Bram

This is an uncorrected proof. WEPJ and the Journal are correcting it. As far as I can judge it will appear in the February or March issue. You will receive 5 complimentary copies. My compliments on your wonderful Report.

Dec 2. 1969 Lance

(Steps have been taken on being to alert the relevant Bram.)

Preface

WE consider the publication of the Report to be of importance for four reasons:-

(1) It was on the basis of Mr. Mogens Louis Bramson's favourable judgement, as Consulting Engineer, formed against much adverse expert opinion, that the development of the jet engine was originally financed and organised. The Report, which was directed to the original Whittle proposals and the analyses which he had made, was commissioned by one of the signatories (L.L.W.) on behalf of Falk and Partners (O. T. Falk and Sir Maurice Bonham Carter). Together with them, Whittle's friends Messrs. Williams and Tinling adopted the Bramsom Report, and so Power Jets Limited came into being.

(2) The Report, as will be seen, is a model of cleai and and consistent writing and as such deserves to be studied by all technical people whose duties involve reporting. It is remarkable in content, and exemplary in style.

(3) Mr. Bramson, has never, in our opinion, received the credit due to him as one of the constructive early proponents: he remained as an actively participating Consultant to the project for several years.

(4) It has been insufficiently recognised that the Whittle project essentially depended on the marriage of a jet engine with a new subgenus of airframe, and that the boldness and completeness of the total concept (as appreciated by Bramson) went beyond the mere proposal of using a gas turbine to produce a propulsive jet. The third signatory (W.E.P.J.) spent many hours with Frank Whittle (whose Patent Agent he was) discussing the inventive features thus involved, such as boundary-layer control, cabin-pressurisation, by-pass-engine feasibility, and the then revolutionary idea of a squat undercarriage.

It is with pleasure that the three signatories below, all intimately connected with the earliest phase of jet development, have submitted the Report for publication.

....
FRANK WHITTLE (*Hon. Fellow*)

....
LANCELOT L. WHYTE LLW

....
W. E. P. JOHNSON (*Fellow*)

REPORT ON THE WHITTLE SYSTEM OF AIRCRAFT PROPULSION (THEORETICAL STAGE)—OCTOBER 1935

M. L. BRAMSON, ACGI, FRAeS

1. TERMS OF REFERENCE

The purpose of this Report is to record the result of an independent stop by stop check of the theories, calculations and design proposals originated by Flt. Lt. 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. LT. WHITTLE

The inventions and discoveries of Flt. Lt. 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 ft 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.

Those, 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 recognised 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. Lt. Whittle

The system of propulsion proposed by Flt. Lt. 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 ft and at a speed of 500 mph). 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 airstream 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 relating 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 airstream 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 mph. Under corresponding flight conditions the speed is, for practical purposes, inversely proportional to the square root of the air density. Therefore, at 69 000 ft, for example, where the relative density is 1/16, the corresponding speed of such an aircraft is $125 \times 1/16 = 500$ mph. 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 mph 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 mph at the operating height of 69 000 ft, and the other set represents starting conditions at no airspeed 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 lb/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 lb/sq in. It is thereupon further compressed adiabatically in the centrifugal compressor to 6·76 lb/sq in which gives a theoretical temperature increase to 420°C abs. All losses in the compressor (assumed efficiency 80 per cent) 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 1092 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 lb/sq in to 2·3 lb/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/s up to a velocity of 2500 ft/s. They thereupon impinge on and pass through the turbine blades, to which, assuming a 75 per cent turbine efficiency, they give up 75 per cent of their kinetic energy. The losses, amounting to 25 per cent 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 875°C absolute.

The second expansion stage takes place in the propulsion nozzle, the presure drop being from 2·3 lb/sq in absolute to 0·702 lb/sq in abs and the corresponding temopreature drop from 873° abs to 623°abs. The result of this expansion is that the gases are accelerated to a velocity of 2230 ft/s. The thrust per lb of air per second flowing through the nozzle is given by the formula $V—u/g$, where V is the velocity of the jet, u is the speed of the aircraft in ft/s., and g is the accelera-tion due to gravity. In the particular case under consideration this thrust would therefore be $2\,320—77/32 \cdot 2=49 \cdot 3$ lb. per lb. of air per second. (*Note.* 500 mph = 733 ft/sec).

