FUNDAMENTAL DESIGN PROBLEMS OF AIRCRAFT WITH BOUNDARY LAYER CONTROL FOR MAINTAINING LAMINAR FLOW

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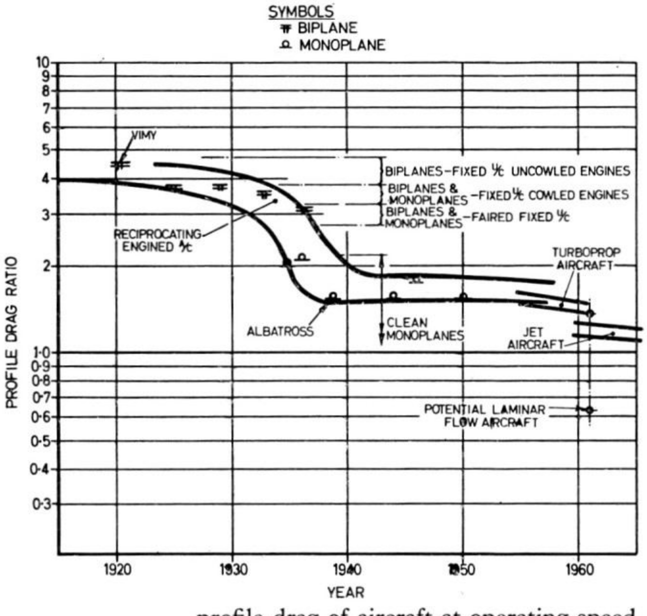
INTRODUCTION

"THERE is more to life than merely increasing its pace", said Gandhi. Equally there is more to flying than merely the increase of flight Mach numbers.

This applies particularly to civil aviation where economy, safety and reliability are paramount in contrast to military aviation where absolute performance regardless of cost is the predominant need.

The outstanding factor in the improvement of commercial aircraft has been the reduction of drag in relation to the payload due to the improvement of lift-to-drag ratio.



$$\text{Profile drag ratio} = \frac{\text{profile drag of aircraft at operating speed}}{\text{turbulent skin friction drag of flat plate at same speed}}$$

Flat plate area = Gross wetted area of A/C
Flat plate chord = A/C mean wing chord

FIG. 1. Historical survey of profile drag reduction.

