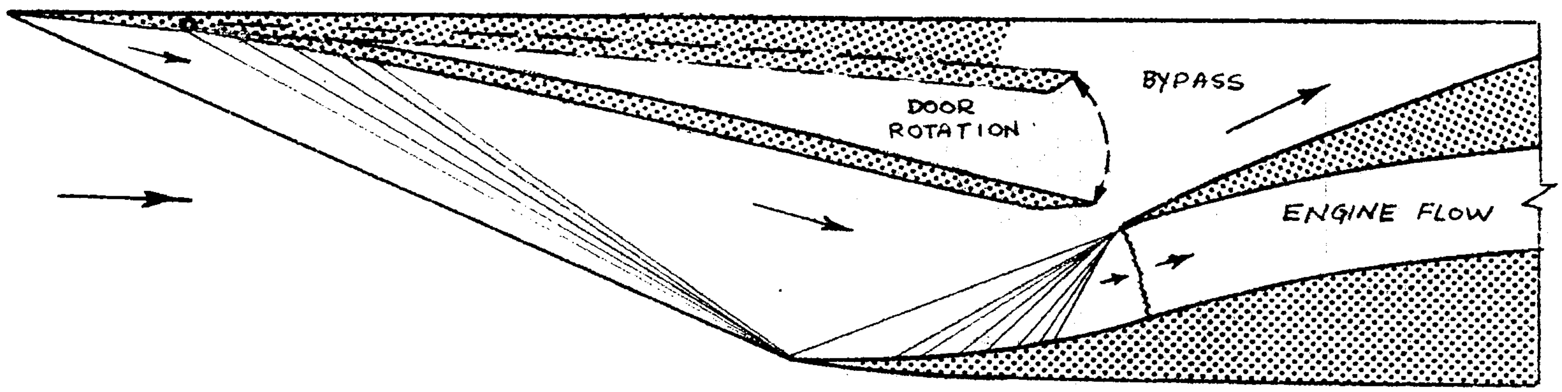
FIG. 3



SCHEMATIC OF 2-DIM. EXTERNAL-INTERNAL COMPRESSION INLET

FIG. 3

FORM 9767A-1

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Fig. 4

