LARGE LIGHTWEIGHT TURBOJET ENGINES

by

C.A. GRINYER

VICE-PRESIDENT ENGINEERING

and

CHIEF ENGINEER

ORENDA ENGINES LIMITED

MALTON ONTARIO CANADA

(MEMBER: A.V. ROE CANADA LTD and HAWKER SIDDELEY GROUP)

"This paper deals with the general performance of turbojet engines in both subsonic and supersonic flight in relation to aircraft requirements. The advantages and disadvantages of the bypass engine and small turbojet in relation to the large straight jet engine of 18000 pounds thrust are examined. Also discussed is the improvement in thrust/weight ratio of large engines made possible by the use of titanium, and the potentialities of ejector nozzles and turbine blade cooling."

ARMSTRONG (5)
72.116-57/04

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INTRODUCTION

The importance of low engine weight in its effect on range of a given aircraft or on the weight of an aircraft to accomplish a certain mission has been adequately emphasized by previous writers. It is particularly important in the case of short range aircraft.

In this paper we will first review the performance characteristics and requirements of turbojets with particular regard to operation at high Mach numbers. The problems associated with the design of turbojets required to operate over a wide speed range and possible implications with respect to aircraft range are discussed. Brief commentary is made on the current controversial topics "Bypass vs. Turbojets" and "Large vs. Small Engines".

In the second portion of the paper we will examine the effect of titanium on the mechanical design of large engines and work directed towards engine operation at higher turbine temperatures.

GENERAL PERFORMANCE CHARACTERISTICS OF TURBOJETS

While the general performance characteristics of turbojets have been previously discussed several times and are fairly well known, a review is not considered out of place at this time in the light of interest in higher speeds.

Figures 1 to 6 show the effect of compression ratio and turbine inlet temperature of specific thrust and specific fuel consumption at sea level static and Mach number 0.9 and 2.5 in the stratosphere. Additional effect of afterburning is shown at Mach 2.5.

Briefly, at sea level static and Mach 0.9, to attain high specific thrust, we require high turbine inlet temperatures irrespective of compression ratio, while to attain low specific fuel consumption requires high compression ratio and fairly moderate turbine inlet temperatures of the order of 1,000 degrees K (1340 degrees F). We will see later that the performance requirements of the aircraft considered are such that only moderate specific thrusts are required at these flight conditions which is compatible with low specific consumption.

At Mach 2.5 without afterburning, high turbine inlet temperatures are, of course, still required to attain high specific thrusts; however, much lower compression ratios are required to attain minimum specific fuel consumption. While the minimum specific fuel consumption which can be attained is of the order of 1.35, the maximum specific thrust, even at the turbine inlet temperature of 1,300 degrees K (1880 degrees F) is only about 35 pounds per pound per second.

At Mach 2.5 with afterburning, both maximum specific thrust and minimum specific fuel consumption are attained by the use of high turbine inlet temperatures and low compression ratios of the order of six. Most noticeable, of course, is the tremendous increase in specific thrust possible when using afterburning at this speed. While specific thrusts of the order of 110 pounds per pound per second should be attainable with stoichiometric afterburning and theoretical full expansion - since they would require final nozzle throat and exit diameters considerably greater than that of the main engine - it is probable that in practice, specific thrust at this speed will be limited to about 70 pounds per pound per second with consequent increase in specific fuel consumption[1]. Nevertheless, the specific thrust attainable with afterburning at this speed is about twice that available without afterburning, so that after-burning engines would need have only half the air flow of non-afterburning engines and should be considerably lighter. This, of course, is only achieved at the expense of a 50 percent increase in fuel consumption for a given thrust.

The relationship between engine weight and aircraft weight will be considered later.

THE EFFECT OF FORWARD SPEED ON ENGINE OPERATION

Using the simple assumptions given in Appendix I we can predict approximately the effect of forward speed at constant mechanical rpm on compression ratio and non-dimensional flow. The results are plotted in Figure 7 for an engine of 12:1 pressure ratio under standard NACA atmosphere conditions.

Two curves are shown, one assuming constant turbine throat area and turbine inlet temperature and the other assuming constant compressor temperature rise.

The assumption of constant compressor temperature rise at constant rpm necessitates a fall off in turbine inlet temperature with constant throat area or an increase in turbine throat area with constant turbine temperature at temperatures greater than 15 degrees C. We therefore use the first assumption of constant turbine throat area and turbine inlet temperature as the basis of approximate prediction of the change in pressure ratio and flow of a turbojet. Using general performance curves such as Figures 1 to 6, we can therefore obtain the variation in thrust and specific fuel consumption with forward speed as shown in Figure 8.

It is emphasised that the above method only gives the approximate performance of an engine. Accurate performance prediction can only be obtained by the use of appropriate compressor and turbine characteristics.

With respect to the problem of surging at high forward speeds, no conclusions can be drawn from Figure 7.

In general the operating line at constant rpm and turbine inlet temperature will lie slightly below that associated with part rpm operation at constant inlet temperature.

In any high compression ratio engine with fixed turbine geometry some variable geometry such as some variable angle stators or two spooling will be required in the compressor for normal test bed accelerations.

THE EFFECT OF FORWARD SPEED ON TURBOJET PERFORMANCE

Figure 8 shows the effect of forward speed on turbojet performance. Here we have considered an engine of compression ratio of 12 at sea level static and assumed constant rpm and constant turbine inlet temperature of 1,200 degrees K (1700 degrees F) operation. It will be seen that the compression ratio decreases with forward speed, in a manner compatible with the requirements of maximum specific thrust. With no afterburning, the specific thrust becomes zero at about Mach 3. With afterburning 1,750 degrees K (2700 degrees F), the afterburner contributes a large proportion of the thrust with increasing speed. At Mach 2.5 a specific thrust of about 65 is available with full expansion and in subsonic flight the engine is capable of the same specific thrust of 65 without afterburning.

GENERAL PERFORMANCE REQUIREMENTS OF TURBOJET ENGINES

It is, unfortunately for the engine designer, impossible to talk about the performance requirements of turbojet engines without some reference to aircraft characteristics. Figure 9 shows typical maximum lift/drag ratios likely to be attained(2). For the purpose of the following discussion it has also been assumed that the maximum lift/drag ratio of a particular aircraft over its speed range follows this curve. It is known that in practice this is not quite the case, particularly with aircraft designed for very high supersonic speed, in that a compromise is necessary in high speed performance if reasonable low speed performance is required, and vice versa. The extent of this compromise obviously depends on the mission, and values must be substituted for any particular case. Also shown in Figure 9 are the assumed values of lift co-efficient occurring at these values of lift/drag.

Now the altitude corresponding to these values of maximum lift/drag is a function of wing loading, lift co-efficient and Mach number (Appendix 2). In Figure 10 the altitude corresponding to maximum lift/drag is shown for wing loading of 50 pounds per square foot. A value of 50 pounds per square foot seems to be reasonable, at least for some current aircraft(3), and gives values of cruise altitude which appear to be of the right order. Using the values of engine intake efficiency shown in Figure 10 it is, therefore, possible to determine the variation of engine intake pressure with Mach number at the best cruise altitude (Appendix 2). This, together with the variation of intake temperature are also shown in Figure 10. Since a supersonic fighter will have to operate at times at altitudes lower than these best altitudes, the intake pressures may be considered as minimum values.

If we assume that the ratio of sea level static air flow to engine weight is about 0.060 pounds per second per pound weight for an afterburning engine, it is possible to determine (Appendix 2) the ratio of engine weight to aircraft weight required for any flight speed using the aircraft characteristics of Figure 9. We find in general that with non-afterburning engines the weight of engine to aircraft weight reaches a very high value of about 0.5 for level supersonic flight and it therefore appears that afterburning engines are necessary for high supersonic speeds, notwithstanding the higher fuel consumption. Employing afterburning engines, the engine weight required is almost constant at about 16 percent of aircraft weight for all speeds. With respect to the specific thrust required for the subsonic cruise of a supersonic aircraft, we find that quite a low value of the order of only 40 to 50 is required, which would require a high temperature engine to be throttled back even without afterburning.

It is apparent from the foregoing that the design of an engine optimized for both subsonic and supersonic operation poses quite a problem. At high speeds we require afterburning and high turbine inlet temperatures, while at subsonic speeds we require high compression ratio and moderate turbine inlet temperatures. Apart from the previously mentioned difficulty of designing aircraft to fly efficiently over a wide speed range, it is apparent that there is some difficulty in designing engines to do likewise.

In this respect, we might consider the question of range. For a fixed ratio of fuel load to aircraft weight, the range of an aircraft is directly proportional to the lift/drag ratio and Mach number and inversely proportional to the specific fuel consumption (Appendix 2). Comparing values at Mach 0.9 and 2.5, taking data from Figure 9 we find that the range factor at Mach 0.9 is 10.8 while at Mach 2.5 it is 6.25. While this indicates that maximum range will be obtained by keeping some portion of the mission subsonic, a study of a particular aircraft and engines may indicate that continuous supersonic flight will give maximum range. This would happen if, in optimizing for supersonic flight, the aircraft lift/drag and engine specific fuel consumption were both 30 percent worse at Mach 0.9 than the best possible at this speed.

EJECTOR NOZZLES

We have already seen the growing importance of the final nozzle of the engine, both on its performance at high speeds and also on the weight of the engine. For an engine to be capable of satisfactory performance both at subsonic and supersonic speed, and using afterburning when desirable, would ideally require a Laval type nozzle with a fully variable throat in order to accommodate afterburning and a fully variable exit diameter to give correct expansion at every engine rpm and forward speed. Such a nozzle is an extremely heavy and mechanically complex device. Moreover, in actual aircraft practice we have to deal not only with the main engine flow but also the cooling air flow which passes around the engine plus whatever bypass flow is necessary for intake matching. It is therefore, apparent that the final nozzle of an engine must have ejector characteristics in order to adequately scavenge the engine compartment and cool the afterburner during take-off when the ram pressure ratio is low.

Fortunately, the flow phenomenon in the simple ejector type nozzle used on subsonic aircraft is such as to allow a certain amount of divergent expansion of the main jet and this idea can be developed in order to obtain divergent expansion at high speeds with considerable weight saving.

Figures 11(a) and (b) show ejector type nozzles for use at Mach numbers up to about 1.5. In (a) without afterburner, it will be seen that as the secondary flow is accelerated through the final nozzle, the annulus which it occupies diminishes in area thus allowing the main jet to expand. In illustration (b) with the afterburner in operation, since the throat area of the main jet (D_p) is now greater, a correspondingly greater diameter (D_j) must be allowed for its expansion. This is achieved by throttling the secondary flow.

This basic principle can be extended to engines required to fly up to Mach numbers of 2.5 as shown in illustrations 11(c) and (d). At low speeds, it may be necessary to bring in some of the secondary flow from outside of the aircraft in order to fill the final nozzle when the main jet pressure ratios are low and divergent expansion is not desirable. This would be done by means of bleed doors. At high speeds the bleed doors are automatically closed allowing the main jet to expand to the full diameter of the exit nozzle. An alternative method of obtaining the necessary amount of divergent expansion at high speeds would be to equip the shroud as well as the primary jet nozzle with an iris type nozzle as shown in 11(e) and (f).

THE BYPASS ENGINE

No paper dealing with large lightweight turbojet engines would be complete at this time without some comments on the advantages and disadvantages of the bypass engine. In the bypass engine, the diameters of the first stages of the compressor are greater than that of the latter stages. Some of the compressor air flow is bled off, bypassing the combustion chamber and turbines, and can either be exhausted separately or re-mixed with the turbine exhaust stream, the whole exhaust flow then being ejected through a single final nozzle.

In Figure 12 a comparison has been made between the design point performance of a straight jet engine and bypass engines of the same overall compression ratio. The flight conditions assumed were Mach 0.85 in the stratosphere. The overall compression ratio of 12 relates to this flight condition. To permit mixing of the bypass air and turbine exhaust gas with minimum pressure loss, it was arranged for the bypass delivery pressure and turbine exhaust pressure to be equal. For all turbine inlet temperatures investigated, the bypass flow was assumed to be 33 percent of the total flow. Thus, we are considering a different bypass engine at each turbine inlet temperature in that the compression ratio of the bypass air increases with turbine inlet temperature. A bypass air flow of 33 percent of the total air flow was assumed since this value seems likely to permit satisfactory engine operation at reduced rpm by using a two spool arrangement and without the additional complication of variable compressor stators which may be necessary, by analogy with a turboprop, at higher bypass flows.

On the basis of these detailed assumptions, the following conclusions can be drawn from the results shown in Figure 12. At a given specific thrust, the bypass engine requires a higher turbine inlet temperature than the straight jet engine. As the bypass flow is increased, the bypass engine has to operate at higher and higher turbine inlet temperatures in order to achieve a reasonable specific thrust. The bypass engine for the example shown has a minimum specific fuel consumption at a turbine inlet temperature of about 1,000 degrees K (1340 degrees F). It gives a specific thrust of about 30 pounds thrust per pound of air flow per second. At this same specific thrust, the straight jet engine has a specific fuel consumption of about four percent higher but operates with a turbine inlet temperature of only 900 degrees K (1160 degrees F). At a specific thrust of 47 pound per pound per second the situation is reversed, the straight jet engine having a specific fuel consumption of about three percent better and this does not allow for blade cooling losses which would be required at the turbine inlet temperature of 1400 degrees K (2060 degrees F) required by the bypass engine at this specific thrust. Thus, for a given specific thrust there is probably very little to choose between the straight jet and bypass engine, as far as fuel consumption is concerned. Consideration of the reduced rpm performance of a given engine may modify this conclusion somewhat, but probably not to any great extent, provided the bypass engine uses separate exhausts.

If we consider a straight jet engine and bypass engine of the same specific thrust, it would also seem reasonable that they would have the same specific thrust at take-off at the same rpm. Since specific thrust is a direct function of jet velocity, we would, therefore, expect both engines to emit the same amount of noise at take-off. The jet engine has an advantage in that, because of its lower inlet turbine temperature, it can be over-speeded at take-off (if suitably stressed), and the extra thrust (30 to 40 percent) can be used to get a greater load airborne within a given distance or alternatively would prove useful under hot day conditions. Only when over-speeded, however, will it be noisier than the bypass engine.

While much has been said about bypass engines in terms of better propulsive efficiency it appears that the only real advantage of the bypass engine may lie in higher thrust/weight ratio. In Figure 13 a comparison is made between a straight jet engine and a bypass engine of the same thrust. Both are two spool arrangements and both have the same intake diameter. However, in the bypass engine the high pressure portion of the compressor can be made smaller in diameter. With respect to the turbines, the low pressure turbine power per pound of turbine flow is greater in the bypass engine, necessitating a turbine diameter somewhat larger than in the straight jet engine if the same number of turbine stages are to be employed. Another important point is that the high pressure compressor, combustion chamber and high pressure turbine are shorter, leading to reduced shaft lengths. This reduction in length may be somewhat offset by the necessity of reasonable distance between the low pressure compressor outlet and high pressure compressor inlet to accommodate changes in bypass ratio at reduced rpm operation.

Nevertheless, an overall reduction in engine length would appear to be possible in the bypass engine.

In conclusion, it appears that the application of the bypass principle offers the possibility of a certain reduction in engine weight for a given thrust at the expense of higher turbine operating temperatures. Against this, the straight jet engine, while perhaps slightly heavier, affords the advantage of a reserve in thrust and longer turbine operating life.

With regard to application, since the advantages of the bypass principle diminish with forward speed, the bypass engine shows to best advantage at low speed. Because of low specific thrust, the bypass engine and low temperature straight jet require pod type installation since high airflows are an embarrassment in buried installations.

MECHANICAL DESIGN

Figure 14 shows the trend in thrust/weight ratio of large and small turbojets as predicted by Gregory[4]. Our work indicates the possibility of an 18,000 pound thrust engine with a thrust/weight ratio which lies about mid way between the trend for small and large engines. How is this large thrust/weight ratio achieved?

With the introduction of new materials, such as titanium, the weight of rotating components of turbojet engines can be considerably reduced as shown in Figure 15. It will be seen that the weight of a compressor stage in titanium is only about 45 percent of the same stage designed in steel while the density of titanium is about 60 percent that of steel. This considerable reduction in the weight of dynamic parts in turn allows us to use a very much simplified rotor bearing system as shown in Figure 16 with even further weight reduction. Figure 16 shows a four bearing arrangement for a large two spool engine. In order to keep the low pressure shaft as light as possible a steady bearing has been added where the low pressure shaft passes through the high pressure thrust bearing. Nevertheless, there are only two oil sumps, one at the front of the engine where the thrust bearings are located and one within the tailcone bullet. The usual centre bearing associated with turbojet construction together with its oil sump and supporting structure has been completely eliminated. In a two spool engine this leads to futher simplification since the couplings and other components are unnecessary and leads to a comparatively low cost engine.