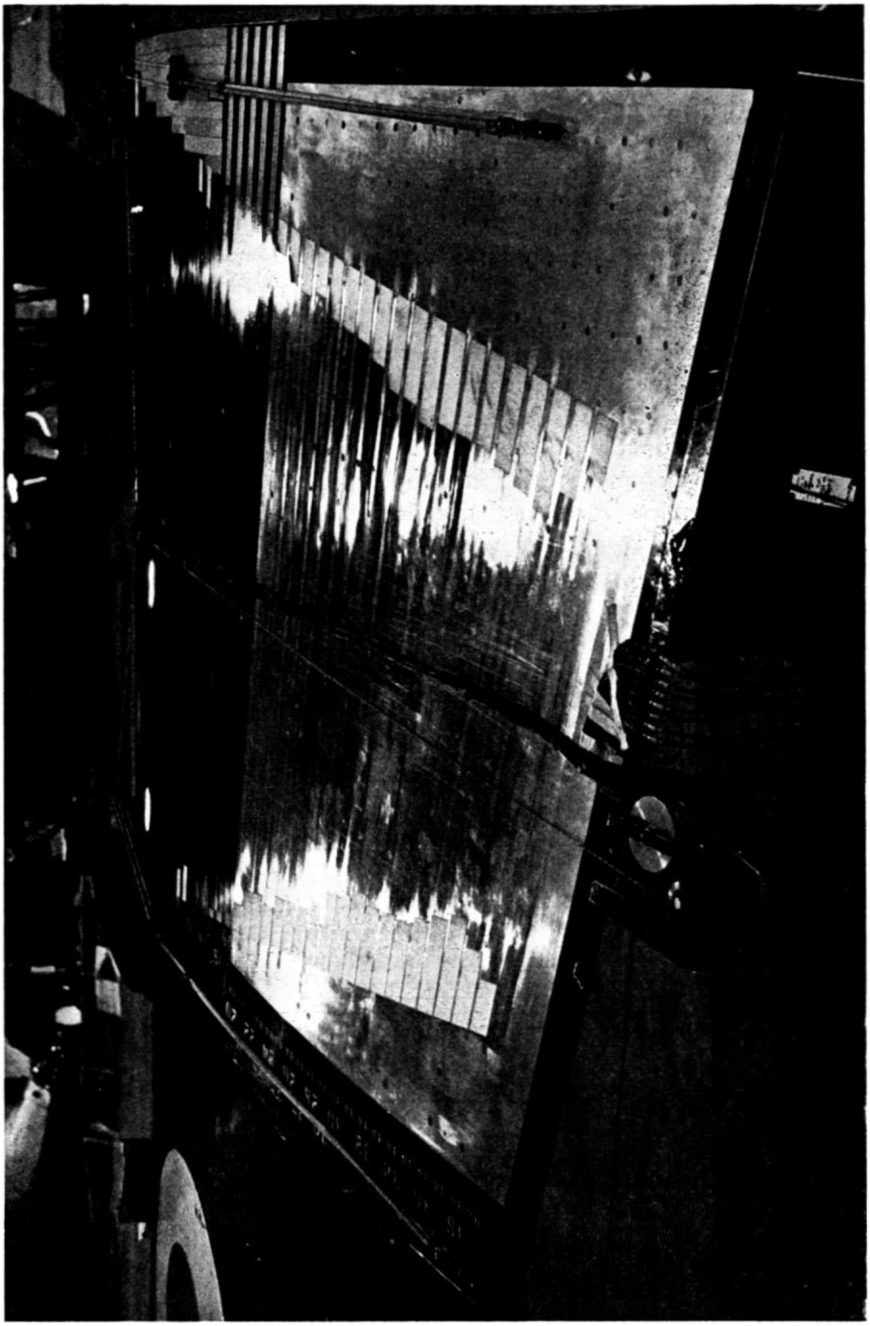


FIG. 2. Glove with Handley Page type suction surface on a VAMPIRE. (A pitot comb for measuring boundary layer profiles in flight is shown *in situ*. The comb can be moved, during flight, in chordwise direction.)

Figure 1 gives an historical survey showing the relationship over the years of the profile drag of subsonic aircraft at their operating speed to that of the turbulent skin friction drag of a flat plate of the same wetted surface and the same average Reynolds number. The effect of various steps in the process of cleaning up the design is also indicated[1].

It is significant that since about 1938–1940 no major advance in reducing profile drag has been made, although, with the advent of jet propulsion and thanks to the elimination of cooling drag the ideal set by Sir Melvill Jones in his classical paper "A Streamline Aeroplane"[2] has almost been reached.

Achievement of full chord laminar flow in flight and the corresponding powerful reduction of skin friction drag constitute a major aerodynamic break through of the same order as that due to the replacement of biplanes with fixed undercarriages by the clean cantilever monoplane.

1. THE STATE OF THE ART

Boundary Layer Research in Flight 1952–1958

During the last six years practical and theoretical research, especially flight research, has been performed mainly by three groups in two countries: in Great Britain by the Handley Page Research Department and the Royal Aircraft Establishment[3], continued later by Dr. Head at Cambridge University. Wind tunnel research work was carried out at the National Physical Laboratory (N.P.L.) and at R.A.E.

In the United States flight research on jet powered aircraft has been carried out by the group led by Dr. W. Pfenninger at Northrop Aircraft Inc.[4]. Earlier flight research work by a team led by Professor A. Raspet of the Mississippi State College, had already in 1951 established the possibility of maintaining full chord laminar flow on the wing of a sailplane.

For flight research Vampires were used in England, and in the United States a F.94 was modified for this purpose. The test equipment consisted in both countries of a glove built on one wing (Fig. 2). The surface of the glove could be sucked by means of a pump driven by an air turbine with the necessary compressed air tapped off the engine compressors (Fig. 3). The first successful demonstration of full-chord laminar flow on a jet aircraft took place in 1953 at the R.A.E.

It is estimated that since then over 200 flying hours with laminar flow have been achieved in Great Britain and the U.S.A.

The most important conclusions derived from this flight research work are summarized in Appendix I.

From the point of view of practical application of boundary layer control for low drag the outstanding achievements were:

(a) the development of engineering solutions for suction surfaces;

