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PROJECT HAZEL

PROPULSION, STRUCTURAL HEATING AND PRESSURIZATION

REPORT NO. ZJ-026

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CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO, CALIF.

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FOREWORD

This report is presented as one of a set describing the Project "Hazel" study performed by the Convair San Diego Division of the General Dynamics Corporation. The entire set of reports, listed below, represents Convair's fulfillment of the publications obligation specified in Contract NOas-58-812 (SS-100) and Amendment #1, issued 14 August 1958 by the Bureau of Aeronautics.

ZP 252	Summary (Brochure of Charts with Text)
ZP 253	Aircraft Design
ZA 282	Aerodynamics
ZJ 026	Propulsion, Structure Heating, and Pressurization

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INTRODUCTION

This report presents the results of studies of engine and inlet performance, structural heating problems, and structural pressurization systems, carried out by the Thermodynamics Group.

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SUMMARY AND GENERAL CONCLUSION

Several inlets were examined by Convair, Pratt & Whitney and Marquardt for this application. It was concluded that the fixed isentropic spike diffuser with slight internal contraction if necessary would be best.

Engine performance as presented by Pratt & Whitney, and Marquardt is exhibited in the report. This performance was checked by Convair and found correct with the reservation that the combustion efficiencies assumed will have to be verified by testing.

Hydrogen appears to be better than pentaborane from a propulsion and handling standpoint. Both fuels are adequate for the altitude of this mission.

Both engine companies have facilities that will be available by 1960 that can handle the engines they propose. Government facilities are also available at NACA and A.R.D.C.

Structural temperatures were found to be within operating limits of the materials proposed, with the exception of some sections of the engine, where additional materials study is indicated. Fuel heating will not be a major problem for the fuels proposed. Wing surface temperatures will vary from 630° F at the leading edge to 400 and 300° F one and ten feet, respectively, from the leading edge.

Minimum structural pressurization system weight is obtained by utilizing helium, stored in the liquid state and heated after evaporation by mixing with hydrazine exhaust products from the auxiliary power unit.

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PROPULSION SYSTEM

INLETS

Early proposals by Pratt and Whitney and Marquardt were somewhat conservative on pressure recovery, both using values of about .70. Boost and range considerations indicated that the best system would most probably dictate ram jet take over at or near the design Mach number. This allowed better diffuser design point selection. Current peak pressure recoveries used were about .77 - .79 at Mach 3.0. Under these conditions the best type of diffuser appeared to be the fixed isentropic spike. The nearest contender was the Internal Compression Inlet which may well have been selected on a total thrust minus drag basis but was not because of higher weight. This resulted from its longer design and moveable spike. The fixed isentropic spike inlets selected gave a total external drag coefficient of .11 based on engine area. Of this, .06 was wave drag and .041 was skin friction of the engine external surface.

The inlets had to be placed with respect to the wing in a way that satisfied radar visibility restrictions. An over wing location resulted and two arrangements were found as satisfactory compromises. Two engines located outboard about mid half span can be situated over the drooped leading edge so that the upper wing surface with a minimum of flattening can give zero angle of attack with respect to the inlet. One engine centrally located can be placed behind the apex of the delta planform with a portion of the surface made plane at zero angle of attack to the inlet.

It was found that the recovery penalty suffered from expansion over the resulting flat surface when the vehicle was operated at higher than design angle of attack was less than that suffered from the inlet in free stream at the same off design angle. This is because the flow expands over the flat surface parallel to the axis of the inlet. For a 2° positive angle of attack the loss in pressure recovery is about 4% behind the flat surface and 6.6% in free stream.

Both Pratt and Whitney and Marquardt claim to have adequately tested the inlets selected at the required design Mach number. Neither has matched the Reynolds number of the flight condition, however.

FUELS

A table of physical and handling characteristics is given below

TABLE I

PENTABORANE VS SF-1 FUELS

GROUND HANDLING & LOGISTICS

Property	Pentaborane	SF-1
Price $/lb. Today	20	1-10
Large Scale Production	3	.2

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Property	Pentaborane	SF-1
Explosive Hazard	Pyrophoric	Air mixtures can be ignited by a spark
Toxicity	Extremely toxic either by inhalation or contact.	None. Can suffocate if it displaces all oxygen. Cold "burns" because of low temperature.
Corrosion	OK with anodized alum., copper, steel. NACA RM E54 E12 has data on materials	Non corrosive. Can cause low temperature embrittlement. 18-8 steels; low C, high Nc steel; monels are OK. Plastics will have to be checked out.
Storage	Inert atmosphere. No leaks can be tolerated	Dewar tanks. Boil off must be ventilated.
Hazards	Toxicity and prophoric properties require inert atmosphere transfer system and protective clothing with respiratory protection.	Protective clothing to protect against cold "burns".

AIRPLANE PERFORMANCE CONSIDERATIONS

Property	Pentaborane	SF-1
Heating Value BTU/lb	Approx. 29,300 (lower) 30,300 (higher)	51,500 (lower)
Boiling Point °R	Approx. 600	37
Density at B.P. lb/ft^3	Approx. 37	4.4
Use as Cooling Fluid	Decomposes at temp. 260° F	Excellent coolant
Tanks and Lines	Inert transfer system Insulation may be needed on tanks to prevent thermal decomposition.	Insulation needed

It can be seen that hydrogen appears to have advantages in the handling area since it is neither toxic nor pyrophoric. Also considerable experience with handling and pumping cold liquids has been gained with liquid rockets in recent years. On performance it has higher heat content and clean exhaust. It does not have a solid-liquid vapor phase as does pentaborane.

Pentaborane has advantages in its ability to produce a strong stable flame at high altitudes. It has better volume characteristics for tankage.

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Both fuels appear adequate for the combustion conditions anticipated for the Hazel vehicle. Hydrogen would seem to be somewhat marginal at altitudes above 150,000 feet at the chosen design Mach number. This is based on preliminary results from Marquardt and depends partially on combustion efficiency assumptions. This does not seem to be a problem, however, as the present mission does not attain this altitude. Approximate combustion pressures and inlet temperatures are shown on Figure 1, together with Marquardt and Pratt and Whitney test data available on the two fuels. The test data is at or near Mach 2.0 but combustion conditions may be expected to improve at the same pressures and higher temperatures encountered at Mach 3.0. No combustion efficiency data was derived from these tests, a fact that has led to a marked difference in design of combustion chamber lengths as will be brought out later.

ENGINES

As was requested by the Navy, Marquardt and Pratt and Whitney were the only engine companies approached for performance and design data. The results received from them are presented at the design points chosen for each engine. These are also substantiated by calculations made by Convair in the region of the selected design points.

The engine size range was established by an interchange of estimated L/D's, gross weights and flight conditions between Convair and the engine companies. Latitude on either side of the estimated design sizes was given to allow for changes produced by more detailed calculations. Contacts with both companies were made regularly by visit and mail to resolve design problems and interchange data.

Somewhat optimistic engine performance and weight data was given earlier by Marquardt, while the reverse was essentially true of Pratt and Whitney. Subsequent results received are in much better agreement between the two companies.

The mission performed starts at 125,000 feet and ends at approximately 140,000 feet. It is assumed that the vehicle will be boosted to the design Mach number of 3.0 and follow a Breguet range path at constant Mach number. Mach 3.0 was necessary to keep within the structural limits of the Marquardt engine, as well as the plastic airframe.

The Pratt and Whitney engine is shown on Figure 2. It is constructed of high temperature steels throughout. The fuel system is designed to vaporize the fuel within the cowl surfaces and center body. This general fuel system approach is proposed for both hydrogen and pentaborane. The same basic geometry was held using pentaborane as SF-1 except that the exit nozzle throat diameter was adjusted. This was done to match the higher combustion temperature considered optimum by Pratt and Whitney for pentaborane. The inlet shown is not the final Pratt and Whitney design. In place of the two step cone, an isentropic spike was used and the cowl lip geometry altered to match.

The use of the engine was limited earlier to 54.8" exit diameter by Pratt and Whitney facility capability. This was relaxed to 104" diameter as later

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facility availability data revealed possible. A graph showing Pratt and Whitney facility capability by 1959 and 1960 is shown in Figure 14. The subject of facilities is discussed later.

The Marquardt engine is shown on Figure 3. The proposed construction is plastic honeycomb except for the fuel system, flame holder, cooling shroud, and engine mounts. The plastic is a ceramic fiber impregnated with a high temperature phenolic. Plastics of this type are marketed under the trade name of "Refrasil". The maximum skin temperature allowable is 800 - 900°. Very little data is available for strength at these temperatures for time periods typical of the Hazel mission. The critical point is at the exhaust nozzle throat where the double skin area surrounding the exit nozzle has to be perforated to allow cooling by radiation leakage.

Marquardt is facility limited to 8' diameter as shown on Figure 15. They do not look at the scaling problem for this application as being a great risk, however. This is backed with considerable experience in the ram jet field.

The Marquardt engine, as was Pratt and Whitney's, is designed for vaporized fuel. In this case, too, the nose cone and recirculation zone walls are utilized but additional heat exchanger surface supplied by Convair is required. This is described in more detail under final tankage study results elsewhere in the report.

PERFORMANCE

Marquardt

Data presented by Marquardt for design point selection is shown on Figure 4 for the pentaborane fuel and Figure 5 for the SF-1 fuel. The "net jet" thrust coefficients are based on "A_3" as shown on the inset sketch Figure 5. These data are based on the combustion efficiency variation assumed for a 16' combustion chamber length and shown on Figure 6. Shorter lengths were examined but there was no real requirement. The weight of the additional length of combustion chamber was negligible compared to the loss in range caused by a reduction in length.

The engine selected has a geometry peculiar to the design points shown on Figures 4 and 5. The exit nozzle throat and exit areas "A5" and "A6", are given as ratios to a reference area A_3. These ratios are held for the entire graph while the inlet area "Ac" is allowed to vary to place the diffuser always at design pressure ratio. Thus, each point on the graphs represent a single engine geometry. The basis for choice of the particular engine in each case was a compromise between engine size and best specific fuel consumption. The choice of the particular set of A5/A3 and A6/A3 ratios resulted from the exchange of vehicle L/D, and gross weight data with Marquardt, which led to a narrower field of engine geometries giving best range of the total vehicle. The curves supplied by Marquardt and used for engine weights are shown on Figure 7. Here diameter and combustion chamber length is given along with the effect of altitude on the design weight of the engine at a combustion chamber length of 16 feet. Off design per-

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formance variation with Mach number is given for two altitudes with pentaborane fuel on Figures 8 and 9 . Effect of angle of attack is also shown on these figures.

Pratt and Whitney

Data presented by Pratt and Whitney for design point selection is presented in Tables 2 through 5 . The size of this engine was fixed at what Pratt and Whitney considered reasonable for full scale testing. Given the same general input on mission requirements, a basic engine geometry was established by Pratt and Whitney. A possible alternate was provided for the SF-1 engine only. These are designated SRJ-43D for both the SF-1 and pentaborane engines, while the SF-1 alternate is the SRJ-43E. The engines were scaled down in size where necessary but not up, as this would exceed facility limits. The scaling curves provided by Pratt and Whitney are given in Figure 10.

Performance and basic physical data of the engine is given at altitudes from 80,000 feet to 150,000 feet on a standard day at the design Mach number of 3.0 and for 80,000 to 135,000 feet at the off design condition of Mach 2.5. The off design data was requested of Pratt and Whitney for turns and/or climb. Angle of attack effect on performance was also provided by Pratt and Whitney to determine the effect of a 2° trim error on angle of attack. This is also shown on the tables.

Comparative Data

The engines selected are arranged in tabular form below with pertinent physical dimensions and performance. All engines are for a design point of Mach 3.0 on a standard day. The range for all cases is 3240 nautical miles.

MFGR.	Engines	Fuel	Dimension, In. Diameter Inlet	Dimension, In. Diameter Exit	Dimension, In. Length	Total Engine Weight Pounds	Total Thrust Pounds	Fuel-air Ratio	Specific Fuel Consumption
P & W	(Vehicle: PC 22; Altitude: 135,000 ft. at start of cruise)								
	2	SF-1	73.0	89.3	226	1710	2382	.0148	.880
	(Vehicle: PC 20; Altitude: 125,000 ft. at start of cruise)								
	2	PB	68.3	82.7	221	1590	3520	.0366	1.950
MAR-QUARDT	(Vehicle: MC22; Altitude: 139,000 ft. at start of cruise)								
	2	SF-1	84.7	104.7	808	920	2260	.0150	.970
	(Vehicle: MC10; Altitude: 125,000 ft. at start of cruise)								
	1	PB	125.2	153.5	786	1460	3640	.0175	1.460
	(Vehicle: MC20; Altitude: 125,000 ft. at start of cruise)								
	2	PB	85.7	105.3	808	1340	1695	.0175	1.460

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As can be seen, Marquardt presented pentaborane engine data at lower fuel - ratios favoring the plastic construction. The lower fuel - air ratios dictated a larger engine. Marquardt engine weights are lower despite the larger size by a considerable margin. This is due to two factors; (a) the plastic construction with a more liberal use of honeycomb structure and (b) the size restriction by Pratt and Whitney which required that two smaller engines be used instead of one larger engine, with higher weights as shown in Figure 10.

An obvious difference in the two engines is in the over-all lengths. This is due to the difference in combustion chamber lengths. Marquardt used 16 feet while Pratt and Whitney used 4 feet. Marquardt may be quite conservative but the question of which is correct can only be resolved through adequate testing. As yet, neither company has measured combustion efficiency accurately enough. This may not be an important issue, however, as very little engine weight is involved and the space requirements of most vehicles studied will permit both engine lengths.

A check was made by Convair on the performance estimates of the two companies. This was done at comparative performance points using Convair methods and without knowledge of the complete cycle assumptions made by either engine company. The results are shown in Figures 11 through 13. The agreement was very good in all cases on both engine geometry and performance, and is considered adequate to substantiate both companies estimates. The differences that do exist may be caused by slightly differing diffuser efficiencies, assumptions of combustion total head loss and degree of dissociation and recombination in the exhaust nozzle.