With regard to the vibration characteristics of such a rotor system, Figure 17 shows the type of vibration pattern likely to be obtained with this type of rotor system. The usual design practice is to assume rigid bearing supports and design both shafts to have critical speeds above maximum speed. Because of the flexibility of the bearing supports we find that both critical speeds will lie within the engine speed range. However, the damping capacity of the engine frame is such that the amplitude of shaft vibration at this critical speed does not attain any appreciable magnitude, and moreover, the amplitude of vibration of the engine frame itself is well within the allowable tolerances.

Figure 17 was obtained with the shaft weight distribution and joint efficiencies as designed. In the case of compressor blade failures, weight distribution can vary considerably providing out of balance forces that may increase the amplitude at the critical speed leading to permanent distortion at the shaft joints, rotor instability and failure.

To check that this condition did not arise, an engine has been built with rotor weight distribution as designed but without blades in the compressors or turbines. Each rotor was driven from external sources at rpm well above maximum design speed. When normal vibration characteristics had been obtained, out of balance was introduced equivalent to the worst case likely to arise. Dimensional checks subsequently disclosed no movement of the rotor joints and that complete stability had been achieved. Figure 18 shows the engine and external drives used for this investigation.

In order to reduce the weight of rotating components so that the engine structure can be correspondingly lightened, stress studies require particular emphasis when dealing with a new material such as titanium. The low modulus of elasticity for instance presents difficulties in load distribution when designs approach ultimate values. Figure 19 shows some photoelastic studies carried out on compressor disc design.

TURBINE DESIGN

Figure 20 shows the development of turbine blade fixings. The well known "fir tree" type of root (a) was one of the earliest types of turbine blade fixing. This type of design was necessitated by the large number of blades required in the single stage turbines of early turbojets, which were highly loaded.

With engine development to higher compression ratios it was eventually necessary to use a two stage turbine. This immediately gave reduced blade loadings and permitted consideration of the simplest form of fixing, namely the single branch fixing shown in (b). This fixing is the simplest possible to manufacture and inspect and permits a rocking motion thereby giving some damping. Nevertheless the width of the stub between blades is undesirably narrow and in the event of shear of a branch the complete blade could leave the disc. This in turn would permit all blades to be shed.

A compromise between ease of manufacture and safety is therefore attained by the use of a two branch fixing (c).

With regard to the permissible limit of output of a turbine stage it has been our experience that a two stage turbine of the same total output as a single stage turbine at its maximum output is only slightly heavier but has greater efficiency leading to lower specific fuel consumption.

TURBINE DISC MATERIALS

The development of new turbine disc materials is also contributing to the reduction of weight in turbines. In Figure 21 is given the properties of early disc materials such as H-46 as compared with the more recent A-286 and Greek Ascoloy and the most recent Inconel 901. In changing from H-46 to A-286 it is possible to achieve a reduction in disc weight of about 50 percent. Further decrease should be possible when Inconel 901 is used. Since the turbine of a turbojet accounts for something like 10 percent of its total weight, it will be appreciated that any reduction which can be effected in weight of this rotating component will be extremely beneficial towards reducing the total engine weight.

TURBINE BLADE COOLING

With the continuous improvement of turbine blade materials up to the present time, there has been very little interest in attainment of even higher temperatures by the use of turbine blade cooling and its advantages and disadvantages are still subject to controversy. As we saw in Figures

3 and 4 for non-afterburning engines, while higher turbine inlet temperatures do not give very much benefit as regards specific fuel consumption they can considerably increase the specific thrust and thus avoid the use of afterburning except at very high speeds and during manoeuvre. In Figure 22 the effect of air cooling of turbine blades on the performance of an afterburning engine is shown. In this example a three stage turbine is employed and, in view of the high compressor delivery temperature at Mach 2.5, very high cooling air flows would have to be employed. Assuming that it were possible to pass these air flows through the blades, it will be seen that the quantities which have to be bled from the compressor, and therefore perform very little useful work in the turbine, are such that the improvement in performance is only slight. In actual practice with a three stage turbine, probably about 10 percent is the maximum air flow which could be used, which would give an upper limit of turbine inlet temperature of about 1,350 degrees K (1970 degrees F). In addition to this aerodynamic limitation of air cooling blades, the difficulty of fabrication is quite considerable. Figure 23 shows some early types of fabricated turbine blades which were run in engines for the purpose of evaluating different methods of construction and fabrication. While all three types of blade completed a 150 hour qualification test with a turbine inlet temperature of 1,200 degrees K (1700 degrees F), cracks occurred in the cover plates or in the joint in all cases. The results of fatigue tests in the laboratory at room temperature indicated that blades (a) and (b) have an endurance limit of about 40 percent of that of a solid blade in the same material. Blade type (c) made of two

machined blade forgings joined on the convex surface near the neutral bending axis was slightly better. Improvement is expected with development.

When the cooling air flow exceeds about two percent per stage it appears that it will be necessary to increase the pressure of the cooling air over that of the compressor cooling air in order to obtain the necessary pressure drop to drive the air through the blades.

To summarize the position as we see it, it appears possible to attain turbine inlet temperatures of about 1,350 degrees K (1970 degrees F) at forward speeds up to about Mach 1.5. This requires cooling air flow of about two percent per stage.

It is apparent from the foregoing that if we wish to achieve higher turbine inlet temperatures or even moderate increases in turbine inlet temperature at speeds of the order of Mach 2.5 we will have to use methods other than air cooling. Considerable development has already taken place in Canada on alternative methods of cooling. Ideally, of course, it would be desirable to obtain these higher temperatures without cooling. In Figure 24 the properties of molybdenum are compared with Inconel 700, a turbine blade material now in fairly common use. The tremendous improvement in the creep properties of molybdenum are apparent; however, use of this material is dependent upon the development of satisfactory methods of "cladding" the molybdenum blade to prevent high temperature corrosion to which the bare material is very susceptible. It should not be forgotten, of course, that the use of higher inlet turbine temperatures also imposes combustion development problems.

COMPARISON OF 2,250 POUND AND 18,000 POUND THRUST ENGINES IN LARGE AIRCRAFT

As we have seen in Figure 14 Gregory predicts that small jet engines of simple construction of about 2,000 pounds thrust will be available in 1960 with a thrust/weight ratio of 7.5. This agrees with our own estimates. We have also shown that large subsonic engines of 18,000 pounds thrust with a thrust/weight ratio of 6.0 should be available at that time. This seemingly large weight advantage of the small engine, is, however, considerably diminished, as far as use in large aircraft is concerned, when we come to consider the installation and control problems associated with small engines. In Figure 25 we have considered the installation of both types of engine to an aircraft requiring 72,000 pounds take-off thrust. This aircraft would require four 18,000 pound engines, or alternatively, thirty-two small engines. Since an aircraft of this size would probably have a wing chord of the order of 25 feet, it will be seen that the buried installation (Figure 25b) necessitates very long intakes and jet pipes. Thus, with sixteen engines in each wing, the wing structure would be penalized and space which might conceivably be used for fuel tanks would be considerably reduced. No value of relative frontal area has been given for this installation but it is apparent if the frontal area required to swallow the air approaches that of the wing then the engines cannot be buried. At the cruise altitude of such an aircraft, the engine performance and intake efficiency will certainly deteriorate due to Reynolds number effect.

In Figure 25c we have considered a podded arrangement of small engines. The largest number of pods which one can use in order to avoid interference drag effects is probably eight. Thus, each pod will contain four engines. Nevertheless, making allowance for external piping and control systems the frontal area of such a pod will be approximately 1.8 times that of the 18,000 pound engine pod. In order to attain a satisfactory aerodynamic shape for the pod and to obtain good flow into the engines and avoid base drag, the pod would probably have to be about the same length as for a single engine pod.

If interference effects demand the use of only four pods with eight engines in each, the relative frontal area increases to 2.2. Again the overall length of the pod will be equal to or greater than that of an 18,000 pound engine pod.

With regard to the problem of control and supply of fuel to a large number of engines, it is presumed that a single control system would be used to control at least all the engines in one pod. Each engine would have to contribute an equal share of bleed air for driving the single turbine pump to avoid over temperature on any one engine. Satisfactory fuel distribution although necessitating extensive piping could probably be achieved in podded engines but it would be very difficult with buried engines, particularly at high altitude when fuel flows and burner pressures are very low.

In considering the application of small engines to interceptor aircraft which today, ignoring the lightweight fighter, appear to require take-off thrusts of 18,000 pounds or greater, clusters of small engines appear undesirable because of frontal area considerations.

In comparing large and small engines, another consideration which cannot be ignored is the matter of cost. While it is unlikely that the small engine can incorporate the necessary degree of aerodynamic refinement to achieve the performance possible with large engines, if we assume that it could, then it would contain approximately the same number of parts. Because of the finer tolerances required on the parts for the small engine, the manufacturing cost, even though the amount of machining is less, would probably be equal to that of the large engine. The cost of the large engine would therefore, only be greater by the amount of the increased material cost. It is therefore, difficult to avoid the conclusion that eight 2,250 pound thrust engines would cost several times that of one 18,000 pound engine.

We therefore conclude that the small jet engine is not competitive with the large engine as the main power plant for large aircraft since it will be deficient in performance by reason of its necessarily simpler construction and Reynolds number effects. While its installed weight will be about the same, its frontal area will be greater and its cost per pound of thrust several times greater.

CONCLUSIONS

The use of titanium in turbojet engines has permitted an improvement in thrust/weight ratio of a greater mangitude than to be expected by straight material substitution. This is because the use of titanium in rotating parts permits the use of a two bearing rotor system with elimination of centre bearings. Furthermore, with smaller dynamic forces the engine structure can be correspondingly lighter.

Small turbojet engines, despite their slightly higher bare thrust/weight ratio are not competitive with large engines for large aircraft, for one or more of the following reasons:- higher installed weight, inferior performance, installation and control complication, higher cost per pound of thrust.

A bypass engine and turbojet of the same specific thrust have about the same specific fuel consumption. In addition to having a lower turbine operating temperature, the turbojet at the same specific thrust is no noisier. Moreover, the turbojet can be overspeeded giving a thrust reserve of about 30 percent.

In order to operate over a wide speed range at maximum rpm with maximum combustion temperature indicates a small compromise in subsonic performance which would appear to be the desirable alternative to variable turbine geometry.

High speed aircraft using afterburning have to accommodate bypass flow for intake matching and cooling flow for the afterburner. The use of a lightweight ejector nozzle fulfills this purpose and also permits satisfactory divergent expansion within the range of exit diameter to engine diameter likely to be used in practice.

Higher turbine inlet temperatures are most beneficial in non-afterburning engines. For high speeds because of high compressor delivery temperatures, the use of air cooling of turbine blades becomes limited and the development of other methods of cooling or of materials, which do not require cooling, is necesary.

ACKNOWLEDGEMENTS

I wish to thank Orenda Engines Limited for permission to give this paper and to my colleagues for their help; in particular to Mr. W.H. Gibson who has been responsible for preparation of much of the detail.

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NOMENCLATURE

A	Area, A_t turbine throat area
C_L	Aircraft lift co-efficient
D	Aircraft drag, diameter
F	SF_n net specific thrust
L	Aircraft lift
M	Mach number
N	RPM
P	Total pressure, absolute
p	Static pressure
q	Dynamic pressure
R	Compression ratio, RF range factor
S	Wing area, SFC specific fuel consumption lbs/hr/lb thrust
T	Total temperature degrees absolute
t	Static temperature degrees absolute
W	Air weight flow
w	Weight, w_a aircraft weight, w_e engine weight
Δt	Compressor temperature rise - degrees
γ	Ratio of specific heats for air
η	Efficiency, η_i intake efficiency, η_{poly} polytropic compressor efficiency

Other Subscripts (not defined in text)

1	Compressor inlet
3	Turbine inlet

APPENDIX I

The Effect of Forward Speed on Engine Operation

If we assume that the flow $\frac{W_1\sqrt{T_1}}{P_1}$ into the compressor is a linear function of $N/\sqrt{T_1}$ then for constant rpm (N)

$$\frac{W_1 \, T_1}{P_1} = \text{constant} \qquad (1)$$

If we further assume that the turbine nozzles are choked then

$$\frac{W_1\sqrt{T_3}}{A_t \;\; P_3} = \text{constant}$$

$$\therefore \frac{\sqrt{T_3}}{T_1} \times \frac{P_1}{P_3} \times \frac{1}{A_t} = \text{constant} \qquad (2)$$

Hence for fixed A_t and T_3 $\frac{P_3}{P_1} \propto \frac{1}{T_1}$ i.e. the compression ratio is inversely proportional to the inlet temperature. Alternatively, we may assume that the compressor temperature rise Δ_t remains constant giving

$$R \propto \left(\frac{T_1 + \Delta_t}{T_1} \right)^{3.5 \, \eta_{poly}}$$

and substituting in (2) we can obtain variation of T_3 for constant A_t or vice versa.

APPENDIX II

Relationships Between Engine and Aircraft Performance

1. Best Cruise Altitude (h)

Let $\left(\frac{w_a}{S}\right)$ = aircraft wing loading

Then $\left(\frac{w_a}{S}\right) = C_L q = C_L \frac{\gamma}{2} p_a M^2$

$\therefore$ Altitude ambient pressure $(p_a) = \left(\frac{w_a}{S}\right) \frac{2}{C_L \gamma M^2}$ (1)

2. Engine Intake Pressure (P_d)

$P_d = P_o \times \eta_i$

$= p_a (1 + 0.2M^2)^{3.5} \times \eta_i$ (2)

3. Engine Intake Temperature (T_d)

$T_d = t_a (1 + 0.2M^2)$ (3)

4. Ratio Engine Weight/Aircraft Weight (w_e/w_a)

Let subscript "s" refer to sea level static condition.

"d" to design conditions

Then $\frac{W_s \; T_s}{P_s} = \frac{W_d \; T_d}{P_d}$ (equation 1 of Appendix I)

or $W_s = W_d \frac{P_s}{P_d} \frac{T_d}{T_s}$

Let SF_n = specific thrust

Then $W_d = \dfrac{W_a}{L/D\ (SF_{nd})}$

Giving $W_s = \dfrac{W_a}{L/D\ (SF_{nd})} \times \dfrac{P_s}{P_d} \times \dfrac{T_d}{T_s}$

and taking $\dfrac{W_s}{W_e}$ = 0.06 lbs airflow/sec. per lb weight for

an afterburning engine at sea level static we obtain:-

$$\left(\frac{W_e}{W_a}\right) = \frac{0.55}{L/D\ (SF_{nd})} \times \frac{T_d}{P_d} \tag{4}$$

for P_s = 14.7 psia and T_s = 288 degrees K (59 degrees F)

NOTE: The value of $\dfrac{W_s}{W_e}$ of 0.06 is for an engine of compression ratio 12 stressed for supersonic flight.

5. The Ratio of Specific Thrusts for a Given Engine at Different Flight Speeds

Let M_1 and M_2 be the two speeds and assume as in Appendix I that a constant rpm

$$\frac{W_1 T_1}{P_1} = \frac{W_2 T_2}{P_2}$$

or $\dfrac{W_a}{(L/D)_1\ (SF_n)_1} \times \dfrac{T_1}{P_1} = \dfrac{W_a}{(L/D)_2\ (SF_n)_2} \times \dfrac{T_2}{P_2}$

$$\therefore \frac{SF_{n1}}{SF_{n2}} = \frac{(L/D)_2}{(L/D)_1} \text{x2} \frac{P_2}{P_1} \times \frac{T_1}{T_2} \tag{5}$$

Example: Let $M_1 = 0.9M$, $M_2 = 2.5M$ then from Figure 10

for $\frac{w_a}{S} = 50\#/ft^2$, $T_1 = 260$ degrees K, $T_2 = 480$ degrees K,

$P_1 = 2.5$ psia, $P_2 = 6.5$ psia

$(L/D)_1 = 12$, $(L/D)_2 = 5.5$

$$\therefore \frac{SF_{n1}}{SF_{n2}} = \frac{5.5}{12} \times \frac{6.5}{2.5} \times \frac{260}{480} = 0.65$$

Therefore, if the maximum specific thrust at Mach 2.5 was 70 (say) then the specific thrust required from the same engine for subsonic cruise at Mach 0.9 would be 0.65 x 70 = 45.5.