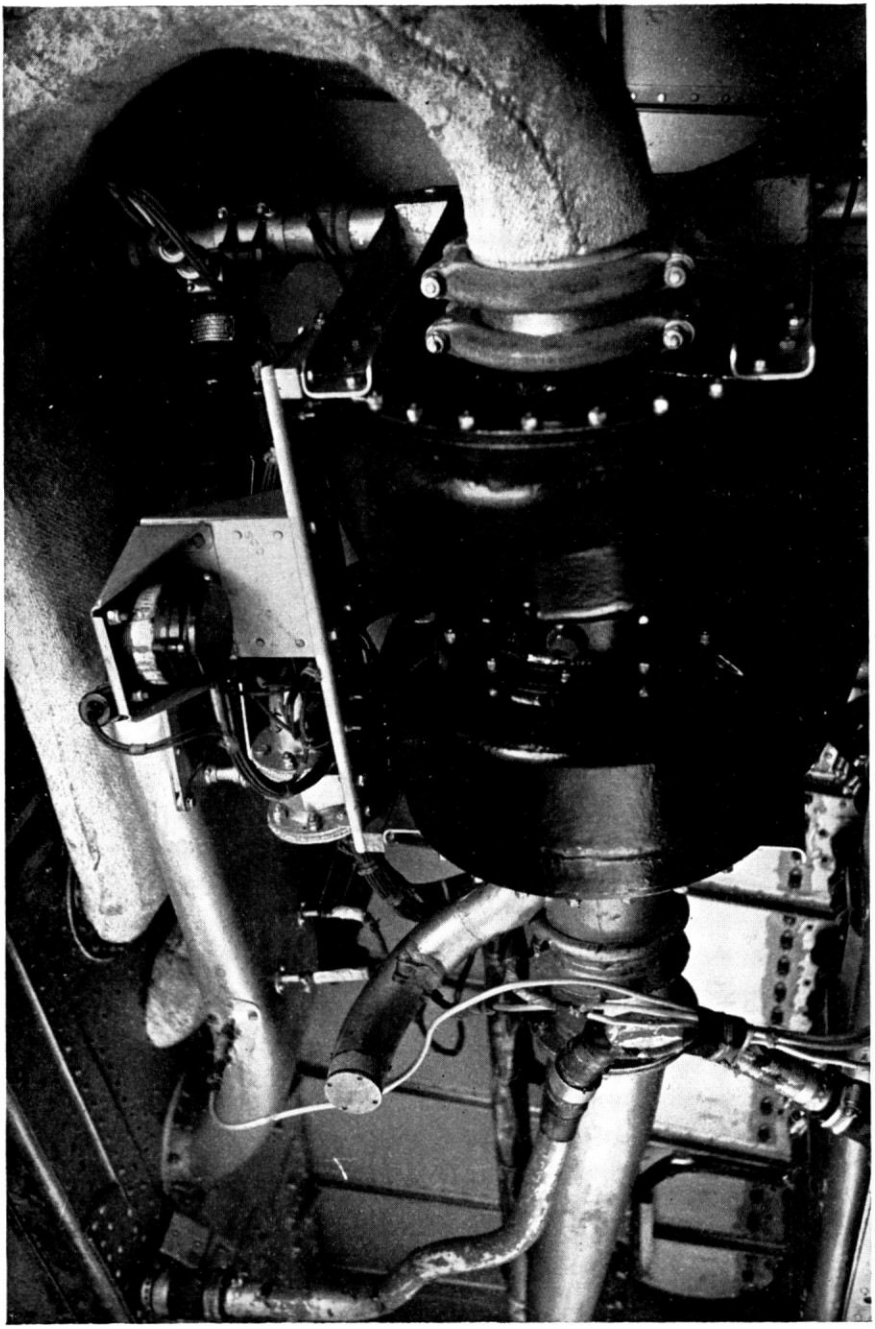


FIG. 3. Suction plant installation developed by Handley Page Ltd., in co-operation with Sir George Godfrey Ltd., installed in inner tank bay of VAMPIRE (seen from below). The air turbine is fed by the compressor and drives a centrifugal pump.

(b) the discovery and subsequent theoretical and experimental investigations of instability caused by crossflow;
(c) the insight gained into roughness effects as function of unit Reynolds number, i.e. Reynolds number per unit length.

Crossflow

Let us first consider crossflow.

In 1952 Gray of R.A.E. discovered, during flight experiments, that sweepback could cause transition very near the leading edge through the development of secondary flow in the laminar boundary layer. If this secondary flow reached a sufficient magnitude streamwise striations were formed which broke down rapidly causing transition at the leading edge.

Crossflow is created by a pressure gradient which causes curvature of the streamline in a plane parallel to the surface. Flow near the surface is turned more sharply than the outer flow and the boundary layer profile becomes twisted (Fig. 4). The profile can be resolved into two component

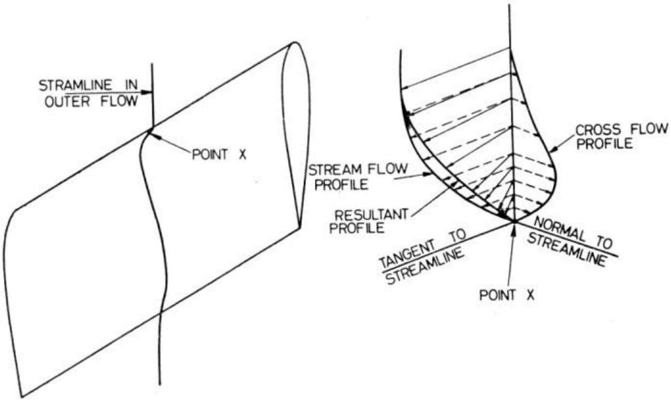


FIG. 4. Twisted profile in three-dimensional boundary layer flow.

profiles, one in the direction of flow at a large distance from the surface where viscous effects are unimportant, and the other perpendicular to the outer flow. The first component resembles a two-dimensional profile and is called the streamwise profile; the second component is called the crossflow profile; its shape is characteristically different and always has a point of inflection. Owen and Randall[5] developed a criterion for this type of instability based on the local thickness of boundary layer and maximum value of crossflow velocity. Stuart[6] derived the disturbance equations of the laminar boundary layer with crossflow.