ESTIMATED INLET PERFORMANCE

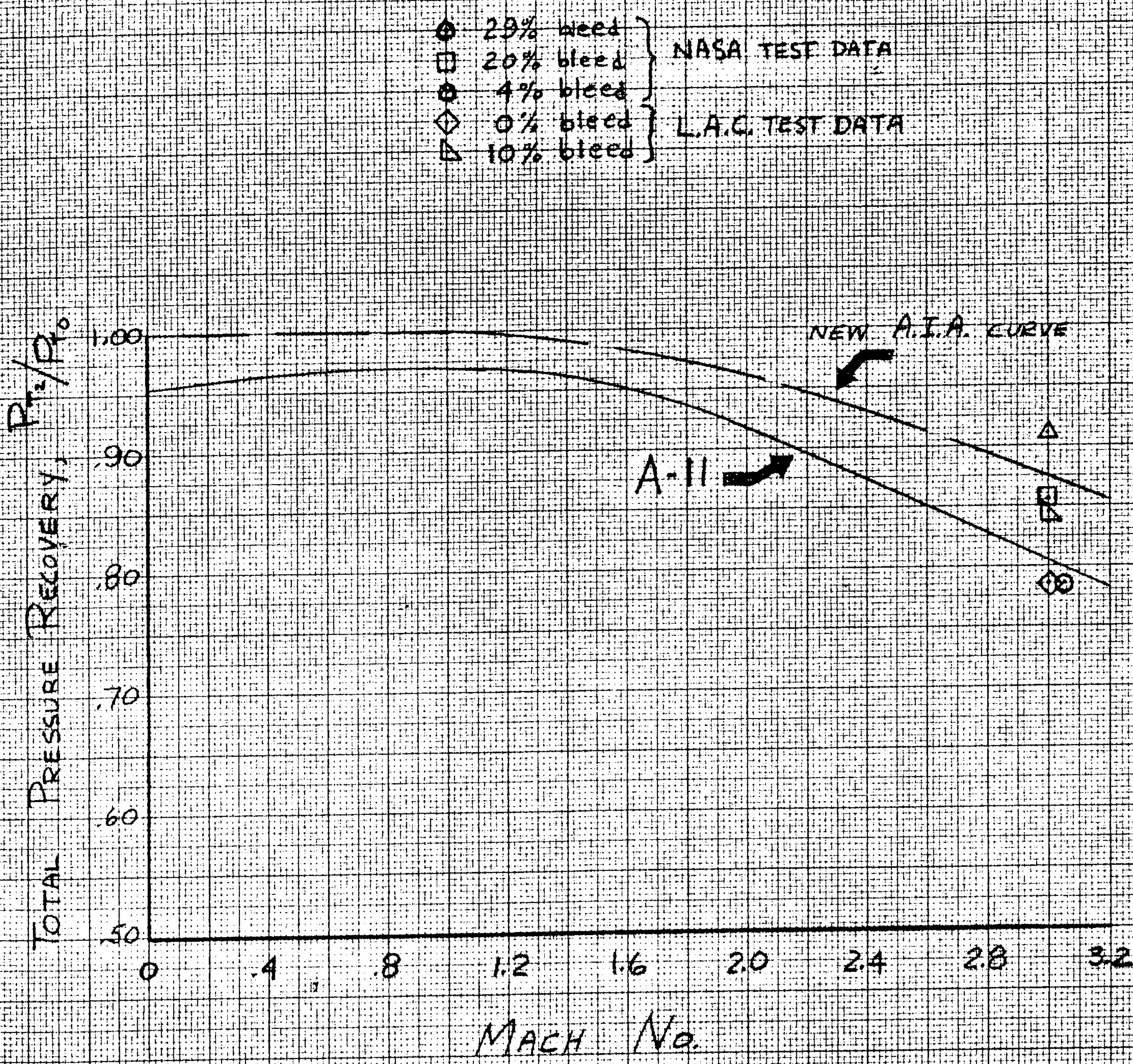

FIG. 4

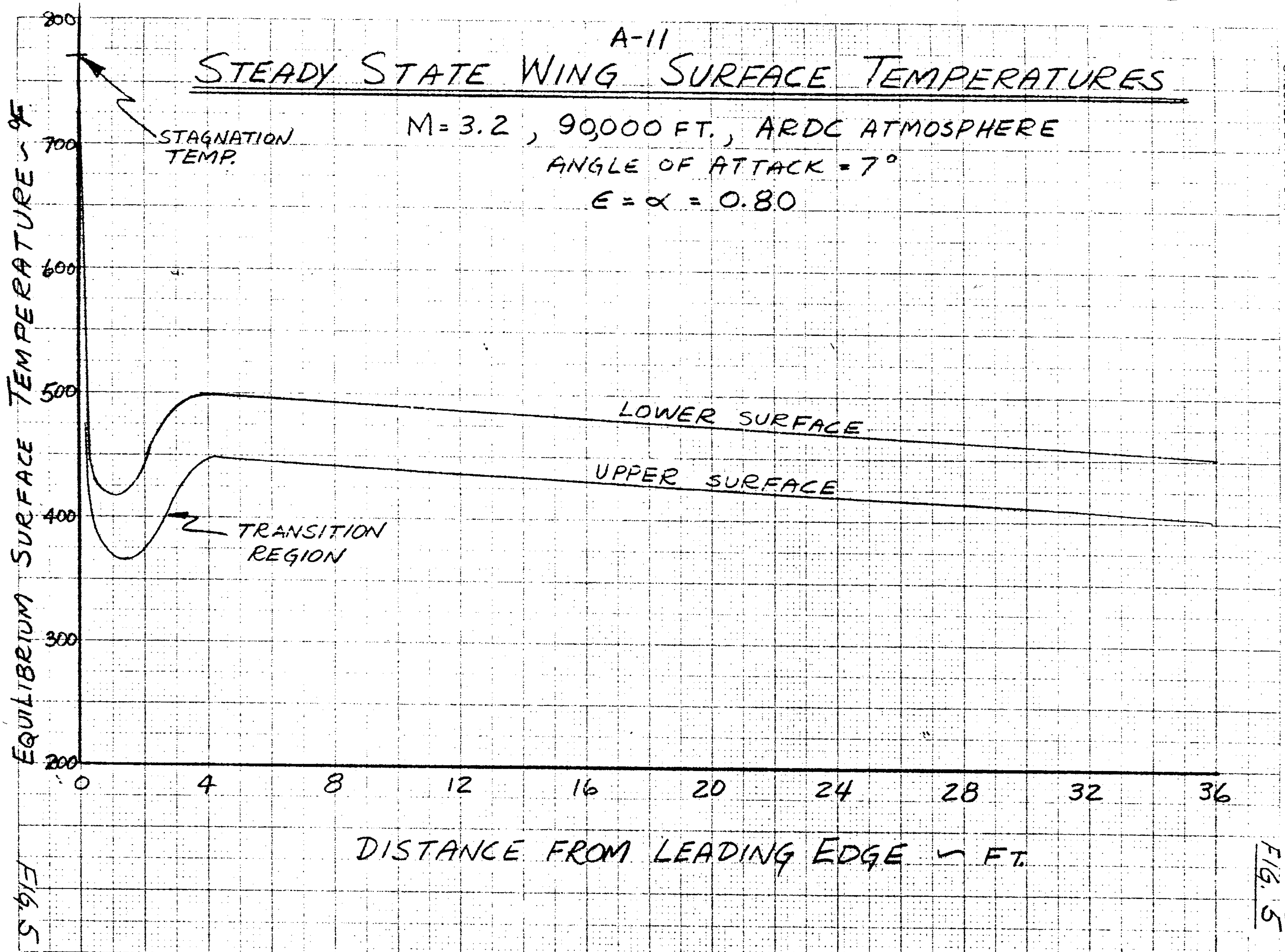

FIG. 5

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FIG. 6