TESTING AND FACILITY REQUIREMENTS

Both Marquardt and Pratt and Whitney have adequate home facilities for ram jet testing. Pratt and Whitney will have capacity by 1960 for testing its 86 inch engine as shown on Figure 14. At Mach 3.0 there appears to be sufficient margin to operate with the exit nozzle throat sonic. Marquardt facility capacity is shown on Figure 15. The 8 foot diameter engine can be operated with the exit nozzle throat sonic simulating the Mach 3.0 case at 125,000 feet. Certainly, the two engine versions of both Pratt and Whitney and Marquardt's engines can be tested with the facilities available in the time period. It is also very probable that Marquardt's single engine is within scaling distance of the 8 foot diameter engine which would be simulated with sonic exit at 125,000 feet altitude and with subsonic exit and combustor Mach number matching at 140,000 feet.

Capacity is also available at the present time at NACA to handle a 10.5 foot engine to 117,000 at Mach 3.0 with sonic exit. A.E.D.C. plans within one year to handle an 8 - 9 foot diameter engine at 135,000 feet under the same conditions.

CONCLUSIONS AND RECOMMENDATIONS

The estimates of engine performance appear to be correct depending upon the validity of the combustion efficiencies assumed. It is recommended that

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testing be directed toward substantiating the values assumed as this affects range directly.

There appears little question that Pratt and Whitney can build the engine presented using the materials selected. Similarly, Marquardt could build their plastic design but it is obvious that more development work would be required to obtain the advantage that plastic offers in weight saving. It is also not entirely clear that by use of honeycomb structure the metal engine could not have been made lighter.

Both fuels have undesirable logistics characteristics but are considered essential to do this high altitude mission. Hydrogen appears to present the least over-all problems from the propulsion standpoint. Its volume characteristics appear to require a two engine vehicle.

Facilities apparently can be made available in the required time period that will satisfy the basic engine needs either at home facilities or Government test laboratories.

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STRUCTURAL HEATING ANALYSIS

INTRODUCTION

The basic problems of fuel and structural heating have been evaluated. Aerodynamic heating and heating effects from the engine, contribute to increase the temperature of the basic structures. Both factors are considered in the analysis.

SUMMARY

Structural temperatures will not exceed reasonable operating limits for the materials proposed, with the exception of some sections of the engine case. Some areas of the engine case may require additional materials study for an optimum design.

Fuel heating will not be a major problem if a liquid fuel system is selected. No insulation will be required for a liquid pentaborane system. A hydrogen fuel system would only require insulation to avoid icing conditions.

The wing surface temperature will vary from 630° F at the leading edge to 400 and 300° F at one-foot and ten-feet from the leading edge respectively.

RECOMMENDATIONS

During early stages of development, run heat flow test across simulated engine walls to ascertain thermal transmission. Radiation properties are of prime importance.

Determine rates of decomposition and deposits within fuel controls and the heat exchanger if vaporized pentaborane is used as fuel.

DISCUSSION OF RESULTS

Fuel System Heating

Two types of fuel systems were investigated, gaseous and liquid injection. Pentaborane and SF-1 were considered for both systems. With the liquid injection systems, the possibility of fuel losses by evaporation, and malfunction of the fuel system due to vapor entrainment, are two major problems. The major problems in a gaseous distribution system are the correct sizing of generators, or heat exchangers, to vaporize the liquid, and considerably larger flow controls than normally needed for liquid systems.

Pentaborane fuel can be used as a liquid in the proposed tank arrangements without requiring insulation to avoid overheat, or boiling, at a 15 psi fuel system pressure. This requires a fuel temperature at take-off below 60° F, which is not considered restrictive. This system will result in lesser maintenance problems due to the absence of deposits of decomposed fuel elements. The analysis was based upon cylindrical fuel cells. If the internal wing volume were used to store fuel in bulk, a small amount of insulation may be required on the lower surface to maintain fuel temperatures below boiling. A schematic diagram of a liquid system is shown on Figure16. Due to a possible fire hazard, the fuel tank pressure relief line must be vented downstream of the vehicle. Fuel decomposition is negligible but the system should be flushed after each flight. Deposits in the fuel system that occur, due to temperature, are absorbed by fuel at temperatures below 100° F and sea level pressure. Therefore, fuel may be used to flush the system after each flight.

Figure 16 describes a vapor feed system for pentaborane. With a vapor feed system a large amount of energy is absorbed by the fuel during vaporization. It is evident from the analysis that a minimum of 1500 sq.ft. of external surface would be required to evaporate the fuel, at the required rate, by aerodynamic heating. A heat exchanger may be made as an integral part of the engine wall using only 75 sq.ft. of surface. Any deposits within the heat exchanger can be removed by flushing after each flight. The fuel flow diagram shows the gaseous fuel bubbling through the liquid fuel. This will minimize the deposits within the flow controls and spray nozzles.

Sufficient vapor for starting must be stored within the tanks. to minimize the storage volume the pressure at light-off must be at a system maximum, and the temperature must be at the boiling point. This will allow vapor generation by lowering the tank pressure during the time the heat exchanger is becoming operative. The required vapor boil-off rate is maintained by controlling the pumping rate through the heat exchanger.

SF-1 fuel has the inherent problem of boil-off at very low temperature. While this is helpful in flight, in reducing heat exchanger size, it creates high fuel losses and icing problems during and previous to launch.

Approximately two-inches of insulation will be required to avoid icing. A weight saving of the vehicle may result by developing rapid fuel handling techniques and accepting the icing penalties encountered during last minute ground check out of launch. A minimum weight exchanger would probably be one that is an intergral part of the engine wall. This arrangement would require a heat exchanger of approximately 50 sq.ft., based on a fuel consumption of 4000 #/hr.

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STRUCTURAL HEATING

The basic airframe heating is caused by the usual aerodynamic and solar heating. Some areas are also effected by radiation from the engine surfaces.

The selection of materials is a major factor in the thermal analysis of the engine case. Due to the high temperature of the combustion gases, 3100° R, the inner surface of the engine absorbs large quantities of heat, both by radiation and convection. The amount of structural cooling done by inner passage air flow, or external flow, is limited by the high energy level of the ambient air stream (approximately a 1250° R boundary layer and a 1350° R stagnation temperature) and the low convective heat transfer coefficients. This means that engine structures must rely on thermal radiation to the atmosphere for cooling. Preliminary investigation shows that the materials selected by the engine manufacturer can be surfaced to control thermal emissivity and, therefore, the temperatures can be maintained within the limits to which the materials can perform.

Limited information is available on the deposits of combustion products on the engine walls. Additional data is also required on the gaseous radiation to the engine walls. Both of these areas will have to be investigated for an optimum design of the engine structure.

Placing the heat exchanger, for vaporizing the fuel, on the inside engine wall will result in lower structural temperatures in a local area. Some advantage may be gained by this in the detailed design.

The maximum heating during the cruise portion of the trajectory of the proposed Hazel vehicle occurs at its beginning (M = 3 @ 125,000 ft.)

The temperatures of the wing were determined from steady state equilibrium heat balances by equating the engine, solar, aerodynamic, and terrestrial heating, to the radiation to space. The flow field at this condition would be laminar.

The aerodynamic heating for the flat portion of the wing was evaluated by the Reference Temperature Method (2) and the predicted temperatures are 400 and 300° F at one-foot and ten-feet respectively, from the leading edge.

The nose stagnation temperature of the vehicle was determined by the method of Sibulkin (1). The temperature determined by this method was 725° F, considering two inch radius.

The temperature of the stagnation line of the leading edge was also determined by the method of Sibulkin, but modified by the cosine of the effective sweep angle in order to account for the sweep of the leading edge. The temperature determined by this method was 630° F, considering a two inch radius.

None of the above predicted temperatures have been found to be prohibitive for the proposed structure.

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STRUCTURAL PRESSURIZATION SYSTEM

INTRODUCTION

The inflatable configuration of the vehicle consists of a rigid pilot's capsule and engine structure supported by a pressurized airfoil. This report evaluates various systems for supplying this pressurization and outlines those found most promising. From these, selection is made on the basis of minimum weight and operational suitability.

CONCLUSIONS

1. Minimum system weight is afforded by a system utilizing helium stored in the liquid state and heated by direct mixing with hot gas from the monopropellant APU hot gas generator. Total weight of the proposed system is 142 pounds.

2. Pure helium free of the hydrazine decompositon products can be supplied to the structure by an alternate system for a 19 pound weight penalty. Alternate system weight is 161 pounds.

3. Tests should be conducted to determine compatability with structure and explosive hazard of gas mixture containing hydrazine decomposition products at operating conditions.

4. Data on leakage rates for materials and construction employed should be obtained and all possible steps taken to reduce these quantities.

SYSTEM REQUIREMENTS

Initial pressurization is supplied on the ground prior to take-off. Means must be provided for the controlled escape of a portion of this gas with (1) decreasing ambient pressure as the vehicle is lifted to 45,000 feet and boosted to 125,000 feet and (2) increased internal temperature due to aerodynamic heating during cruise. Following this loss and stabilization at cruise conditions, gas must be added to offset leakage and maintain the given 15 psig pressure differentail as increasing ambient pressures are encountered during let-down from altitude. The inlet gas must be injected at such temperatures as to preclude thermal damage to the structure and to minimize total system weight. The pressurizing medium chosen must remain a gas over the temperature and pressure range encountered within the structure.

SYSTEMS CONSIDERED

The requirements on the pressurizing medium of (1) remaining a gas over the operating temperature range of the structure, and (2) having low weight, suggest the use of the low molecular weight gaseous elements. Table I below lists some properties of these gases.

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TABLE I

Properties of Pressurizing Gases

Gas	Density Referenced to H_2	Critical Temperature, °F	Critical Pressure atm	Remarks
Hydrogen (H_2)	1	-400	12.2	Highly Inflamable
Helium (H_e)	2	-450	2.3	Inert
Nitrogen (N_2)	14	-233	33.5	Inert
Oxygen (O_2)	16	-182	49.7	Reactive with Structural Material
Neon	10	-380	26.9	Inert
Argon	20	-88	48.0	Inert

Hydrogen has the lowest density and thus affords the minimum weight penalty for the pressurizing gas itself but presents an explosive hazard. Oxygen is heavy and could react chemically with structural members at elevated temperatures. Neon offers no advantages over Helium, is heavier and less readily available. Similarly, Argon affords no advantage over Nitrogen. Thus, the choice from this group for the pressurizing gas is between Nitrogen and Helium. Table II below lists the advantages and disadvantages of these two gases as the pressurizing medium.

TABLE II

Comparison of Helium and Nitrogen as Pressurizing Medium

Gas	Advantages	Disadvantages
Helium	Density 1/7 as great	Must be transported to point of use; higher leakage rate; liquifies only at extremely low temperature.
Nitrogen	Can be produced at site of use in either liquid or gaseous form; liquifies at higher temperature.	For equal volume leakage, 7 times weight of H_e required.

Also to be considered are the low molecular weight compounds existing as gases at the temperatures and pressures considered. Those include such compounds as ammonia (NH_3, M = 17), methane (CH_4, M = 16), and others. Some of these, such as ammonia, offer the advantage of remaining a liquid at ambient temperatures and only moderate pressures and would thus require a simpler and lighter container system. However, almost all of these compounds are toxic and/or inflammable and afford no over-all weight advantage as seen below.

Based on the vehicle requirements as outlined in the following section of this report, the required weights of various gases for the let-down re-pressurization with zero leakage were calculated and listed in Table III below. This

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weight is for gas alone and includes no allowance for container weight.

TABLE III

Required Weights of Gases for Let-Down Re-Pressurization

Helium	34 lbs
Nitrogen	238 "
Ammonia	144 "
Methane	136 "

The following calculation demonstrates the necessity of using a low molecular weight gas for the pressurization of vehicles of this size. Assuming the required weight of helium to be 34 lbs., data from Reference 3 gives the weight of a suitable storage container for the gas in liquid form as

$$\left(\text{container weight}\right) = 16 + 1.53 \times (\text{weight of } H_e)$$
$$\left(\text{container weight}\right) = 68 \text{ pounds}$$

or a combined gas and container weight of 102 pounds. Considering the gas weight alone, the density of a second gas must be less than $\frac{102}{34} = 3$ times greater than that of helium to show a weight saving. This second gas must therefore have a molecular weight less than 12. This condition is met by only three substances besides helium which are gases under the operating conditions; hydrogen fluoride, hydrogen, and neon. Hydrogen fluoride is dropped from consideration due to its extreme corrosiveness while hydrogen and neon were discussed and rejected previously.

PROPOSED SYSTEM

The configuration and conditions assumed are listed below in Table IV.

TABLE IV

Configuration and Assumed Conditions

Configuration	MC-10
Cruise duration	98-minutes
Descent duration	10-minutes
Assumed gas temperature at cruise	300° F
Assumed gas temperature at landing	-50° F
Assumed gas temperature at take-off	60° F
Wing area	1985 ft^2
Total area of pressurized sections	3967 ft^2
Inflatable volume	1720 ft^3
Structural pressure differential	15 psig
Launch altitude (boost)	45,000 feet
Cruise altitude	125,000 - 137,800 feet

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The proposed system is shown schematically in Figure 17 and an alternative system in Figure 18. Both systems utilize initial pressurization before take-off with helium gas, with make-up gas for leakage and let-down re-pressurization supplied from a storage bottle of liquid helium. Both systems utilize hot gas from the monorpopellant (hydrazine) auxiliary power supply hot gas generator for heating the very cold helium prior to its use in the structure. They differ only in the method by which this heating is accomplished. The former utilizes direct mixing of the hot and cold gases while the latter passes the gases through a heat exchanger allowing only pure helium to enter the structure. The proposed direct mixing system requires less total gas and does not require the added weight of the heat exchanger. However, the compatibility with structural materials and the safety of the resulting gas mixture containing the hydrazine decomposition products must be proven by tests. As shown in Table V, the maximum concentration of hydrogen within the structure is 5% by volume which is within the explosive limits of hydrogen in air (4.1 to 47.2% by volume). Oxygen within the inflated volume will however, be limited, due to the positive pressure differential above the ambient, to leakage from the pilot's capsule. In addition, it should be noted that due to adding the helium gas cold for leakage make-up as noted below, the structure contains only pure helium during all phases up to and including cruise with hydrazine gas added only during the let-down phase.