Thus in the case of an aeroplane weighin 2 000 lb and requiring a thrust of 1/18 of that weight, *i.e.* 111 lb, $111/49 \cdot 3=2 \cdot 25$ lb/s of air will be the necessary capacity of the reaction engine to provide the thrust required.

The interesting case of non-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° abs.
2. The smaller amount of heat added per lb of air not to exceed the same maximum temperature of the cycle and of the cycle and the same blade temperature as that adopted for the high altitude conditions.
3. The greatly increased throughput in lb/s 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 results of these changes in conditions is, it will be seen, that the *thrust per lb of air* per second is slightly greater, namely 53·2 lb but the thrust due to the 36 lb's throughout is $36 \times 53 \cdot 2 = 1915$ lb 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.)

The overall efficiency of a reaction engine of this type is the thrust horse-power 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.* kinetic energy given to the working fluid divided by the heat energy input) would be 48 per cent giving an overall efficiency 17·13 per cent. For the sea-level conditions, and assuming a flying speed of 125 mph the thermal efficiency would be 22·9 per cent and the overall efficiency would be 4·5 per cent.

Engineering

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 lb of air per second at 69 000 ft, the impellor diameter would be 19 in. and its speed would be 17 850 rpm giving a linear tip speed of 1470 ft/s. (The overall diameter of the compressor would be 43 in.)

The compressor has a double inlet and its designed capacity would be 470 cu ft/s giving an inlet velocity of 400 ft/s. 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 2500 ft/s. The turbine blade speed should therefore be 1250 ft/s and the effective diameter of the turbine wheels 16·15 inches.

The turbine exhaust bases pass straight to the propulsion nozzle where, as already mentioned, the speed of the gases is accelerated to 2 320 ft/s. The volume per lb of gas has at this point expanded to 591 cu ft/lb giving a total of 591 × 2·25 = 1330 cu ft/s in the particular case considered. This gives a propulsion nozzle outlet diameter of 10·25 in.

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 lb/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 sq in (in the particular case considered), which, assuming 4 ft 6 in to be the diameter of the sealed portion, gives a width of the annular opening of only 0·6 in. (In actual practice the width of this opening would be made greater 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 lb/sq ft. As there is no propeller there is no need for large ground clearances during the landing and take-off, and the retractable undercarriage can be short. Furthermore, as the machine is designed and intended for flight 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 atmosphere pressure into the sealed cabin.

The inventor claims that by fitting a correctly designed sleeve of venturi shape over the propulsion nozzle, an increased propulsion efficiency can be obtained, but any such possible advantage has been ignored in his calculations.

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 lb
	Wing loading	19·3 lb/sq ft
	Most economical speed (speed of minimum drag)	125 mph at ground level
	Minimum drag	111 lb
	Corresponding speed at 69 000 ft altitude (where effective efficiency = 1/16 is 500 mph)	733 ft/s
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 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.)

(*Note* 2. The figure of 219 units is derived from the stipulation that the turbine must be able to drive the compressor.)

Theoretical heat drop in turbine 219/0·75	292 units

Deduced

Final compression temperature 245°C + 219°C	464°C abs.
Maximum temperature of cycle 800°C + 292°C	1092°C abs.
Overall temperature ratio of compression—420/220	1·91
Temperature ratio of first stage (turbine) of expansion 1092/800	1·366
Therefore temperature ratio of final expansion 1·91/1·366	1·40
Temperature before final expansion 1092—219 = 873°C abs.	
Temperature at end of final expansion 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	
Therefore effective output of engine = 250—25	225 units
Heat addition = 1092°C—464°C = 628 units	
Therefore thermal efficiency = effective heat drop/heat addition = 225/628 = 35·8 per cent	
The jet velocity resulting from a useful heat drop of 250 units is 146·7 × /250 = 2320 ft/s	

(*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/s
Therefore propulsive thrust per pound of gas per second = 1587/32·2 = 49·3 lb
Therefore weight of air throughput required per second = 111/49·3 = 2·25 lb
Propulsive efficiency = 48 per cent
Overall efficiency = 17·2 per cent
Jet horse-power = 308 hp
The thrust horse-power = 148 hp

AERODYNAMICS

Flt. Lt. 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 lb/sq ft may, under conditions of *minimum drag*, be expected to have a lift/drag ratio of 21. He adopts, however, the figure of 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. Lt. Whittle's figure for drag, and for speed at which such drag will be experienced at 59 000 ft may, therefore, be accepted unreservedly.