6. Range Factor

$$\text{Range Factor} = \left(\frac{L}{D}\right) \times \frac{\text{Mach No.}}{\text{Specific Fuel Consumption}} \qquad (6)$$

Example: At 0.9M, $L/D = 12$ (Figure 9) and SFC = 1.0 (Figure 8)

at 2.5M, $L/D = 5.5$ and SFC = 2.2

Hence $RF_{0.9} = \frac{12 \times 0.9}{1.0} = 10.8$

$RF_{2.5} = \frac{5.5 \times 2.5}{2.2} = 6.25$

Suppose, however, that in order to attain $L/D = 5.5$ at 2.5M the maximum L/D at 0.9M was only 9 and that in order to attain a specific fuel consumption of 2.2 at 2.5M the exhaust was over-expanded and compression ratio lower than 13 at 0.9M giving a specific fuel consumption of 1.3, then $RF_{0.9} = 6.25$.

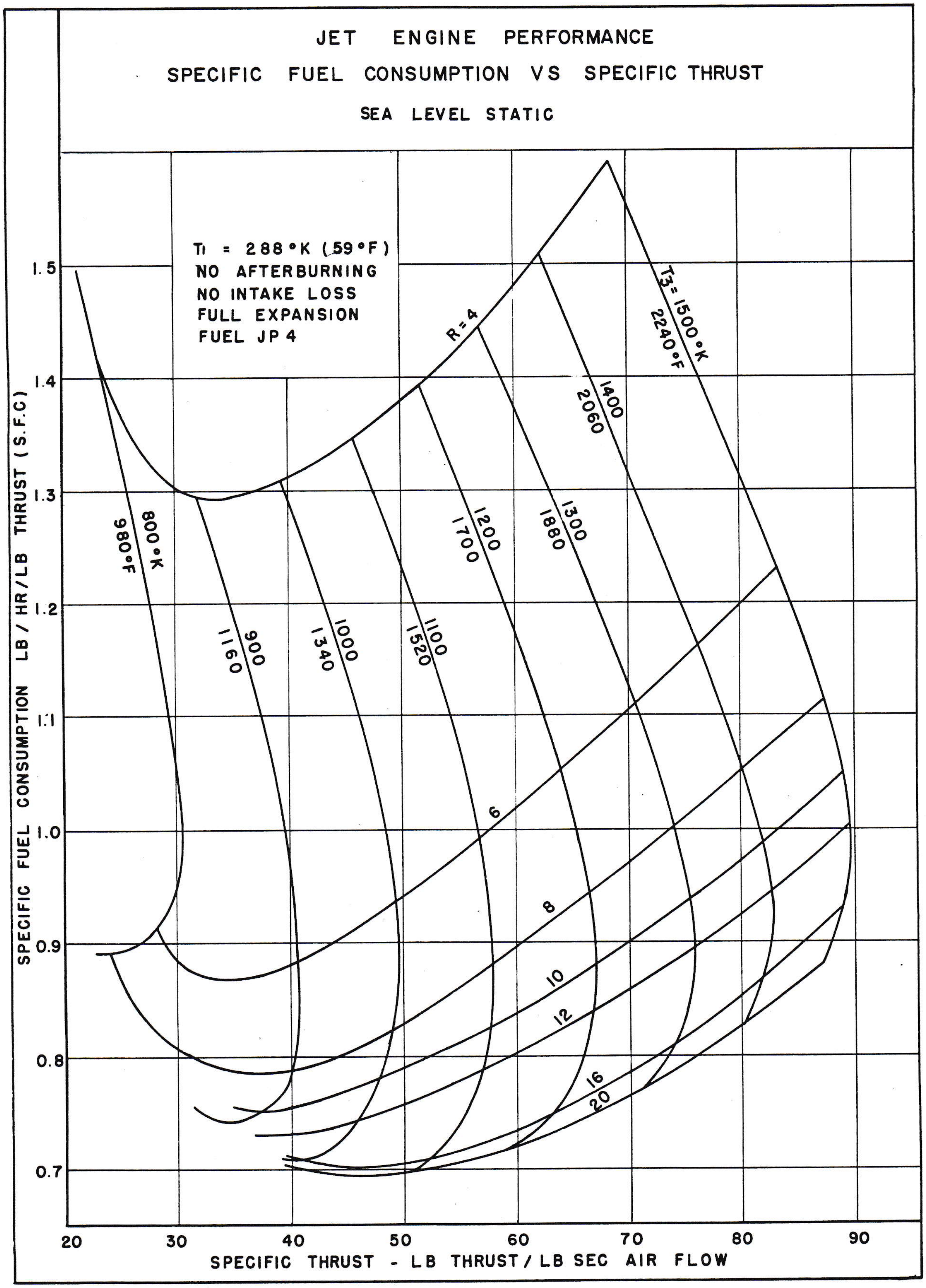
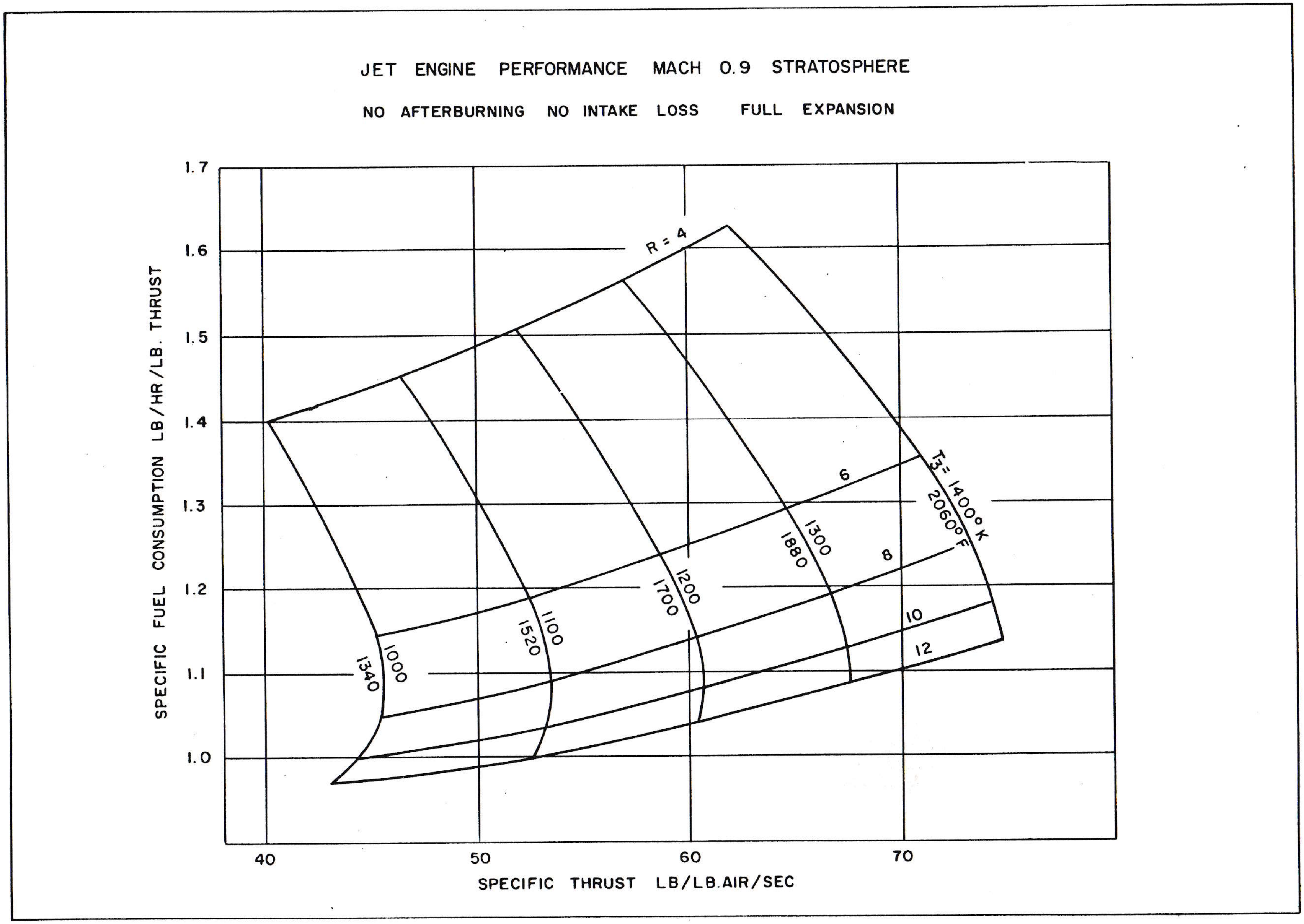

JET ENGINE PERFORMANCE
SPECIFIC FUEL CONSUMPTION VS SPECIFIC THRUST
SEA LEVEL STATIC
T1 = 288 °K (59 °F)
NO AFTERBURNING
NO INTAKE LOSS
FULL EXPANSION
FUEL JP 4
R = 4
6
8
10
12
16
20
800 °K 980 °F
900 1160
1000 1340
1100 1520
1200 1700
1300 1880
1400 2060
T3 = 1500 °K 2240 °F
SPECIFIC FUEL CONSUMPTION LB / HR / LB THRUST (S.F.C)
1.5
1.4
1.3
1.2
1.1
1.0
0.9
0.8
0.7
20
30
40
50
60
70
80
90
SPECIFIC THRUST - LB THRUST / LB SEC AIR FLOW

Fig. 1



JET ENGINE PERFORMANCE MACH 0.9 STRATOSPHERE
NO AFTERBURNING NO INTAKE LOSS FULL EXPANSION
SPECIFIC FUEL CONSUMPTION LB/HR/LB. THRUST
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
SPECIFIC THRUST LB/LB.AIR/SEC
40
50
60
70
R = 4
6
8
10
12
T3 = 1400° K
2060° F
1300
1880
1200
1700
1100
1520
1000
1340

Fig. 2

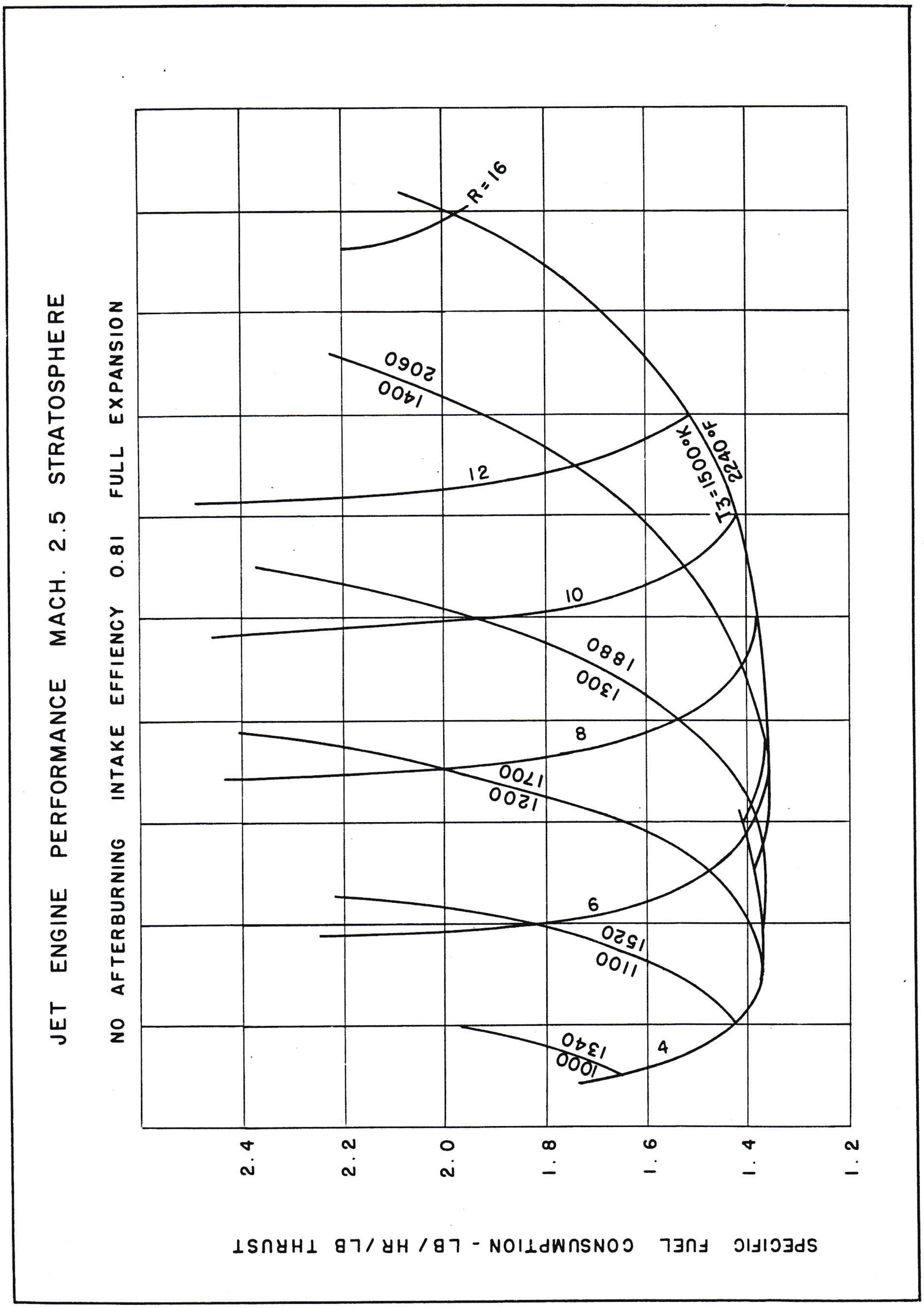
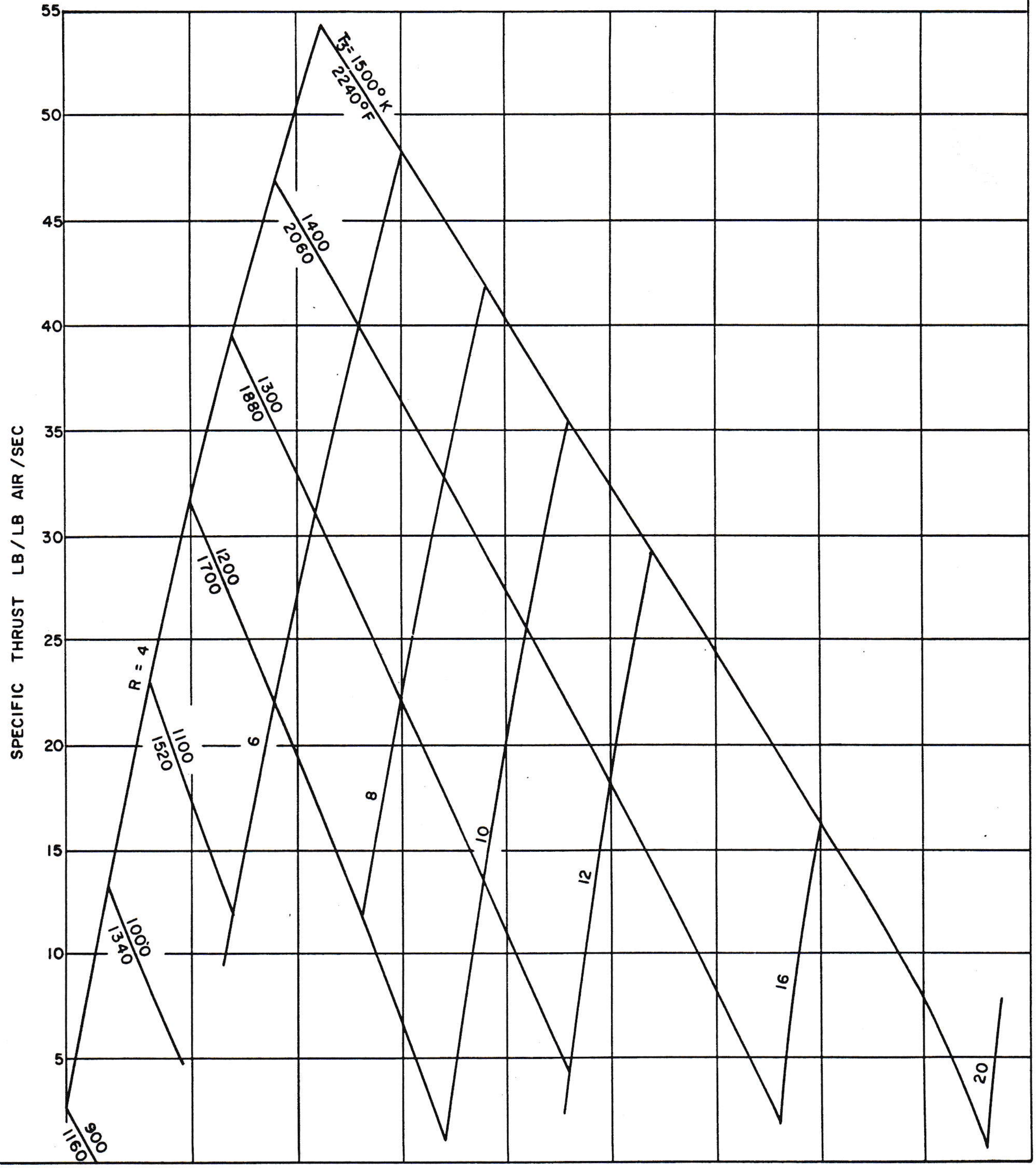