Both systems make use of electrical heaters within the liquid helium storage tank for maintaining internal pressure as gas is withdrawn. Operation of both is based on the assumption that make-up for leakage during cruise would require very low flow and could be made with unheated gas direct from the liquid tank with heating supplied from the hot structure.

Table V shows the amount and composition of gas present in the structure at various phases of the flight for both systems.

TABLE V

Structural Gas Content and Composition

Phase of Flight	Item	Proposed System Direct Mixing	Alternate System Pure H_e in Struct.
Take-off from Sea Level	Gas in Structure (H_e)	37 lb.	37 lb.
	Air Displaced	132 lb.	132 lb.
	Net Lift	95 lb.	95 lb.
Start of Boost	Gas in Structure (H_e)	28 lb.	28 lb.
Stabilized Cruise	Gas in Structure (H_e)	13 lb.	13 lb.
Landing	Gas in Structure	58 lb.	47 lb.
	H_e	42 lb.	47 lb.

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TABLE V
(continued)

Phase of Flight	Item	Proposed System Direct Mixing	Alternate System Pure He in Struct.
Landing (continued)	H_2	1 lb.	0 lb.
	N_2	10 lb.	0 lb.
	NH_3	5 lb.	0 lb.
	Air Displaced	132 lb.	132 lb.
	Net Lift	74 lb.	85 lb.
	Gas Composition % by Volume		
	H_e	72 %	100 %
	H_2	2 %	
	N_2	17 %	
	NH_3	9 %	

A weight breakdown of the two systems is given in Table VI. This weight includes a 20% safety factor on required amounts of gas.

TABLE VI

System Weight Breakdown

	Proposed System Direct Mixing	Alternate System Pure He in Structures
Liquid Helium	35 [1,2] lbs.	40 [1,2] lbs.
Helium Dewar	70 [1,2] lbs.	77 [1,2] lbs.
Mixing Chamber	2 lbs.	
Heat Exchanger		5 lbs.
Valves	7 lbs.	7 lbs.
Ducting & Miscellaneous	5 lbs.	5 lbs.
Sub Total	119 lbs.	134 lbs.
Hydrazine (Hot Gas)	19 lbs.	22 lbs.
Sub Total	138 lbs.	156 lbs.
Batteries for Electric Heater	4 lbs.	5 lbs.
TOTAL	142 lbs.	161 lbs.

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1 - Weight includes a 20 percent safety factor on required gas weight.
2 - These weights will be increased due to leakage as outlined in following paragraph.

At the time of writing of this report no data was available on leakage rates for the materials and construction proposed for use. An expression for the weight penalty was therefore derived on the basis of known flight parameters and presented as a function of the leakage rate. For this purpose it was assumed that all leakage consisted of, and was replaced by, pure helium gas. For a given leakage rate the total gas weight lost is given by

$$(\text{gas weight lost}) = \phi \rho_{STP} \tau A (\Delta p) \text{ lbs.}$$

where:

ϕ = leakage rate - cfm_{stp}/ft^2 psig

ρ_{stp} = gas density at t = 0°C, p = 1 atm - lb/ft^3

τ = flight duration- minutes

A = surface area for leakage – ft^2

Δp = pressure differential - psig

By data from Reference 1

$$(\text{additional weight dewar}) = 1.53 (\text{gas weight lost})$$

or

$$(\text{total added weight}) = 2.53 \phi \rho_{STP} \tau A (\Delta p) \text{ lbs.}$$

From the configuration data of Table IV

$$(\text{weight gas lost}) = 71.5 \phi \times 10^3 \text{ lb.}$$

$$(\text{added bottle weight}) = 109 \phi \times 10^3 \text{ lb.}$$

$$(\text{total added weight}) = 181 \phi \times 10^3 \text{ lb.}$$

If the structure is assumed to consist of rubber 4-mils in thickness, extrapolation from International Critical Tables data gives

$$\phi = 33.4 \times 10^{-6} \text{ cfm/ft}^2 \text{ psig}$$

$$(\text{weight gas lost}) = 2.4 \text{ lb.}$$

$$(\text{added bottle weight}) = 3.6 \text{ lb.}$$

$$(\text{total added weight}) = 6.0 \text{ lb.}$$

However, the leakage rate stated for a somewhat similar material yields results 50-times the above. In addition, it should be noted that this additional weight calculated above compensates for leakage by diffusion through the material only and not through any holes which may be present due to construction

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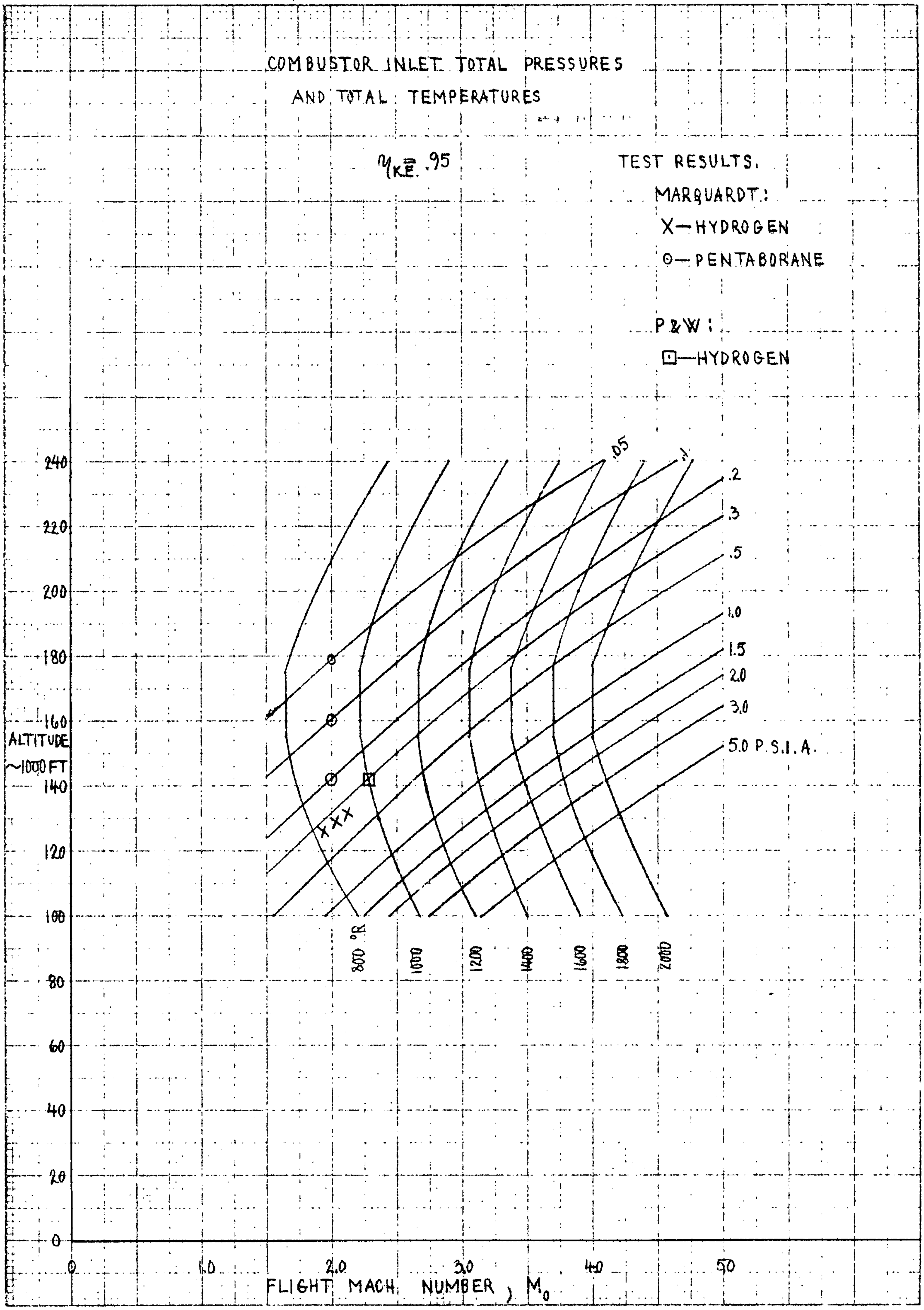
or damage. Loss of Helium alone due to this latter cause could run to the order of 6.5 pounds/minute for one-square inch of hole. This would give a total weight penalty of 16.5 pounds for each square inch of holes per minute of leakage time.


COMBUSTOR INLET TOTAL PRESSURES
AND TOTAL TEMPERATURES
$\eta_{KE} = .95$
TEST RESULTS.
MARQUARDT:
X—HYDROGEN
⊙—PENTABORANE
P&W:
⊡—HYDROGEN
.05
.1
.2
.3
.5
1.0
1.5
2.0
3.0
5.0 P.S.I.A.
800 °R
1000
1200
1400
1600
1800
2000
ALTITUDE ~1000 FT
240
220
200
180
160
140
120
100
80
60
40
20
0
1.0
2.0
3.0
4.0
5.0
FLIGHT MACH NUMBER, M_0

