ORENDA PS 16

PRELIMINARY PERFORMANCE ESTIMATES AND MECHANICAL DESCRIPTION

SE 8-58

ORENDA

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ORENDA ENGINES LIMITED

MALTON ONTARIO

(Member A.V. Roe Canada Limited and Hawker Siddeley Group)

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INTRODUCTION

Project Study 16 is a small supersonic turbojet engine suitable for small engine applications. The engine incorporates a combination of the most advanced design features of the Orenda and Iroquois engines and reflects the latest turbojet design philosophies at Orenda Engines Limited. The single spool, eight stage, axial flow compressor is driven by a two stage turbine and the engine has a fully annular vapourizing combustor. There is an afterburner and a variable area final exhaust nozzle.

The engine is designed to meet the requirements of United States Military Specification MIL-E-5007A dated July 27, 1951.

This report is prepared to present the preliminary design specification and performance estimates together with a brief mechanical description of the engine. Two different afterburner designs and two exhaust nozzle configurations are described.

ORENDA P.S.16

PRELIMINARY PERFORMANCE ESTIMATES

AND

MECHANICAL DESCRIPTION

1. ENGINE SPECIFICATIONS AND PRELIMINARY PERFORMANCE ESTIMATES

	20 inch AB	23 inch AB
Weight (Including all engine accessories such as controls, oil tank, ignition, etc.)	560 pounds	560 pounds
Diameter (over compressor inlet flange)	19.8 inches	
Diameter (over combustor)	22.0 inches	
Diameter (over afterburner shroud)	20.0 inches	23.0 inches
Overall Length, to end of conical nozzle	113 inches	99.2 inches
Rotor Speed (sea level static, standard day)	16,000 rpm	
Mass Flow (sea level static, standard day)	50 lb/sec	
Pressure Ratio (sea level static, standard day)	8:1	
Turbine Inlet Temperature (sea level static, standard day)	1275°K	
Thrust - dry (sea level static, standard day)	3587 pounds	

	20 inch AB	23 inch AB
Thrust with Afterburner (sea level static, standard day)	4825 pounds	
Limiting Mach Number (max. allowable inlet temperature 556°K)	2.8	
Operational Altitude	75,000 feet	
Polar Moment of Inertia of Rotor	27 lbs ft^2	
Moment of inertia about any axis through C of G perpendicular to engine axis	3200 lb ft^2	
Moment of inertia about engine axis	220 lb ft^2	
Specific Fuel Consumption - dry	0.973 lb/hr/lb	
Specific Fuel Consumption with afterburner	1.84 lb/hr/lb	1.89
Oil Consumption	1.0 pt/hr	
Fuel Specification	MIL-F-5624C MIL-F-5616 MIL-F-5572A	
Oil Specification	MIL-L-7808	

2. MECHANICAL DESCRIPTION

2.1 General

This section is devoted to a description of the mechanical configuration of the engine and to information on installation and starting. The engine is described on a fore to aft basis.

2.2 Front Frame

The front frame is a cast magnesium structure comprising a conventional casing with four equally spaced faired struts supporting an integrally cast hub. The hub forms a housing which supports the front main rotor thrust bearing; it also forms a casing for the internal gears and acts as the main oil sump. The front main rotor thrust bearing is sealed on the rear face by a two-stage floating seal of the composite steel and carbon type. The cavity between the two-stage seal is pressurized with compressor third stage air.

A bevel gear mounted at the forward end of the compressor rotor front shaft, drives a mating gear on a shaft which runs (at reduced engine speed) through the lower frame strut to the engine auxiliaries gearbox which is mounted on the bottom of the front frame. The power take-off gearbox or hydraulic unit may be located on the forward side of the accessories gearbox. A drive is also provided at the front of the engine for a starter-generator should one be required for some aircraft installations.

Small amounts of air, that pass through the front and rear bearing seals into the sumps, are exhausted to atmosphere through a centrifugal separator built into the auxiliaries gearbox drive.

Precision cast variable incidence inlet guide vanes are mounted on bushings and operated by a unison ring that is mounted externally on the engine.

2.3 Compressor

The rotor of the eight stage axial flow single spool compressor is attached directly to the turbine on a single shaft assembly with the entire rotor assembly being supported by a main bearing at each end.

All stages of rotor blades and discs are steel. The stator blades in all stages are solid steel, brazed into outer blade rings which are mounted in "T" slots in the steel split outer casings.

2.4 Combustion Chamber

The combustion system is a fully annular vapourizing type. An external fuel manifold is employed to feed fuel through individual lines to sixteen vapourizing tubes. Two surface gap igniters pass through the outer combustion liner for ignition.

An air take-off, to supply air for aircraft use, is welded to the outside of the casing just aft of the front flange on the top centreline of the engine.

An optimum combination of standard stainless and high strength weldable stainless materials is employed for this welded fabrication.

2.5 Turbine

The complete two-stage turbine stator, rotor shroud and rear frame are clamped together by a single row of bolts. Both stator stages are nicrobrazed segments of precision cast blades, brazed to inner and outer mounting shrouds.

The turbine rotor is joined directly to the compressor rotor. The complete assembly is balanced as a unit and then broken at the joint at the front of the first stage turbine disc prior to final engine assembly.

The two turbine discs are steel forgings with extended necks and integrally machined air baffles. Precision cast blades with firtree roots are mounted in the discs. The rear bearing support shaft is attached to the rear face of the first stage disc. There is an insulating sleeve assembled between the shaft and the bearing race.

Compressor delivery air is used to cool the two turbine discs and the interstage spacer. Seal diameters are such as to keep the thrust load on the front bearing within limits.

2.6 Rear Frame

The rear bearing housing is supported by four fabricated, sheet metal, tangentially mounted struts to a sheet metal stiffening ring on a conventional fabricated sheet metal outer casing. The bearing housing and struts are enclosed in an integrally stiffened pressed sheet metal "bullet and vane" weldment. The outer ends of the vanes are supported to the casing by the struts near their leading edge, and by a pin arrangement near their trailing edge. The forward diameter of the bullet is connected, via a support diaphragm, to a flange on the bearing housing.

2.7 Afterburner

The 20 inch outside diameter afterburner incorporates a double fuel manifold and a double annular gutter stabilizer to obtain a peak combustion efficiency of 93 percent. The overall length and weight of this engine with a variable convergent nozzle are 113 inches and 560 pounds respectively.

For some installations, it may be preferable to shorten and increase the diameter of the afterburner. Approximately the same peak afterburner efficiency can be obtained from a 23 inch outside diameter afterburner shortened 4.8 inches. The overall length and weight of this engine configuration would be 108.2 inches and 571 pounds.

For installations in which length is critical, the 23 inch diameter afterburner can be shortened a further 9 inches at a cost of 5 percent in peak afterburner efficiency. At low jet pipe pressures, the efficiency of this afterburner also falls off more rapidly than either of the preceding ones. The overall length and weight of an engine with this afterburner are 99.2 inches and 560 pounds with variable convergent nozzle.

The graphs of afterburning S.F.C. are valid for the 113 inch engine with the 20 inch afterburner and for the 108.2 inch engine with the 23 inch afterburner.

2.8 Lubrication System

The lubrication system is primarily a cooling system with lubrication being automatically provided as a somewhat secondary requirement. Relatively high rates of oil circulation are used, which although increasing the overall heat load on the oil, reduces the more critical temperature rise from the supply to the scavenge.

The lubrication pump is a four element gerotor type comprising an oil supply element, front bearing rear sump, and auxiliary gearbox oil scavenge elements. The pump is driven from the engine auxiliaries gearbox and supplies oil to the front and rear main bearings, the internal gears in the front frame and the gears in the engine auxiliaries gearbox. Oil from the rear bearing is scavenged back to the oil tank through a pump element in all flight or ground attitudes, while oil from the front bearing, internal and auxiliary gearing is, in horizontal engine attitude, gravity scavenged through the lower front frame strut to the gearbox and is then pumped directly to the oil tank. Continuous operation in nose up attitudes is catered to by the front bearing scavenge element of the pump which picks up both bearing and gear fed oil from behind the front bearing and returns it to the tank. A gravity valve in the oil tank permits continuous operation in nose up and nose down attitudes, and in addition meets all the flight attitudes specified in MIL-E-5007A.

Oil is cooled, in an oil to fuel heat exchanger integrally mounted with the oil tank, before the oil enters the supply pump. It is filtered downstream of the pump and distributed to gears and bearings without pressure regulation, the individual oil jet flows being regulated by jet orifice and line resistance and the total jet flows being the full pump capacity.

2.9 Control System

The control system consists of three basic sections, the main engine control, the afterburner control, and the nozzle area control.

The pilot's power lever selects an engine speed. A coarse schedule of fuel flow is a function of compressor delivery pressure. Acceleration is scheduled on non-dimensional rotor speed. A non-electric jet pipe temperature over-ride is included. The fuel pump and governor are engine driven. Inlet guide vane operation is controlled by a fuel powered actuator. Fuel enters the engine from an eight port distributor.

The afterburner control modulates pump output on a schedule of fuel flow as a function of compressor outlet pressure. A power lever over-ride permits modulated afterburner performance.

A single positive displacement variable stroke fuel pump supplies both main engine and afterburner fuel.

The nozzle area control schedules nozzle area as a function of turbine pressure ratio. A spool valve in the control meters hydraulic fluid to the final nozzle actuating jacks.

As the system consists of comparatively simple and reliable mechanical hydraulic units, no emergency system is planned.

2.10 Ignition

The ignition system consists of a dual circuit high energy condenser discharge exciter and two shunted surface gap igniter plugs with the necessary leads. An additional igniter plug lights the afterburner.

If necessary, an oxygen torch igniter is available to permit relights at extreme altitudes and forward speeds.

2.11 Hydraulic System

Hydraulic fluid flow and pressure to operate the final nozzle jacks is supplied from an engine driven (aircraft supplied) hydraulic pump. A reservoir, filters and relief valve complete the system. A continuous bleed through the jacks prevents the fluid temperature from becoming too great during steady state conditions.

2.12 Installation

2.12.1 Engine Mounting: The complete unit is five point mounted to distribute the thrust and flight loads tangentially into the engine casing.