It is only recently that satisfactory methods have been evolved for calculating the laminar boundary layer with suction in the case of swept wings where three-dimensional effects predominate.

A method by Lindfield and Pinsent of the Handley Page Research Department gives crossflow velocity profiles with considerable accuracy and, used in conjunction with a suitable stability criterion, makes it possible to calculate the necessary distribution of suction to maintain laminar flow over a swept wing. Calculations based on this method have shown that for thin wings the additional suction requirements made necessary by sweep are not excessive. Thus the potential field of application of boundary layer control for low drag is greatly widened. The theoretical basis of this method is described in greater detail in Appendix II.

Surface Roughness

It has long been realized that the great sensitivity to surface roughness presented one of the greatest practical difficulties to laminarization.

In view of this some recent investigations made in the Langley low turbulence tunnel[(7)] are of great significance and importance and confirm certain flight observations*. The object of these tests was to determine the effect of size and location of a sandpaper type of roughness on the Reynolds number for transition.

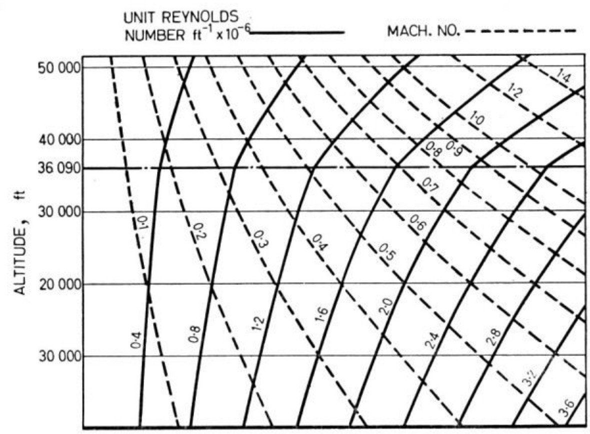


FIG. 5. Mach and corresponding unit Reynolds number U/ν ft^{-1} as function of altitude.

It was found that with this type of roughness turbulent spots began to appear immediately behind the roughness when the product of unit Reynolds number, that is Reynolds number per foot chord, and roughness height exceeded a value of $R_K = 680$.

The importance of the unit Reynolds number becomes clear in contrast to the more conventional Reynolds number R_c based on chord length. In Fig. 5, values of unit Reynolds number for various flight Mach numbers are plotted as function of height. (For example: at a height of 50,000 ft and flying at a Mach number of 1·0 unit Reynolds number is only $1{\cdot}2 \times 10^6$ which corresponds to $M = 0{\cdot}17$ at sea-level.)

* It has been observed that in spite of turbulent wakes, caused by insects which had impinged during take-off on the nose of the glove, laminar flow was re-established at greater altitudes. The original wakes had become visible since the surface of the wing nose had been sprayed before taking-off with a solution of naphthalene in petrol ether.

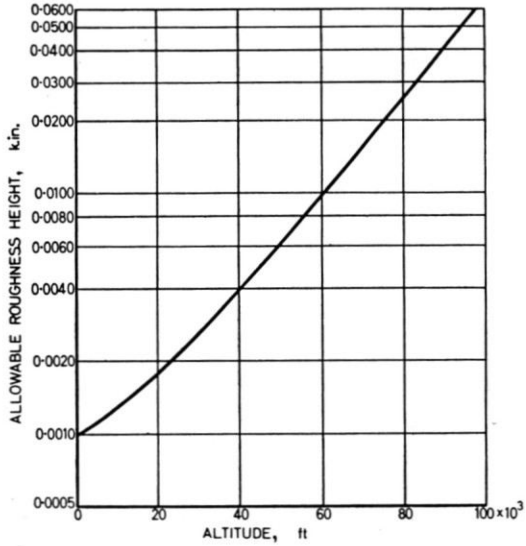


FIG. 6. Allowable roughness height with sandpaper type of roughness as function of altitude for $M=1$. (Critical Roughness Reynolds number $R_K=680$.) (From N.A.C.A. T.N. 3858.)