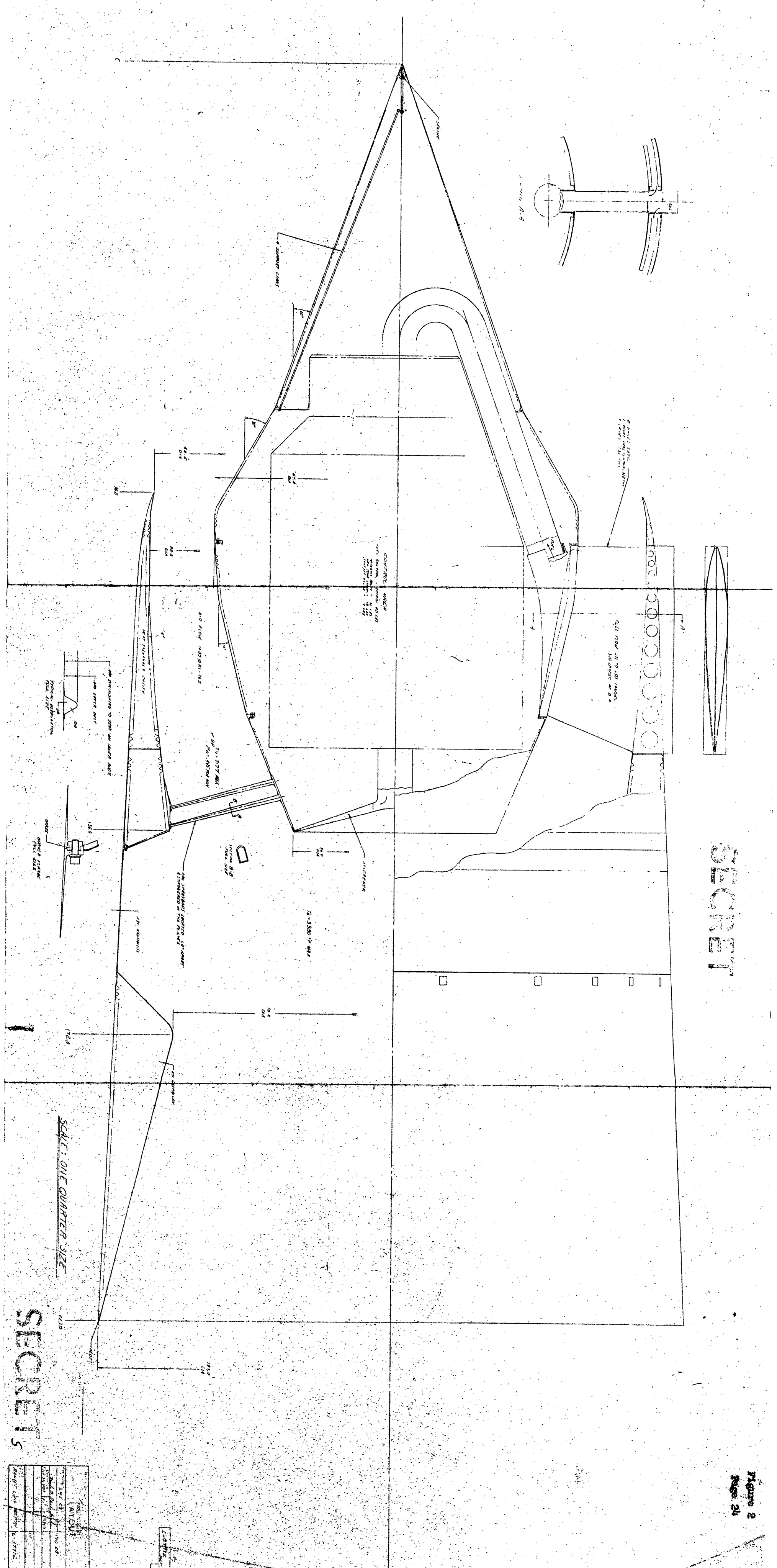


Figure 2
Page 24

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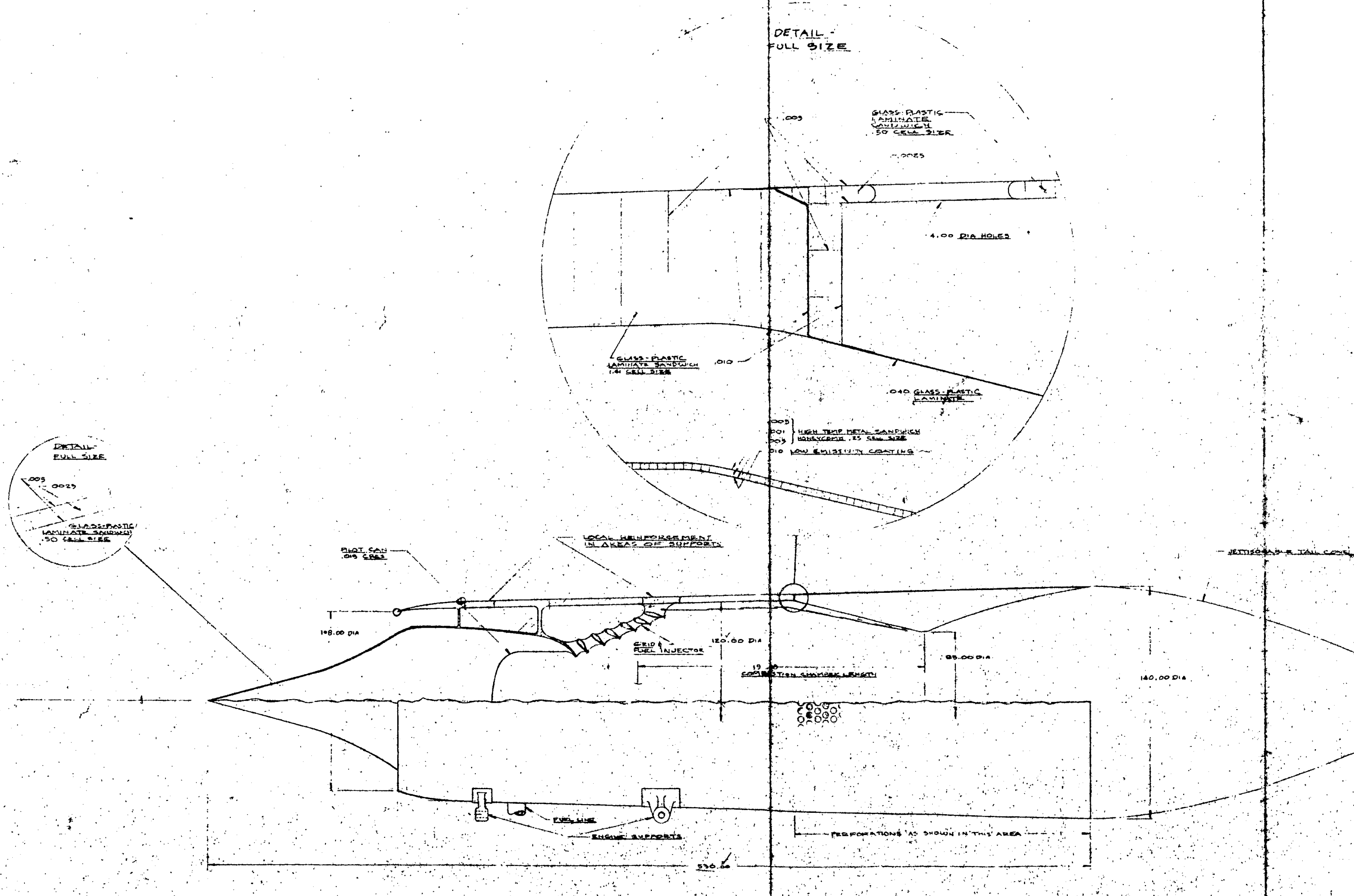

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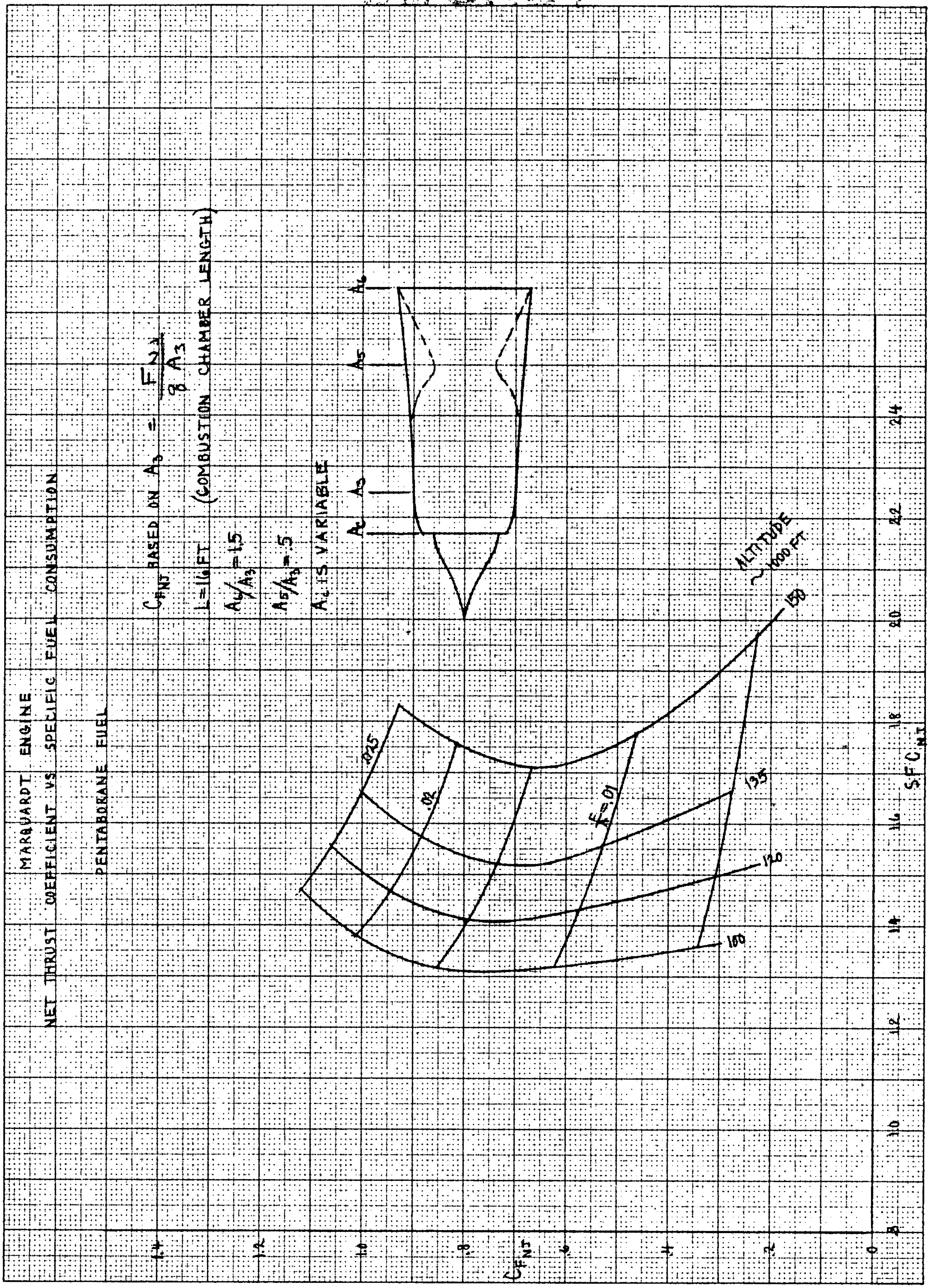
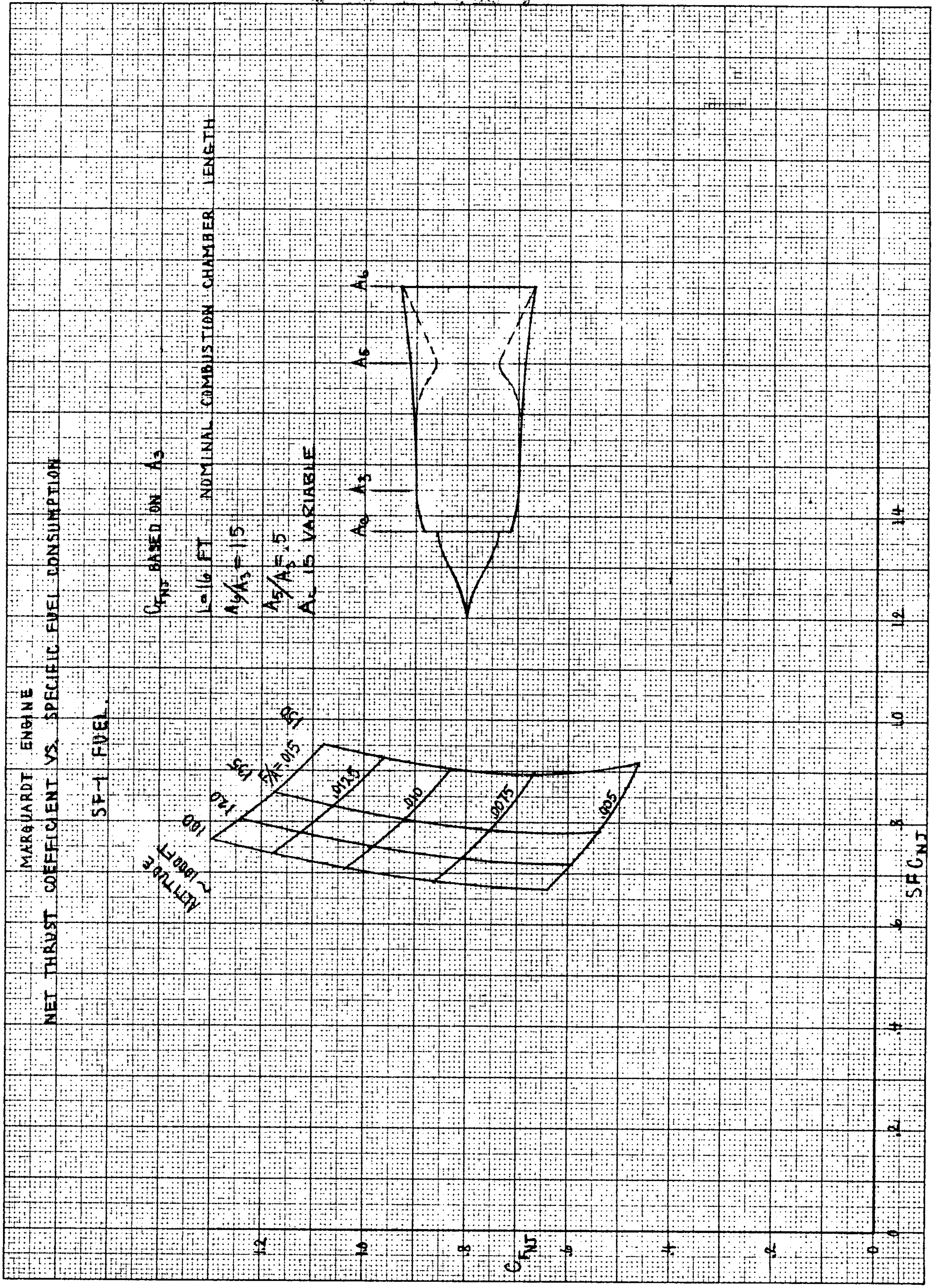
Marquardt
Ram Jet

5

Declassified and Approved For Release 2012/05/31 : CIA-RDP89B00709R000400810001-5


MARQUARDT ENGINE
NET THRUST COEFFICIENT VS SPECIFIC FUEL CONSUMPTION
PENTABORANE FUEL
$C_{F_{NT}}$ BASED ON $A_3 = \frac{F_{NT}}{q A_3}$
L=16 FT (COMBUSTION CHAMBER LENGTH)
$A_6/A_3 = 1.5$
$A_5/A_3 = .5$
A_2 IS VARIABLE
A_2
A_3
A_5
A_6
ALTITUDE ~1000 FT
150
135
120
100
.025
.02
$\frac{F}{A} = .01$
$C_{F_{NT}}$
SFC_{NT}
0
.2
.4
.6
.8
1.0
1.2
1.4
8
10
12
14
16
18
20
22
24