At the front frame, a mount on the top centreline takes axial and side leads which a single mount at the horizontal centreline, either on one side of the frame or the other, depending on installation requirements, takes vertical loads only. At the rear trunnion ring, two mounts above the horizontal centreline take vertical loads only, while a third mount at the top centreline takes side loads only.

If a second mount at the front horizontal centreline is required for vertical loads, then the aircraft support at the rear mounting ring must be designed so that no torque can be transmitted through the engine by airframe deflections, etc.

2.12.2 Exhaust Systems: The basic engine weights are given for the fully modulated convergent nozzle alone. Engine performance, Figures 1 to 8, assumes expansion either to 21.5 inches diameter or atmospheric pressure if that occurs at a smaller diameter. A velocity coefficient of 0.98 has been applied to the calculated exit momentum. The plotted performance thus assumes a variable convergent-divergent nozzle or ejector with maximum exit diameter of about 21 or 22 inches.

Figure 15 shows the preliminary design of a short, variable divergent ejector which adds 90 pounds and 11.8 inches to the basic engine. About 5% by-pass flow would be required and this could serve as cooling air for the engine bay. Good thrust performance over the whole flight range should be achieved. Afterburning at Mach 2.5 for instance, this nozzle should realize 97% of theoretical full expansion thrust. This is perhaps two percent less than the very best, and biggest, convergent-divergent nozzle or ejector. At this point, the plotted performance is 97.4% of theoretical full expansion hence is in good agreement with the variable ejector.

A longer, heavier, larger-exit-diameter nozzle could pick up one or two percent gross thrust at high Mach numbers. A lighter and simpler nozzle could be built which would give the same thrust over part of the flight envelope. Such a nozzle might give, for instance, the same take-off and high Mach number afterburning thrusts at the expense of some thrust loss non-afterburning.

The convergent nozzle segments are positioned by unison ring rollers riding on cammed surfaces of the segments. Since the main engine mountings are not designed to take the overhung moment of an afterburner, the afterburner duct is connected to the engine exhaust through a quick disconnect flexible coupling which basically imparts only axial loads to the engine structure. Jet pipe vertical loads are taken out by sliding trunnions provided on the sides of the jet pipe.

2.12.3 Accessories and Servicing: The basic engine has all of the accessories on the bottom of the engine. The oil refilling connection and the dipstick are accessible from the top and all engine servicing points and adjustments and engine-airframe connections can also be made from below.

To suit other installations, the engine could with some external redesign be supplied with all accessories on either the side or the top. All of the servicing points and engine-airframe connections would then be accessible from above or below the engine.

2.12.3.1 Engine Mounted Accessories: The tachometer-generator is a standard percentage type driven at 4200 rpm at 100 percent engine speed.

A starter or starter-generator could be mounted on the forward centre drive.

2.12.3.2 Aircraft Air Supply: A pad is provided on the top engine centreline, behind the compressor rear flange, for supplying aircraft pressurizing air. The engine will supply a maximum of 5.0 lb/sec at sea level static conditions or ten percent of mass flow at any condition at a performance penalty indicated by Figure 11 and 12.

2.12.3.3 Fuel Connection: The aircraft fuel connection is a standard flared tube connection and can be located at any convenient position.

2.12.3.4 Power Lever Connection: The power lever movement is according to the military specification for thrust versus throttle angle.

2.12.3.5 Electrical Connection: The engine electrical connection is through a standard M.S. connector. D.C. power only is required.

2.12.3.6 Firewall: A flange is provided near the front of the combustion chamber for attachment of a firewall.

2.12.3.7 Power Take-off: Provision for power take-off (mechanical, hydraulic or electrical) is made at the forward face of the auxiliaries drive gearbox on the bottom of the engine. The engine can be started through this drive. Limiting static torque on the shaft is 650 inch/pounds. Other positions for power take-off can be provided, if necessary.

2.12.3.8 Combustion Drains: Combustion drain valves and a single aircraft drain line are incorporated on the underside of the combustion chamber. The aircraft connection is to AND10056-6.

2.12.3.9 Slinging: The slinging points are provided by means of brackets: two of these are located on the front flange of the front frame, approximately 45 degrees either side of the top centreline, and a third is located on a flange near the turbine area on the top centreline.

2.13 Starting

Dependent on the aircraft installation, a start can be accomplished through the aircraft accessories gearbox at either side of the engine, or, through the forward centre drive as mentioned in item 2.2. Gear trains are designed for normal electric starter torques for side starting, but a higher load can be taken through the forward centre drive which can be a direct connection to the rotor.

3. PERFORMANCE CURVES

3.1 Assumptions

Performance curves are presented in Figures 1 to 8 showing engine performance with and without afterburner from sea level to 75,000 feet. In the calculation of these performance estimates the following assumptions were made:

(a) I.C.A.O. standard atmosphere (I.C.A.O. Doc. No. 7488).

(b) A.I.A. intake pressure recovery.

$$P_1/P_o = 1 - 0.1\ (M-1)\ 3/2$$

(c) Engine rotor speed limit = 16,000 rpm.

(d) Maximum combustion temperature = 1275 degrees K.
= 2300 degrees R.

(e) Afterburner temperature = 1850 degrees K.
= 3330 degrees R.

(f) Exhaust nozzle throat effective area (non-afterburning) = 105 square inches

(g) Allowable exhaust diameter for jet expansion = 22 inches

(h) Expanded effective area five percent less than given by 22 inch diameter to allow for engine by-pass ejector flow.

(i) Jet velocity coefficient of 0.98 at exit plane of expanded flow.

(j) No power extracted or air bled for external services, except as noted.

3.2 Fuel Thermal Properties

The combustion temperature rise properties of JP4 fuel were taken from NACA RM E55G27a.

3.3 Ratings

Maximum with augmentation rating represents the engine operating at maximum engine conditions with full afterburning. Time limit at this rating is fifteen minutes.

Maximum rating represents the maximum non-afterburning power which may be maintained for fifteen minutes.

Military rating represents the maximum power which may be maintained for thirty minutes.

Normal rating represents the maximum power which may be maintained continuously.

Since no control system schedules were defined for this presentation, these ratings and their time limitations represent turbine inlet temperature limits rather than throttle settings. In addition, engine rotor speed is limited to 16,000 rpm and at lower flight Mach numbers this limits temperatures for maximum rating to values below rated values.

3.4 Expanded Jet Performance

The thrust values shown in Figures 1 to 8 assumed expansion of the jet to an area corresponding to a 22 inch nacelle outlet diameter, less five percent for ejector flow. When this area would be great enough to over-expand the flow, it is assumed that the outlet area closes until the flow is just fully expanded. This occurs only at the low rpm low flight speed conditions.

3.5 Overboard Air Bleed

The effect of bleeding up to 10% of compressor delivery air in sea level static thrust has been shown in Figures 11 and 12. Figure 11 shows thrust as percentage bleed for maximum rating (non-afterburning) and Figure 12 shows thrust as percentage bleed and afterburner temperature.

3.6 Reynold's Number Corrections

Performance data shown in Figures 1 to 8 have been presented with no deterioration at high altitude due to reduction in Reynold's number. The effect of Reynold's number on the engine performance is shown in Figures 9 and 10 and should be applied to the data of Figures 1 to 8. The corrections apply accurately only to ratings above that corresponding to a cruise condition.

3.7 Inlet Duct Pressure Loss Correction

Inlet duct correction co-efficient is shown in Figure 13. This data has been calculated for flight operation in the stratosphere as well as the sea level static. The correction is defined according to the equation:-