JET ENGINE PERFORMANCE MACH. 2.5 STRATOSPHERE
NO AFTERBURNING INTAKE EFFIENCY 0.81 FULL EXPANSION
SPECIFIC FUEL CONSUMPTION - LB / HR / LB THRUST
2.4
2.2
2.0
1.8
1.6
1.4
1.2
R=16
12
10
8
6
4
2060
1400
1880
1300
1700
1200
1520
1100
1340
1000
T3=1500°K
2240°F

Fig. 3

JET ENGINE PERFORMANCE
MACH 2.5 STRATOSPHERE

NO AFTERBURNING
INTAKE EFFICIENY 0.81
FULL EXPANSION



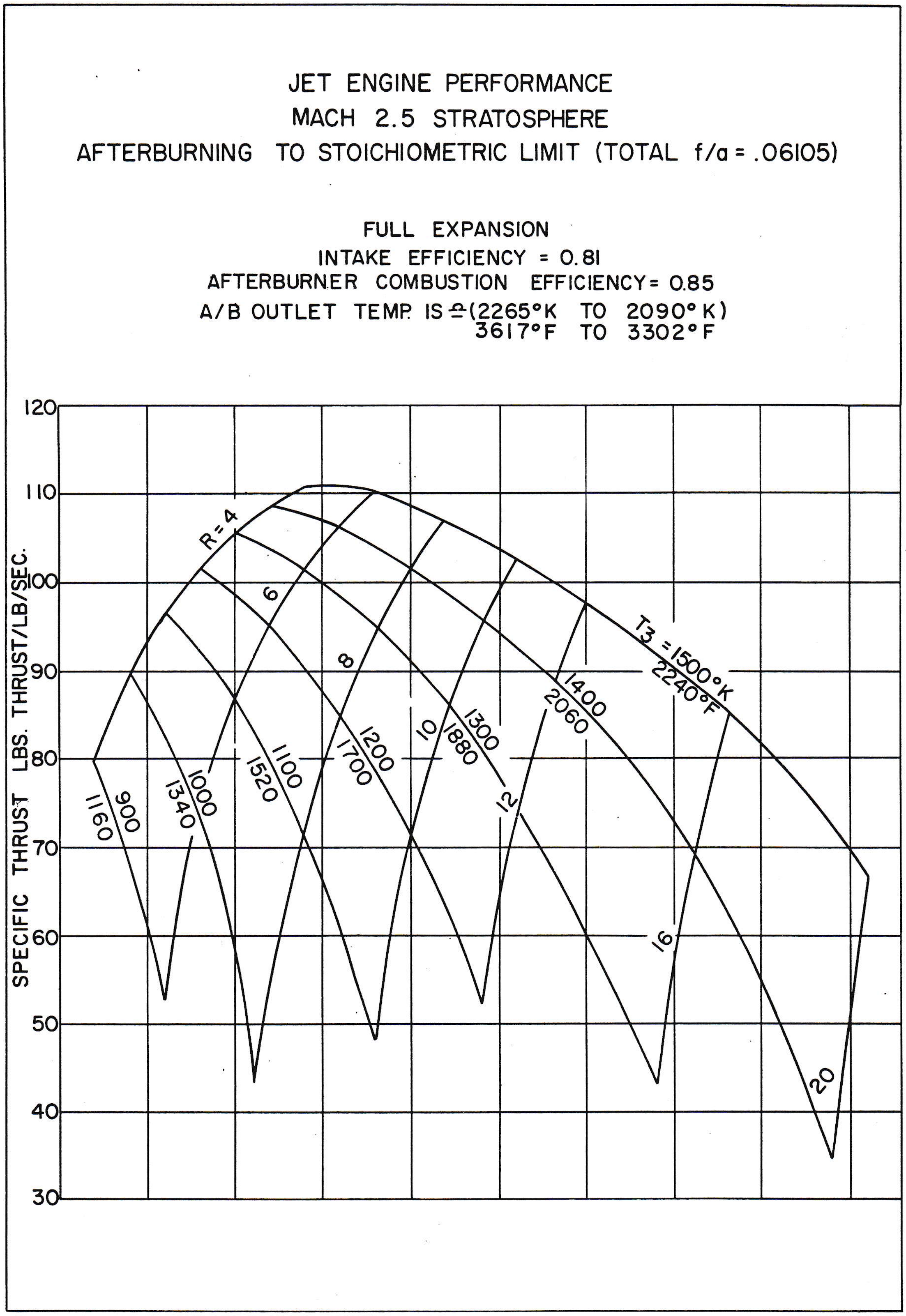
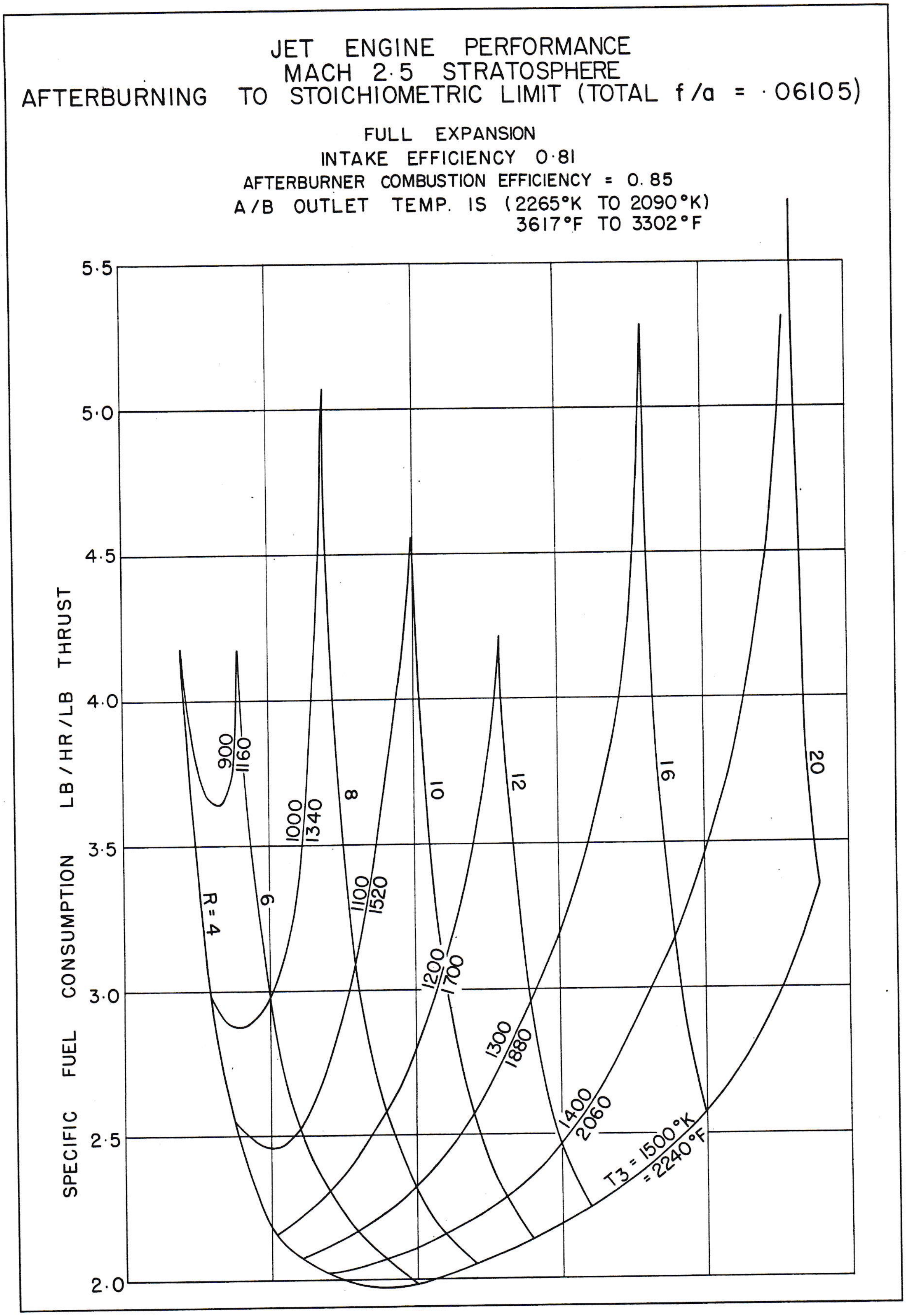

JET ENGINE PERFORMANCE
MACH 2.5 STRATOSPHERE
AFTERBURNING TO STOICHIOMETRIC LIMIT (TOTAL f/a = .06105)
FULL EXPANSION
INTAKE EFFICIENCY = 0.81
AFTERBURNER COMBUSTION EFFICIENCY = 0.85
A/B OUTLET TEMP. IS ≏ (2265°K TO 2090°K)
3617°F TO 3302°F
SPECIFIC THRUST LBS. THRUST/LB/SEC.
120
110
100
90
80
70
60
50
40
30
R = 4
6
8
10
12
16
20
T3 = 1500°K
2240°F
1400
2060
1300
1880
1200
1700
1100
1520
1000
1340
900
1160

Fig. 5



JET ENGINE PERFORMANCE
MACH 2·5 STRATOSPHERE
AFTERBURNING TO STOICHIOMETRIC LIMIT (TOTAL f/a = ·06105)
FULL EXPANSION
INTAKE EFFICIENCY 0·81
AFTERBURNER COMBUSTION EFFICIENCY = 0.85
A/B OUTLET TEMP. IS (2265°K TO 2090°K)
3617°F TO 3302°F
SPECIFIC FUEL CONSUMPTION LB/HR/LB THRUST
5·5
5·0
4·5
4·0
3·5
3·0
2·5
2·0
R = 4
6
8
10
12
16
20
900
1160
1000
1340
1100
1520
1200
1700
1300
1880
1400
2060
T3 = 1500°K
= 2240°F

Fig. 6

THE EFFECT OF FORWARD SPEED ON TURBOJET OPERATION AT MAXIMUM R.P.M

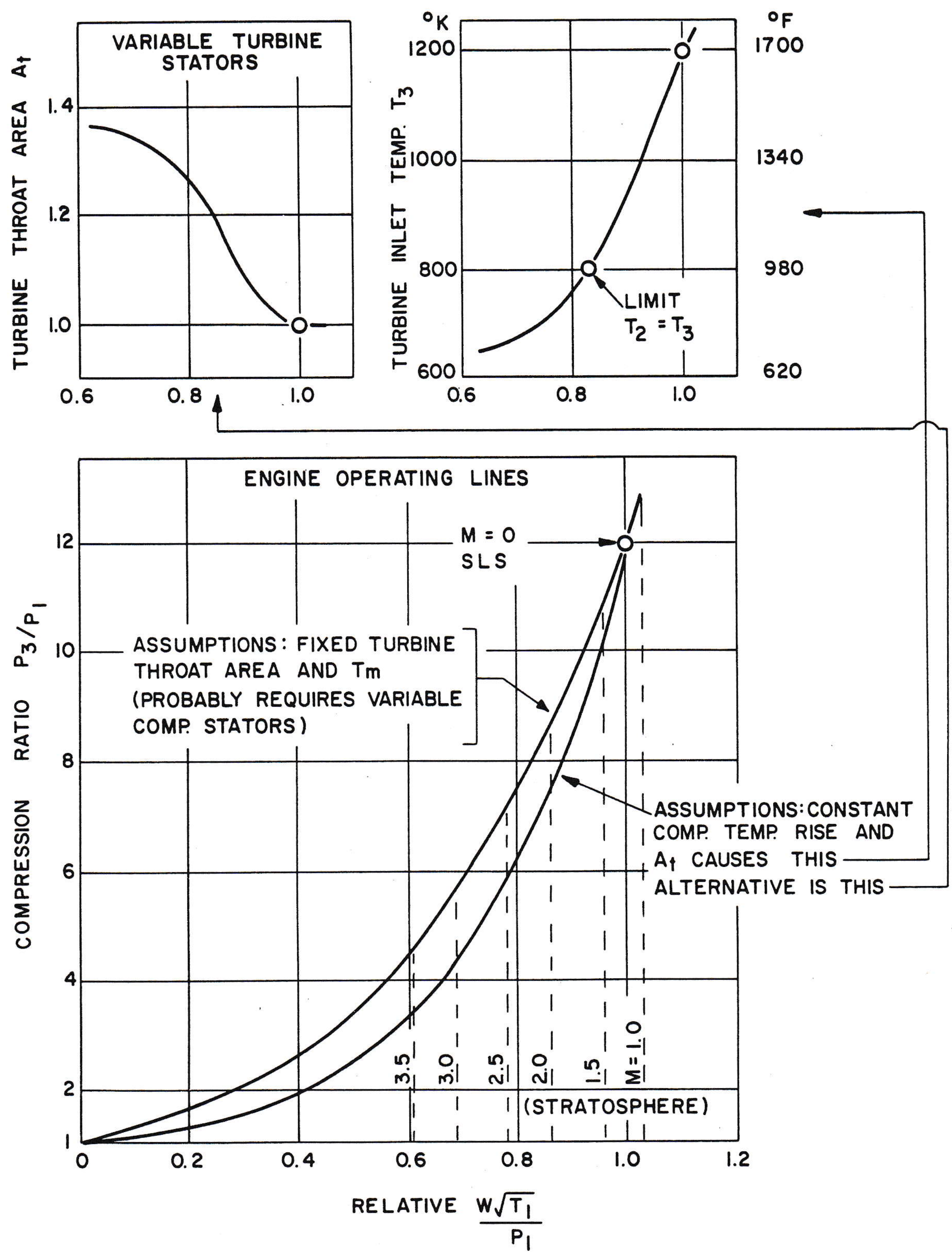
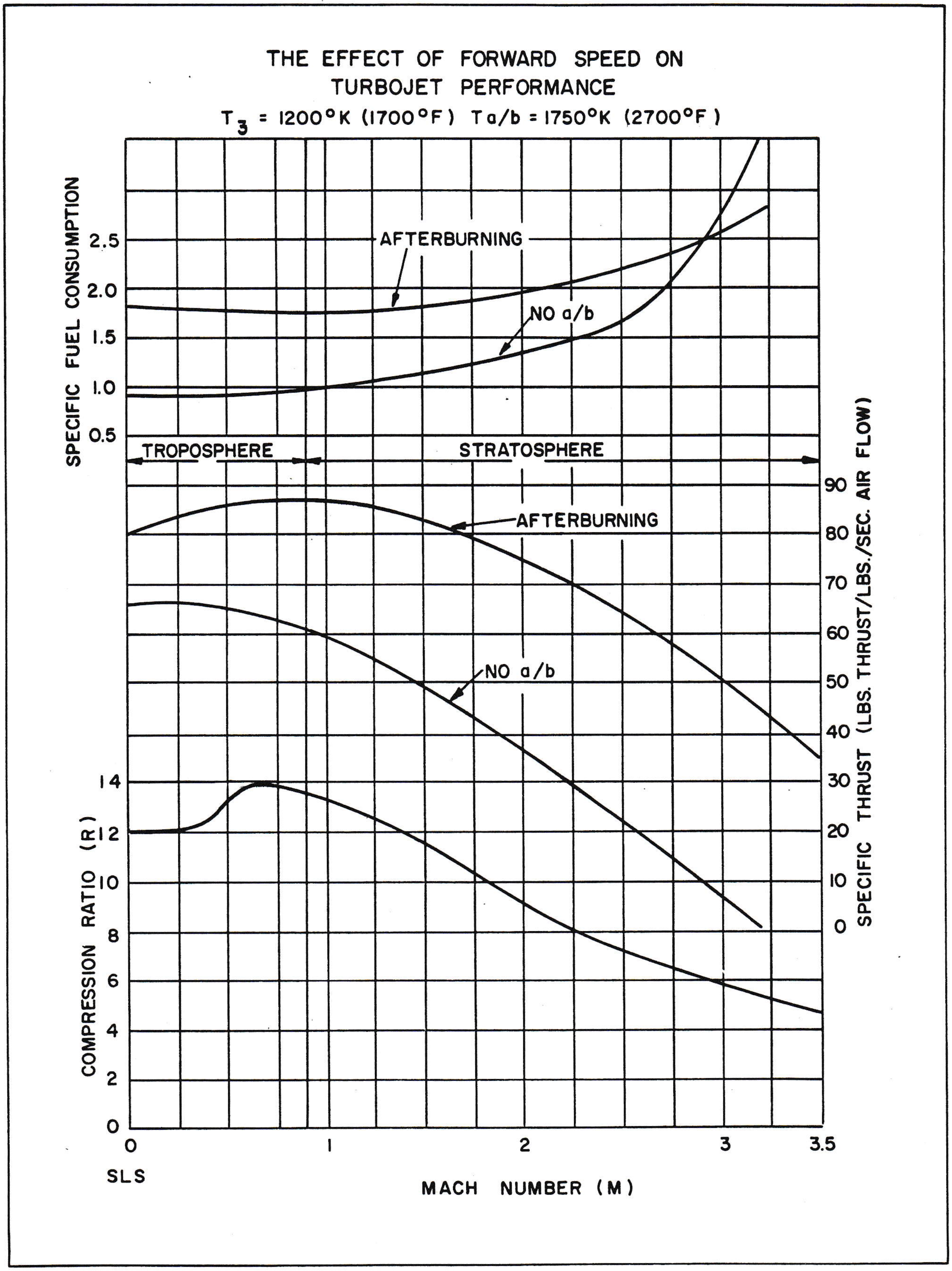


