REPORT

on the

WHITTLE SYSTEM OF AIRCRAFT PROPULSION
(THEORETICAL STAGE)

by

M. L. Bramson, A. C. G. I., F. R. Ae. S.

Bush House
Aldwych
London, W. C. 2

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1. TERMS OF REFERENCE.

The purpose of this Report is to record the result of an independent step by step check of the theories, calculations and design proposals originated by Flt. Lieut. Whittle, and having for their object the achievement of practical stratospheric transport. No investigation of the patent situation has been attempted.

2. MATERIAL SUBMITTED BY FLT. LIEUT. WHITTLE.

The inventions and discoveries of Flt. Lieut. Whittle have not yet reached the experimental stage, and so the material available for investigation is, necessarily, confined to a reasoned statement of the principles involved, coupled with justifying aerodynamic and thermodynamic calculations and design proposals.

3. DESCRIPTION.

A. The Problem.

The desirability of stratospheric flight arises from the rapid decrease of air-resistance (drag) experienced at high altitudes, due in turn to the low air densities obtaining there. For example, the density at an altitude of 69,000 feet is only one-sixteenth of normal atmospheric density at sea level. The principal difficulty to be overcome for this purpose is the maintenance of power notwithstanding the rarification of the atmosphere available for combustion. This difficulty has, in a moderate measure, been overcome by supercharging aero-engines of orthodox type, but it can be shown that supercharging to the extent which would be necessary for maintenance of adequate power in the stratosphere would not be feasible on account of the power which would be consumed by such a supercharger even if the size and weight were not prohibitive.

Furthermore, even assuming that means were found to maintain adequate power at the altitude mentioned, the orthodox means, to wit the aircraft propeller, of applying that power effectively to the atmosphere for propulsive purposes would need to be of impracticable dimensions.

These, briefly, are the reasons underlying the search for some alternative mode of propulsion which shall both maintain and apply propulsive power with adequate efficiency at those altitudes where such power will produce the greatest speed, economy and range.

It has long been recognized that for this purpose some form of jet propulsion would be necessary. Many suggestions have been made to that end, mostly based on the use of some explosive as propellant, but none have been practical. They have failed to provide a solution in the main, either because they involved carrying in the aircraft not only the fuel but also the oxygen required for combustion, or because though theoretically capable of functioning in the stratosphere, the means proposed were incapable of raising the aircraft to the stratosphere.

B. Solution Proposed by Flt. Lieut. Whittle.

The system of propulsion proposed by Flt. Lieut. Whittle falls, and will be treated, under three headings: Aerodynamic Principles, Thermodynamic Principles and Engineering. The general scheme is as follows:

An aeroplane of the "cleanest" possible aerodynamic form is provided with a circular or annular forward orifice facing the air stream. (NOTE: In this report all dimensions, temperatures, pressures and other figures given will, unless otherwise stated, relate to the particular case of an aircraft and reaction engine designed to operate at an altitude of 69,000 feet and at a speed of 500 m.p.h.) This orifice communicates directly with what may be termed the engine room in which will exist a pressure exceeding that of the surrounding atmosphere by the pressure head corresponding to the kinetic energy of the air-stream meeting the said orifice. The air thus partly compressed is drawn into a centrifugal air compressor which delivers into a heat insulated combustion chamber. Oil fuel is admitted into this combustion chamber where it burns, thus raising the temperature of the air which is allowed to expand at constant pressure in the combustion chamber. The mixture of air and products of combustion then flows to the intake nozzle or nozzles of an impulse turbine so designed that the temperature and pressure drop is only sufficient to enable the turbine to drive the centrifugal compressor to which it is directly coupled. The energy contained in the air-stream issuing from the turbine exhaust is thereupon converted into kinetic energy in a propulsion nozzle which delivers a high velocity jet of air and combustion gases through an orifice in the tail of the aeroplane. The total forward thrust imparted to the aeroplane is equal to the force required to accelerate the mass of air flowing through the machine in unit time from rest to the velocity (absolute) of the propulsion jet.

AERODYNAMIC PRINCIPLES.