$$\frac{\Delta F_N}{F_N} = C_{D_2} \frac{\Delta P_{T_2}}{P_{T_2}}$$

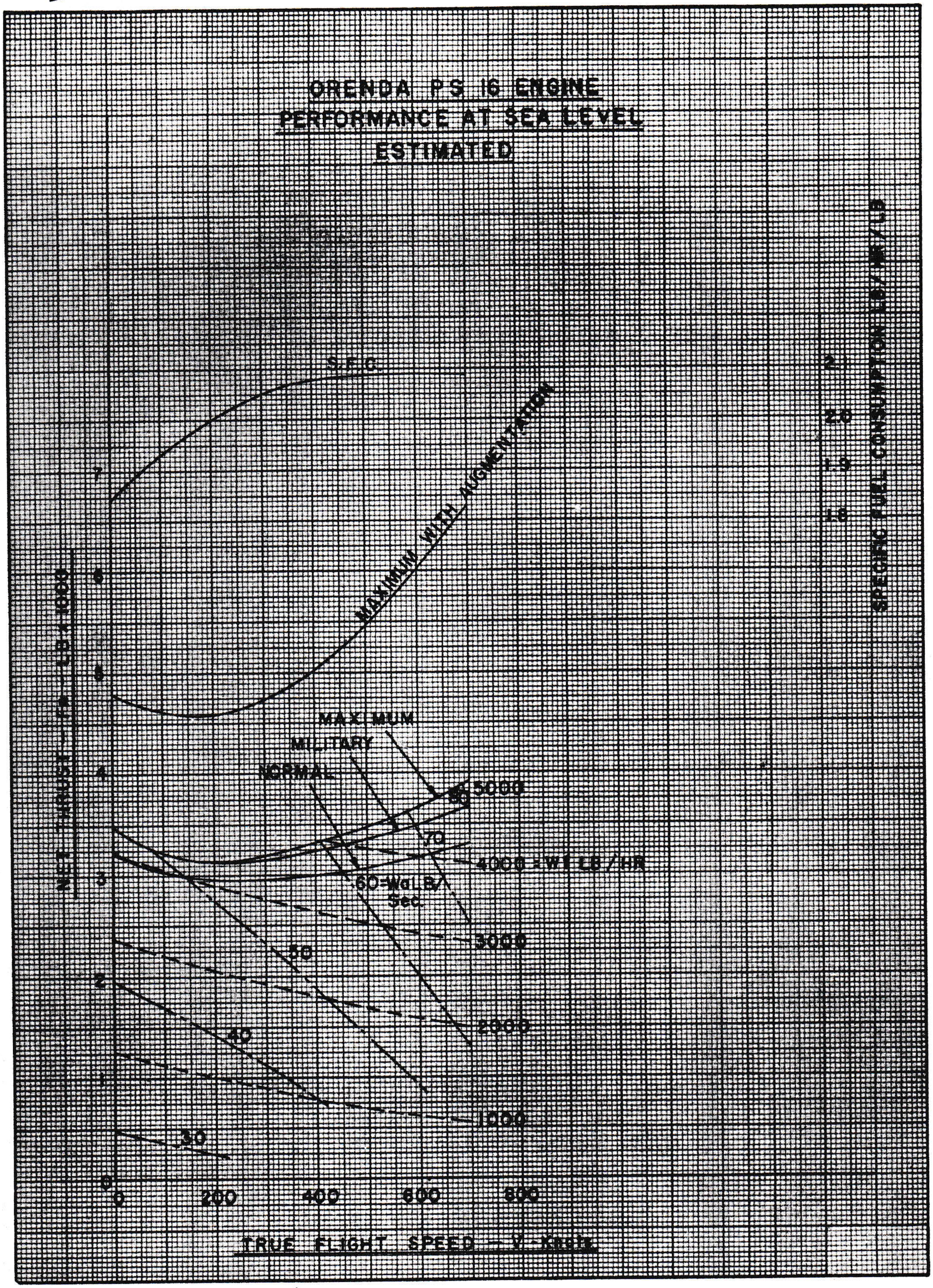
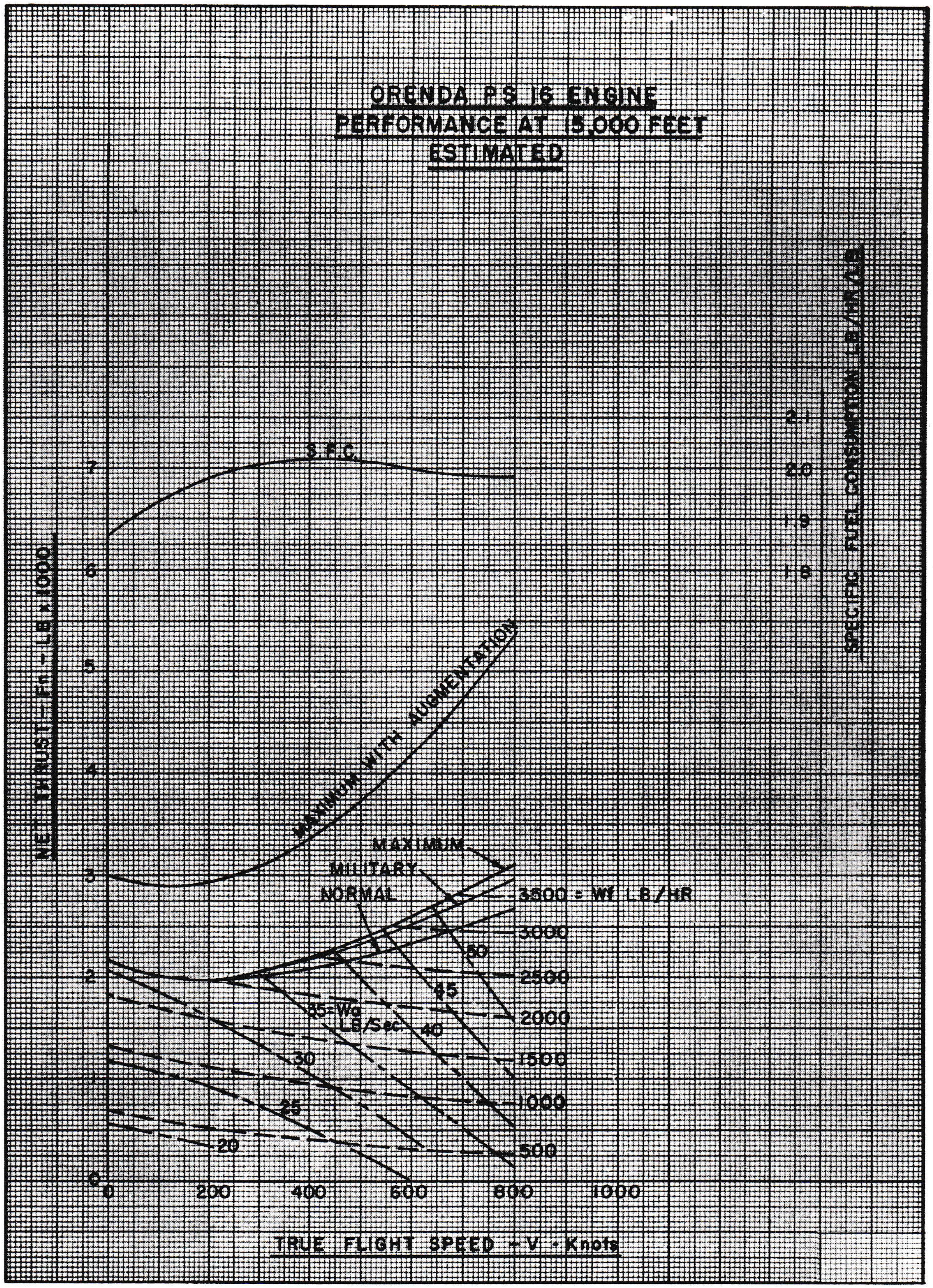
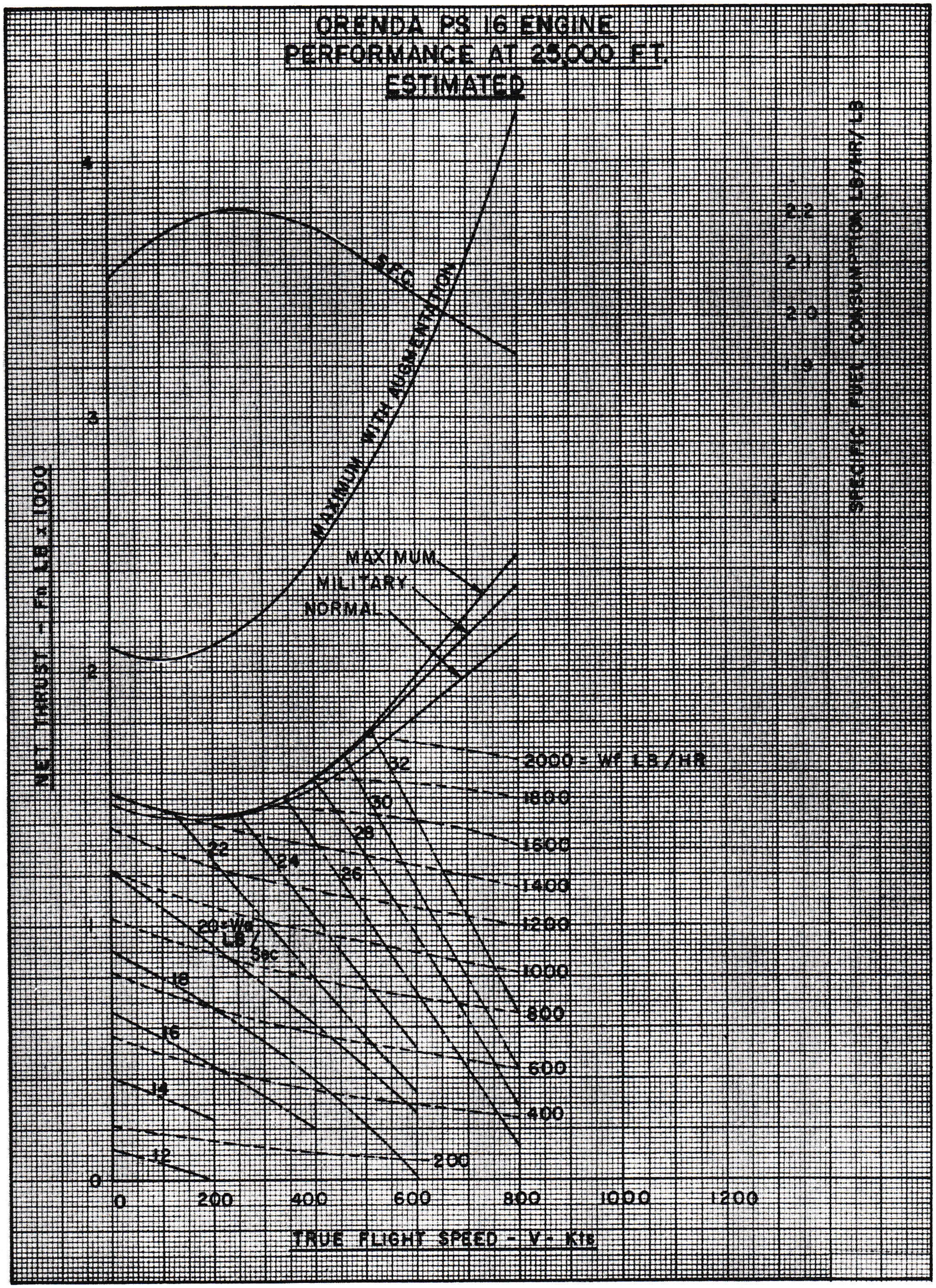
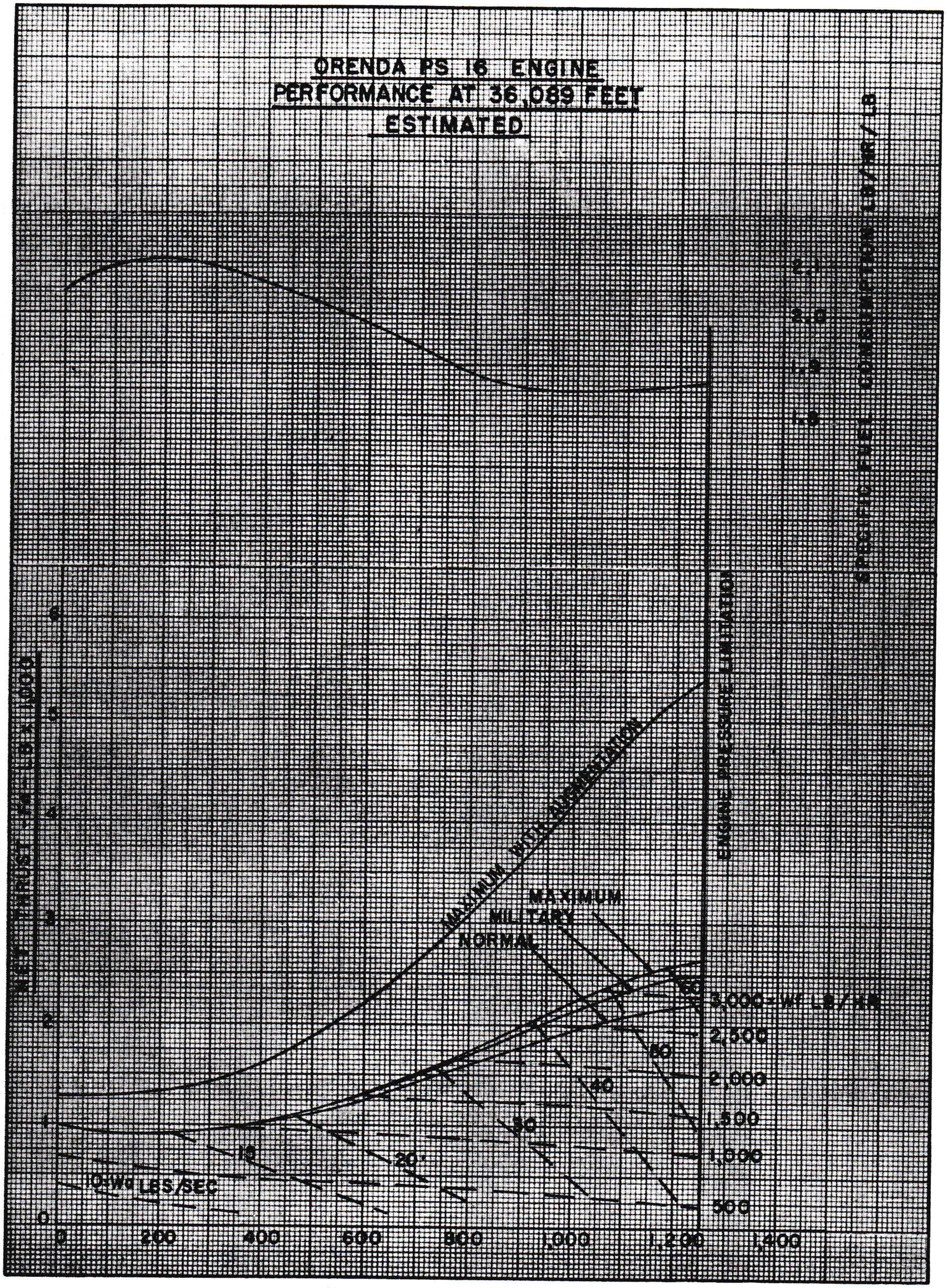