SECRET



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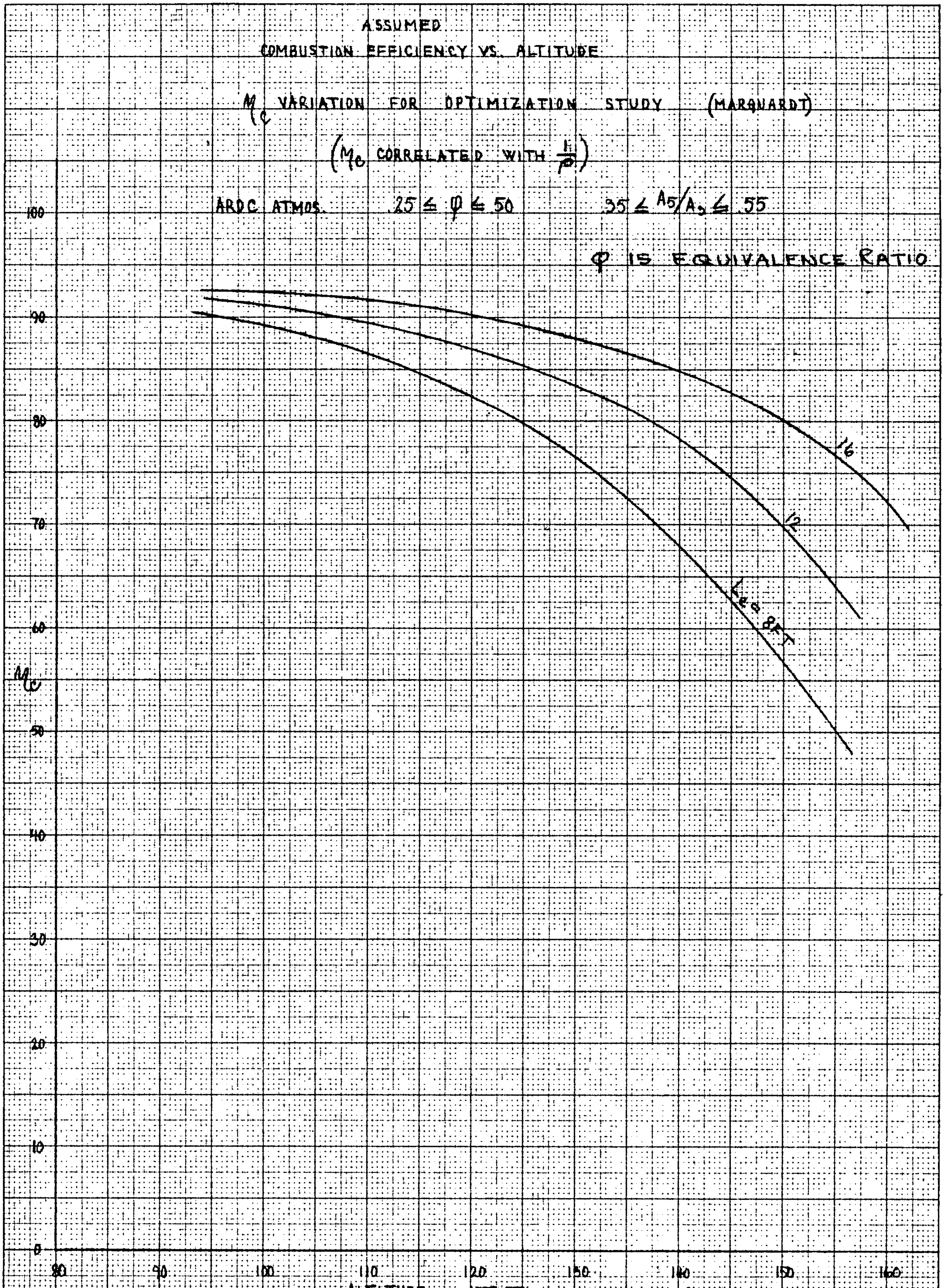
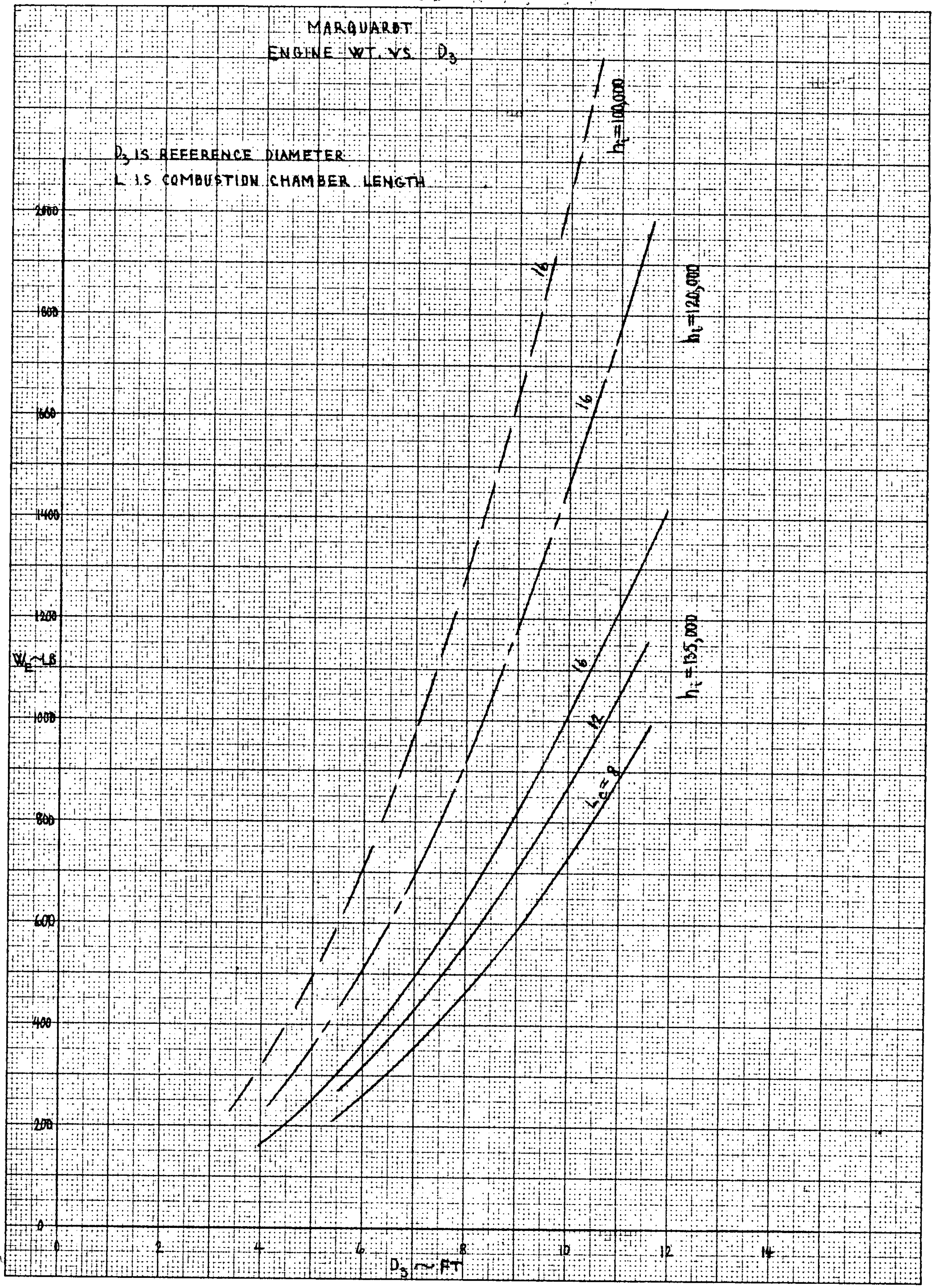
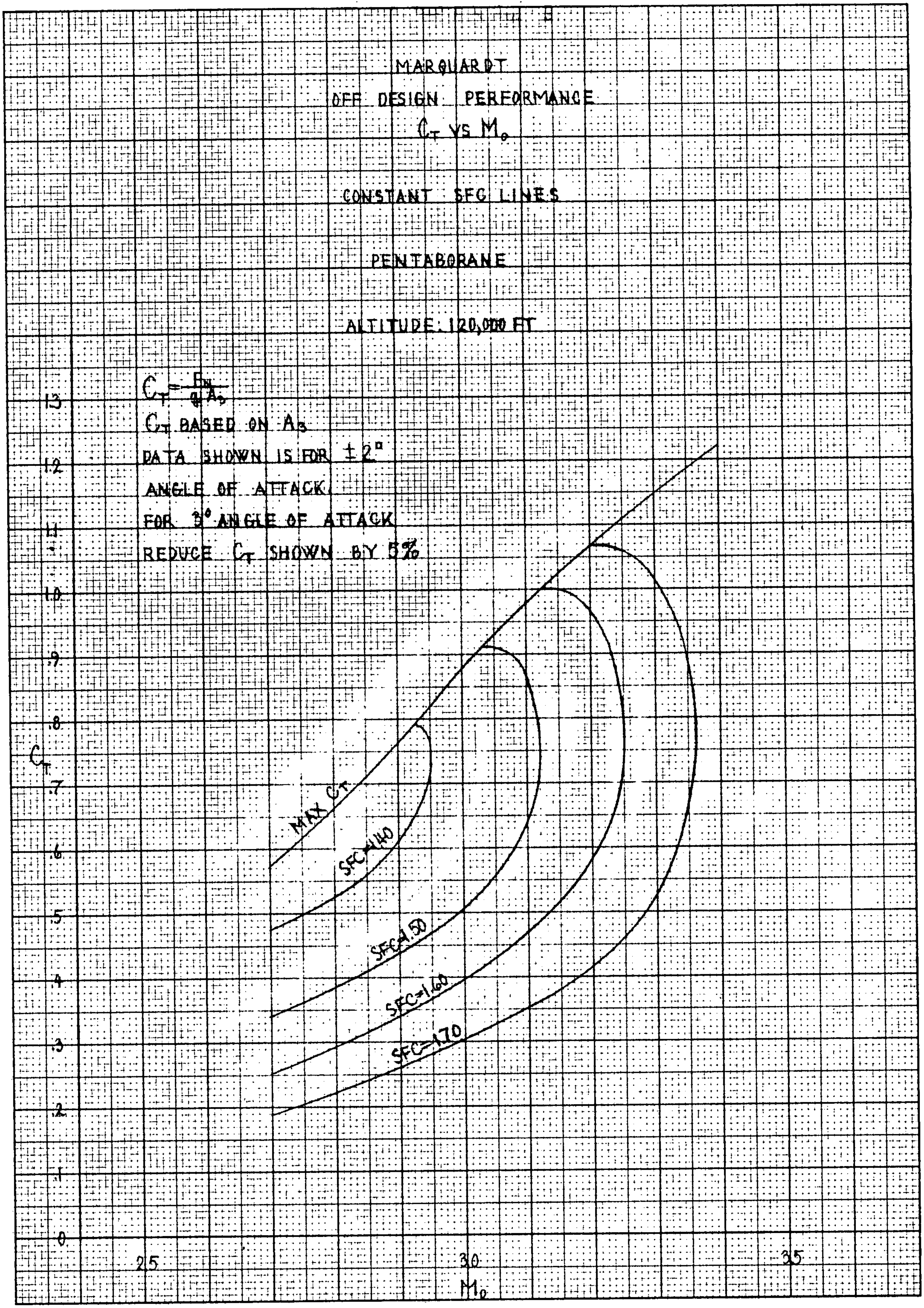
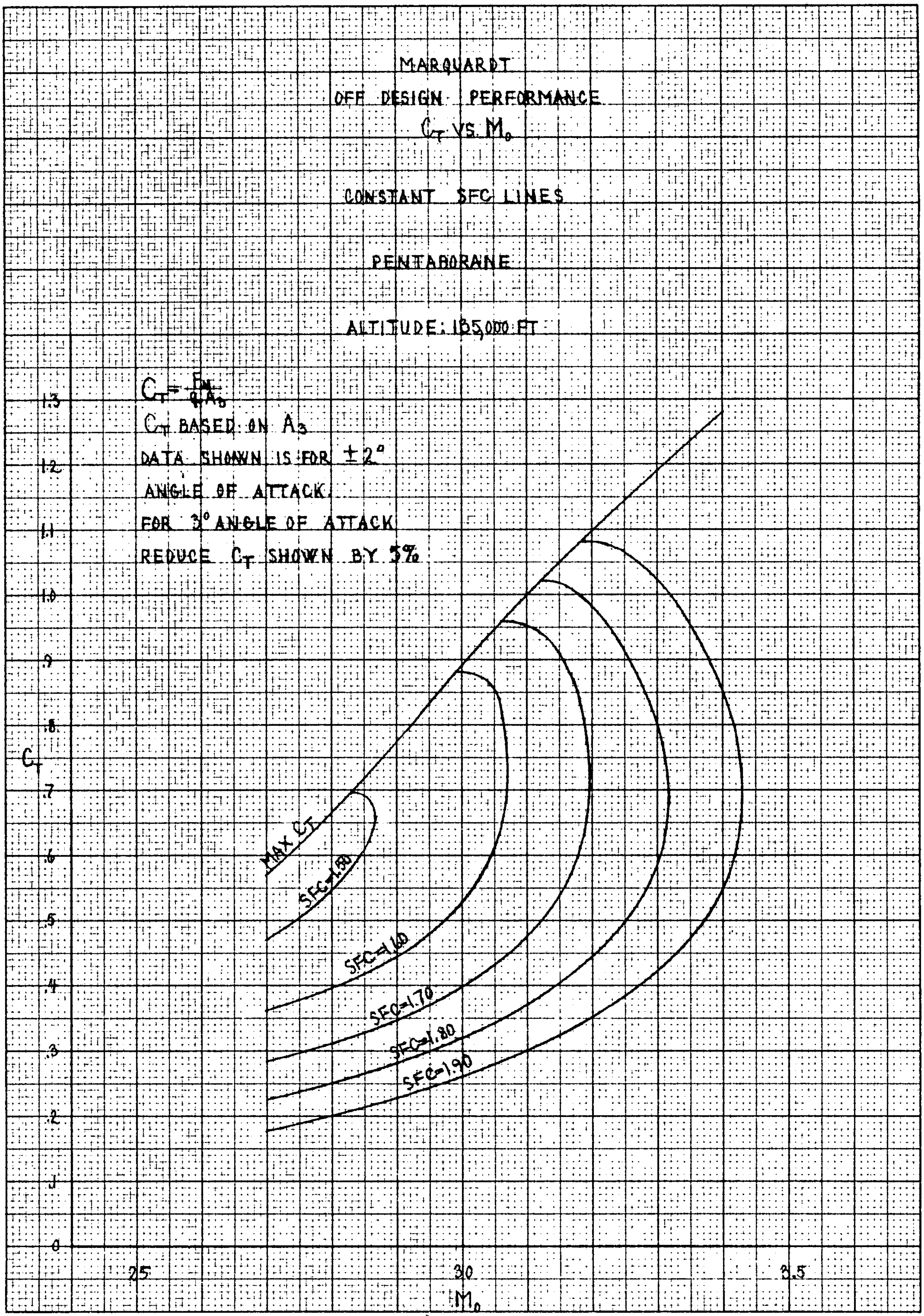
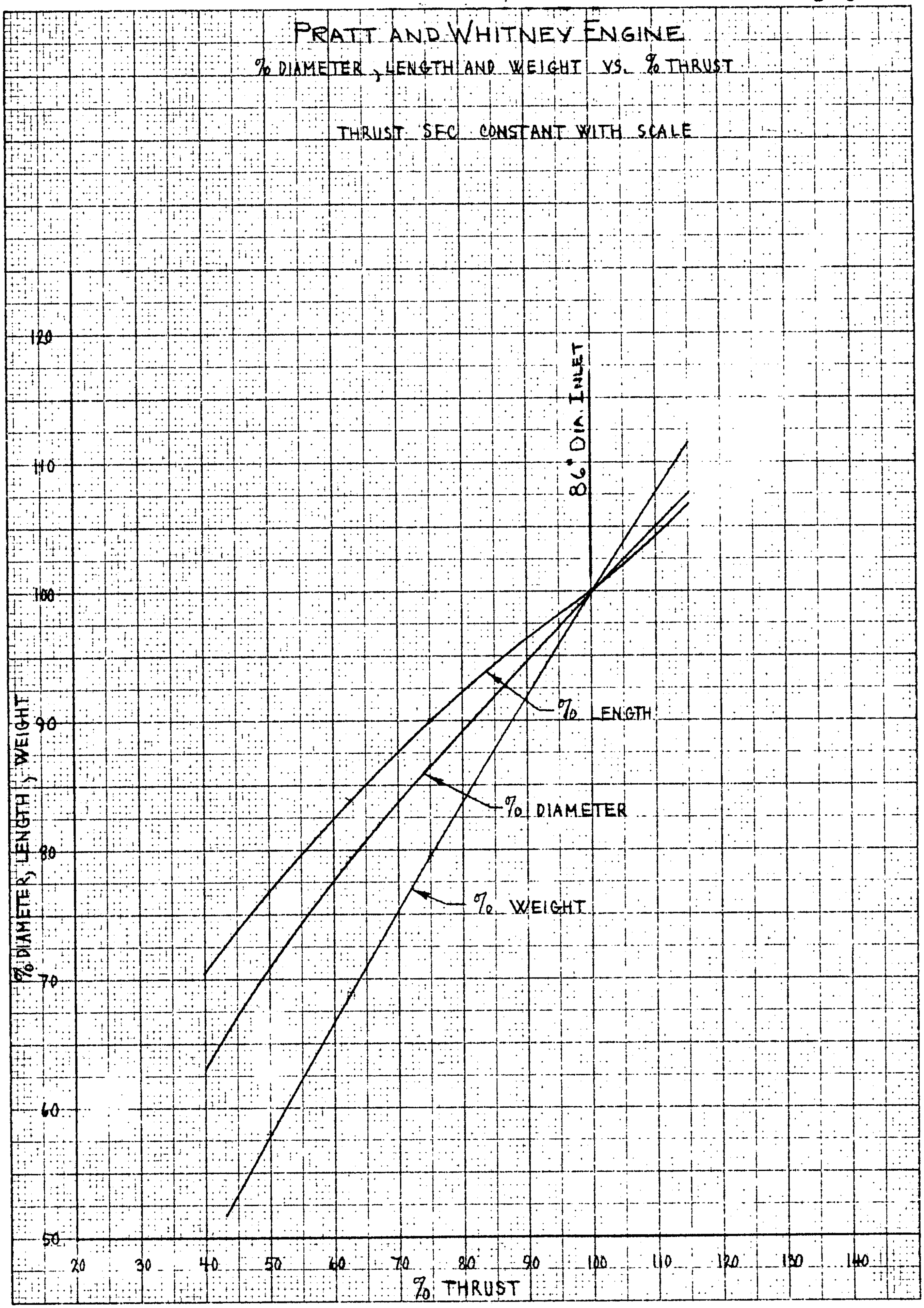