When an aeroplane is in steady horizontal flight the force of propulsion must be equal to the total air resistance or drag. For a very clean streamline aeroplane without excrescences such as undercarriages and the like, the drag may, under the most economical flight conditions, amount to one-eighteenth of the total weight of the aeroplane. Such an aeroplane might be designed to have its most economical speed at ground level at, say, 125 m.p.h. Under corresponding flight conditions the speed is, for practical purposes, inversely proportional to the square root of the air density. Therefore, at 69,000 feet, for example, where the relative density is 1/16, the corresponding speed of such an aircraft is $125 \times \sqrt{16} = 500$ m.p.h. At this speed drag will still be the same fraction (1/18) of the total weight of the aeroplane, and if therefore, a thrust of equal amount can be maintained the speed of 500 m.p.h. will be achieved. Any excess of thrust available between ground level and the operating altitude can be used for climbing in overcoming the component of gravity which lies along an inclined flight path.

THERMODYNAMIC PRINCIPLES.

The Whittle Reaction Engine is based upon a heat engine cycle of the "combustion at constant pressure" variety. The inventor has prepared pressure-volume diagrams and entropy diagrams for the particular case of his reaction engine which forms the subject of the critical discussion below. One set of diagrams represent conditions at 500 m.p.h. at the operating height of 69,000 feet, and the other set represents starting conditions at no air-speed and at sea level. These diagrams have been attached as Appendix No. I to this Report.

The thermodynamic cycle is as follows: Air at 220° C absolute and 0.702 lbs./sq. in. absolute pressure, (atmospheric conditions at 69,000 feet), is compressed adiabatically (due to the forward speed of the aircraft as mentioned above) to 245° C absolute and 1.04 lbs./sq. in. It is thereupon further compressed adiabatically in the centrifugal compressor to 6.76 lbs./sq. in. which gives a theoretical temperature increase to 420° C absolute. All losses in the compressor (assumed efficiency 80%) with the exception of bearing and radiation losses which are negligible, will be transformed into heat contained in the air so compressed. This produces a further temperature rise from 420° C to 464° C absolute. Heat is then added at constant pressure, (fuel oil is introduced and burnt in the combustion chamber) raising the temperature to 1,092° C absolute. Expansion now takes place in two stages: the first stage takes place in the turbine nozzles through which there is a pressure drop from 6.76 lbs./sq. in. to 2.3 lbs./sq. in., the corresponding theoretical temperature drop being from 1,092° C to 800° C absolute. During this expansion the gases accelerate from 300 ft./sec. up to a velocity of 2,500 ft./sec. They thereupon impinge on and pass through the turbine blades, to which, assuming a 75% turbine efficiency, they give up 75%

of their kinetic energy. The losses, amounting to 25% of the said kinetic energy, are essentially fluid friction losses, and are therefore transformed into heat which raises the temperature at the turbine exhaust from 800° C absolute to 873° C absolute. The second expansion stage takes place in the propulsion nozzle, the pressure drop being from 2.3 lbs./sq. in. absolute to 0.702 lbs./sq. in. absolute and the corresponding temperature drop from 873° C absolute to 623° C absolute. The result of this expansion is that the gases are accelerated to a velocity of 2,320 ft./sec. The thrust per pound of air per second flowing through the nozzle is given by the formula $\frac{V - u}{g}$, where V. is the velocity of the jet, u. is the speed of the aircraft in ft./sec. and g. is the acceleration due to gravity. In the particular case under consideration this thrust would therefore be $\frac{2,320 - 733}{32.2} = 49.3$ lbs. per pound of air per second. (NOTE: 500 m.p.h. = 733 ft./sec.)

Thus in the case of an aeroplane weighing 2,000 lbs. and requiring a thrust of 1/18 of that weight, i.e. 111 lbs., $\frac{111}{49.3} = 2.25$ lbs./sec. of air will be the necessary capacity of the reaction engine to provide the thrust required.

Sea Level Conditions. The interesting case of sea level conditions at no forward speed of the aircraft is dealt with in the inventor's second Pressure-Volume Diagram in Appendix I. The main differences are:

1. The initial temperature which is 288° C abs. instead of 220° C abs.

2. The smaller amount of heat added per lb. of air in order not to exceed the same maximum temperature of the cycle and the same blade temperature as that adopted for the high altitude conditions.

3. The very greatly increased throughput in lbs./sec. of the reaction engine due to the increased density of the atmosphere.