Fig. 7



THE EFFECT OF FORWARD SPEED ON
TURBOJET PERFORMANCE
T_3 = 1200°K (1700°F) Ta/b = 1750°K (2700°F)
SPECIFIC FUEL CONSUMPTION
2.5
2.0
1.5
1.0
0.5
AFTERBURNING
NO a/b
TROPOSPHERE
STRATOSPHERE
AFTERBURNING
NO a/b
SPECIFIC THRUST (LBS. THRUST/LBS./SEC. AIR FLOW)
90
80
70
60
50
40
30
20
10
0
COMPRESSION RATIO (R)
14
12
10
8
6
4
2
0
0
1
2
3
3.5
SLS
MACH NUMBER (M)

Fig.8

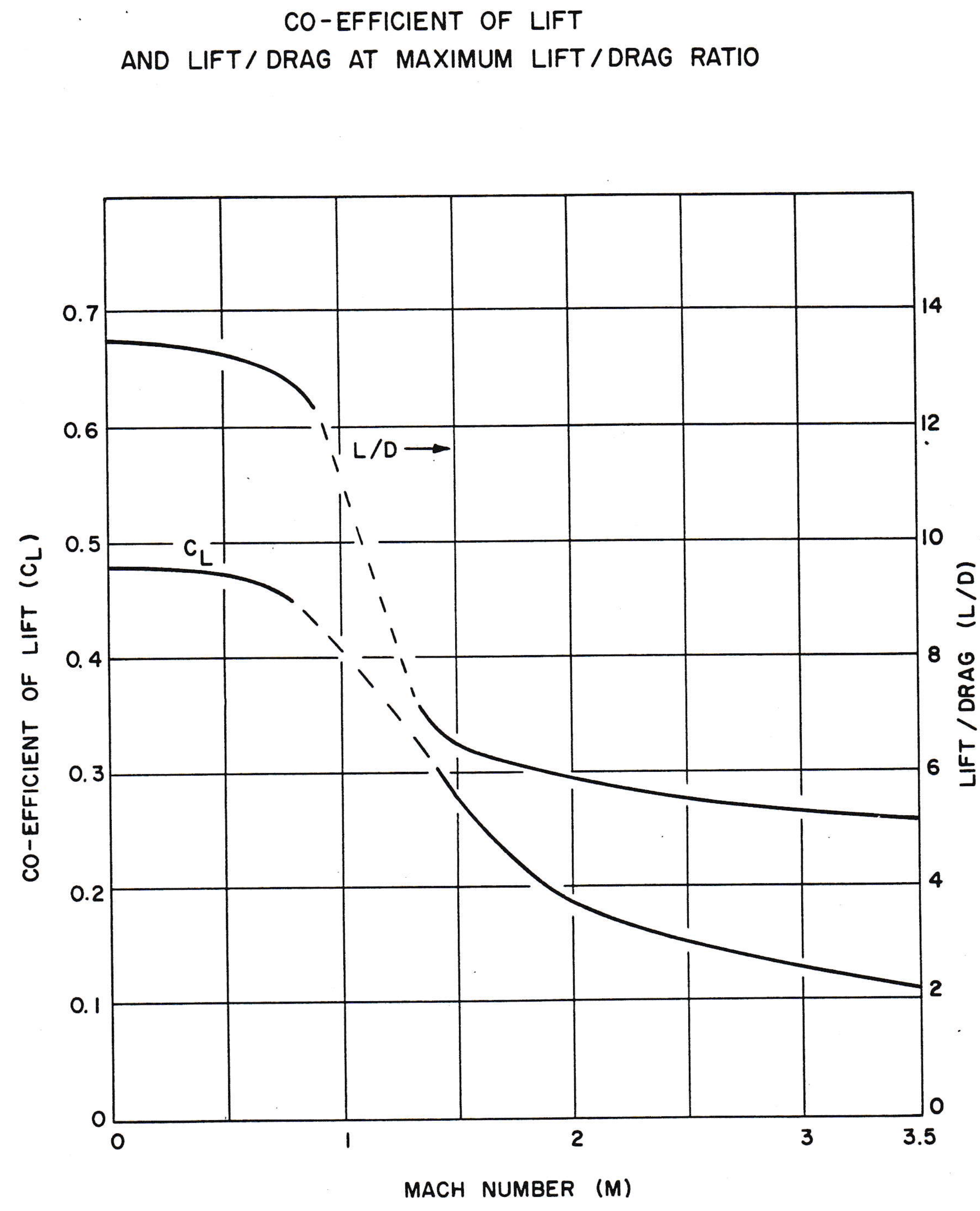
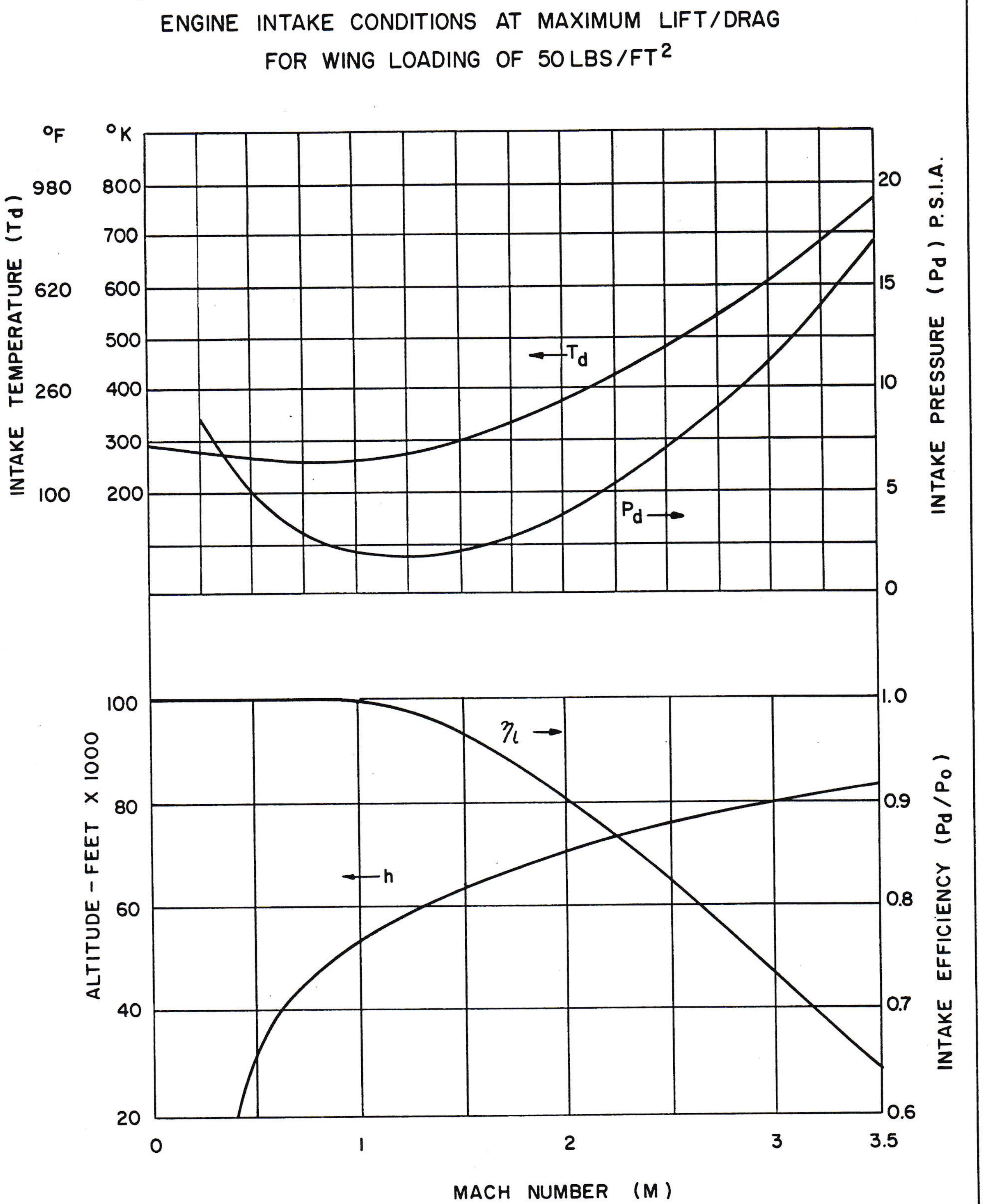
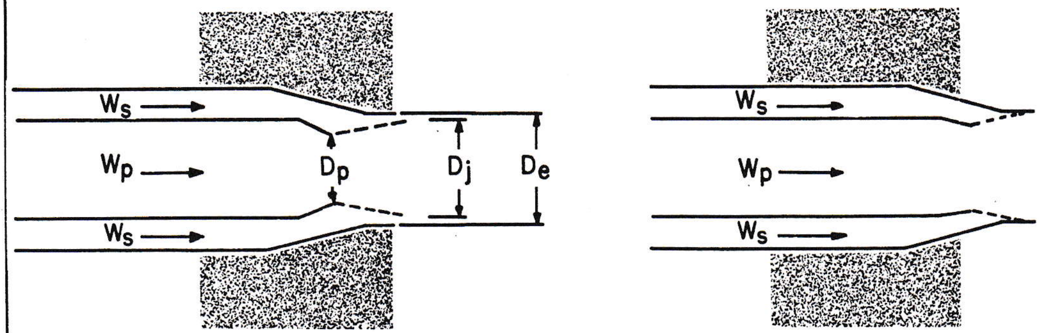

CO-EFFICIENT OF LIFT
AND LIFT/DRAG AT MAXIMUM LIFT/DRAG RATIO
CO-EFFICIENT OF LIFT (C_L)
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
L/D
C_L
LIFT/DRAG (L/D)
14
12
10
8
6
4
2
0
0
1
2
3
3.5
MACH NUMBER (M)

Fig. 9



ENGINE INTAKE CONDITIONS AT MAXIMUM LIFT/DRAG
FOR WING LOADING OF 50 LBS/FT²
INTAKE TEMPERATURE (Td)
°F
°K
980
800
700
620
600
500
260
400
300
100
200
Td
Pd
INTAKE PRESSURE (Pd) P.S.I.A.
20
15
10
5
0
ALTITUDE - FEET X 1000
100
80
60
40
20
h
ηi
INTAKE EFFICIENCY (Pd/Po)
1.0
0.9
0.8
0.7
0.6
0
1
2
3
3.5
MACH NUMBER (M)

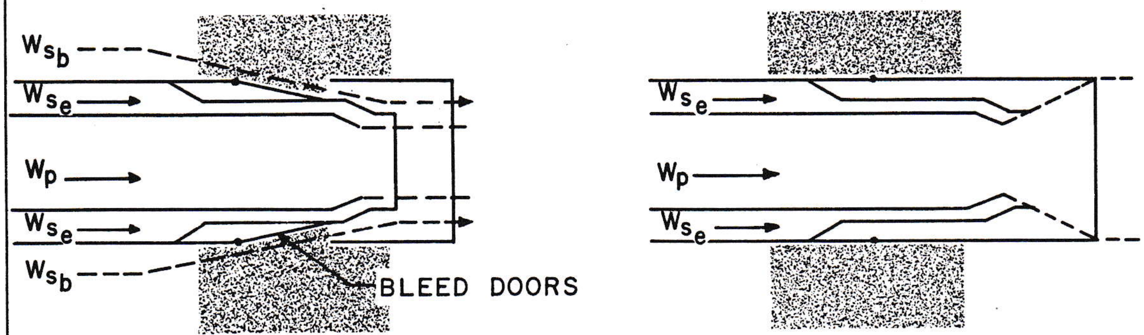
Fig. 10



A. CYLINDRICAL EJECTOR
NO AFTERBURNER

B. CYLINDRICAL EJECTOR
WITH AFTERBURNER
W_s THROTTLED

LOW MACH NUMBER (<1.5) EJECTOR NOZZLES



C. AT LOW MACH NUMBER (<1.0)

D. AT HIGH MACH NUMBER (2.5)

HIGH MACH NUMBER (>1.5) EJECTOR NOZZLES

E. NO AFTERBURNER

F. WITH AFTERBURNER

ALTERNATIVE HIGH MACH NUMBER EJECTOR NOZZLE
(DOUBLE IRIS)