ORENDA PS 16 ENGINE
PERFORMANCE AT SEA LEVEL
ESTIMATED
S.F.C.
MAXIMUM WITH AUGMENTATION
MAXIMUM
MILITARY
NORMAL
5000
80
70
4000 - Wf LB/HR
60 - Wa LB/Sec.
3000
50
2000
40
1000
30
NET THRUST - Fn - LB x 1000
SPECIFIC FUEL CONSUMPTION LB/HR/LB
TRUE FLIGHT SPEED - V - Knots
0
200
400
600
800
1
2
3
4
5
6
7
2.1
2.0
1.9
1.8

Fig. 1


ORENDA PS 16 ENGINE
PERFORMANCE AT 15,000 FEET
ESTIMATED
NET THRUST - Fn - LB x 1000
SPECIFIC FUEL CONSUMPTION LB/HR/LB
TRUE FLIGHT SPEED - V - Knots
S.F.C.
MAXIMUM WITH AUGMENTATION
MAXIMUM
MILITARY
NORMAL
3500 = Wf LB/HR
3000
2500
2000
1500
1000
500
35=Wa
LB/Sec
30
25
20
40
45
50
0
1
2
3
4
5
6
7
200
400
600
800
1000
1.8
1.9
2.0
2.1


ORENDA PS 16 ENGINE
PERFORMANCE AT 25,000 FT.
ESTIMATED
NET THRUST - Fn LB x 1000
SPECIFIC FUEL CONSUMPTION LB/HR/LB
TRUE FLIGHT SPEED - V - Kts
MAXIMUM WITH AUGMENTATION
SFC
MAXIMUM
MILITARY
NORMAL
2000 = Wf LB/HR
1800
1600
1400
1200
1000
800
600
400
200
20 = Wa LB/Sec
0
200
400
600
800
1000
1200


ORENDA PS 16 ENGINE
PERFORMANCE AT 36,089 FEET
ESTIMATED
SPECIFIC FUEL CONSUMPTION LB/HR/LB
2.1
2.0
1.9
1.8
NET THRUST Fn - LB x 1,000
ENGINE PRESSURE LIMITATION
MAXIMUM WITH AUGMENTATION
MAXIMUM
MILITARY
NORMAL
3,000 - Wf LB/HR
2,500
2,000
1,500
1,000
500
10 Wa LBS/SEC
0
200
400
600
800
1,000
1,200
1,400