4. The fact that due to the aircraft being stationary, there is only one compression stage.

The net result of these changes in conditions is, it will be seen, that the thrust per pound of air per second is very slightly greater, namely 53.2 lbs. but the thrust due to the 36 lbs./sec. throughput is 36 x 1,915 lbs., which is nearly equal to the weight of the aircraft. (Should these figures actually be obtained it is clear that both acceleration and climb will be very rapid indeed.)

Efficiencies. The overall efficiency of a reaction engine of this type is the thrust horsepower divided by the input of heat energy in unit time. (This corresponds to the thermal efficiency of an aero engine multiplied by the propeller efficiency.)

In the particular case referred to, thermal efficiency (i.e. the kinetic energy given to the working fluid divided by the heat energy input) would be 48% giving an overall efficiency of 17.13%. For the sea level conditions, and assuming a flying speed of 125 m.p.h. the thermal efficiency would be 22.9% and the overall efficiency would be 4.5%.

ENGINEERING

1. The Power Unit. The Whittle Reaction Engine consists of a single-stage turbo-compressor directly coupled to and driven by a gas turbine of the pure impulse type. Taking the case of a unit capable of a throughput of 2.25 lbs. of air per second at 69,000 feet, the impellor diameter would be 19 inches and its speed would be 17,850 r.p.m. giving a linear tip speed

of 1,470 ft./sec. (The overall diameter of the compressor would be 43 inches.)

The compressor has a double inlet and its designed capacity would be 470 cu. ft./sec. giving an inlet velocity of 400 ft./sec. The turbine may, alternatively, consist of a double row velocity compounded impulse wheel or of two single row impulse wheels working in parallel. The latter arrangement is probably preferable as it permits direct coupling between the compressor and the turbine. (The two row turbine wheels would have to be geared down in relation to the compressor; this complication might, however, be balanced by the advantage of lower peripheral speed of the turbine wheels.) For efficiency, the linear speed of the single row turbine blades should be one half that of the gases issuing from the turbine nozzle (See Appendix III) which is 2,500 ft./sec. The turbine blade speed should therefore be 1,250 ft./sec. and the effective diameter of the turbine wheels 16.15 inches.

The turbine exhaust gases pass straight to the propulsion nozzle where, as already mentioned, the speed of the gases is accelerated to 2,320 ft./sec. The volume per pound of gas has at this point expanded to 591 cu. ft./lb. giving a total of 591 x 2.25 = 1,330 cu. ft./sec. in the particular case considered. This gives a propulsion nozzle outlet diameter of 10.25 inches.

2. <u>The Aeroplane.</u> The aeroplane consists of a fuselage of correct streamline form, the forward portion of which is a sealed air reservoir capable of withstanding an internal pressure of 15 lbs./sq. in. and containing the pilot, passengers

and controls. An annular opening facing the air-stream is formed between the circumference of this sealed portion and the monocoque shell of the rest of the fuselage. The total cross-sectional area of this annular opening need only be about 100 square inches (in the particular case considered), which, assuming 4'-6" to be the diameter of the sealed portion, gives a width of the annular opening of only 0.6 inch. (In actual practice the width of this opening would be made greater in order to make certain of getting the full necessary flow.)

A cantilever monoplane wing of 52 sq. ft. area would be fitted giving a wing loading of 19.3 lbs. sq./ft. As there is no propeller there is no need for large ground clearances during the landing and take-off, and the retractable under-carriage can be short. Furthermore, as the machine is designed and intended forflight at its most economical angle of incidence, the wings can be set at a larger angle of incidence in relation to the fuselage than is the present practice. Thus, even from the point of view of getting correct wing incidence for take-off and landing, a high undercarriage is not necessary. An auxiliary compressor would be fitted drawing air from the pressure side of the main compressor and delivering at normal atmospheric pressure into the sealed cabin.

4. CRITICAL DISCUSSION.

The following critical discussion is based entirely upon the particular case to which reference has already been made and which has been worked out by the inventor. The calculations have been checked. The data are as follows:

	Given:	
Aeroplane.	Weight	2000 lbs.
	Wing loading	19.3 lbs./sq. ft.
	Most economical speed (speed of minimum drag)	125 m.p.h. at ground level
	Minimum drag	111 lbs.
	Corresponding speed at 69,000 ft. altitude (where effective efficiency = 1/16 is 500 m.p.h.)	733 ft./sec.
Engine Thermal Cycle	Assumed compressor efficiency	80%
	Assumed turbine efficiency	75%
	Total theoretical temperature rise due to compression	200° C
	Maximum turbine blade temperature	527° C = 800° C abs.
	Actual temperature rise due to 1st stage of compression (pitot head)	25° C
	$\therefore$ Rise of temperature due to 2nd stage of compression (compressor)	175° C
	Actual temperature rise in compressor $\frac{175}{0.8}$	219° C
	Initial air temperature	220° C abs.
	Temperature after pitot head compression	245° C abs.
	Effective heat drop in turbine	219 units

(NOTE: The unit of heat chosen in these calculations is the pound calorie multiplied by the specific heat of air at constant pressure, or, in other words, the quantity of heat required to raise one pound of air one degree centigrade at constant pressure. The above figure of 219 units is derived from the stipulation that the turbine must be able to drive the compressor.)

Theoretical heat drop in turbine $\frac{219}{0.75}$	292 units
Deduced:	
Final compression temperature 245° C plus 219° C	464° C abs.
Maximum temperature of cycle 800° C plus 292° C	1092° C abs.
Overall temperature ratio of compression $\frac{420}{220}$	1.91
Temperature ratio of 1st stage (turbine) of expansion $\frac{1092}{800}$	1.366
$\therefore$ Temperature ratio of final expansion $\frac{1.91}{1.366}$	1.40
Temperature before final expansion 1092 - 219	873^0 C abs.
Temperature at end of final expansion $\frac{873}{1.40}$	623° C
$\therefore$ Useful heat drop is 873° C - 623° C	250 units

To obtain the effective output of the engine the heat equivalent of the pitot compression must be deducted.

∴ Effective output of engine = 250 - 25. . 225 units

Heat addition = 1092° C - 464° C . . . . . . 628 units

∴ Thermal efficiency =

$\frac{\text{effective heat drop}}{\text{heat addition}} = \frac{225}{628}$ 35.8%

The jet velocity resulting from a useful heat drop of 250 units is $146.7 \times \sqrt{250}$ . . . . 2320 ft./sec.

(NOTE: The constant 146.7 is derived from the mechanical equivalent of the pound calorie.)

Hence, net change of gas velocity produced = 2320 - 733. . . . . .1587 ft./sec.

∴ Propulsive thrust per pound of gas per second = $\frac{1587}{32.2}$ . . . . 49.3 lbs.

∴ Weight of air through-put required per second = $\frac{111}{49.3}$ . . . . 2.25 lbs.

Propulsive efficiency . . . . = 48%
Overall efficiency . . . . = 17.2%
Jet horse-power . . . . = 308 H.P.
The thrust horse-power . . . . = 148 H.P.

AERODYNAMICS

Flt. Lieut. Whittle shows, by the application of Professor Melville Jones' formulae for induced power and profile drag, that a well streamlined aeroplane with the proposed wing loading of 19.3 lbs./sq. ft. may, under conditions of minimum drag, be expected to have a lift/drag ratio of 21. He adopts, however, the figure 18, which more nearly corresponds to flight conditions giving maximum range.

There is no serious doubt that an aircraft of the type contemplated could be made to approach the ideal streamline aeroplane as closely as, for example, the modern glider, whose best lift/drag ratio has been known to attain the figure of 23 and over. Flt. Lieut. Whittle's figure for drag, and for speed at which such drag will be experienced at 69,000 feet, may, therefore, be accepted unreservedly.

THERMODYNAMICS

A Consideration of Basic Assumptions.

a) Compressor Efficiency. The compressor efficiency assumed of 80% is unusually high. There are published test results (Aeronautical Research Committee R. & M. 1336) showing adiabatic temperature efficiencies for a single phase centrifugal compressor up to 73%. Dr. A. Rateau, probably the greatest authority on exhaust driven turbo-compressors states, in an article in the Revue Generale des Sciences of the 15 January, 1930 and reproduced in the Genie Civil of the 15 February, 1930 as follows: (The subject of the article is the supercharging of Diesel Engines by means of exhaust driven turbo-compressors)

> . . . . . "On the other hand in designing the compressor exactly for the required throughput of air, efficiencies of 82% for the compressor and 78% for the turbine, or in other words, an overall efficiency of 64% can be counted on . . . . ."