FUEL AND WING SKIN TEMPERATURES
INTEGRAL TANKS

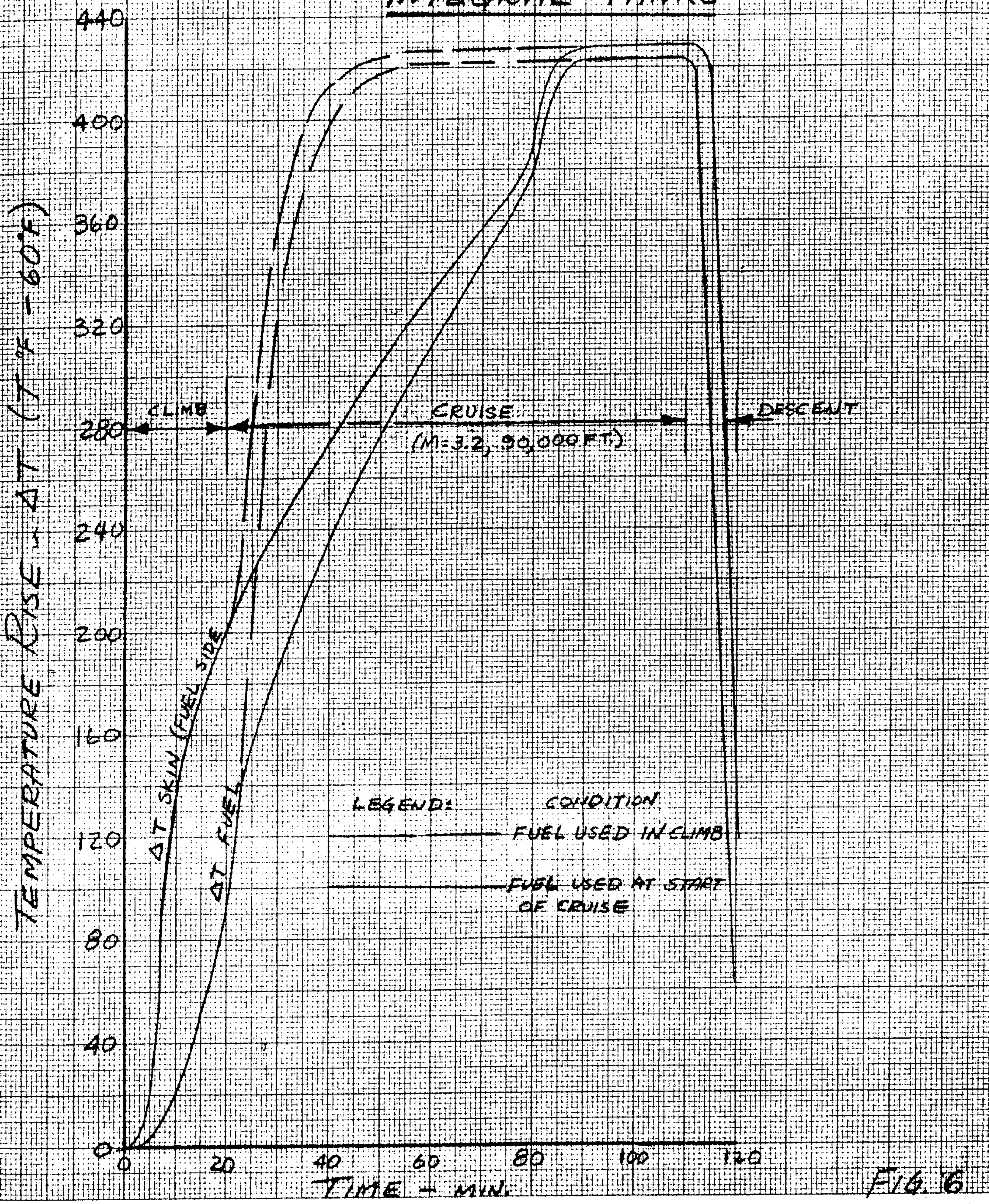

FIG. 6

ESTIMATED FUSELAGE FUEL AND WALL TEMP.

FUEL IN BAGS — NO INSULATION
M = 3.2 CRUISE, 90,000 FT.