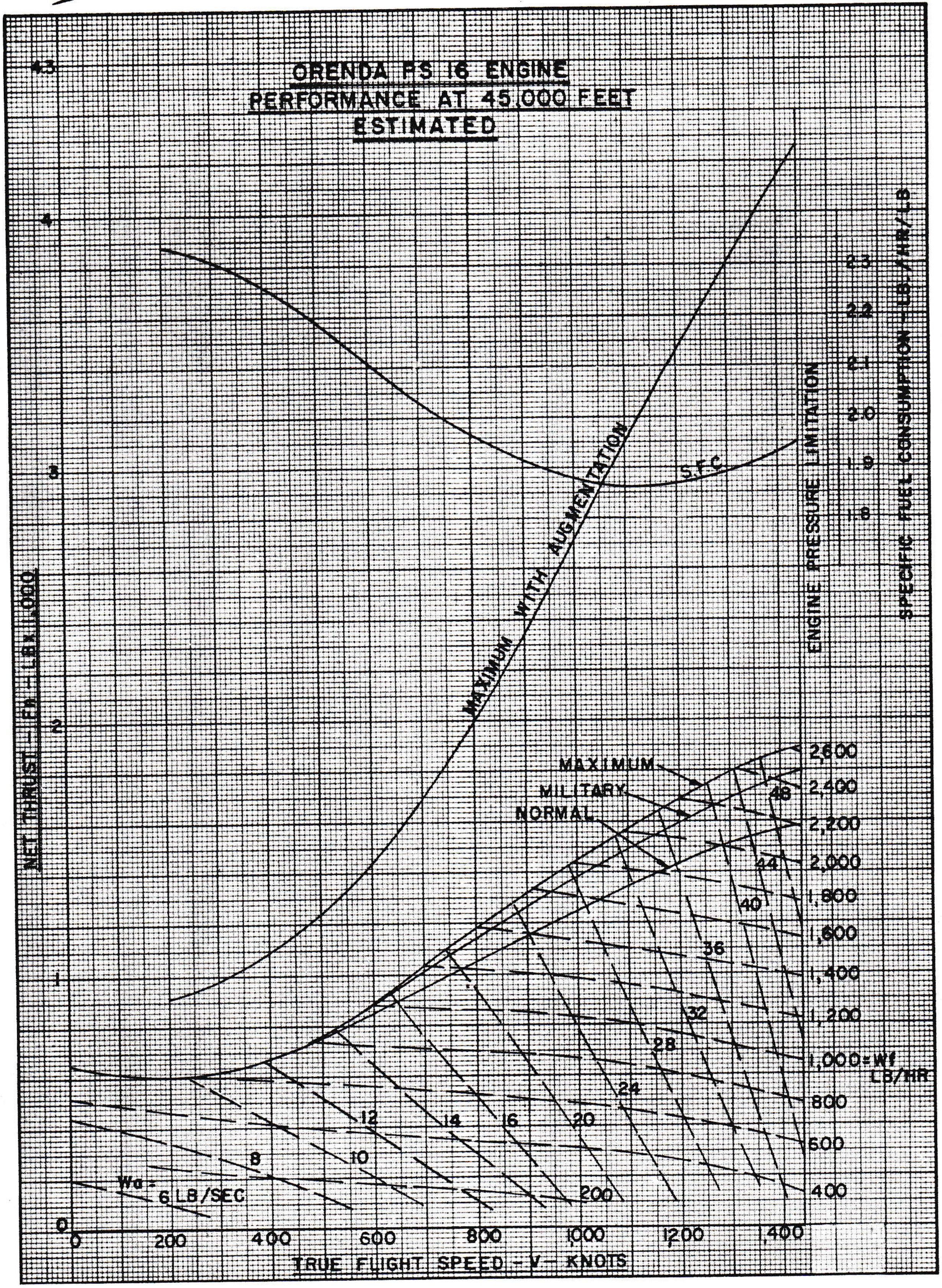
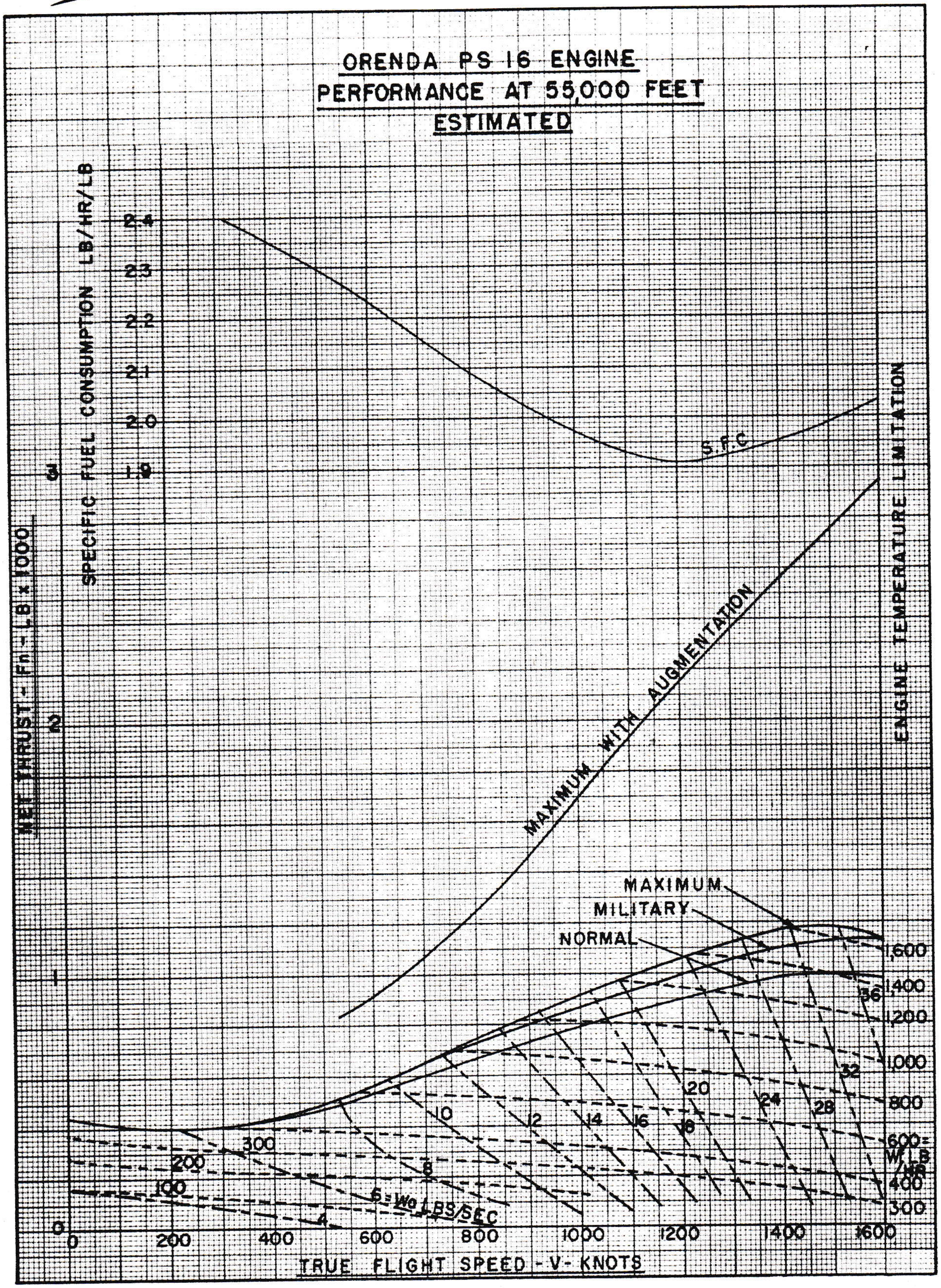
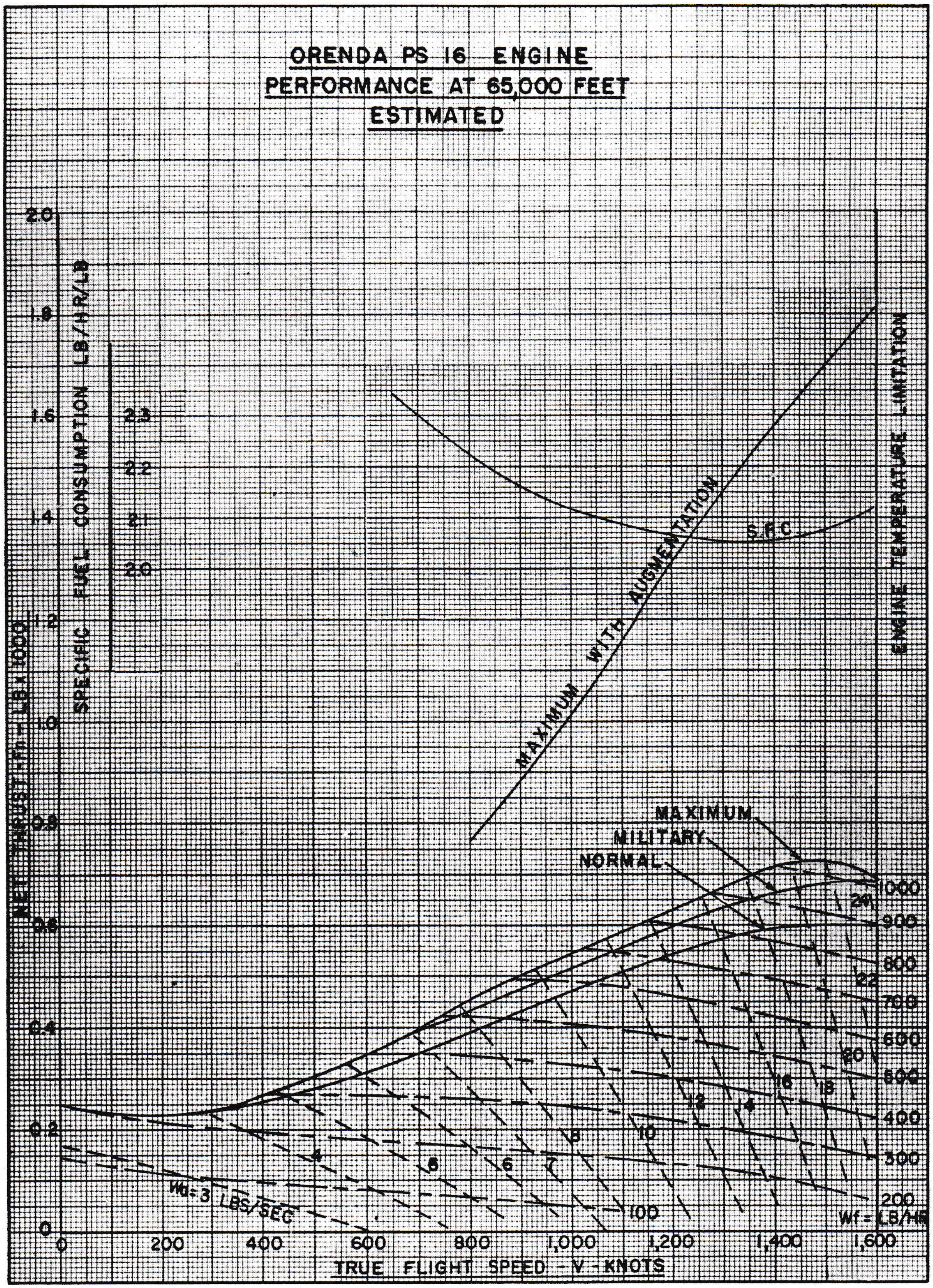

ORENDA PS 16 ENGINE
PERFORMANCE AT 45,000 FEET
ESTIMATED
NET THRUST - Fn - LB x 1,000
TRUE FLIGHT SPEED - V - KNOTS
SPECIFIC FUEL CONSUMPTION - LB./HR/LB
ENGINE PRESSURE LIMITATION
MAXIMUM WITH AUGMENTATION
S.F.C.
MAXIMUM
MILITARY
NORMAL
Wa = 6 LB/SEC
1,000 = Wf LB/HR

Fig. 5


ORENDA PS 16 ENGINE
PERFORMANCE AT 55,000 FEET
ESTIMATED
SPECIFIC FUEL CONSUMPTION LB/HR/LB
NET THRUST - Fn - LB x 1000
ENGINE TEMPERATURE LIMITATION
S.F.C.
MAXIMUM WITH AUGMENTATION
MAXIMUM
MILITARY
NORMAL
W/LB/HR
δ = Wa LBS/SEC
TRUE FLIGHT SPEED - V - KNOTS


ORENDA PS 16 ENGINE
PERFORMANCE AT 65,000 FEET
ESTIMATED
NET THRUST Fn - LB x 1000
SPECIFIC FUEL CONSUMPTION LB/HR/LB
ENGINE TEMPERATURE LIMITATION
MAXIMUM WITH AUGMENTATION
S.F.C
MAXIMUM
MILITARY
NORMAL
Wa=3 LBS/SEC
Wf= LB/HR
TRUE FLIGHT SPEED - V - KNOTS
0
200
400
600
800
1,000
1,200
1,400
1,600
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
2.0
2.1
2.2
2.3
100
200
300
400
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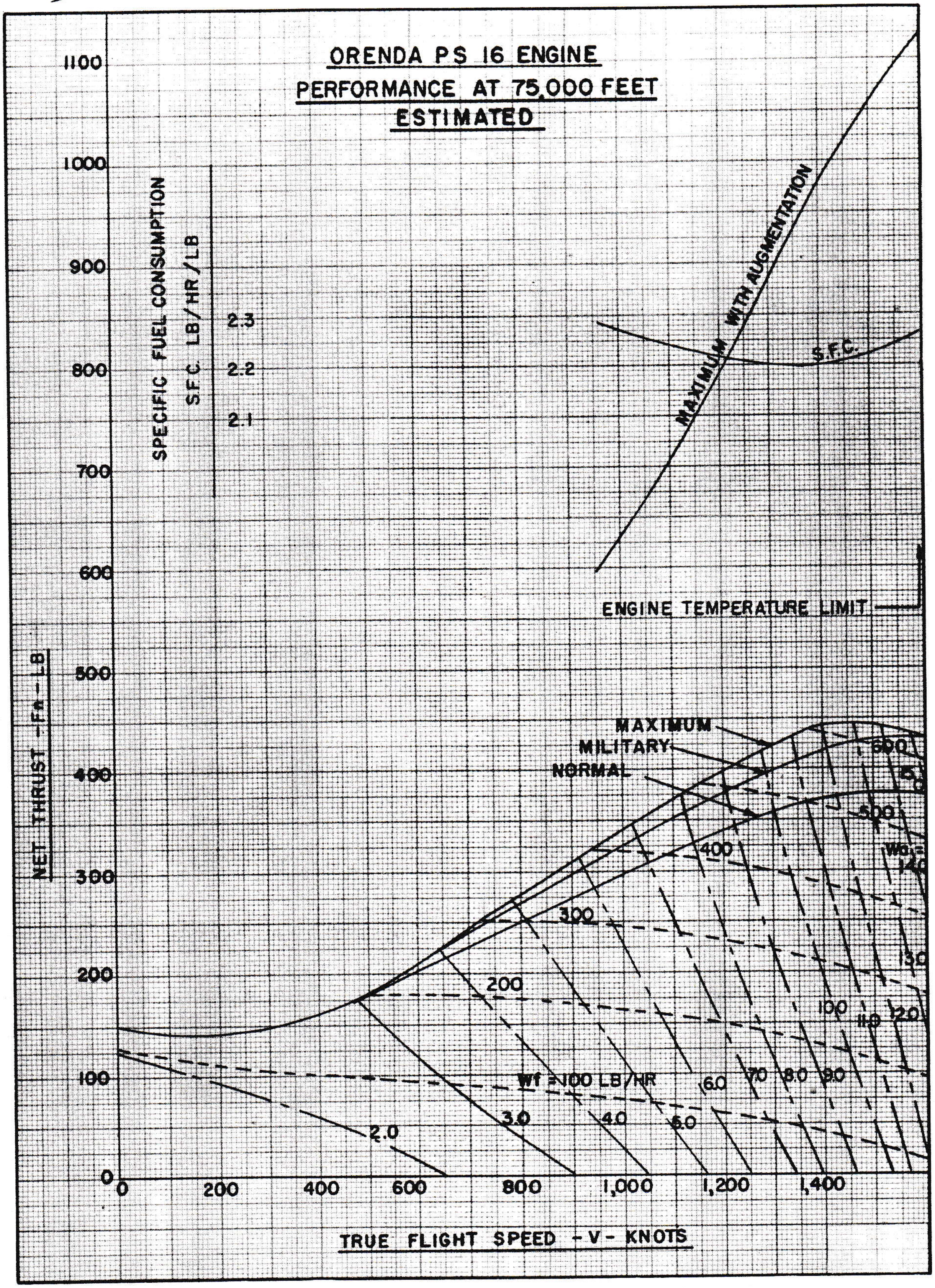
16
18
20
22
24