The inventor supplies the following interesting information which he has obtained from the compressor experts of the British Thomson-Houston Co., Ltd. This company is said to have obtained compressor efficiencies of 76% and 82% on actual test. Moreover, they have, on the basis of their experience, established a non-dimensional figure of merit for centrifugal compressors according to which Flt. Lieut. Whittle's proposed compressor should compare favourably with their best existing examples.

The main features distinguishing the inventor's compressor from normal practice is the high pressure ratio obtained in a single stage and the high volumetric output. The former is almost a natural function of the peripheral impellor speed and should be realised. The latter is obtained mainly by the double intake arrangement and without adopting excessive intake velocities (400 ft./sec.) A blower built by British Thomson Houston for Messrs. Charles Nelson & Company had a maximum intake velocity well in excess of this figure.

The very large mass flow obtained through the double intake should tend towards increased efficiency since the fluid friction losses cannot increase proportionately.

In view of these considerations, I regard an 80% efficiency as a probability, but by no means a certainty. I do feel confident, however, that with skillful design a compressor efficiency between 70% and 75% will be obtained.

b) <u>Turbine Efficiency</u>. Referring again to the quotation from Dr. Rateau's paper, it will be seen that 73% is given as obtainable in practice. Having regard to the fact that the blade speed adopted approaches one half the gas speed at the turbine nozzle, (which is a condition for maximum efficiency), it is probable that 75% will be achieved.

c) <u>Temperature Rise Due to Compressor</u>. For practical purposes adiabatic compression may be assumed with negligible error. The theoretical temperature rise required is 175° C. For a compressor having a sufficient number of vanes to make the peripheral component of the gas speed discharging from the impellor equal to the peripheral speed of the impellor, the relationship between impellor speed and temperature raise is given by $U^2 = 32.2 \times 333 \times$ the temperature rise, from which can be derived the impellor speed required $U = 1380$ ft./sec. The inventor's figure is 1470 ft./sec. which is, therefore, in excess of the speed theoretically required. The actual temperature rise in the compressor, $\frac{175}{0.8} = 219^\circ$ C, is a measure of the actual power required per pound of throughput. Therefore, actual power required equals $\frac{2.25 \times 333 \times 219}{550} = 300$ H.P. For adiabatic compression, the pressure ratio equals the (temperature ratio)$^{3.5}$. Therefore, as the overall temperature ratio of compression is $\frac{420}{220} = 1.91$, the compression ratio will be $(1.91)^{3.5} = 9.6$ (this ratio, of course, includes the pitot compression). I see no reason why these temperatures and compression ratios should not be obtained in practice.

d) Turbine Blade Temperature. A turbine blade temperature of 800° C absolute (527° C) has been taken as the basis of the turbine design. Since the effective power of the turbine must be equal to the power absorbed by the compressor, it follows that the effective heat drop in the turbine must be equal to the actual temperature rise in the compressor which is 219° C. We therefore have actual heat drop in turbine nozzles $\frac{219}{0.75} = 292$ units. In order, therfore, that the blade temperature shall be 800° C absolute, the temperature at the beginning of the first expansion stage through the turbine must be 800 plus 292 = 1092° C absolute.

This, therefore, limits the heat addition per pound to 1092 - 464 = 628 units. Herein lies the justification for the assumption, or rather the stipulation, that the blade temperature shall not exceed 800° C absolute.

e) Thrust. All the further thermodynamic deductions enumerated on Page 10 hereof, follow directly from the fact that in a heat cycle of the type adopted, the overall compression ratio is equal to the overall expansion ratio. One or two points should be noted in this connection. Great emphasis is rightly laid by the inventor on the fact that almost the entire losses incurred in the turbine are transformed into heat in the gas stream. Whereas in all existing applications of a gas turbine such heat would be entirely lost, this is not the case in the reaction engine, since part of such heat is recovered in the form of additional kinetic energy in the jet. The inventor has made no allowance for fluid friction losses in the propulsion nozzle. These should, however, be very small. They should be provided for by a slight increase in the throughput capacity of the unit.

The performance of the reaction engine for ground level conditions, has been obtained on the same basic assumptions. The calculations for horizontal flight conditions are given in Appendix II. It will be seen that although the thermal efficiency is low at ground level, the thrust is exceedingly large in relation to the total weight of the aircraft.