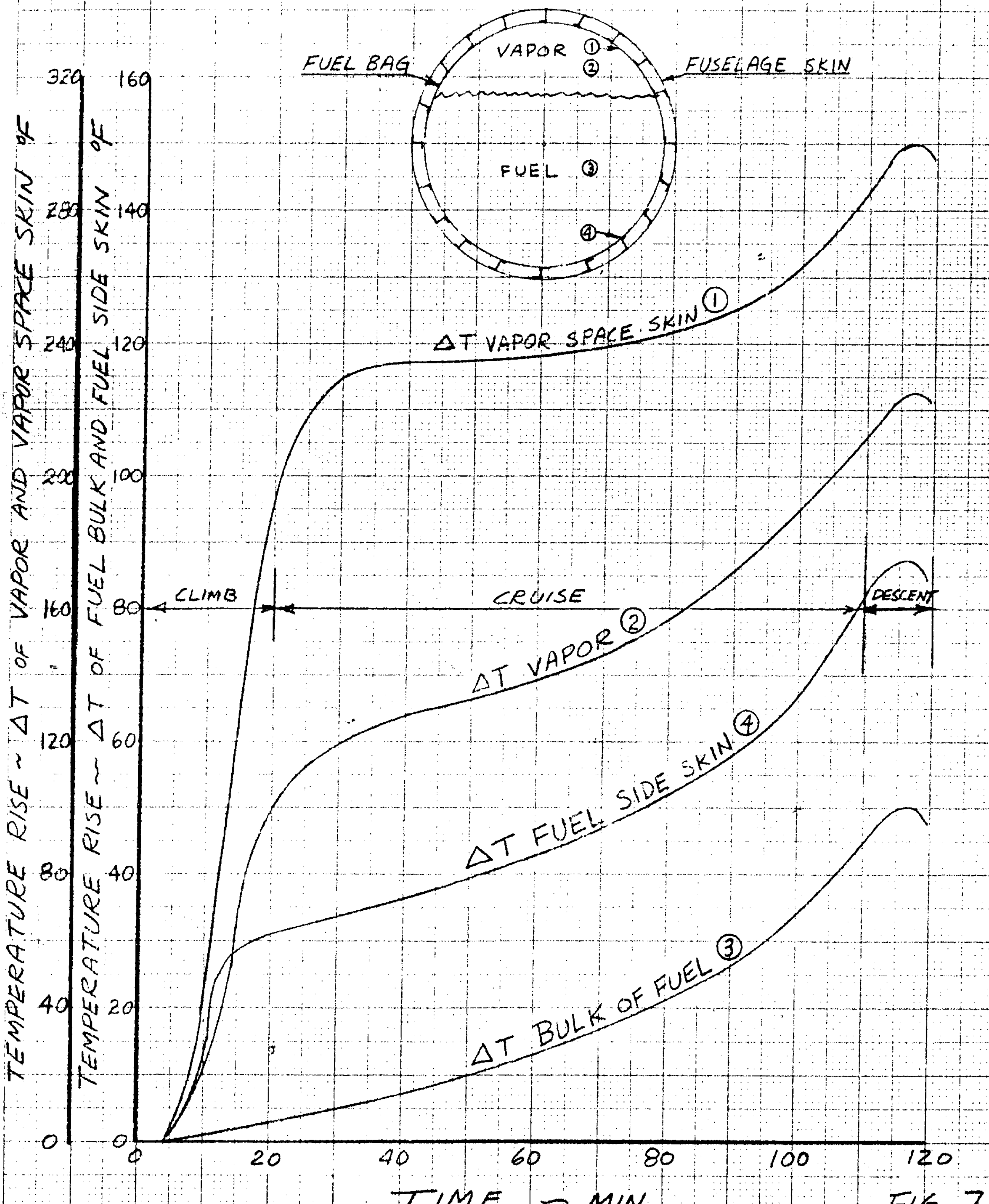

FIG. 7

NAVIGATION SYSTEM

Considering the high speed capability of this aircraft, approximately 14/30 nautical miles per minute, an automatic navigation system becomes almost mandatory. It is impracticable to expect the pilot to determine his position by conventional methods of sun sights and drift readings and perform his normal duties and properly fulfill the mission.

To this end we propose to install an inertial reference guidance system. At this time, several manufacturers such as Nortronics Company, Kearfott Company, Minneapolis Honeywell, and the Autronics Company are designing and developing components and complete navigational systems of the inertial reference and stellar inertial types.

The stellar inertial systems are all basically the same with a daylight star tracking telescope mounted on a gyro stabilized platform which is constantly aligned to the local mass attraction vertical. This type of instrument constantly corrects the gyro stabilized platform drift by a program of star sights; thus, the position error will never exceed 1 nautical mile C.E.P. See Figure 1.

The weight of such a stellar inertial system would be approximately 250 pounds.

The simpler type system herein proposed utilizes a stable platform with a three gimbal-three axis assembly aligned to the local mass

NAVIGATION SYSTEM (CONT.)

attraction vertical. If we utilize the stabilized platform as a reference for determining position, and since this platform will be uncorrected during the mission flight, it will build up a position error at the end of a 2.5 hour flight of 2.9 nautical miles C.E.P.

This error is derived from a position error drift rate of 1 nautical mile per hour and an assumed initial datum error of 2,500 feet. See Figure 1.

As the MK III driftsight will be used for final pin-pointing of a target, a position error of 3 nautical miles will be more than satisfactory.

The pilot will have a display panel showing distance traveled and present position in latitude and longitude; he will also have an instrument showing true heading.