ASSUMED
COMBUSTION EFFICIENCY VS. ALTITUDE
η_c VARIATION FOR OPTIMIZATION STUDY (MARQUARDT)
(η_c CORRELATED WITH $\frac{H}{P}$)
ARDC ATMOS. .25 ≤ φ ≤ .50 .35 ≤ A5/A3 ≤ .55
φ IS EQUIVALENCE RATIO
16
12
L_c = 8 FT
η_c
100
90
80
70
60
50
40
30
20
10
0
80
90
100
110
120
130
140
150
160
ALTITUDE ~ 1000 FT

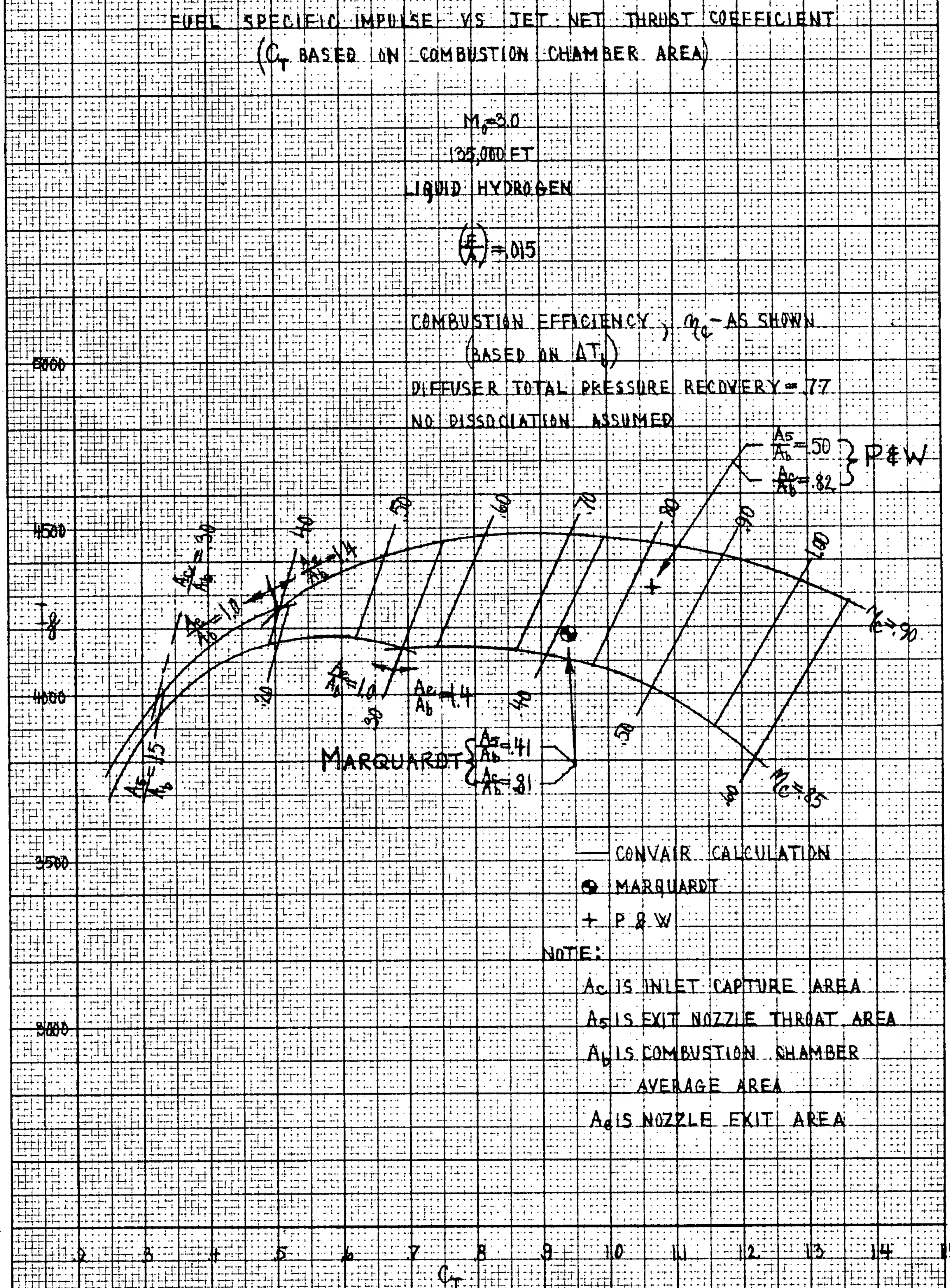

MARQUARDT
ENGINE WT. VS. D_3
D_3 IS REFERENCE DIAMETER
L IS COMBUSTION CHAMBER LENGTH
h_t=100,000
h_t=120,000
h_t=135,000
16
16
16
12
L_c=8
W_E~LB
2000
1800
1600
1400
1200
1000
800
600
400
200
0
0
2
4
6
8
10
12
14
D_3 ~ FT


MARQUARDT
OFF DESIGN PERFORMANCE
C_T VS M_0
CONSTANT SFC LINES
PENTABORANE
ALTITUDE: 120,000 FT
$C_T = \frac{F_N}{q A_3}$
C_T BASED ON A_3
DATA SHOWN IS FOR ±2°
ANGLE OF ATTACK.
FOR 3° ANGLE OF ATTACK
REDUCE C_T SHOWN BY 5%
MAX C_T
SFC=1.40
SFC=1.50
SFC=1.60
SFC=1.70
C_T
M_0
1.3
1.2
1.1
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
0
2.5
3.0
3.5

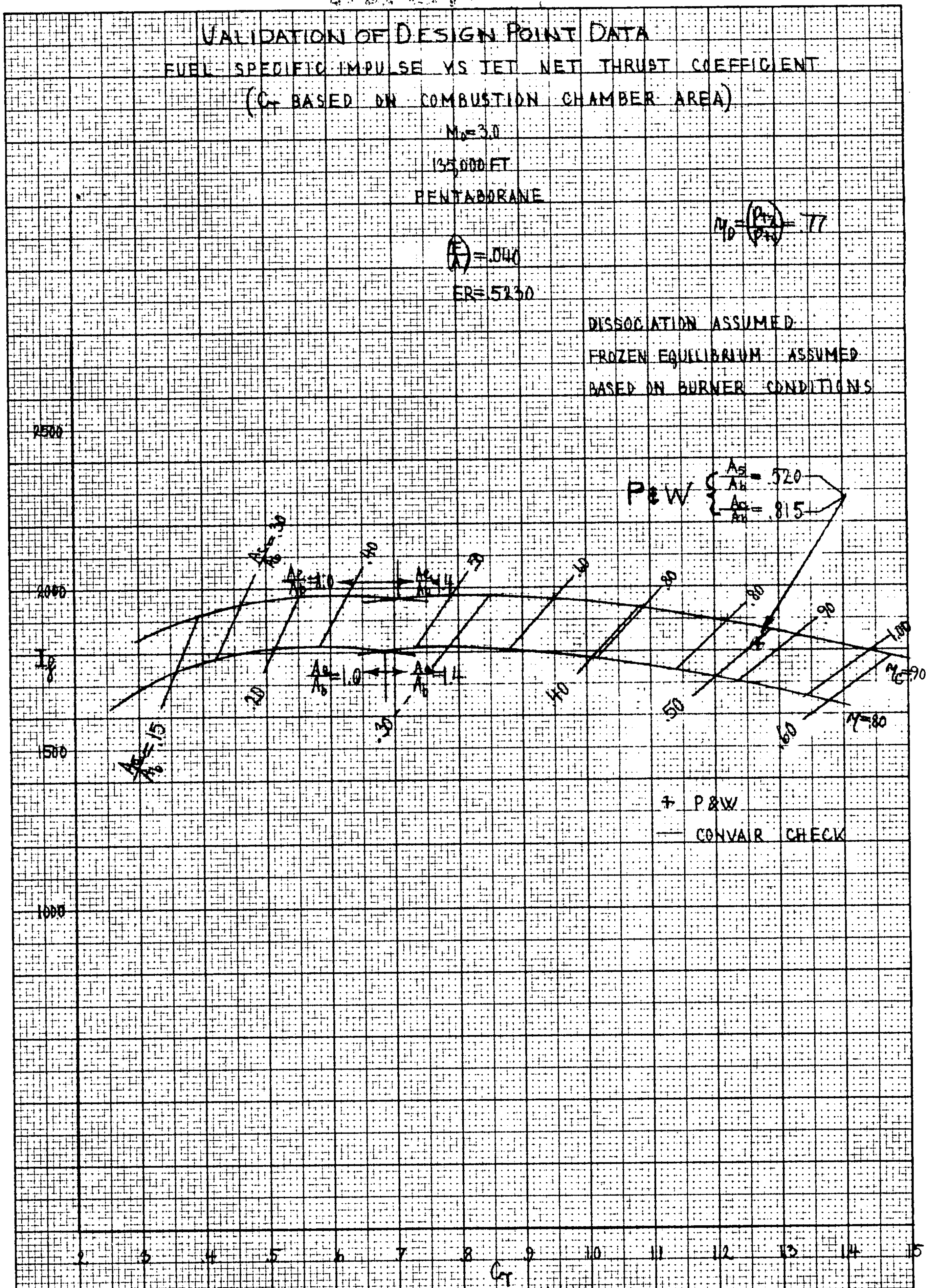

MARQUARDT
OFF DESIGN PERFORMANCE
C_T VS. M_0
CONSTANT SFC LINES
PENTABORANE
ALTITUDE: 135,000 FT
$C_T = \frac{F_N}{q A_3}$
C_T BASED ON A_3
DATA SHOWN IS FOR $\pm 2°$
ANGLE OF ATTACK.
FOR 3° ANGLE OF ATTACK
REDUCE C_T SHOWN BY 5%
MAX C_T
SFC=1.50
SFC=1.60
SFC=1.70
SFC=1.80
SFC=1.90
C_T
1.3
1.2
1.1
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
0
2.5
3.0
3.5
M_0

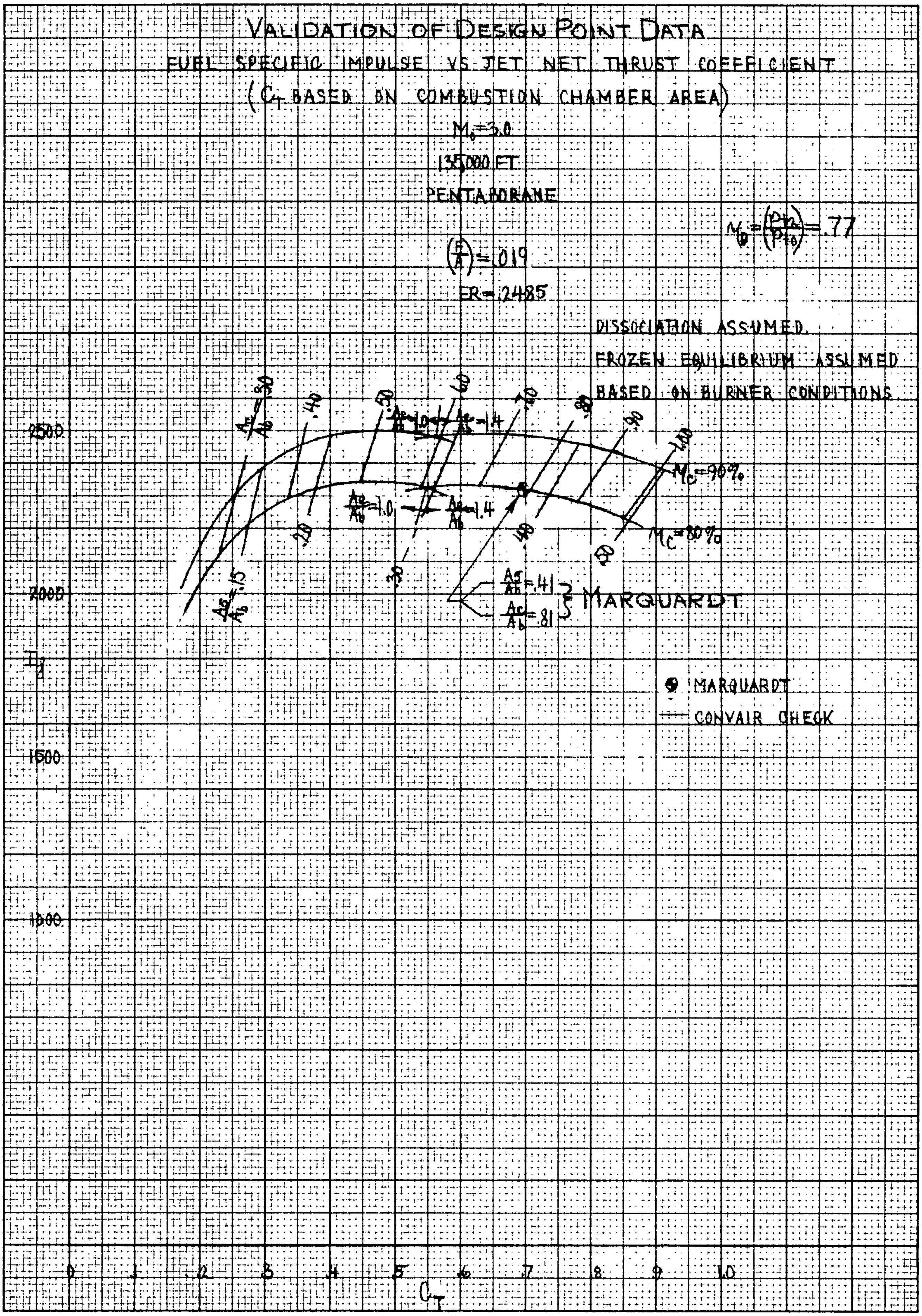

PRATT AND WHITNEY ENGINE
% DIAMETER, LENGTH AND WEIGHT VS. % THRUST
THRUST SFC CONSTANT WITH SCALE
86" DIA. INLET
% LENGTH
% DIAMETER
% WEIGHT
% DIAMETER, LENGTH, WEIGHT
% THRUST
120
110
100
90
80
70
60
50
20
30
40
50
60
70
80
90
100
110
120
130
140