ENGINEERING

The Reaction Engine. The peripheral speed adopted by the inventor for the compressor rotor is 1470 ft./sec. This is considerably in excess of existing practice. (I am informed by Ft. Lieut. Whittle that the British Thomson Houston Centrifugal

Compressor Design Department are aware of certain cases of speeds of 1250 ft./sec.) The speed now proposed involves an increase of stress, all other things being equal, of about 38%. Provided one of the modern high tensile steels is used, I believe that with careful design it will be possible to make an impellor capable of standing up to the peripheral speed proposed. Great care must be taken to avoid the risk of vibration of the impellor blade tips. Much depends upon the skill and care employed in the detail design, and one must attach great importance to the employment of all available expert advice for this purpose. Subject to the foregoing, the engineering features of the compressor should be trouble-free.

Combustion Chamber and Burners. These do not call for special comment and any minor problems arising should yield to ordinary skillful design.

Turbine. As already mentioned on Page 8, the turbine consists of two single-row impulse wheels working in parallel and, in fact, having a common shaft. It will probably be proved desirable to machine the two wheels and their shaft from one solid forging. The turbine blade speed of 1250 ft./sec. gives rise to a stress at the blade root of the order of 12.6 tons per square inch for a blade length of 1.33 inches. At the very reasonable blade temperature of 527° C adopted, this stress would give a "creep rate" of 2×10^{-7} inch per inch per hour if the steel used is Kayser Ellison 965, which creep must be allowed for in the design.

The turbine wheel rim will necessarily be considerably hotter than the rest of the wheel and will suffer tangential compression stresses caused by the centrifugal tension of the blade. The whole design of the turbine discs and blade root and their method of attachment is a very delicate and important matter, and should be submitted to experts. I do not expect trouble due to heat transmission from the rim to the turbine discs. If a small amount of cool air is allowed to enter the turbine casing near the shaft, it will flow outward whilst being "sheared" at an exceedingly high rate in the small clearance space between the turbine wheel and the casing. This will effectively prevent excessive temperatures from reaching the shaft and/or the bearings.

It is appropriate here to point out that should detailed consideration of the engineering design problems lead to unexpected difficulties, the alternative of adopting a two row, 2-stage impulse wheel is available and would lead to considerably reduced peripheral speed. It would, however, be at the expense of simplicity as such an arrangement would necessitate gearing between the turbine and the impellor shaft. In my view, it is worth while going to considerable lengths to avoid such gearing. (Likewise, in the case of the compressor, a 2-stage compression could be resorted to at the expense, however, of increased, but not prohibitive, weight and bulk.)

The design problems and difficulties to be overcome, in their probable order of importance, may be summarised as follows:

1. To make provisions for the combined heat and centrifugal stresses at the turbine blade roots.

2. The design and manufacture of a compressor rotor capable of withstanding the centrifugal and bending loads on the vanes.
3. To guard against turbine blade and compressor blade vibration.
4. Design of main shaft to avoid torsional vibration periods, and to resist gyroscopic couples.

I do not regard any of these problems as insurmountable, but I do consider it possible that they may not all be satisfactorily overcome in the first reaction engine produced.

Weight. The inventor estimates the weight of the complete reaction engine unit at 500 pounds. Unless or until actual designs are available it is impossible to form a reliable opinion on this estimate. This much may, however, be said. The working elements of the engine operate at extremely high velocities, a fact which tends towards decreased size and weight for a given power. The same tendency results from the fact that the working elements are purely rotary as distinct from reciprocating and rotary. Furthermore, the engine has no equivalent to cooling fins, water jackets or radiators, nor has it any airscrew. For these reasons it can be safely assumed that even the first complete engine will not so far exceed weight estimates as to render flight tests impossible or inconclusive.

The Stratospheric Aeroplane. Structurally, the proposed stratospheric aeroplane presents no new problem with the one exception of providing a hermetically sealed cabin with safe and satisfactory doors permitting the crew getting in and out. The problem of flying controls and engine controls to be operated from inside a sealed cabin without having numerous sources of air leakage can be simplified by operating all controls hydraulically. By tapping the pressure side of the compressor, an ample supply of air for breathing is available. It will be noticed that the compression pressure is very nearly one half atmosphere which is ample for most people. An auxiliary booster can, of course, be fitted.

DEGREE OF PERMISSIBLE ERROR IN FUNDAMENTAL ASSUMPTIONS.