The weight of this type of system complete, including electronics and computer, will be approximately 150 pounds.

This automatic navigational system will be supplemented by the following components:

1. MK III Driftsight.
2. MA-1 Compass System.
3. ARN-44 Radio Compass.
4. AN 5766-74 Standby Compass.

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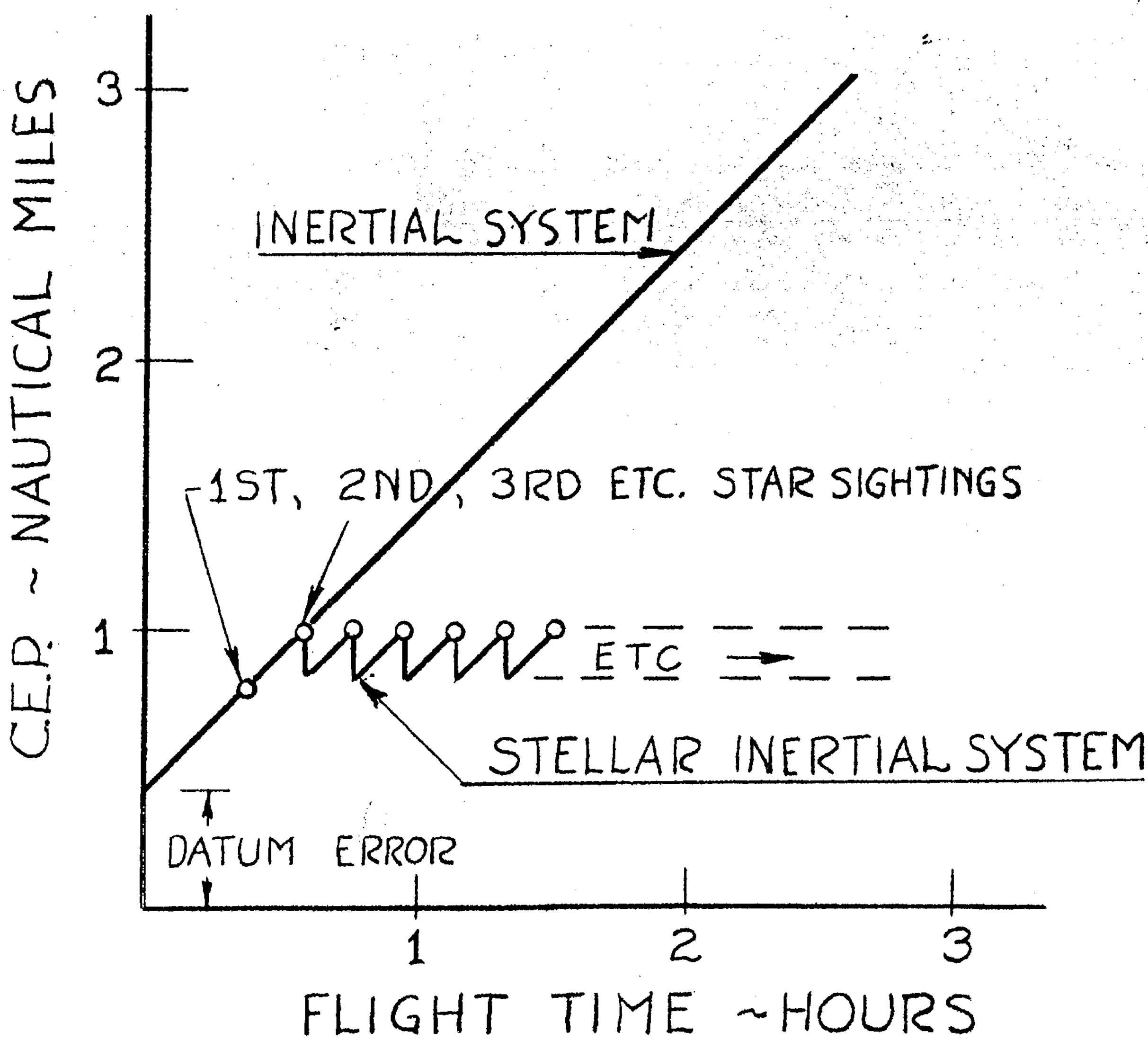
FIGURE 1

POSITION ACCURACY VERSUS TIME



CONTROLS

Cockpit

The flight controls in the cockpit are of the conventional rudder pedal and control stick arrangement. The rudder pedals are also used in the normal manner to apply brakes. Movements of the control stick are mixed mechanically in the cockpit for elevon control. After the control signal is mixed, the control system for each elevon is separate and independent of the other elevon permitting pitch and/or roll control, depending on stick position.

Cable Systems

The pilot forces are transmitted from the cockpit to the boosters by control cables. The rudder system has single cables for each direction of movement. Each elevon has two (2) control cables for each direction of movement; either of the two (2) cables in these dual systems can carry the full pilot load.

These cable systems include tension regulators to maintain a nearly constant rigging tension regardless of changes of the airframe due to variations in temperature.