The graph, Fig. 6, translates the roughness criterion into more easily appreciated terms. The critical size of roughness for an assumed freestream Mach No. 1 has been computed as a function of height by using N.A.C.A. standard atmosphere. At sea-level the critical size is about 0·001 in. This increases to about 0·002 in. at 20,000 ft, and 0·01 in. at about 60,000 ft.

2. ENGINEERING CONSIDERATIONS

The Integrated Powerplant

Reduction of fuel consumption—for a given heat content per pound of fuel—can result from drag reduction or from improved overall efficiency with which heat is converted into thrust work. In either case, some penalty will usually be incurred in the form of additional fixed weight. To achieve a net gain, this weight penalty has to be appreciably less than the consequent reduction of fuel weight. The benefit will increase with the range to be flown, and this means that, generally, for longer ranges more refinement is worthwhile.

A large proportion of the total drag of an aircraft is associated with the wake caused by the growth of boundary layer on its exterior surfaces. The limit of what can be achieved by "streamlining" and the suppression of excrescences has long been approached as was shown in Fig. 1. Further improvements in this vein could only be achieved by reduction of the exterior surface area. The fuselage cannot be improved in this respect, its size being determined mainly by the density of the payload. As far as reduction of wing and tail area goes the limit compatible with take-off and landing performance has almost been reached on modern jet airliners.

Boundary layer control, by maintaining extensive regions of laminar flow with consequent small wakes and, at the same time, small suction quantities, makes possible a new step towards better range payload performance by a combination of drag reduction and improved overall efficiency.

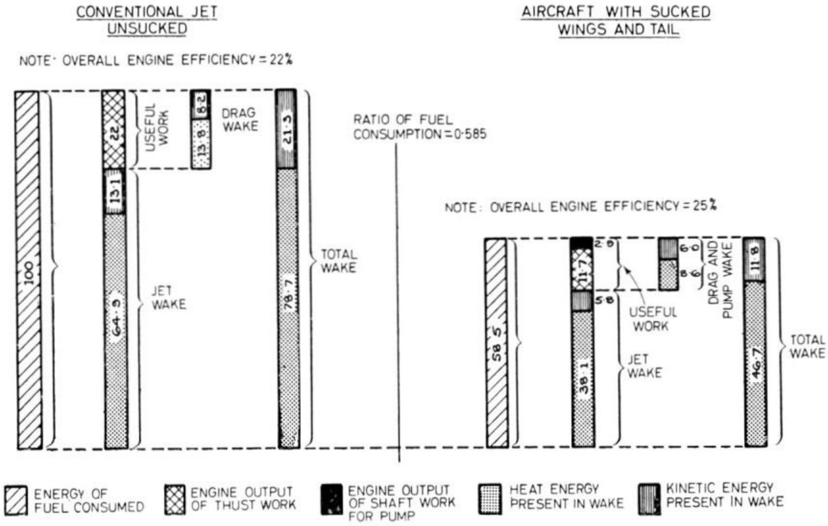


FIG. 7. Fuel-to-wake energy balance.

In Fig. 7 a fuel-to-wake energy balance is presented for a conventional jet airliner and a laminarized one of the same all-up-weight and speed. (A.U.W. = 300,000 lb, $M = 0{\cdot}85$.)

A large proportion of the fuel energy is rejected by the engine cycle. The remainder which is converted into thrust work reappears in the airframe wake, except in the case of suction aircraft when some of the shaft power is absorbed by the pump and reappears in the pump wake.

Wake energy may be in the form of either kinetic energy or heat; that is to say, a wake is composed of air which is in motion relative to its original state of rest before the passage of the aircraft, or which has had its temperature increased.

Diffusion of a wake results in its kinetic energy being decreased with a corresponding increase of heat energy.

Boundary layers have a high rate of diffusion. Thus, most of the acquired energy of a boundary layer wake is in the form of heat even at the instant it is shed.

Figure 7 shows that the laminarized aircraft requires only 58·5% of the fuel consumed by the conventional aircraft with turbulent boundary layer on wings and tail unit*.

This reduction is partly due to increase of overall efficiency† and partly to the reduction of the drag wake.

The overall efficiency is increased from 22% for the aircraft with conventional wings and tail unit to 25% in the case of the aircraft with suction.