ORENDA PS 16 ENGINE
PERFORMANCE AT 75,000 FEET
ESTIMATED
NET THRUST -Fn- LB
SPECIFIC FUEL CONSUMPTION
S.F.C. LB/HR/LB
2.3
2.2
2.1
MAXIMUM WITH AUGMENTATION
S.F.C.
ENGINE TEMPERATURE LIMIT
MAXIMUM
MILITARY
NORMAL
Wf = 100 LB/HR
TRUE FLIGHT SPEED - V - KNOTS
0
100
200
300
400
500
600
700
800
900
1000
1100
0
200
400
600
800
1,000
1,200
1,400

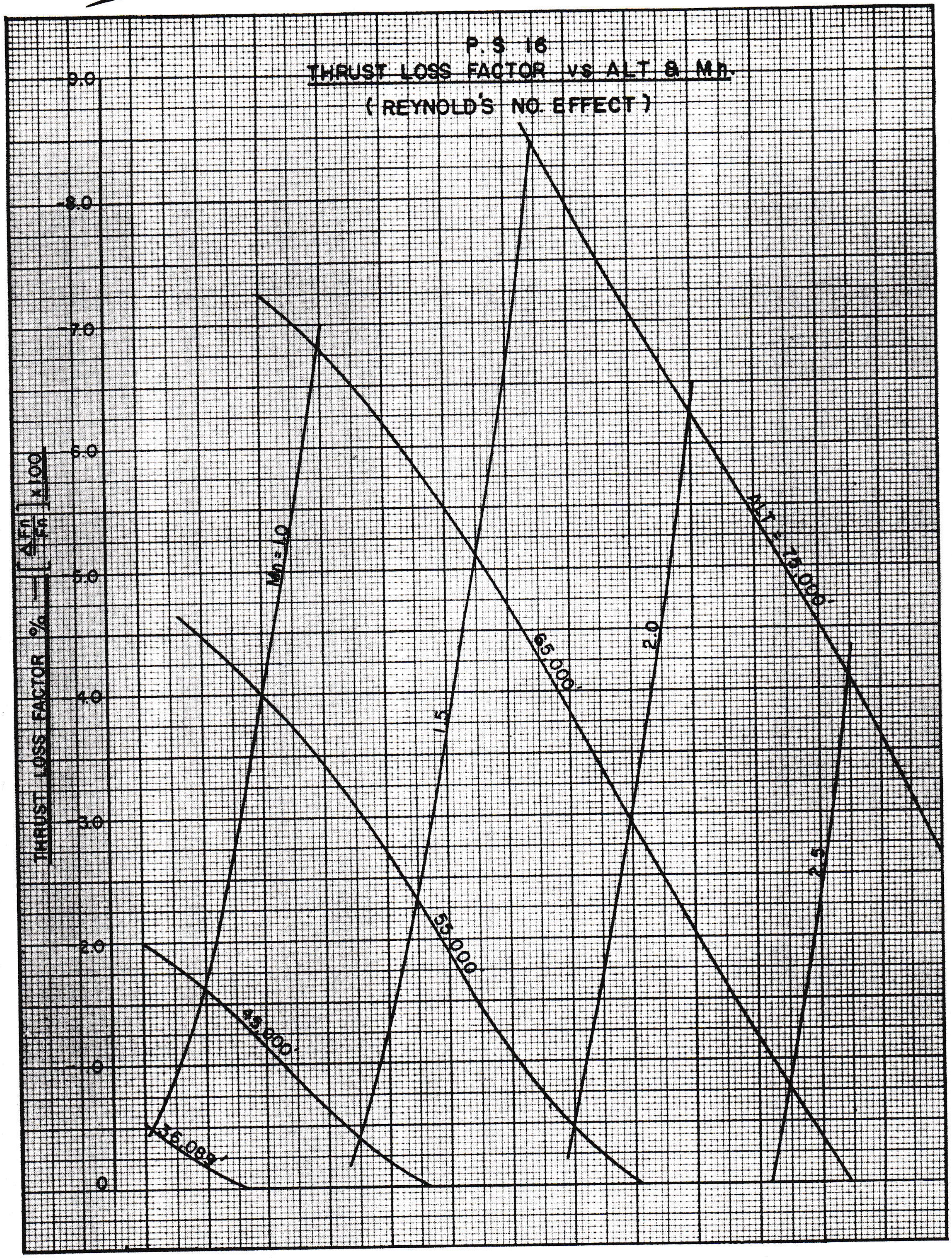
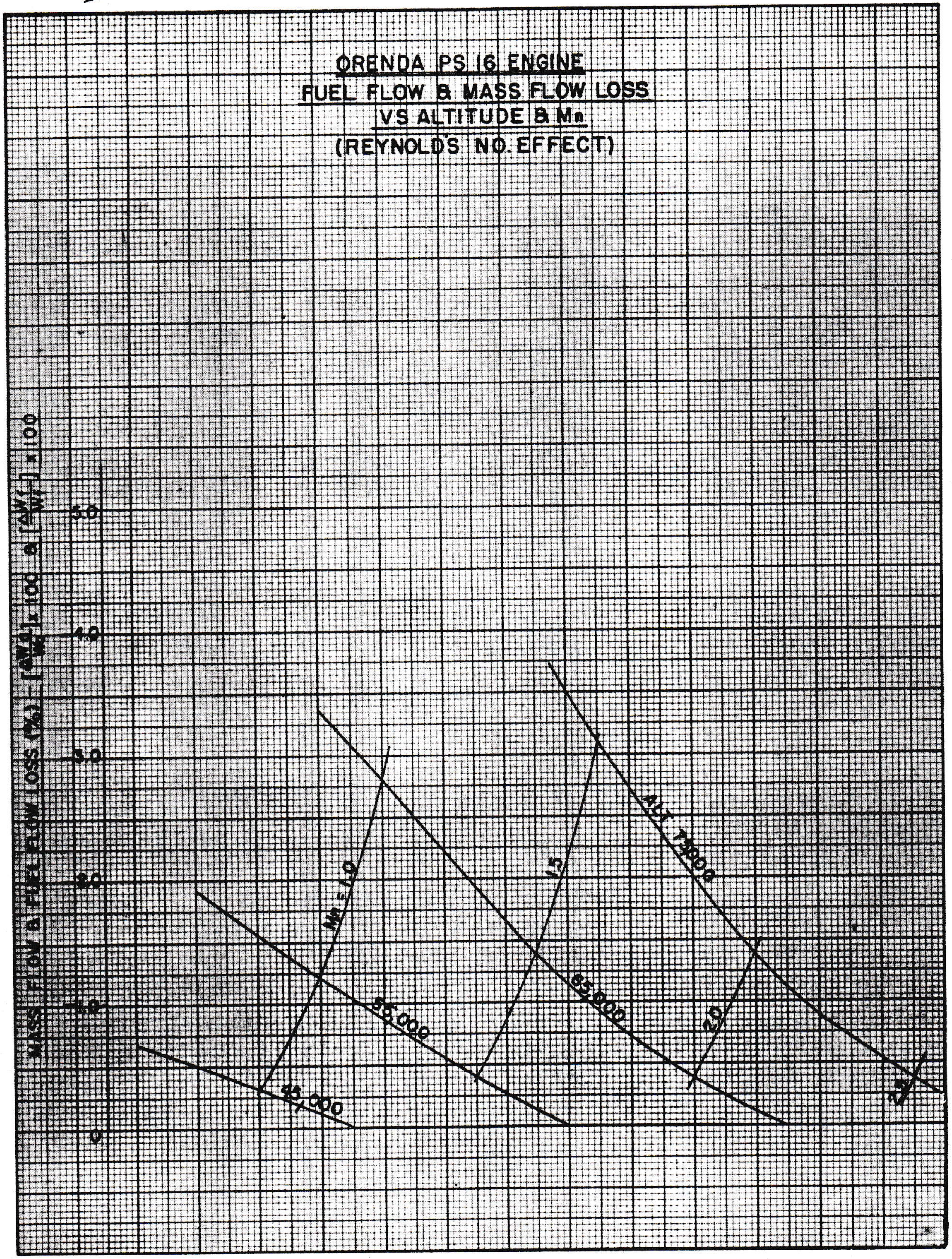

P.S. 16
THRUST LOSS FACTOR vs ALT & Mn.
(REYNOLD'S NO. EFFECT)
THRUST LOSS FACTOR % — [ΔFn/Fn] x 100
9.0
8.0
7.0
6.0
5.0
4.0
3.0
2.0
1.0
0
Mn = 1.0
1.5
2.0
2.5
ALT = 75,000'
65,000'
55,000'
45,000'
36,089'