Boosters

The hydraulic boosters are located at the control surfaces, one at each elevon and one at the rudder. The boosters are all irreversible

CONTROLS (CONT.)

with pilot feel being supplied artificially. Each booster has dual control valves, dual cylinders and is supplied hydraulic pressure from two (2) independent systems. The failure of one (1) control valve, cylinder or hydraulic system will not prevent operation of the control surface with the remaining system.

Control surface trimming is accomplished by actuators at each control surface booster which change the relationship between the zero artificial feel position and the control surface position.

HYDRAULICS

The hydraulic system design shall conform to the requirements of MIL-H-5440B, except the system operating pressure will be 4000 psi. A fuel oil cooler will be provided to maintain a maximum system temperature of +400°F. Low temperature operation will be -65°F.

Monsanto OS-45-1 hydraulic fluid is expected to be used to meet the high temperature (+400°F) requirements. Considerable experience has been gained with this fluid and it is compatible with present hydraulic system components with a minimum of system modification.

The design of the hydraulic system without a cooler will be considered and investigations will be made to determine feasibility of using a +700°F system. Fluids under consideration will be General Electric Versilube f-50 and turbine engine oil.

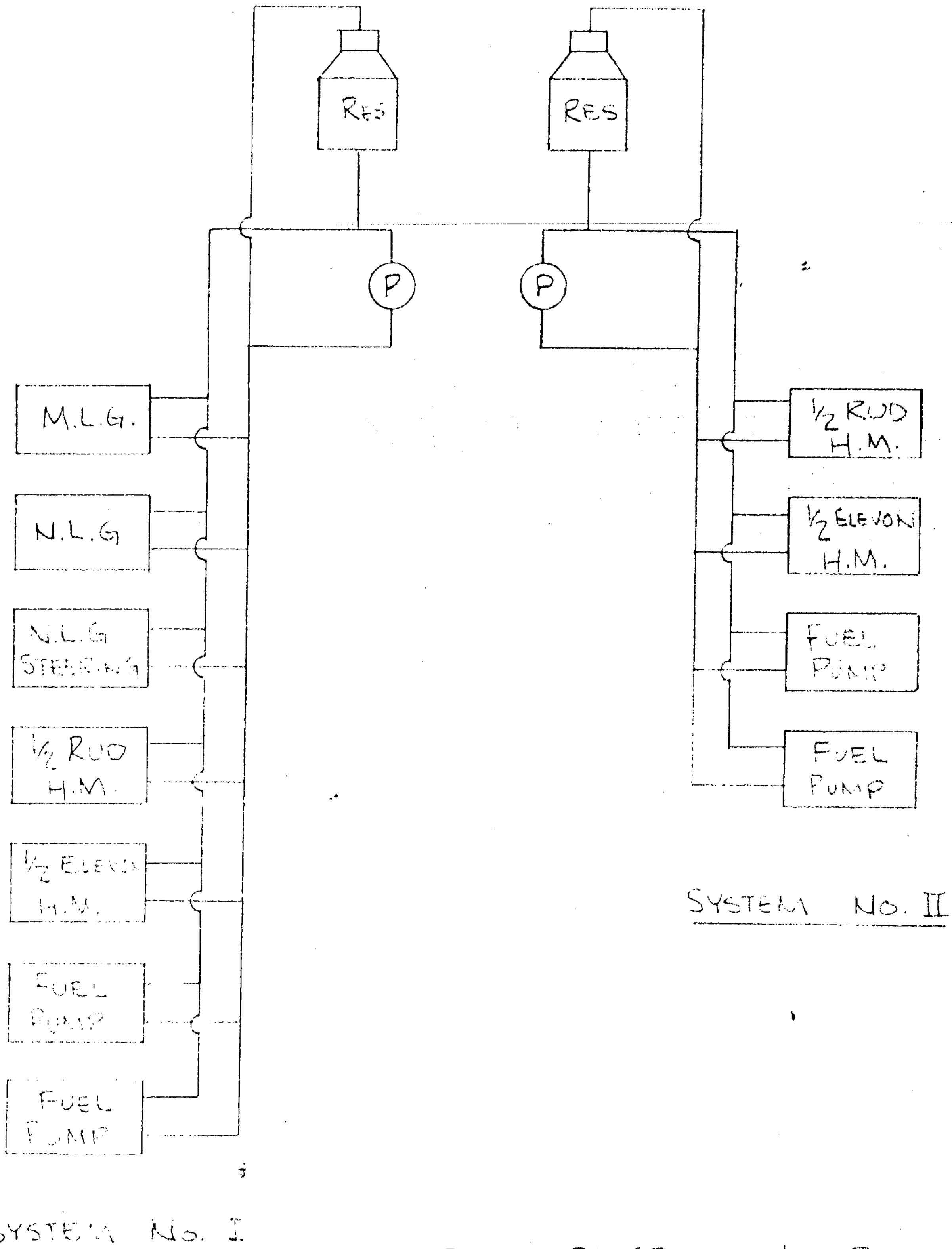
The hydraulic pumps will be engine driven and variable delivery type.