An investigation has been made with the object of ascertaining the lowest compressor and turbine efficiencies at which stratospheric flight would be possible. The result of this investigation shows that even if both the compressor and the turbine efficiencies were each only 60%, then without raising the turbine blade temperature, a thermal efficiency of 14.2% would still be obtained, and an overall (thrust) efficiency of 9.1%. The principal disadvantages experienced in the event of such low efficiencies being obtained would be firstly, that the throughput capacity of the unit would have to be approximately doubled and secondly, that at ground level an adequate thrust would only be obtainable by permitting an increase of the turbine blade temperature.

5. SUMMARY.

The stratospheric aeroplane and the Whittle Reaction Engine have been described in principle and with particular reference to the case of an aeroplane of 2,000 pounds all-up weight capable of a speed of 500 m.p.h. at 69,000 feet altitude.

The inventor's calculations for the aforesaid particular case have been checked and the results are discussed critically from the aerodynamic, thermodynamic and engineering points of view. It is shown that, although the inventor in some respects goes beyond existing experience, he does not appear to go beyond the temperatures, stresses and speeds that are possible with modern technique and materials.

The author's figures are based on a compressor efficiency of 80% and a turbine efficiency of 75%, the attainment of which is considered probable. Nevertheless, it is considered that if these efficiencies are both 60%, stratosperic flight with the Whittle Reaction Engine would still be possible.

6. CONCLUSIONS.

1. Flt. Lieut. Whittle's theoretical calculations and deductions therefrom are substantially correct.

2. His fundamental discovery is that the gas turbine though very inefficient as a prime mover when power is required in the form of shaft horse-power, can be adequately efficient as an auxiliary to the production of a power jet.

3. Should the discovery be successfully put into practice, the points of superiority over existing aeroplanes would be:

a) Economical speeds of 500 m.p.h. and over.
b) Probable ranges of 5,000 miles and over.
c) The use of non-volatile fuel.
d) Freedom from noises and vibration.

4. The proposed development though necessarily speculative as regards time and money required, is so important that it should, if possible, be undertaken.

7. RECOMMENDATIONS.

The "Brief Outline of Development Procedure" appended to this Report (Appendix III) has, by request, been prepared by the inventor.

I recommend the adoption of the procedure therein proposed with the proviso that all designs should be submitted to an independent authority on turbine and compressor design before actual construction is undertaken.

8th October, 1935.

APPENDIX I

PRESSURE-VOLUME AND ENTROPY DIAGRAMS

APPENDIX II

FLIGHT AT SEA LEVEL

APPENDIX II

FLIGHT AT SEA LEVEL

Air temperature 15° C	288° C abs.
Aircraft speed 125 MPH	182 ft./sec.
Assumed compressor efficiency	80%
Assumed turbine efficiency	75%
Theoretical temperature rise in compressor	175° C
Actual temperature rise in compressor $\frac{175}{0.8}$	219° C
Temperature rise due to pitot compression	1.5° C
Total temperature rise due to compression 219 - 1.5	220.5° C
Actual compression temperature 220.5 - 288...	508.5° C
Temperature ratio $\frac{175 - 1.5 - 288}{288}$	1.612
Temperature drop in turbine $\frac{219}{0.75}$	292° C
Blade temperature (stipulated)	800° C abs.
Maximum temperature cycle = 800° C - 292° C .	1092° C abs.
Heat addition 1092 - 508.5	583.5 units
Temperature ratio in turbine $\frac{1092}{800}$	1.365
Temperature ratio of final expansion $\frac{1.612}{1.365}$	1.181
Exhaust temperature of turbine 1092 - 219	873° C abs.
Final temperature $\frac{873^\circ \text{ C}}{1.181}$	739° C abs.
Final temperature drop 873° - 739°	134° C
Jet speed = $146.7 \times \sqrt{134}$	1695 ft./sec.
Thrust = $\frac{1695 - 182}{32.2}$	47.0 lbs. per lb. of air
Capacity of compressor = 16 x 2.25	36 lbs./sec.
Total thrust = 36 x 47	1690 lbs.
Thermal efficiency = $\frac{134}{583.5}$	22.9%
Propulsive efficiency	19.4%
Overall efficiency	4.45%
Jet horse-power	2880 HP
Thrust horse-power	561 HP

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APPENDIX III

BRIEF OUTLINE OF DEVELOPMENT PROCEDURE

HIGH ALTITUDE ENGINE