Fig. 9


ORENDA PS 16 ENGINE
FUEL FLOW & MASS FLOW LOSS
VS ALTITUDE & Mn
(REYNOLD'S NO. EFFECT)
5.0
4.0
3.0
2.0
1.0
0
Mn = 1.0
1.5
2.0
2.5
ALT. 75000
65,000
55,000
45,000

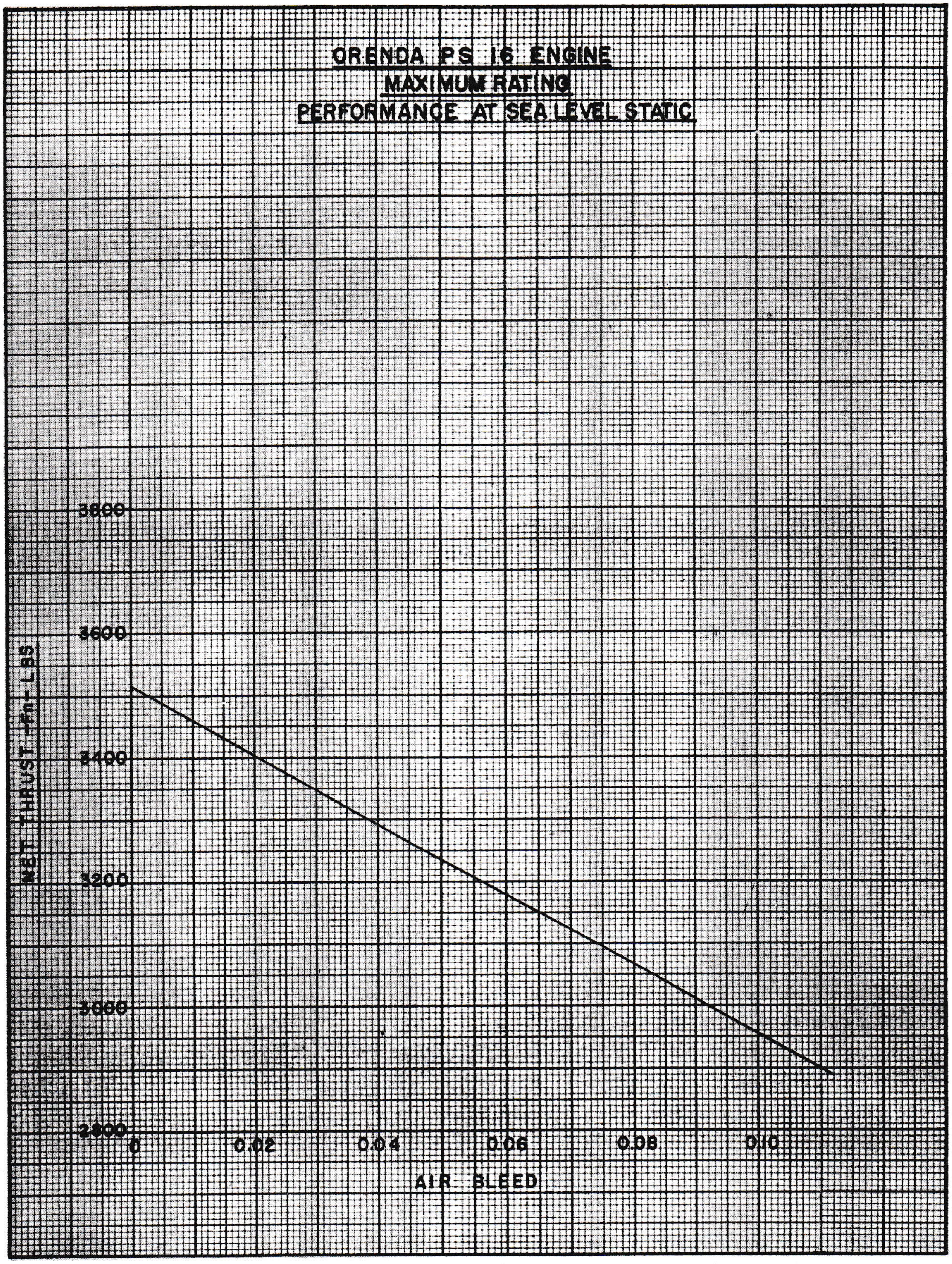
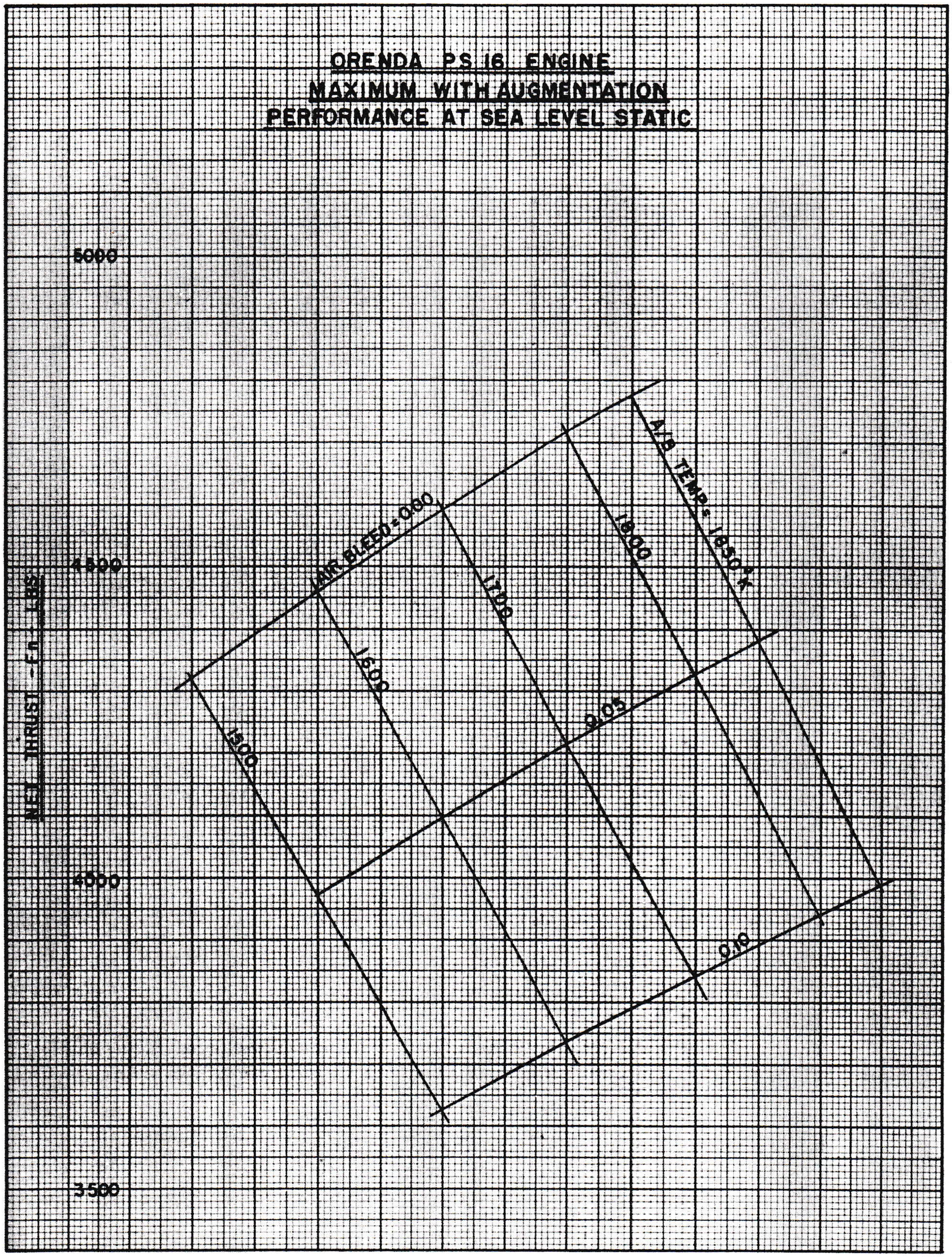
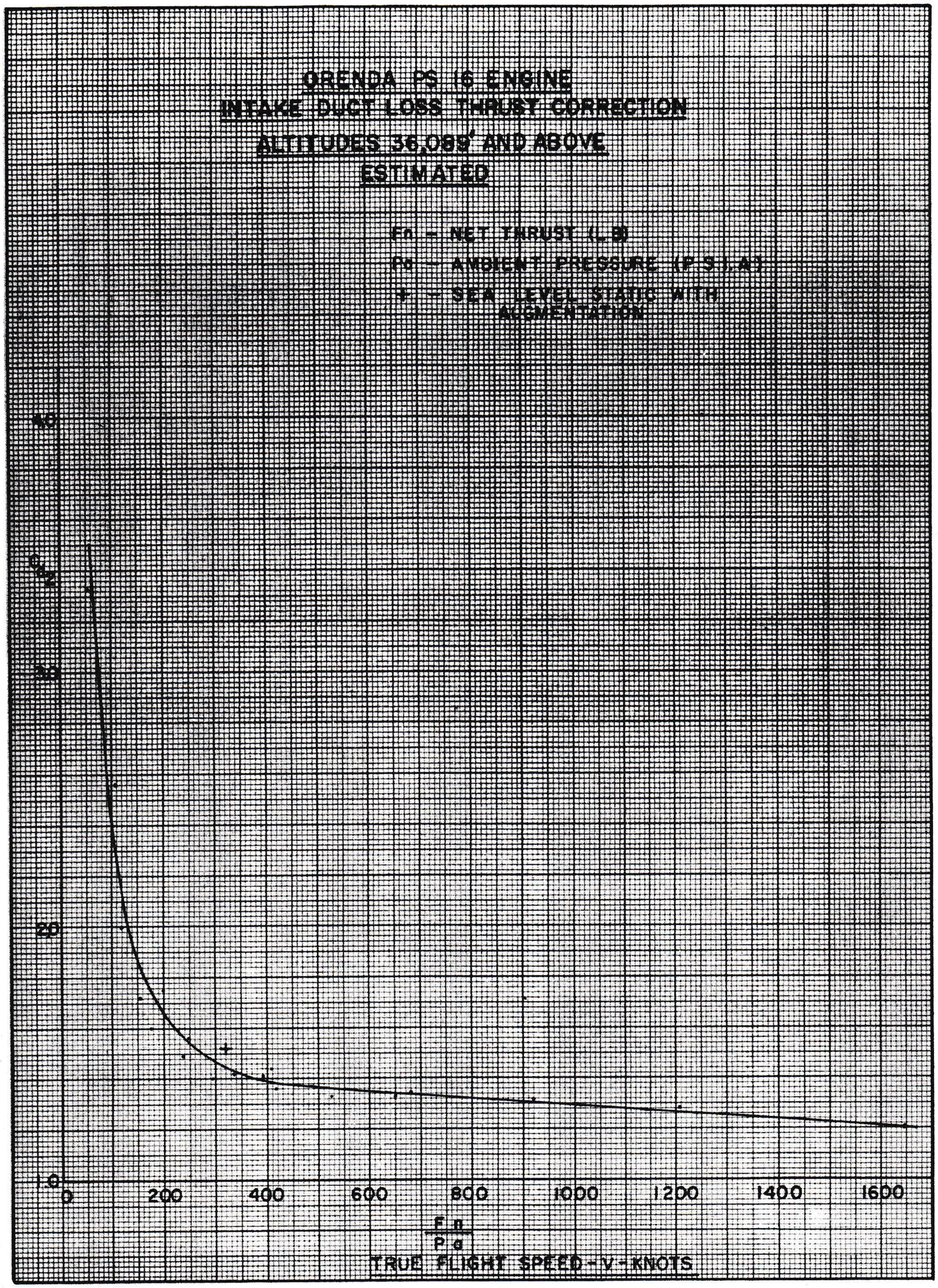
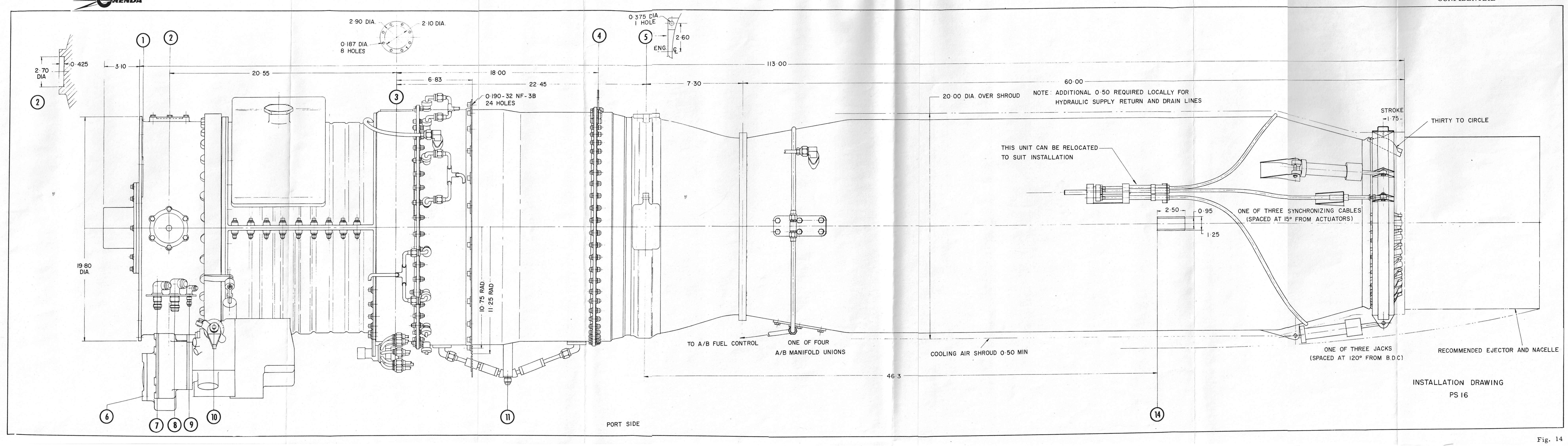
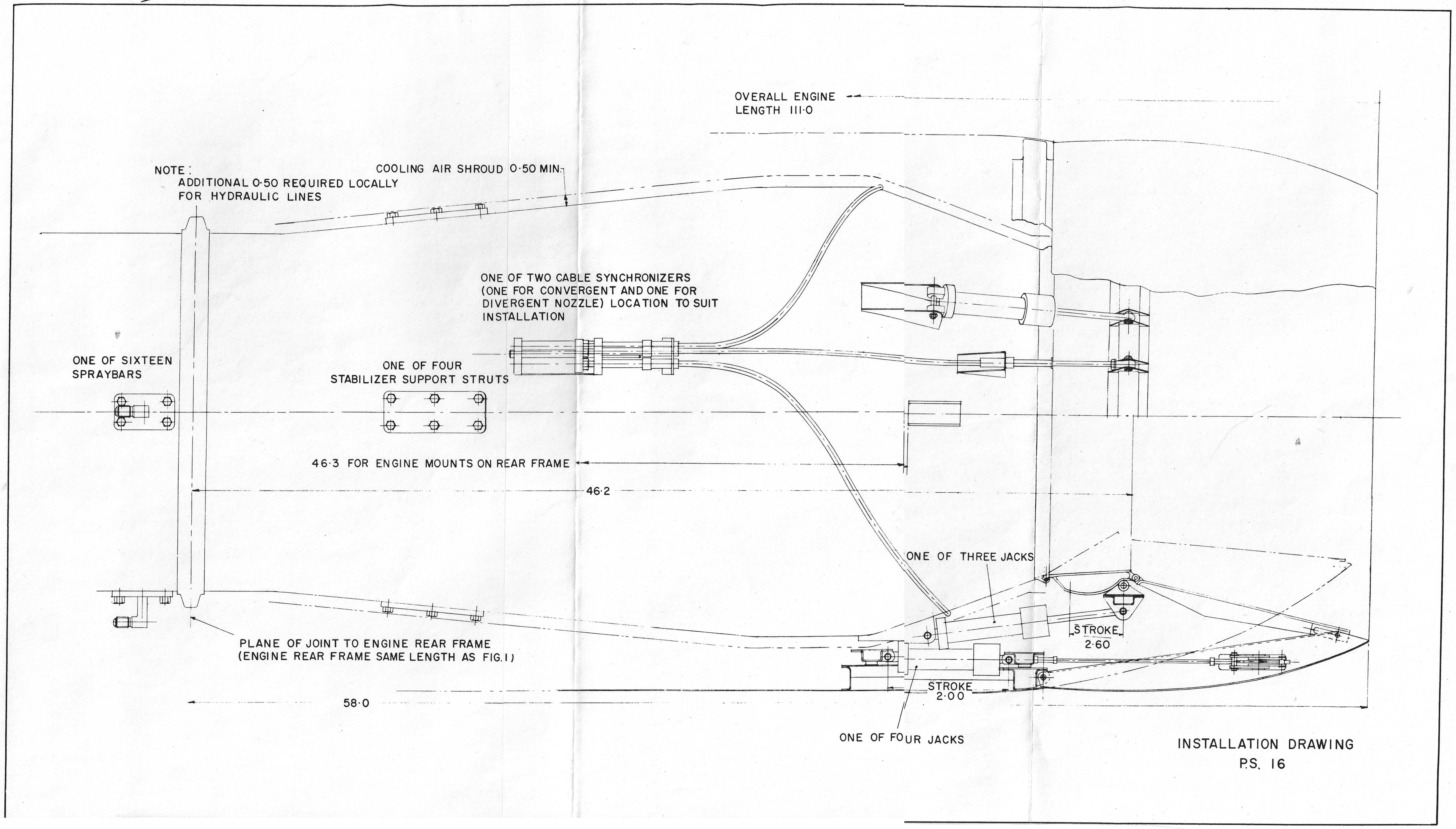