Dual systems will be provided with each system supplying one half the power requirement. System No. I will operate the landing gears, nose wheel steering, two (2) fuel pumps and one half the required hinge moment for the rudder and elevon. System No. II will supply power to two (2) fuel pumps and one half the required hinge moment for the rudder and elevon control surfaces.

HYDRAULICS (CONT.)

The fluid reservoir shall be the airless type. The return system will be closed with returning oil being directed to the pump inlet and the reservoir acting as a low pressure accumulator.

Pressure lines will be 304-1/8 stainless with steel fittings. Line connections will be flareless type fittings in accordance with the MS standard.



BLOCK DIAGRAM - HYDRAULIC SYS.

ELECTRIC SYSTEM

Introduction

Special consideration is given to the design problems of the electric system which exist due to the high operating altitudes and the supersonic flight conditions of the aircraft.

High temperature is the basic electrical problem associated with supersonic speed. It causes physical and/or electrical changes in the materials and equipment used in the system. Wire resistance increases, the volume-resistivity of insulation materials is lowered, and the magnetic characteristics of electrical irons and steels change as the temperature increases.

Since uncooled, high-temperature operation electrical systems are not available, all possible electrical and electronics equipment is installed in pressurized and cooled compartments.

Where required, high-temperature components such as the following will be used:

1. Nickel-clad copper wire and lugs.
2. Teflon fiberglas clamps.
3. High-temperature, environmental type connectors. A special HR series using ceramic inserts and crimped silver-alloy contacts is satisfactory at 1000°F.

ELECTRIC SYSTEM

Introduction (Cont.)

4. MIL-R-25018-Cl.C relays which are miniaturized, hermetically sealed, and rated for continuous operation at 392°F.
5. MIL-R-6106-Type I - Cl. D10 relays which are rated for continuous operation at 482°F and 100,000'.
6. Special high-temperature and hermetically sealed switches.

Another problem associated with supersonic flight which has an effect on the electric system and component design, is that of the so-called "white noise" - the noise level which is estimated at 150db. The basic effect is unusually high induced vibration loads which are minimized by adequate acoustical vibration and insulation techniques.

High altitude operation presents many electric system problems and Lockheed has had considerable experience with high altitude aircraft. Corona has deleterious effects on wire insulations and it increases the hazard of arc-over or voltage breakdown at altitude. Another undesirable side effect of corona is the radio noise problem created. Any damaging effects of ozone concentrations on the materials will be evaluated; however, the associated high stagnation temperatures will considerably reduce this problem. Also, lower ionization potentials are required at altitude.

AC and DC generators are available which are brushless and oil cooled with integral oil pumps. These generators are lightweight and permit high

ELECTRIC SYSTEM

Introduction (Cont.)

temperature and high altitude operation. A 28V DC system has been selected for this aircraft based on known and expected aircraft and military equipment loads. The corona and ionization problems at altitude are considerably reduced by using low voltage. Also, a DC system is simpler than an AC system, since there are no frequency or phasing problems; therefore, it is inherently more reliable,

DC System

Two engine mounted 150A oil cooled, brushless DC generators supply power to the monitored and essential DC busses. Either generator can supply the total electric load. Should both generators fail, a 20 A-H silvercel battery will provide essential DC power for approximately 30 minutes.

COMMUNICATION SYSTEM

AN/ARN-44 Radio Compass

The purpose of the radio compass is to guide the aircraft to a radio transmitting station at its destination, or, as a navigational aid to take bearings on such stations. An ID-250 indicator calibrated in degrees azimuth, continuously indicates the direction of the station with respect to the heading of the aircraft. In addition, it may be used as a radio communications receiver.

The AN/ARN-44 is identical to the AN/ARN-6 except the 100-200 KC band is replaced with a 2.0-3.5 MC band. The 2.0-3.5 MC frequency range covers marine transmissions such as ship-to-shore, Coast Guard communications, and distress calls. It also covers aircraft communications and certain other transmissions.

AN/ARC-62 Command Communications Set

This radio set provides voice and code transmission and reception in the UHF 225-400 MC frequency range, and will probably replace the AN/ARC-34 as the standard military UHF command set. Superior communication capability is available by virtue of the 3500 low-distortion voice channels, and in the extra-range margin provided by the 30 watt heavy-duty transmitter and the highly sensitive receiver. Modular construction allows rearrangement for different configurations in encased widths from 6 1/4" upwards and with a standard height of 6 1/2". The set has a volumetric size of approximately one half cubic foot and weighs less than the AN/ARC-34.

COMMUNICATION SYSTEM

AN/ARC-62 Command Communications Set (Cont.)

An improved control box is used in conjunction with the AN/ARC-62 receiver-transmitter to provide channel selection versatility. Twenty channels can be preset on a memory drum in a matter of seconds and the pilot may choose channels either from the preset number or by setting the five digits of the 3500 possible manual channels. The automatic tuning system operates so rapidly that the pilot is on the air with voice communications completely established in less than 4 seconds after making his selection.