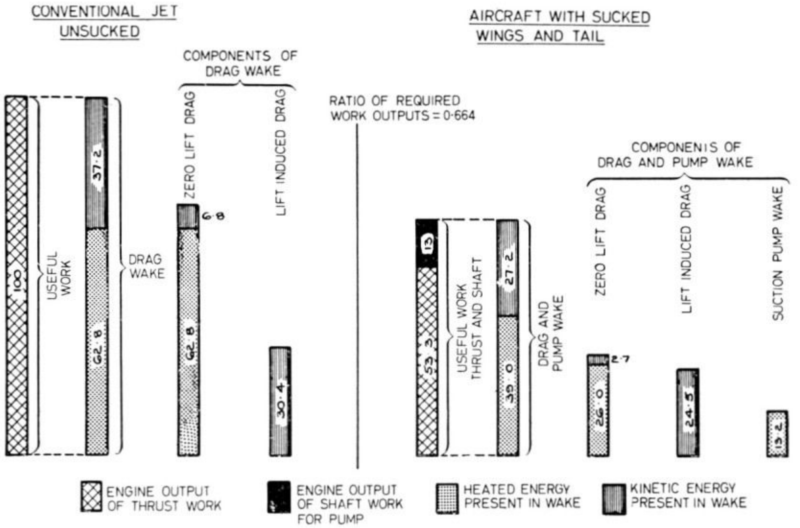


FIG. 8. Components of total aircraft wake.

The reduction of wake drag due to suction follows from Fig. 8 which shows the components of the total aircraft wake for the sucked and unsucked aircraft. Airframe wake comprises the lift induced and boundary layer wakes. The former contains, at least initially, only kinetic energy whilst the latter are composed chiefly of heat and to a smaller proportion of kinetic energy. This, by the way, indicates the rather limited prospects of pressure recovery from a turbulent wake.

The pump wake of a laminarized aircraft contains only heat energy if the sucked air is discharged from the aircraft at flight speed so that it is returned to its original state of rest in the atmosphere.

The total airframe and pump wake energy of the sucked aeroplane amounts to 66·4% of the airframe wake without suction. About 18% of the total power required is used for energizing the pumps.

* Even greater savings will be obtained if aircraft of equal fuselage size were compared. In the present case the laminarized aircraft is assumed to carry a greater payload over the same stage distance than the conventional type (18·3% compared with 10%). Therefore, it has to have a larger fuselage and higher body drag than the aircraft without boundary layer control.

† To the best of my knowledge Dr. W. Pfenninger[8] first drew attention to the improved overall efficiency of a jet engine when integrated into a sucked aircraft.

The exchange rate $\frac{\text{loss of thrust power}}{\text{pump horsepower}} = 0{\cdot}42$

The exchange rate is not a fixed constant. Its value depends on cruise Mach number and will vary for different types of engine. It also depends on how the suction compressors are driven, either by shafts or by second stage turbines. Its value is also affected by the operating conditions of the engines, i.e. the degree of throttling in the cruise.

The exchange rate is obviously an important criterion when selecting an engine—suction blower system for a laminarized aircraft.

Thus, laminar boundary layer control marks a significant step towards engine–airframe integration in a truly functional sense. The integrated jet engine-cum-suction pumps can be viewed as a sophisticated by-pass system.

The Suction System

Suction surfaces. The choice of suction surface construction—distributed suction or suction through discrete sinks (slits or strips)—is intimately bound up with the planform of the wing or tail surfaces to which suction will be applied.

In two-dimensional flow, as it occurs over the greater part of the surface of unswept wings, either type of suction surface can be used without difficulty; there is no need for distributed suction.

By suitable choice of aerofoil section and camber use can be made of the natural laminar flow (without suction) which can be obtained over the forward part of the aerofoil where a favourable pressure gradient exists.

Where crossflow of appreciable magnitude can occur—due to pronounced taper or sweep of leading and trailing edges—the use of distributed suction through some form of porous material becomes imperative, closely spaced slits being an alternative where crossflow is not too pronounced.

Porous surfaces may be made of sintered metal; sintered stainless steel has been used successfully on aircraft where nose suction for the prevention of separation was applied. Chief objection is weight.

An alternative type of porous surface is made of compact fibrous material impregnated with just sufficient bonding material to bind the fibres together but not so much as would cause clogging.

Sheets from such material are light, but the unprotected surface is not very resistant to erosion.

Drilled surfaces have been demonstrated to be effective in maintaining laminar flow with holes of small diameter spaced at large pitch-diameter ratio. The pattern of holes and its orientation to the flow direction has been shown to be critical which excludes this type of surface from application to swept wings, owing to the varying flow directions.

Strips of sintered porous material inserted in the metal surface at intervals were introduced by the author[9]. They were quite satisfactory on wind tunnel models but the edges caused difficulties in flight tests. Better results were obtained with perforated strips.