ORENDA PS 16 ENGINE
MAXIMUM RATING
PERFORMANCE AT SEA LEVEL STATIC
NET THRUST -Fn- LBS
3800
3600
3400
3200
3000
2800
0
0.02
0.04
0.06
0.08
0.10
AIR BLEED

Fig. 11


ORENDA PS 16 ENGINE
MAXIMUM WITH AUGMENTATION
PERFORMANCE AT SEA LEVEL STATIC
5000
4500
4000
3500
NET THRUST - Fn - LBS.
AIR BLEED = 0.00
0.05
0.10
1500
1600
1700
1800
A/B TEMP = 1850°K


ORENDA PS 16 ENGINE
INTAKE DUCT LOSS THRUST CORRECTION
ALTITUDES 36,089' AND ABOVE
ESTIMATED
Fn - NET THRUST (LB)
Pa - AMBIENT PRESSURE (P.S.I.A)
+ - SEA LEVEL STATIC WITH AUGMENTATION
4.0
3.0
2.0
1.0
0
200
400
600
800
1000
1200
1400
1600
Fn/Pa
TRUE FLIGHT SPEED - V - KNOTS


2·90 DIA
2·10 DIA
0·187 DIA
8 HOLES
0·375 DIA
1 HOLE
2·60
ENG ℄
2·70
DIA
0·425
3·10
20·55
18·00
6·83
22·45
7·30
113·00
60·00
0·190-32 NF-3B
24 HOLES
20·00 DIA OVER SHROUD
NOTE: ADDITIONAL 0·50 REQUIRED LOCALLY FOR
HYDRAULIC SUPPLY RETURN AND DRAIN LINES
STROKE
1·75
THIRTY TO CIRCLE
THIS UNIT CAN BE RELOCATED
TO SUIT INSTALLATION
2·50
0·95
1·25
ONE OF THREE SYNCHRONIZING CABLES
(SPACED AT 15° FROM ACTUATORS)
19·80
DIA
10·75 RAD
11·25 RAD
TO A/B FUEL CONTROL
ONE OF FOUR
A/B MANIFOLD UNIONS
COOLING AIR SHROUD 0·50 MIN
46·3
ONE OF THREE JACKS
(SPACED AT 120° FROM B.D.C)
RECOMMENDED EJECTOR AND NACELLE
INSTALLATION DRAWING
PS 16
PORT SIDE
1
2
3
4
5
6
7
8
9
10
11
14



INSTALLATION DRAWING
P.S. 16

Fig. 15