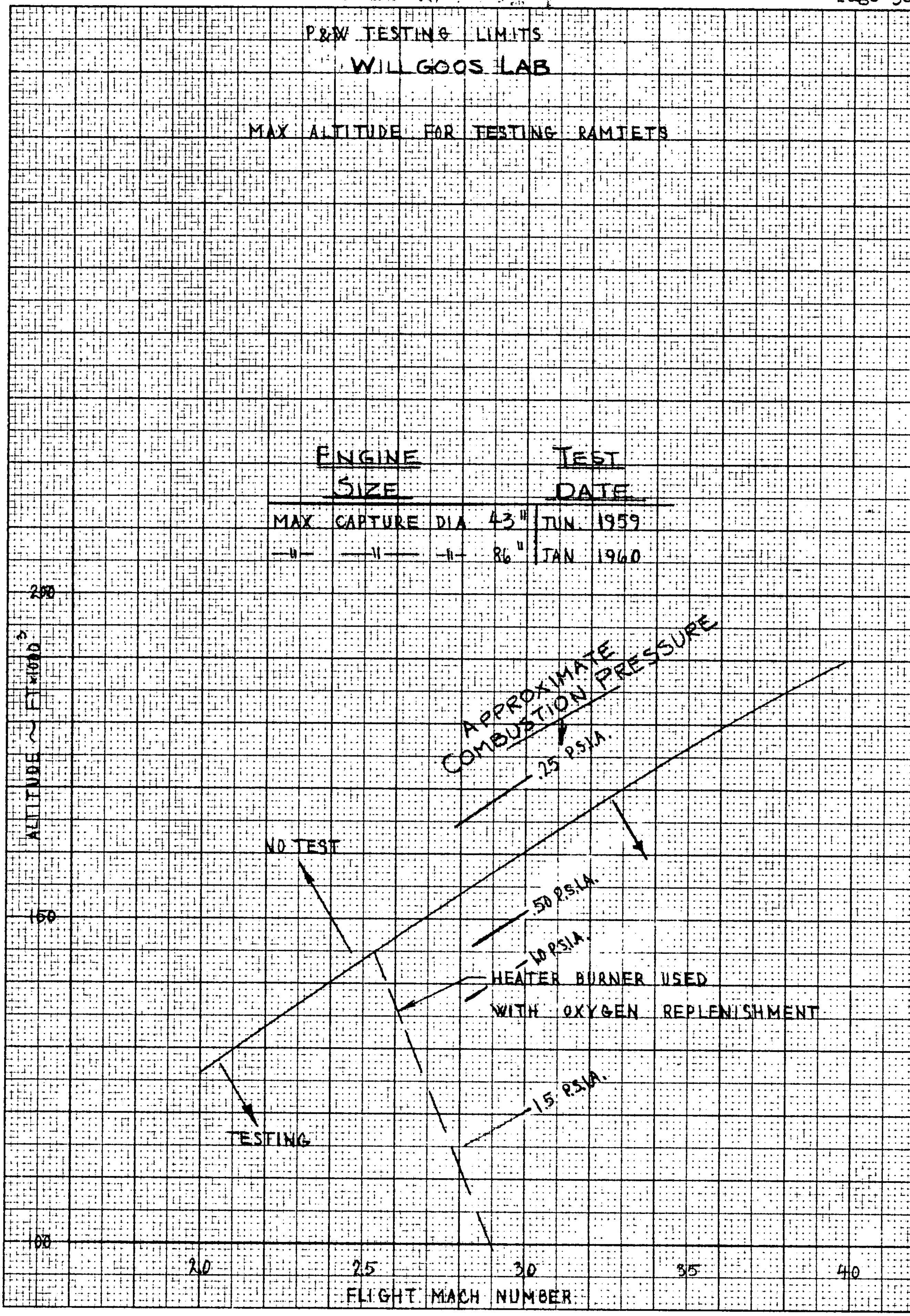
VALIDATION OF DESIGN POINT DATA



359T-11G
K+E 10 X 10 TO THE 1/2 INCH
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE

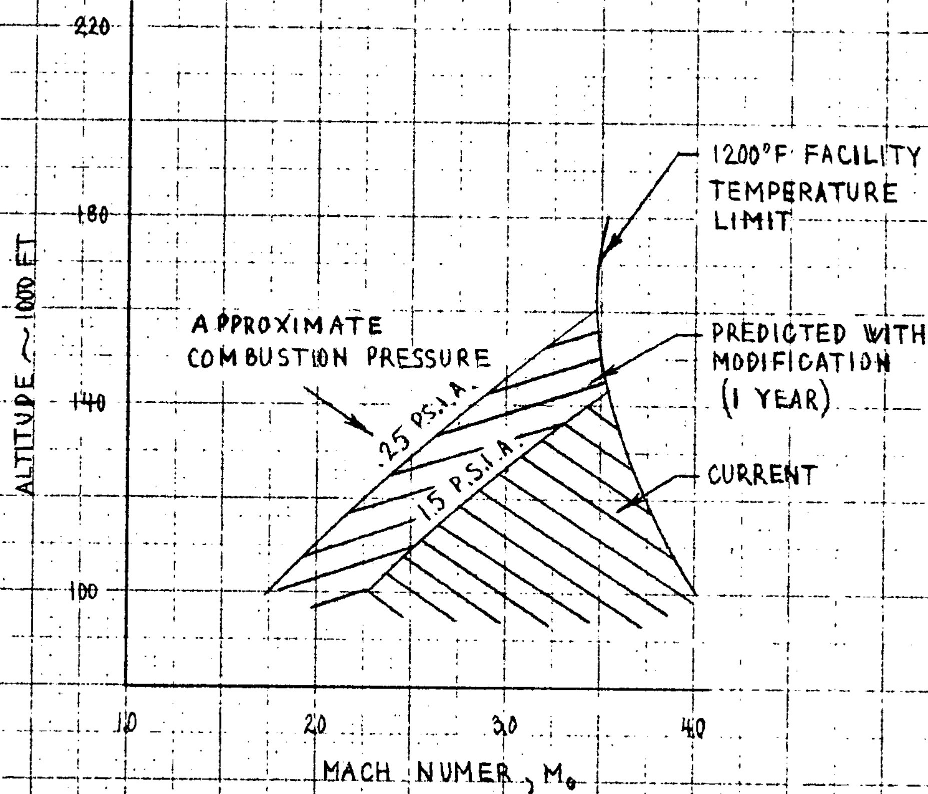

VALIDATION OF DESIGN POINT DATA
FUEL SPECIFIC IMPULSE VS JET NET THRUST COEFFICIENT
(CT BASED ON COMBUSTION CHAMBER AREA)
M0=3.0
135,000 FT
PENTABORANE
ηD=(Pt2/Pt0)=.77
(F/A)=.040
ER=.5230
DISSOCIATION ASSUMED
FROZEN EQUILIBRIUM ASSUMED
BASED ON BURNER CONDITIONS
P&W
Ad/A4=.520
Ae/A4=.815
+ P&W
— CONVAIR CHECK
If
CT
ηC=.90
η=.80
2500
2000
1500
1000
2
3
4
5
6
7
8
9
10
11
12
13
14
15


VALIDATION OF DESIGN POINT DATA
FUEL SPECIFIC IMPULSE VS JET NET THRUST COEFFICIENT
(C_T BASED ON COMBUSTION CHAMBER AREA)
$M_0 = 3.0$
135,000 FT
PENTABORANE
$\eta_D = \left(\frac{P_{t2}}{P_{t0}}\right) = .77$
$\left(\frac{F}{A}\right) = .019$
ER = .2485
DISSOCIATION ASSUMED
FROZEN EQUILIBRIUM ASSUMED
BASED ON BURNER CONDITIONS
$\eta_c = 90\%$
$\eta_c = 80\%$
$\frac{A_5}{A_b} = .41$
$\frac{A_c}{A_b} = .81$
MARQUARDT
MARQUARDT
CONVAIR CHECK
2500
2000
1500
1000
I_f
C_T


P&W TESTING LIMITS
WILLGOOS LAB
MAX ALTITUDE FOR TESTING RAMJETS
ENGINE SIZE
TEST DATE
MAX CAPTURE DIA 43" JUN 1959
—"— —"— —"— 86" JAN 1960
APPROXIMATE COMBUSTION PRESSURE
.25 PSIA.
.50 PSIA.
1.0 PSIA.
1.5 PSIA.
NO TEST
TESTING
HEATER BURNER USED WITH OXYGEN REPLENISHMENT
ALTITUDE ~ FT×1000
200
150
100
2.0
2.5
3.0
3.5
4.0
FLIGHT MACH NUMBER

MARQUARDT
ENGINE FACILITY CAPABILITY
FOR COMBUSTION TESTS

8-FOOT-DIAMETER ENGINE



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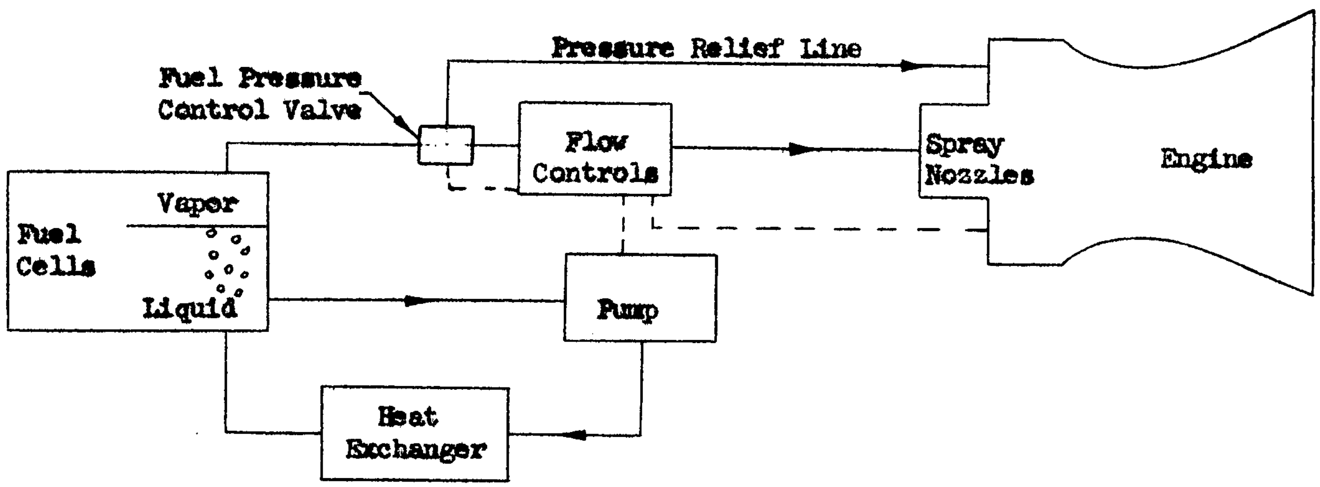
VAPOR FEED FUEL SYSTEM SCHEMATIC