AN/AIC-10 headset, microphone, and interphone control components are installed and used with the UHF communication set.

JP-150 MISSION

It is of interest to determine the effect upon airplane performance of using only hydrocarbon fuel. Flight testing of airframe, engine and equipment and crew training as well as some tactical missions can be conducted on a more economical basis with the less exotic fuel.

To accomplish the identical mission radius of the HEF equipped airplane requires a fuel load of 55,330 pounds with a take-off weight of 92,130 pounds. These numbers are 7,330 pounds greater than the HEF equipped airplane. However, the basic airframe will accommodate the greater weight of fuel at the lesser average density because sufficient fuselage diameter and length have already been established by payload and balance considerations.

The increased take-off weight results in a take-off ground run of 2,900 ft. The landing weight is not affected so that the landing distance remains 2,700 feet. The initial penetration altitude is reduced 1,700 feet and the target altitude is reduced 800 feet, also by virtue of the increased flight weight. The performance is otherwise unaffected by the sole use of JP-150 fuel.

It is noted at this point that the use of JP-150 exclusively does not show up to be as much of a disadvantage as might at first be expected. This comes about because the fuselage size and length required by payload and balance requirements can hold more fuel than is compatible with attaining the highest possible altitude at a 2,000 n. mi. radius using the HEF fuel

JP-150 MISSION (CONT.)

combination. It therefore appears that the basic airplane (Ref. Figure 1 in "Performance Section") could be overloaded with an HEF fuel combination of 55,330 lbs. With this overload of fuel the mission radius will improve to approximately 2,250 n. mi. with about the same altitude profile as attained with JP-150 fuel alone.

Lockheed Aircraft Corporation

CALIFORNIA DIVISION

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PREPARED BY R R Boehme
R. F. Boehme, et al

[redacted]

[redacted]

Approved
[redacted]
Clarence L Johnson
Clarence L. Johnson
Vice President
Advanced Development Projects

REVISIONS

DATE	PAGES AFFECTED

FORM 402-1

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APPENDIX

TABLE OF CONTENTS

A-11A SUMMARY

The A-11A airplane presented in this appendix is proposed ONLY in the event that the more suitable Pratt & Whitney J-58 engines should be unavailable for use in the A-11 airplane. The General Electric J-93 engine is the only other potentially available engine in this speed and altitude regime. While not as outstanding as the J-58, the J-93 nevertheless can be used in the design of a vehicle with quite respectable performance.

The A-11A airplane is designed around two (2) General Electric J-93 afterburning engines using HEF type fuel in the afterburners and JP-150 in the engines. The fuel load is approximately 65% HEF and 35% JP-150. Below 10,000 feet no HEF fuel is burned in order to avoid undesirable smoke and contamination.

The airplane has a 2,000 n.mi. mission radius at Mach 3.2 and crosses the target at 91,000 feet as shown in Figure 1 in the "Performance" section of this Appendix. This target altitude is 3,300 feet lower than for the J-58 powered airplane as shown in Figure 1 in the "Performance" section of the main Report.

The configuration is as shown in Figure 1 in the "General Description" section of this Appendix. This configuration is essentially the same as for the A-11 airplane except that it is scaled down, as practical, so as to

A-11A SUMMARY

be compatible with the smaller J-93 engines. However, the fuselage diameter is not scaled down since the space provisions for the pilot and payload is considered to be a practical minimum on the A-11 airplane.

In the "Alternate Fuel" section of this Appendix it is shown that the A-11A airplane can use JP-150 entirely and accomplish the same 2,000 n.mi. mission radius at approximately 1500 feet less altitude at start of cruise and reaching 90,200 feet over target. This altitude performance with JP-150 fuel is 300 feet less over target than the A-10 airplane presented in February 1959. The A-11A airplane, using only JP-150, is essentially the same as the A-10 airplane. However, the fuselage of the A-11A airplane is 3 1/2" larger in diameter than the fuselage of the A-10, resulting in a slightly lower lift/drag ratio for the A-11A airplane.

A-11A GENERAL DESCRIPTION

The A-11A airplane is a very high altitude Mach 3.2 reconnaisance vehicle designed to perform the same mission as the A-11, but at slightly lower altitudes, using J-93 engines.

The configuration is identical to the A-11, except that wing area is decreased by 200 sq.ft. and fuselage length reduced slightly. Military equipment bay, pilot's compartment and airplane equipment provisions are dimensionally identical to the A-11 airplane.

Structural arrangement and airplane systems are also the same as proposed for the A-11. The lighter and lower thrust J-93 engines result in a lighter airplane, as summarized below.

Weight Empty	32,415
Oxygen, Oil, Unusable Fuel	200
Pilot	285
Payload	500
Zero Fuel Weight	33,400 lbs.
Fuselage Fuel	32,000
Wing Fuel	14,000
Take-off Weight	79,400 lbs.