THERMODYNAMICS

A Consideration of Basic Assumptions

(*a*) *Compressor Efficiency*. The compressor efficiency assumed of 80 per cent is unusually high. There are published test results (ARCR&M 1336) showing adiabatic temperature efficiencies for a single phase centrifugal compressor up to 73 per cent. Dr. A Rateau, probably the greatest authority on exhaust driven turbo-compressors, state, in an article in the Revue Géndrale des Sciences of the 15th January 1930 and reproduced in the Génie Civil of the 15th 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 per cent for the compressor and 78 per cent for the turbine, or in other words, an overall efficiency of 64 per cent 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 per cent and 82 per cent 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. Lt. 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 impeller speed and should be realised. The latter is obtained mainly by the double intake arrangement and without adopting excessive intake velocities 400 ft/s. (A blower built by British Thomson-Houston for Messrs Charles Nelson and Company had a maximum intake velocity well in excess of this figure).

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

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

(*b*) *Turbine efficiency*. Referring again to the quotation from Dr. Rateau's paper, it will be seen that 73 per cent 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 per cent will be achieved.

(*c*) *Temperature Rise due to Compressor*. 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 rise is given by $U^2 = 32\cdot 2 \times 333 \times$ the temperature rise, from which can be derived the inpellor speed required $U = 1380$ ft/s. The inventor's figure is 1470 ft/s which is, therefore, in excess of the speed theoretically required. The actual temperature rise in the compressor $175/0\cdot 8 = 219$°C, is a measure of the actual power required per pound of throughput. Therefore, actual power required equals $2\cdot 25 \times 333 \times 219/550 = 300$ hp. For adiabatic compression, the pressure ratio equals the (temperature ratio)$^{3\cdot 5}$. Therefore, as the overall temperature ratio of compression is $420/220 = 1\cdot 91$, the compression ratio will be $(1\cdot 91)^{3\cdot 5} = 9\cdot 6$ (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 (537°C) has been taken as the basic 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 $219/0\cdot 75 = 292$ units. So that the blade temperature shall be 800°C abs, the temperature at the beginning of the first expansion stage through the turbine must be $800 + 292 = 1092$°C abs. 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 abs.

(*e*) *Thrust*. All the further thermodynamic deductions enumerated previously, follow directly from the fact that in a heat cycle of the type adopted, the overall compression ratio is equal to the overall expansion ration. 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 accumptions. 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/s. This is considerably in excess of existing practice. (I am informed by Flt. Lt. Whittle that the British Thomson-Houston Centrifugal Compressor Desing Departments are aware of certain cases of speeds of 1250 ft/s.) The speed now proposed involves an increase of stress, all other things being equal, of about 38 per cent. Provided one of the modern high tensile steels are 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 for this purpose of all available expert advice. 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 skilful design.

Turbine

As already mentioned previously, 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/s gives rise to a stress at the blade root of about 12·6 tons per sq in, for a blade length of 1·33 in. At the very reasonable blade temperature of 527°C adopted, this stress would give a "creep rate" of $2\times10^{[illegible]}$ 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 in addition to the 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 transmision from the rim to the turbine discs. If a small amount cool air is allowed to enter the turbine casing near the shaft, it will flow outward while 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 worthwhile 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 provision 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 shafts 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 as 500 lb. Unless or until 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 as 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 areoplane 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 nearly ½ 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 compressors and the turbine efficiencies were each only 60 per cent, then without raising the turbine blade temperature, a thermal efficiency of 14·2 per cent would still be obtained, and an overall (thrust) efficiency of 9·1 per cent. 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 2000 lb all-up-weight capable of a speed of 500 mph at 69 000 ft 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 per cent and a turbine efficiency of 75 per cent, the attainment of which is considered probable. Nevertheless, it is considered that if these efficiencies are both 60 per cent, stratospheric flight with the Whittle Reaction Engine would still be possible.

6. CONCLUSIONS

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

2. His fundamental discovery is that the gas turbine although very inefficient as a prime mover when power is required in the form of shat 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 mph and over.
(*b*) Probable ranges of 5000 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 therin proposed with the proviso that all designs should be submitted to an independent authority on turbine and compressor design before actual construction is undertaken.