EJECTOR NOZZLES

Fig. 11

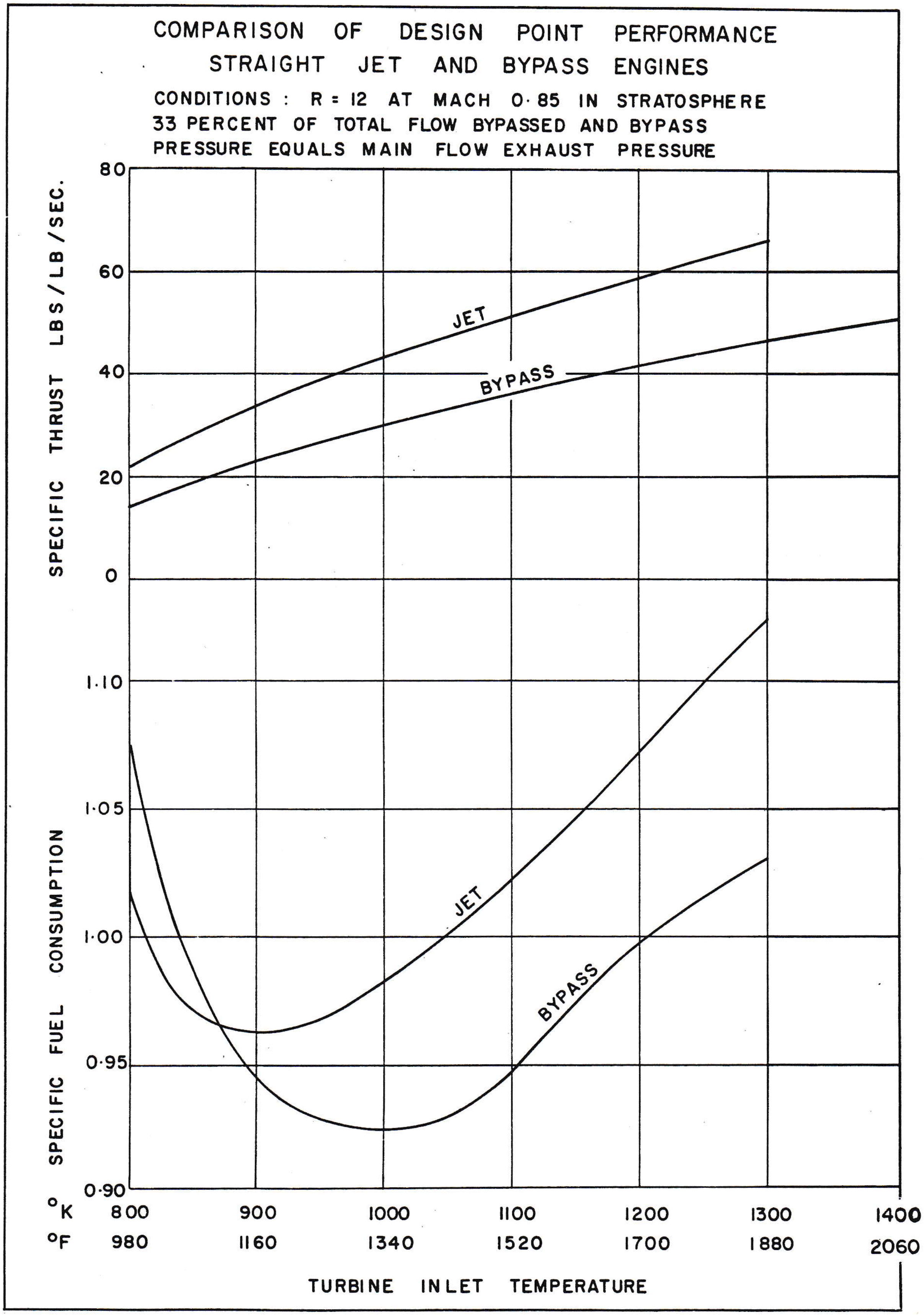
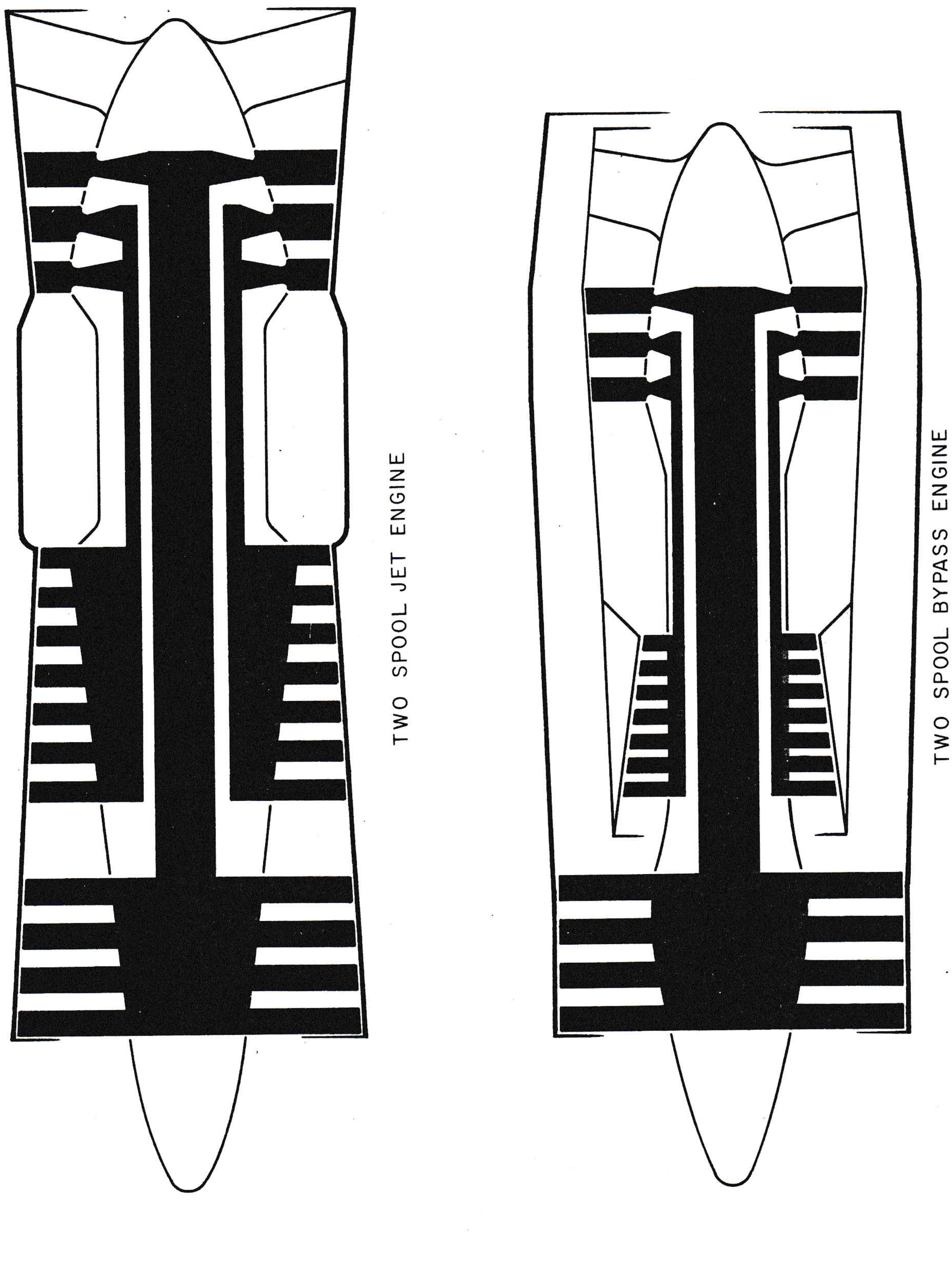

COMPARISON OF DESIGN POINT PERFORMANCE
STRAIGHT JET AND BYPASS ENGINES
CONDITIONS : R = 12 AT MACH 0·85 IN STRATOSPHERE
33 PERCENT OF TOTAL FLOW BYPASSED AND BYPASS
PRESSURE EQUALS MAIN FLOW EXHAUST PRESSURE
SPECIFIC THRUST LBS/LB/SEC.
80
60
40
20
0
JET
BYPASS
SPECIFIC FUEL CONSUMPTION
1·10
1·05
1·00
0·95
0·90
JET
BYPASS
°K 800 900 1000 1100 1200 1300 1400
°F 980 1160 1340 1520 1700 1880 2060
TURBINE INLET TEMPERATURE

Fig. 12



COMPARISON OF JET AND BYPASS ENGINES
OF THE SAME THRUST

Fig. 13

THRUST / WEIGHT RATIO (DRY)

(GREGORY - THE FUTURE OF SMALL TURBOJET ENGINES - 1956)

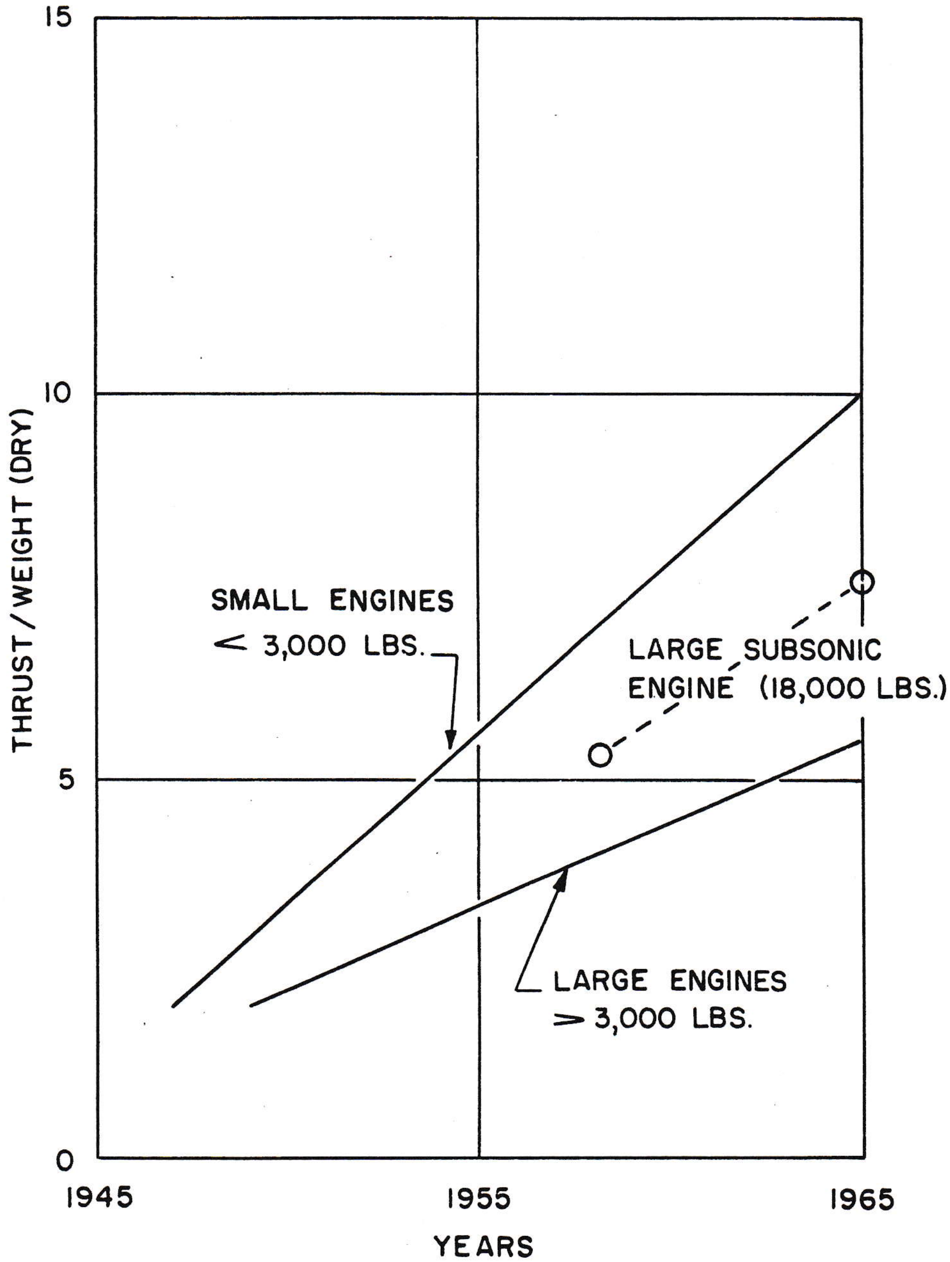
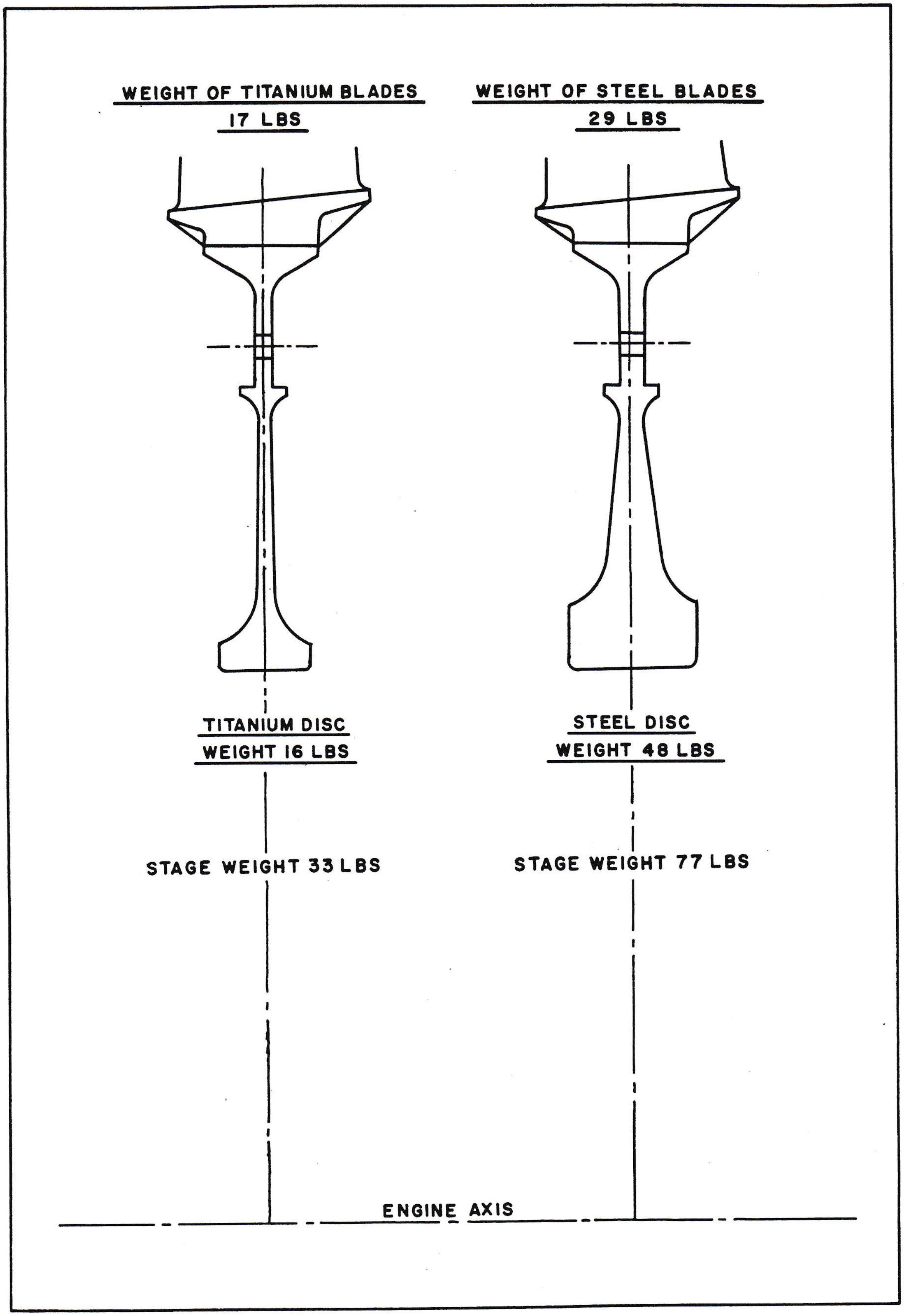
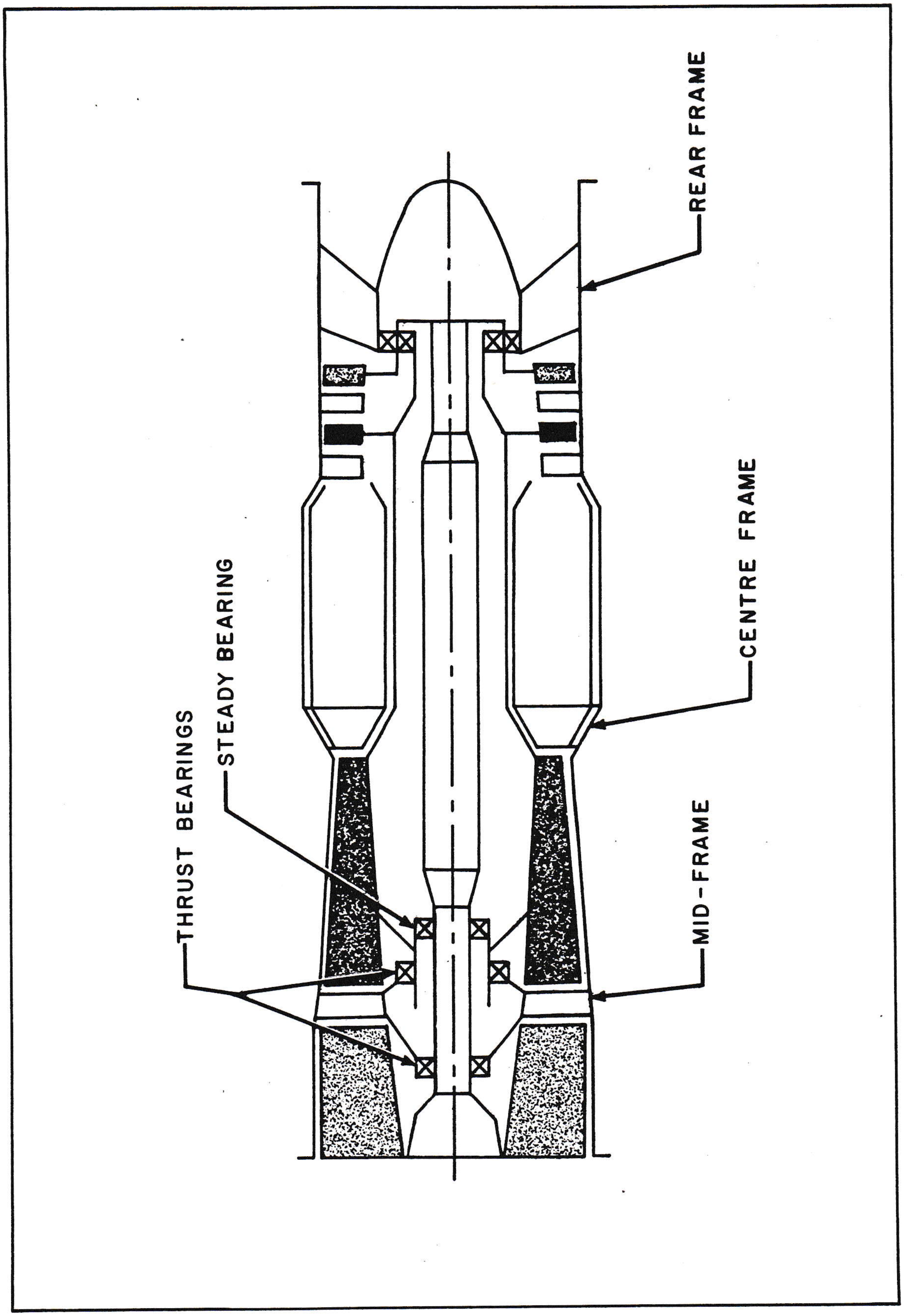