SF-1 or Pentaborane



LIQUID FEED FUEL SYSTEM SCHEMATIC

Pentaborane

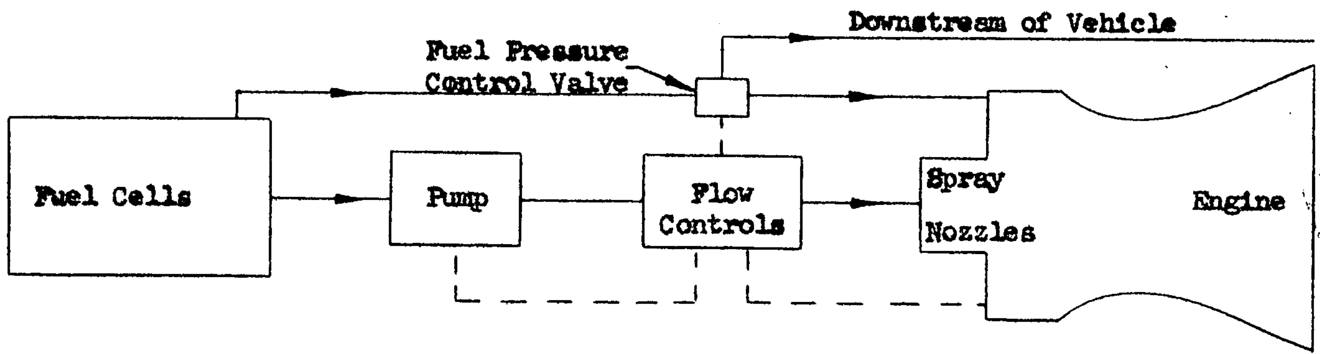


Figure 16

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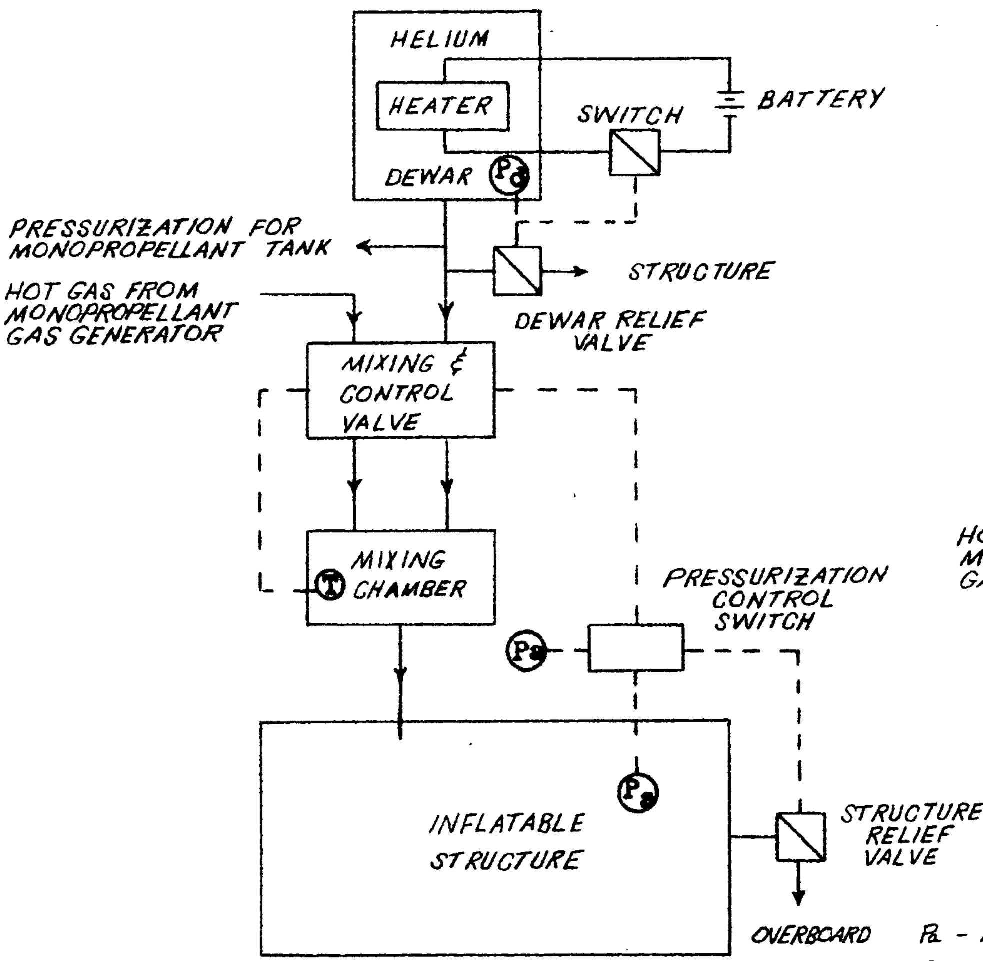

Figure 17: Proposed Pressurization System Schematic

P_a - AMBIENT PRESSURE
P_s - INTERNAL PRESSURE OF STRUCTURE
P_d - INTERNAL PRESSURE OF He DEWAR

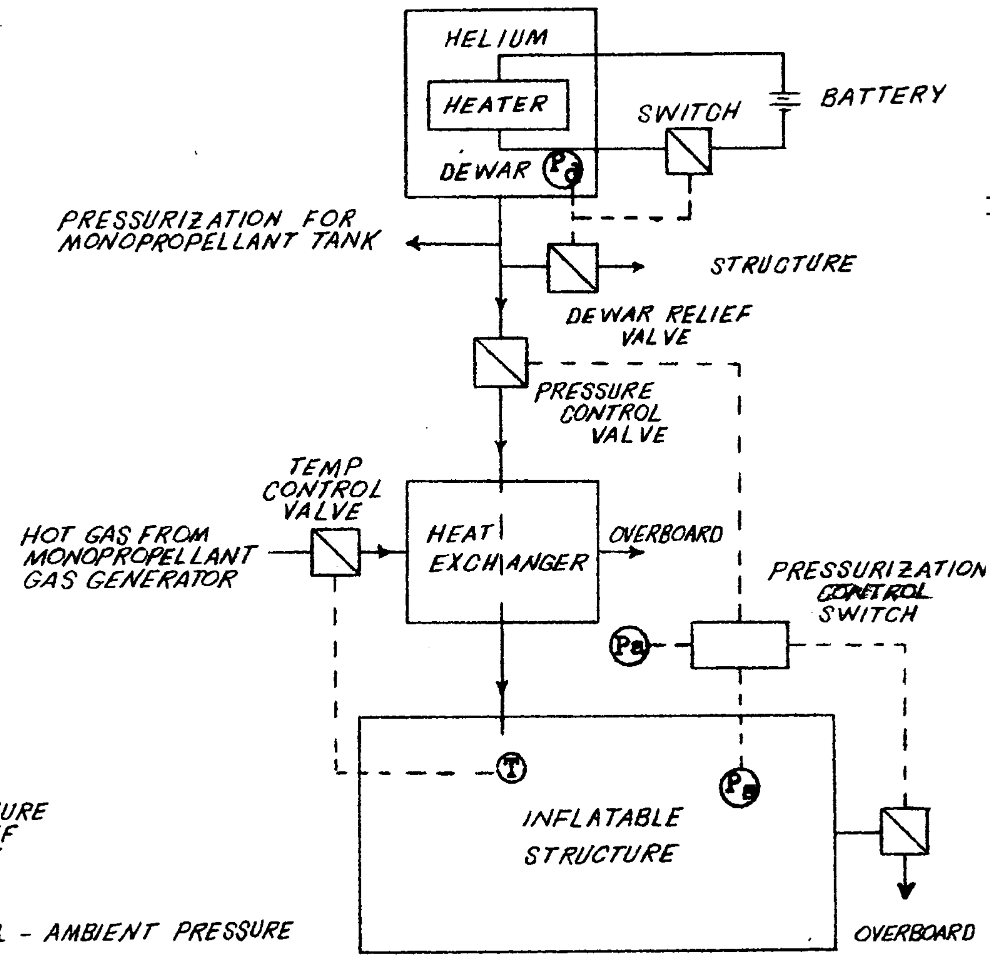

Figure 18: Alternate Pressurization System Schematic

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TABLE 2

Pratt & Whitney		SRJ-43D		SF-1 Fuel
Altitude	80,000	100,000	135,000	150,000
Mach No.	3.0	3.0	3.0	3.0
P_{t2}/P_{to}	0.784	0.784	0.784	0.759
$W_a \sqrt{\theta_{T2}}/\delta t2$	601.9	601.3	599.5	618.17
$\delta t2$	0.780	0.307	0.0709	0.0387
T_{t2} °F	623	703	854	917
T_B °F	2440	2685	2895	2856
f/A	0.0113	0.0133	0.015	0.015
F_N	19,050	7,725	1,732	876
TSFC	0.689	0.760	0.831	0.902
Drag	979	387	99	58
Fi	18,071	7,338	1,633	818
ITSFC	0.726	0.800	0.881	0.967
Weight	1075	1075	1075	1075
Length	256	256	256	256
Weight/Fi	0.048	0.119	0.535	1.070
Engine + Fuel Wt./Fi	1.108	1.239	1.695	2.309

Pratt & Whitney		SRJ-43E (Alternate Design)		SF-1 Fuel
Altitude	80,000	100,000	135,000	150,000
Mach No.	3.0	3.0	3.0	3.0
P_{t2}/P_{to}	0.784	0.784	0.784	0.759
$W_a \sqrt{\theta_{T2}}/\delta t2$	601.9	601.3	599.5	618.17
δ_{t2}	0.780	0.307	0.0709	0.0387
T_{t2} °F	623	703	854	917
T_B °F	2840	3075	3285	3200
f/A	0.0143	0.0168	0.020	0.020
F_N	22,800	9,145	2,059	1,025
TSFC	0.731	0.811	0.933	1.029
Drag	979	387	99	58
Fi	21,821	8,778	1,960	967
ITSFC	0.763	0.846	0.98	1.09
Weight	1057	1057	1057	1057
Length	252	252	252	252
Weight/Fi	0.039	0.098	0.439	0.890
Engine + Fuel Wt./Fi	1.149	1.228	1.729	2.285

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FORM 1812-A-1

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TABLE 3

Pratt & Whitney		SRJ-43D	SF-1 Fuel
Altitude	80,000	100,000	135,000
Mach No.	2.5	2.5	2.5
P_{T2}/P_{to}	0.89	0.89	0.89
$W_a \sqrt{\theta_{T_2}}/\delta_{t2}$	635.8	635.6	635.0
δ_{T2}	0.41	0.162	0.0374
T_{T2} °F	415	481	608
T_B °F	1600	1740	2160
f/A	0.0071	0.00765	0.0104
FN	7980	3110	790
TSFC	0.644	0.679	0.760
Drag	3440	1371	322
Fi	4540	1739	468
ITSFC	1.133	1.212	1.282
Weight	1075	1075	1075
Length	256	256	256
Weight/Fi	0.193	0.503	1.87
Engine + Fuel Wt./Fi	2.173	2.541	3.895

Pratt & Whitney		SRJ-43E (Alternate Design)	SF-1 Fuel
Altitude	80,000	100,000	135,000
Mach No.	2.5	2.5	2.5
P_{T2}/P_{to}	0.89	0.89	0.89
$W_a \sqrt{\theta_{T_2}}/\delta_{t2}$	635.8	635.6	635.0
δ_{T2}	0.41	0.162	0.0374
T_{T2} °F	415	481	608
T_B °F	1950	2094	2530
f/A	0.0093	0.010	0.0138
FN	10,100	3,965	950
TSFC	0.666	0.695	0.835
Drag	3440	1371	322
Fi	6660	2594	628
ITSFC	1.00	1.062	1.262
Weight	1057	1057	1057
Length	252	252	252
Weight/Fi	0.129	0.332	1.370
Engine + Fuel Wt./Fi	1.877	2.117	3.360

For 2° angle of attack reduce thrust by 2% and increase SFC by 2%

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TABLE 4

Pratt & Whitney   SRJ-43D   Pentaborane Fuel

Mach No.	2.5			3.0			
Altitude	80	100	135	80	100	135	150
P_{T2}/P_{to}	0.89	0.89	0.89	.784	.784	.784	.784
δ_{T2}	.412	.163	.0375	.780	.308	.0709	.040
T_{T2} °F	415	479	606	623	703	854	917
W_a /δ_{t2}	635.8	635.6	635	601.9	601.3	599.5	613.9
W_a 16/sec	201	76.5	16.6	324	123	26.7	14.7
f/A	.0158	.0176	.0217	.0245	.0278	.04	.04
	.991	.985	.934	.989	.982	.945	.918
T_B °F	2020	2200	2400	2720	2870	3280	3173
F_N	9440	3821	877	20179	8467	1933	1003
TSFC	1.213	1.272	1.482	1.354	1.453	1.992	2.112
Drag	3440	1371	322	979	387	99	58
IFN	6000	2450	555	20100	8080	1834	945
ITSFC	1.908	1.984	2.342	1.420	1.523	2.100	2.241
SP WT	.1733	.424	1.892	.052	.129	.567	1.101
E + FW/Fi	3.499	3.694	5.582	2.114	.2221	3.324	3.970

DC in	86
A_j Ft^2	25.73
DT In	68.7
L/DT	0.87
LN In	59.8
D_E In	104
L In	266
Wt.Lbs.	1150

For 2° angle of attack reduce thrust by 2% and increase SFC by 2%

SECRET

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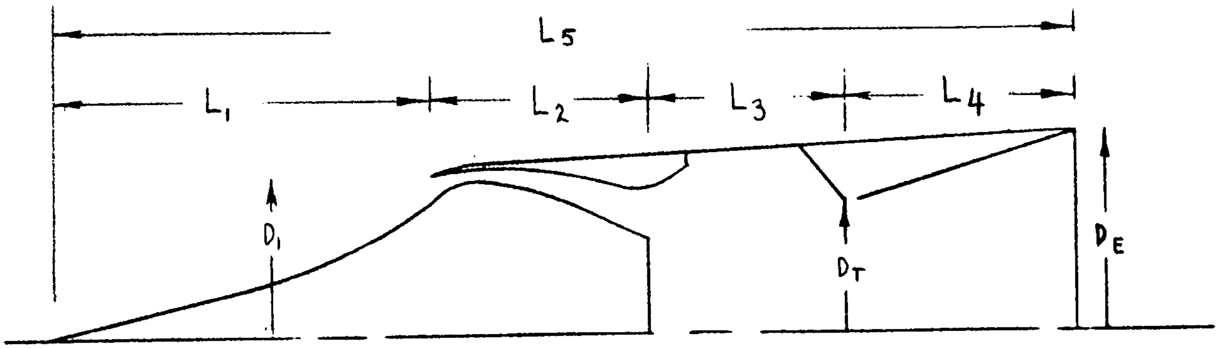
TABLE 5



Pratt and Whitney

Engine Geometry

	SRJ-43D		SRJ-43E Alternate Design
Fuel	Pentaborane	SF-1	SF-1
D_I In.	86	86	86
D_T In.	68.7	67.3	70.1
D_E In.	104	104	104
L_1 In.	98.5	98.5	98.5
L_2 In.	60	60	60
L_3 In.	48	36	36
L_4 In.	60	62	55
L_5 In.	266.5	256.5	252.5

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REFERENCES

1. Sibulkin, M. H., "Heat Transfer Near the Stagnation Point of a Body of Revolution," Journal of Aeronautical Sciences, Vol. 10, P. 570.

2. Echert, Ernest R. G., "Survey on Heat Transfer at High Speeds," WADC Technical Report 54-70.

3. Private communication with Glen E. McIntosh, Group Engineer, Boulder Division, Beechcraft Research and Development, Inc.

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