M. L. BRAMSON
8th October, 1935

Appendices I and III cannot be located. The Society would be grateful to anyone who could give any indication of them.

APPENDIX I

Pressure-Volume and Entropy Diagrams

APPENDIX II

Flight at Sea Level

Air temperature 15°C	288°C abs.
Aircraft speed 125 mph	182 ft/s
Assumed compressor efficiency	80 per cent
Assumed turbine efficiency	78 per cent
Theoretical temperature rise in compressor	175°C
Actual temperature rise in compressor 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 175—1·5—288/288 ...	1·612
Temperature drop in turbine 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 1092/800 ...	1·365
Temperature ratio of final expansion 1·612/1·365	1·181
Exhaust temperature of turbine 1092—219	873°C abs.
Final temperature 973/1·181°C	739°C abs.
Final temperature drop 873—739°C ...	134°C
Therefore jet speed = 146·7 = /134 ...	1695 ft/s
Thrust = 1695—182/32·2	47·0 lb per lb of air
Capacity of compressor = 16 × 2·25 ...	36 lb/s
Total thrust = 38 × 47	1690 lb
Thermal efficiency = 134/583·5	22·9 per cent
Propulsive efficiency	19·4 per cent
Overall efficiency	4·45 per cent
Jet horse-power	2880 hp
Thrust horse-power	561 hp

APPENDIX III

Brief Outline of Development Procedure High Altitude Engine

BACKGROUND COMMENTS BY THE REPORTING ENGINEER 33 YEARS LATER

In 1935 my practice as a consulting engineer was conducted from an office in Bush House, London, and Flt. Lt. Frank Whittle appeared there one day. He wanted financing for the development of a system of jet propulsion of aircraft, which he had invented. About two years earlier he had submitted it to the British Air Ministry, who had turned it down.

His material consisted solely of thermodynamic and aerodynamic calculations and diagrams; there were no engineering designs.

He was a bright, confident, young officer-pilot in the Royal Air Force, who seemed to know what he was talking about. This impression was somewhat qualified by the eyebrow-raising improbability of his basic thesis that aeroplanes could be made to fly without propellers. Moreover, my own retention of thermodynamic theory was rusty, which made me distrust my own judgement. Nevertheless, or perhaps for this reason, as well as because of my general favourable impression of Whittle, I decided to study his theories and proposals thoroughly. This took two weeks.

At the end of that period I got quite excited. First because of the insight, clarity and accuracy of his presentation and calculations; secondly because my scepticism of any project based on internal combustion turbines (which had hitherto resisted all practical development efforts) disappeared when I realised that here, for the first time, was an application where maximum energy was needed in the turbine exhaust, instead of in the shaft. This was, of course, the reverse of all past objectives for such turbines. And, thirdly, because of the dramatic advance in aviation technology implicit in Whittle's theories.

I suddenly felt "This *must* be done!", (which meant 'financed'). A survey of my clients produced a London firm of investment bankers, O. T. Falk and Co., whom I approached. They sent along to see me a man who turned out to be an astonishing professional hybrid, to wit: A financial expert *and* a theoretical physicist! His name was Lancelot L. Whyte, and thanks to his comprehension and sense of perspective, a favorable decision was made. There was, however, one proviso: An independent engineer's report must be produced and must be conclusive.

When this decision was conveyed to Whittle he said, in effect, "Very well, but Bramson must be that engineer, for it is premature and potentially dangerous (he meant nationally) to spread detailed knowledge of this discovery any father".

So, instead of my initial role of capital-raising intermediary (entitled to some minor participation) I had, for the good of the cause, to become an ethically-independent (*i.e.* non-participating) reporting engineer. It pleases me to remember that I did so enthusiastically.

My report was indeed conclusive. And three years later the first Whittle jet engine was tested in Rugby at the British Thomson-Houston Turbine Factory; and the first jet powered aircraft flew in 1941.

Re-reading the report after all these years, I find there are only minor points, of emphasis rather than of substance, that would need amendment. But any temptation to feel smug about that is immediately squelched by one's immense admiration for the originator of one of the most striking and consequential technological revolutions of our time.

M. L. Bramson

12th November 1968