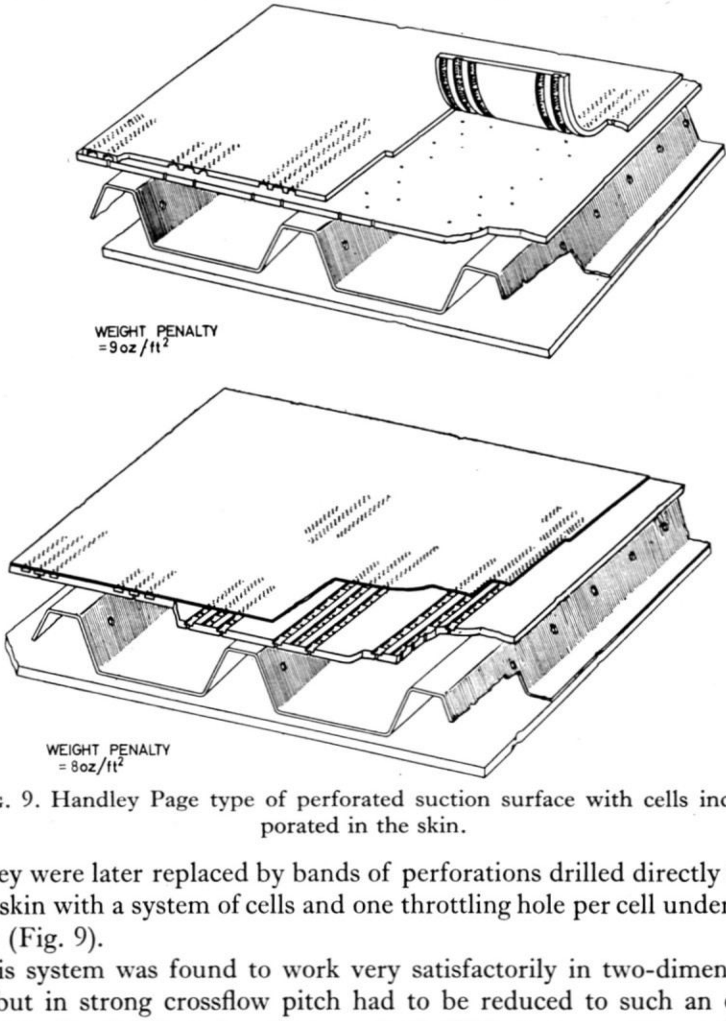


FIG. 9. Handley Page type of perforated suction surface with cells incorporated in the skin.

They were later replaced by bands of perforations drilled directly in the outer skin with a system of cells and one throttling hole per cell underneath [10, 11] (Fig. 9).

This system was found to work very satisfactorily in two-dimensional flow but in strong crossflow pitch had to be reduced to such an extent that virtually distributed suction resulted. With strong suction wakes or rather horseshoe type vortices can be set up behind the holes which can destabilize the flow especially when the wakes emanating from holes line up.

Slitted surfaces were originated by Dr. W. Pfenninger[8]. They are relatively simple to manufacture. Distance for consecutive slits depends on the magnitude of the crossflow. In very strong crossflow pitch may become impracticably close.

Distribution System

The purpose of the suction distribution system is to collect and convey the sucked air from the suction surfaces to the pumps. It must also produce the necessary distribution of inflow over the suction surfaces.

The distribution system can be subdivided into skin ducts, collectors and main ducts (Fig. 10).

Collectors (chordwise) can fulfil a dual function as stressed members (ribs) and components of the distribution system. According to the number and distribution of the collector ribs the suction surface is subdivided into