Fig. 14



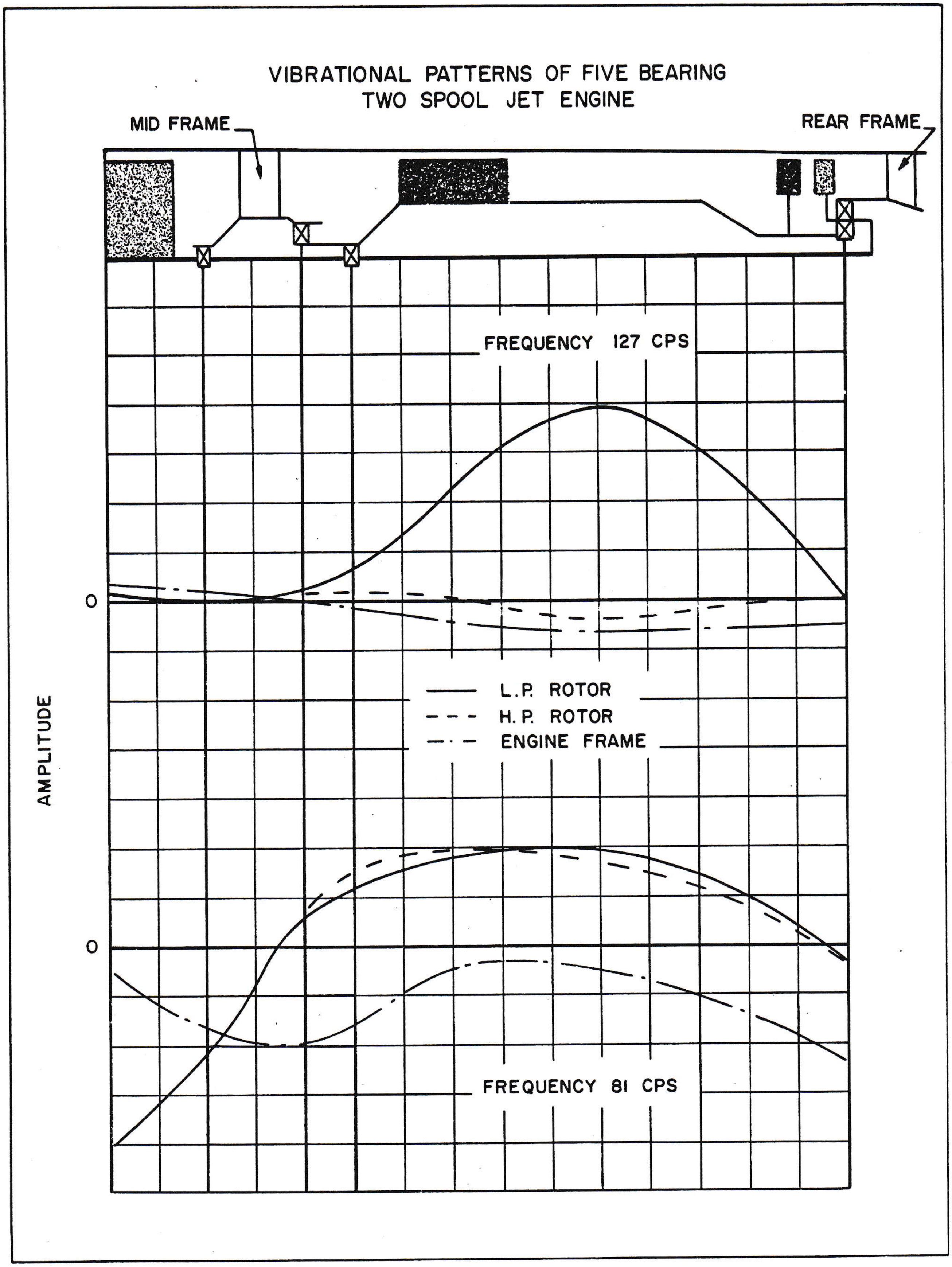
TITANIUM AND STEEL DISCS FOR THE SAME DUTY

Fig. 15



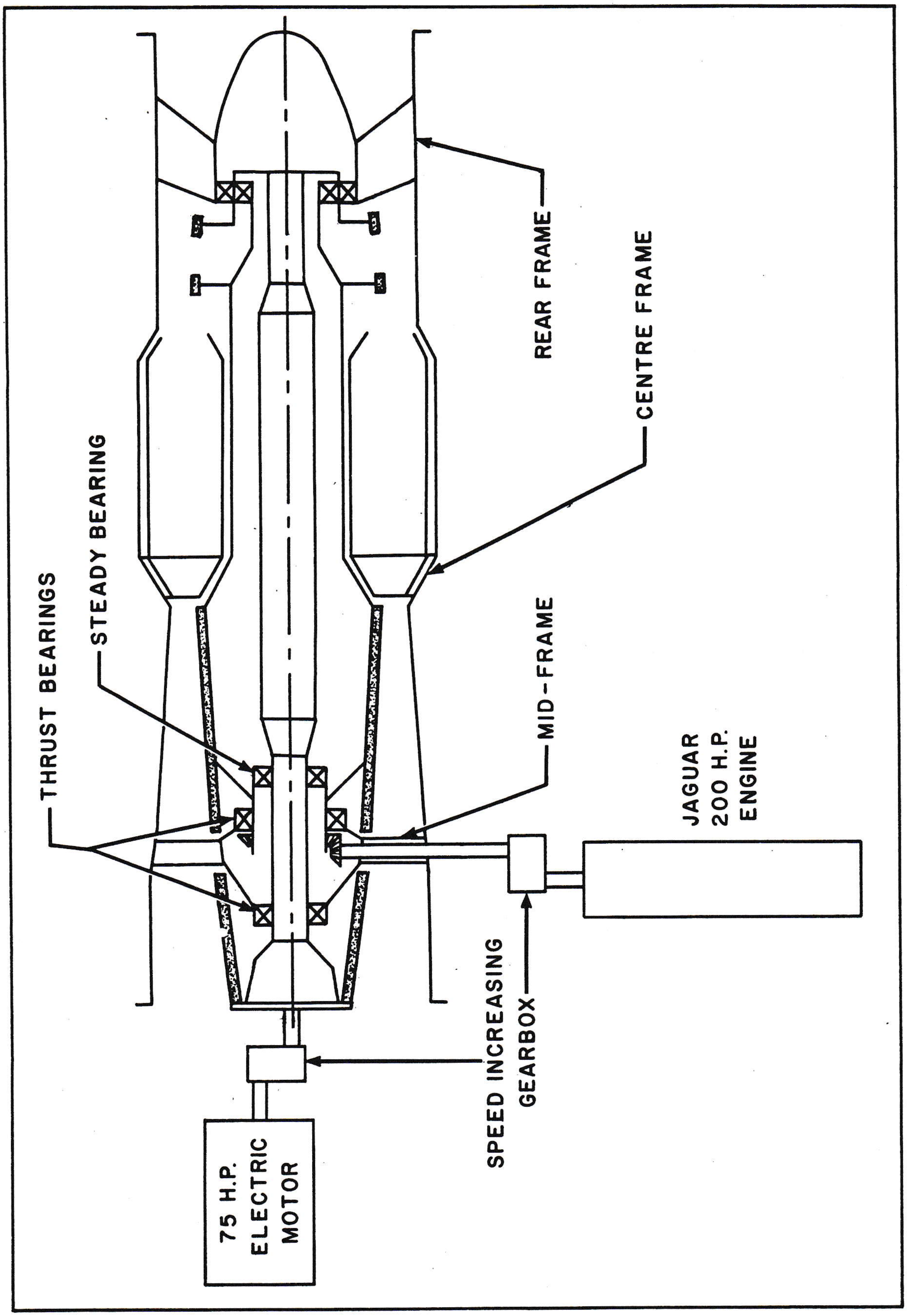
FOUR MAIN BEARING ARRANGEMENT OF
TWO SPOOL JET ENGINE

Fig.16



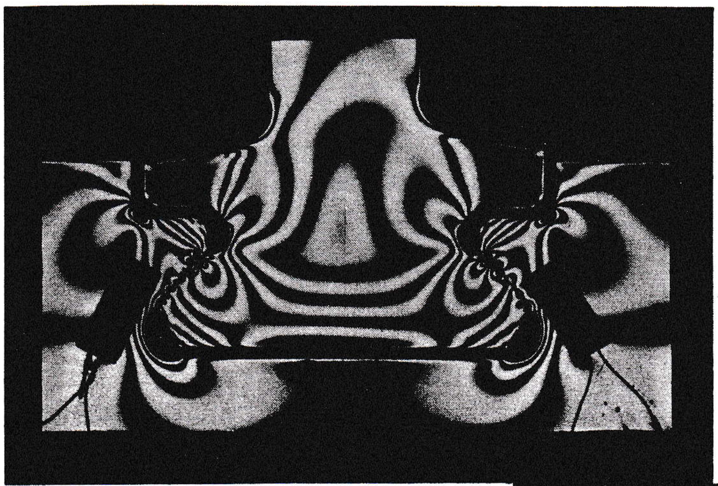
VIBRATIONAL PATTERNS OF FIVE BEARING
TWO SPOOL JET ENGINE
MID FRAME
REAR FRAME
FREQUENCY 127 CPS
0
L.P. ROTOR
H.P. ROTOR
ENGINE FRAME
AMPLITUDE
0
FREQUENCY 81 CPS

Fig. 17



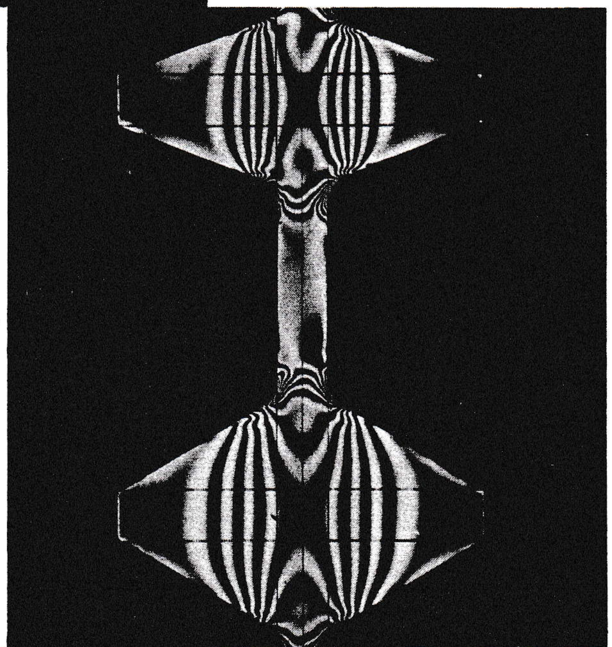
WHIRLING SPEED TEST RIG

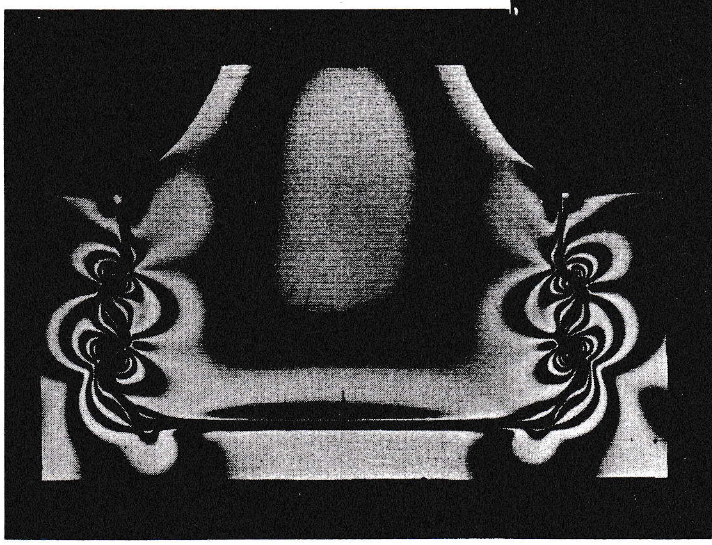
Fig. 18



a) COMPRESSOR BLADE
SINGLE BRANCH

c) AT JUNCTION OF DISC
RIM AND WEB FOR
20 DEGREES AND FOR
30 DEGREES

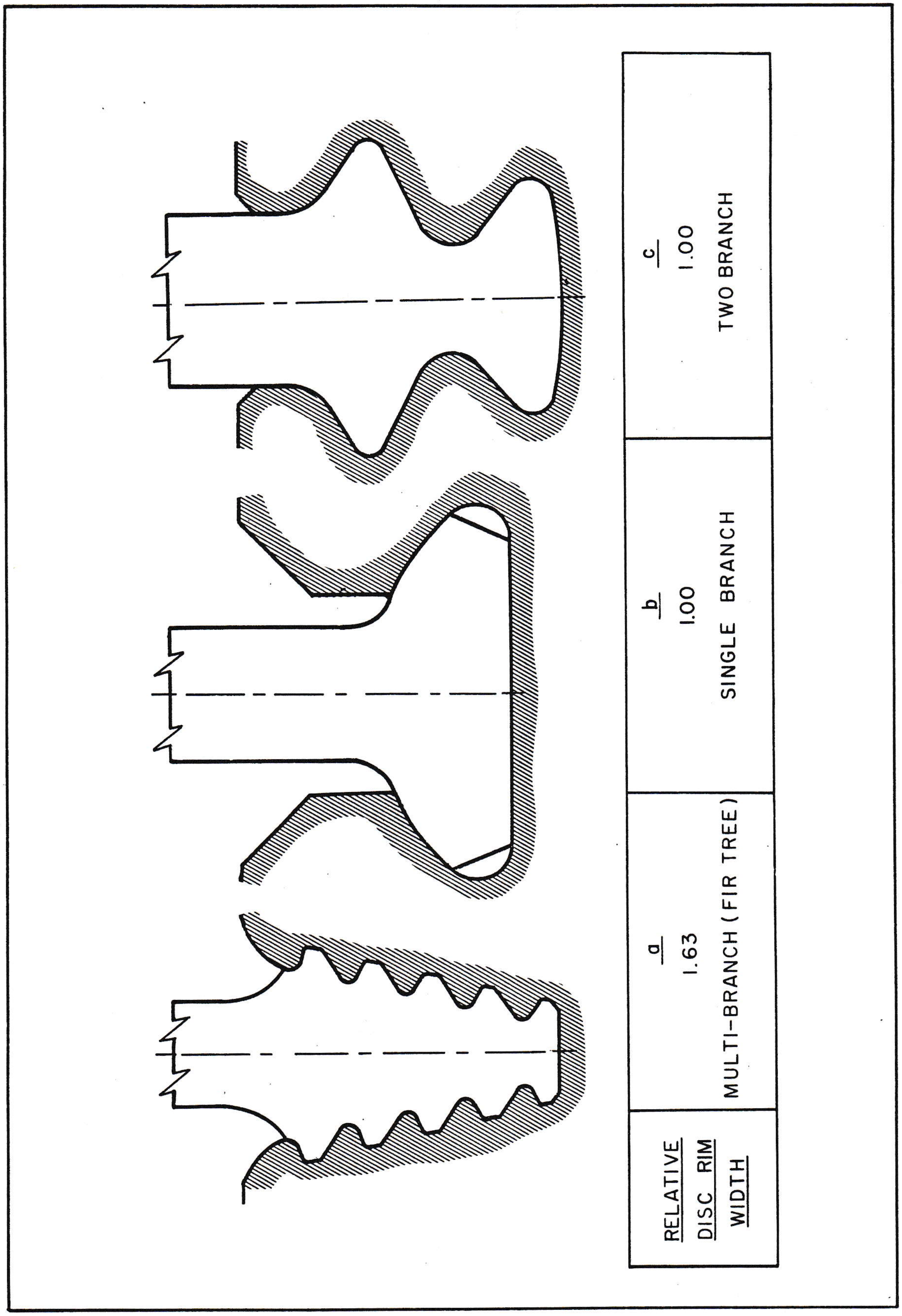




b) COMPRESSOR BLADE
TWO BRANCH

PHOTOELASTIC STUDIES

Fig. 19



DEVELOPMENT OF TURBINE BLADE ROOT FIXINGS
FOR EQUAL LOAD

Fig. 20

COMPARISON OF TURBINE DISC MATERIALS

MATERIALS	A - 286	GREEK ASCOLOY	H-46	INCO 901
1. MAIN ALLOYING ELEMENTS (%)				
CHROME	14·75	13·0	12·0	13·00
NICKEL	25·0	2·0		44·0
TUNGSTON		3·0		
MOLYBDENUM	1·25		0·5	6·00
OTHERS	Ti = 1·90, Al = 0·35 V = 0·25		CB = 0·30 V = 3	Ti = 2·50
2. RELATIVE COST (BASIC)	2·50	1·50	1·50	4·00
3. DENSITY LBS. / CU. IN.	0·286	0·283	0·280	0·297
4. SHORT TIME STRENGTH (MIN.)				
0·2 % Y.S. ROOM TEMP.	100,000	118,000	100,000	100,000
(752°F) 400°C	91,000	107,500	82,500	92,000
(1112°F) 600°C	84,000	67,000	48,500	88,000
UTS ROOM TEMP.	130,000	138,000	124,000	150,000
(752°F) 400°C	114,000	128,000	106,000	130,000
(1112°F) 600°C	101,000	81,000	60,000	120,000
5. CREEP STRENGTH				
RUPTURE IN 1000 HRS. (932°F) 500°C	90,000	50,000	62,000	105,000
(1112°F) 600°C	58,500	21,000	32,500	75,000
6. COEFFICIENT OF THERMAL EXP.				
IN./IN./°C ROOM TEMP.	$16{\cdot}2 \times 10^{-6}$	$9{\cdot}2 \times 10^{-6}$	$9{\cdot}2 \times 10^{-6}$	$13{\cdot}85 \times 10^{-6}$
(752°F) 400°C	17·27	10·9	11·7	14·58
(1112°F) 600°C	17·72	11·6	12·1	15·51

Fig. 21

THE EFFECT OF AIR COOLING OF
TURBINE BLADES ON PERFORMANCE

STOICHIOMETRIC AFTERBURNING MACH. 2.5
COMPRESSOR OUTLET TEMPERATURE = 875 °K (1115 °F)
R = 6.0 THREE STAGE TURBINE FULL EXPANSION

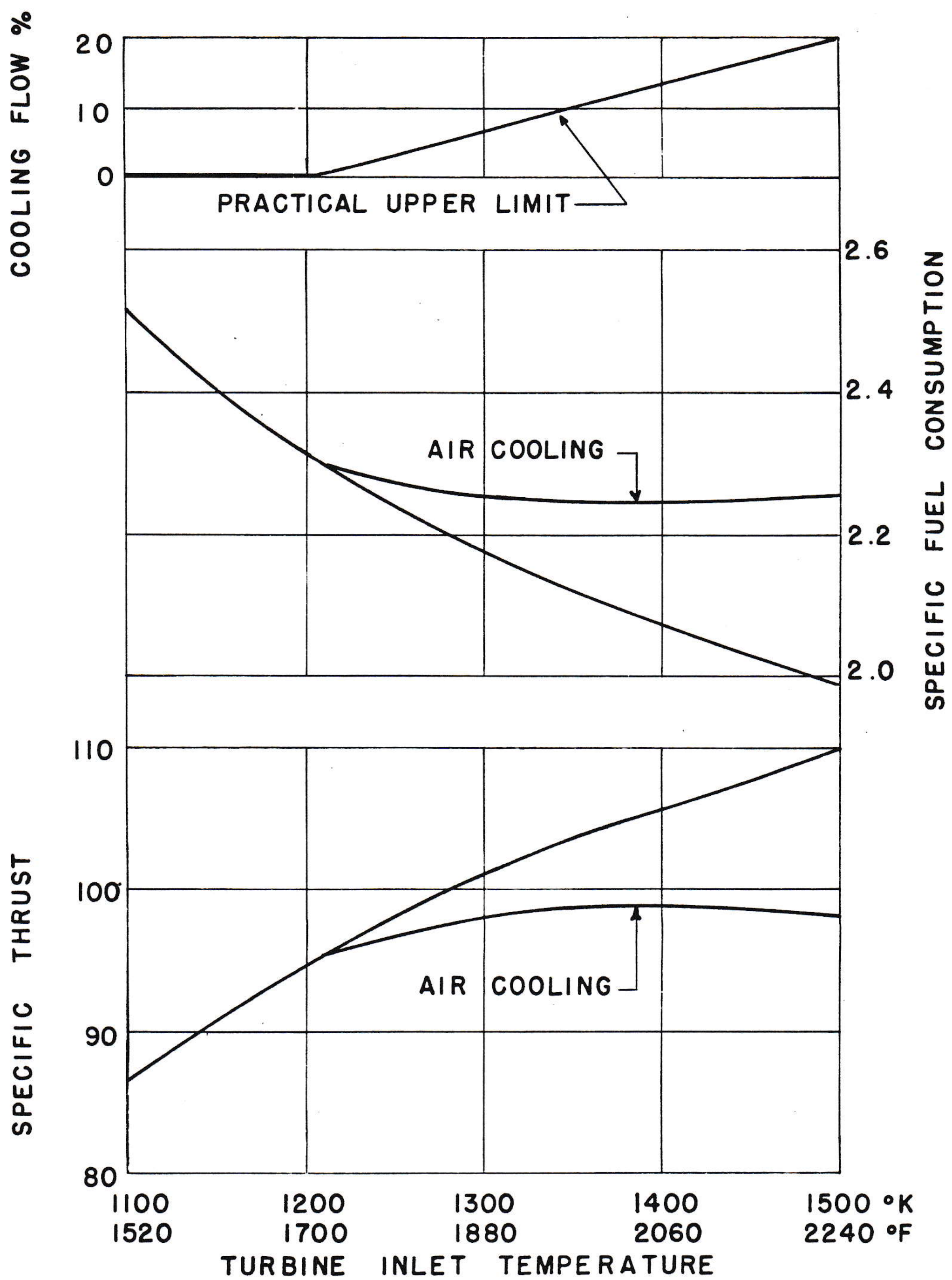
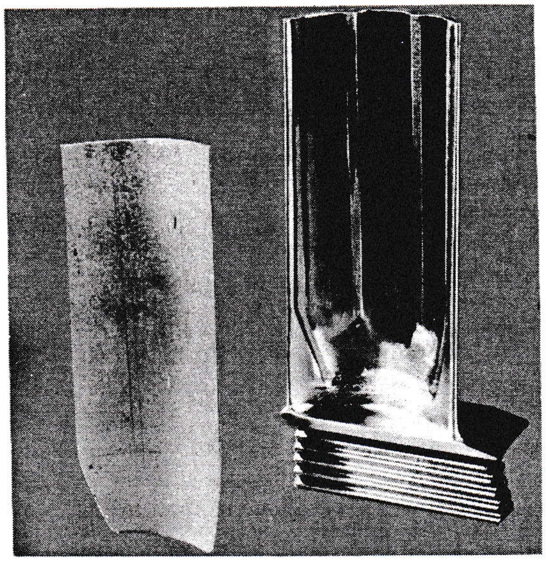
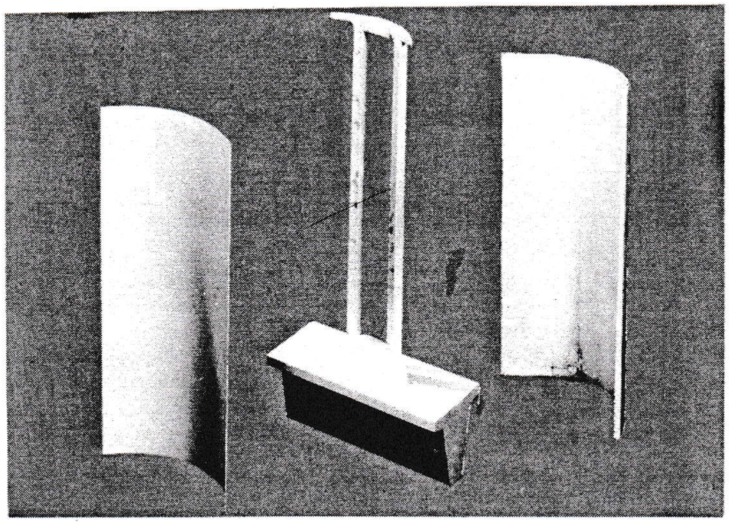


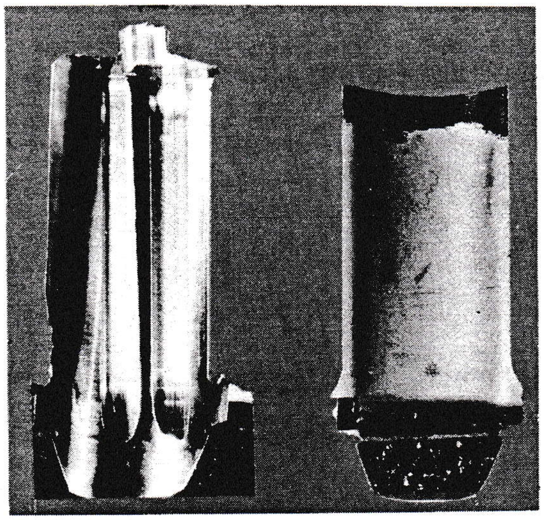
Fig.22



a) SINGLE COVER PLATE JOINED AT LEADING AND TRAILING EDGE

b) DOUBLE COVER PLATE JOINED AT LEADING AND TRAILING EDGE





c) TWO MACHINED BLADE FORGINGS JOINED ON CONVEX SURFACE NEAR NEUTRAL BENDING AXIS

COOLED TURBINE BLADES

(FOR FATIGUE TESTING IN LABORATORY AND ENGINES)

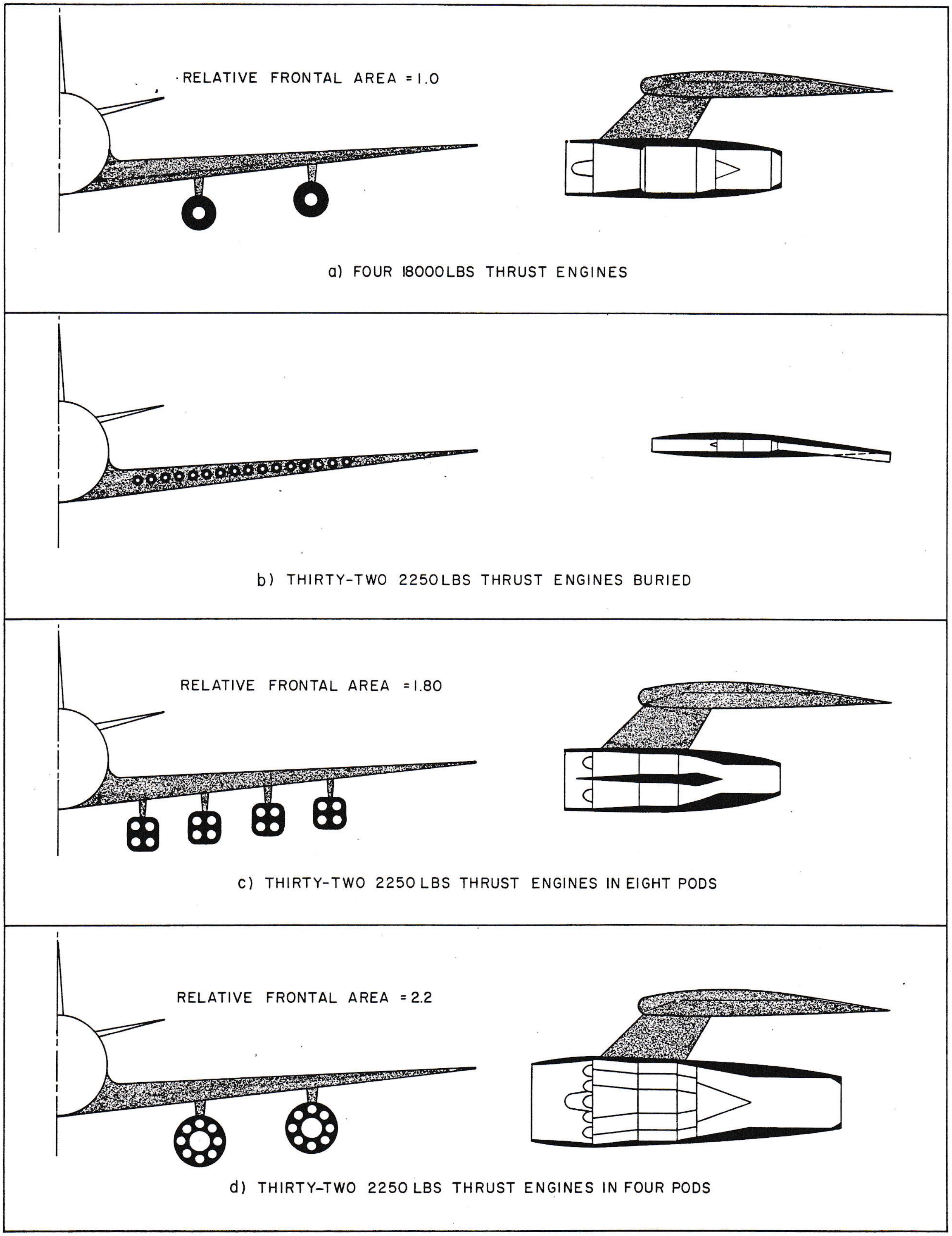
Fig. 23

TURBINE BLADE MATERIALS

PROPERTIES OF MOLYBDENUM (0·5 % Ti) VS. INCO 700

		MOLYBDENUM	INCO 700
1. DENSITY	LBS. /CU. IN	0·369	0·298
2. CREEP PROPERTIES			
100 HRS. TO RUPTURE	800°C (1472°F)	80,500 PSI	43,000
	1,000°C (1832°F)	52,000	4,000
	1,100°C (2012°F)	24,000	
3. SHORT TIME STRENGTH			
UTS	ROOM TEMP.	113,000 PSI	148,000 PSI
	650°C (1202°F)	84,700	113,000
	900°C (1652°F)	77,000	62,000
YS	ROOM TEMP.	87,900 (0.1%)	100,000 (0.2%)
	650°C (1202°F)	62,600	85,000
	900°C (1652°F)	62,800	42,000
4. MODULUS OF ELASTICITY (DYNAMIC)			
	ROOM TEMP.	45.5×10^6 PSI	32.2×10^6 PSI
	800°C (1472°F)	40.0×10^6	23.8×10^6
5. COEFFICIENT OF THERMAL EXP.			
	R.T. to 100°C (212°F)	5.5×10^{-6} IN/IN/°C	12.25×10^{-6} IN/IN/°C
	R.T. to 1,000°C (1832°F)	6.14	19·00 (EST'D.)

Fig. 24



COMPARISON OF 2250 LBS AND 18000 LBS THRUST ENGINES IN LARGE AIRCRAFT

Fig. 25