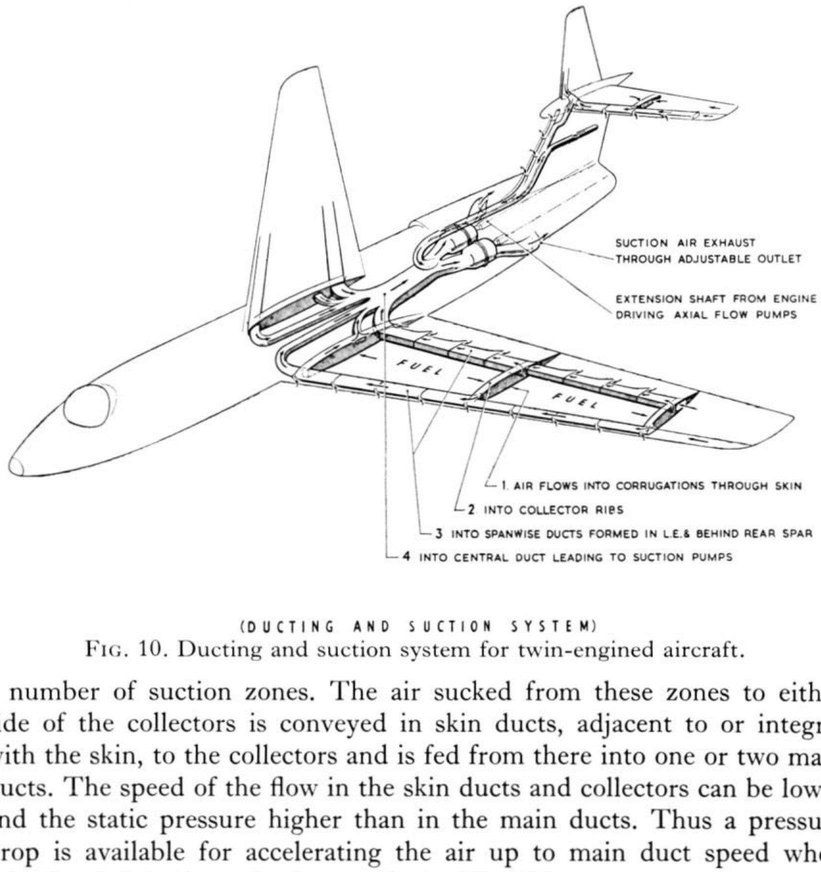


FIG. 10. Ducting and suction system for twin-engined aircraft.

a number of suction zones. The air sucked from these zones to either side of the collectors is conveyed in skin ducts, adjacent to or integral with the skin, to the collectors and is fed from there into one or two main ducts. The speed of the flow in the skin ducts and collectors can be lower and the static pressure higher than in the main ducts. Thus a pressure drop is available for accelerating the air up to main duct speed when injecting it into the main duct or ducts (Fig. 11).

One duct will suffice for a straight wing but two ducts will be found essential on swept wings. The flow velocity in the main ducts should be constant and as low as is compatible with the internal volume of the wings which remains available after allowance has been made for fuel stowage.

If the Mach number of the flow in the duct M_D is restricted to a certain value, say $M_D = 0{\cdot}2$, and with duct area measured perpendicular to the flow in the duct limited to a certain proportion of the total cross-sectional area of the wing profile at any spanwise position, it can be shown that the maximum aspect ratio of the sucked portion of the wing which lies between the tip and the entry of the duct into the pump is limited for given values of flight Mach number, cruise lift coefficient, angle of sweep, taper ratio and suction coefficient C_Q. The higher the taper ratio the

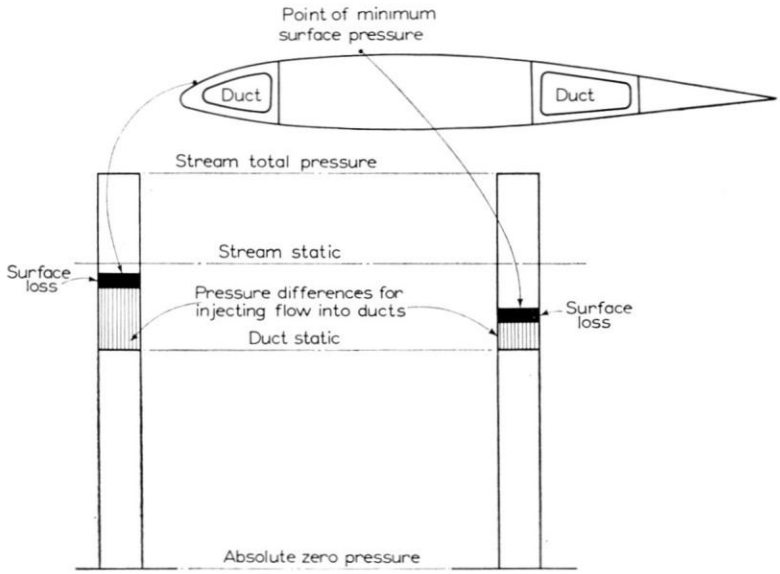


FIG. 11. Pressure relationship in suction system.

higher can be the limiting aspect ratio under otherwise identical conditions.

Valves must be provided for controlling the inflow distribution, especially for "tuning" up the system to obtain laminar flow. Once properly tuned up there should be rarely need for re-adjustment.

Two types of valves are suggested: one to be used to control the flow from each individual wing skin duct into the collector ribs, the other type is used to control the flow from the collector chamber into the main duct or ducts.

Ground adjustment may suffice for these valves.

Pump System

The task of the pumps is to suck the air from the lower strata of the boundary layer through the porous skin or perforations or slits. The pressure drop through the skin is relatively small compared with the opposing outside suction acting on the wing surface. Additional duct losses occur while conveying the air through the collectors and ducts to the pumps where the air is re-compressed to the original ambient pressure. Figure 11 shows the appropriate relative order of external and internal losses which have to be made good by the pumps.

The air is ejected either at flight speed or at somewhat greater speed. When ejected at flight speed, the air is stationary relative to the atmosphere; no wake is formed. If ejected with an efflux speed higher than that of flight a thrust wake is produced. It can be shown that an optimum efflux speed exceeding somewhat that of flight exists, but this optimum is very flat, especially when the gain is expressed in terms of range, and when increase of pump weight with compression ratio is taken into account.

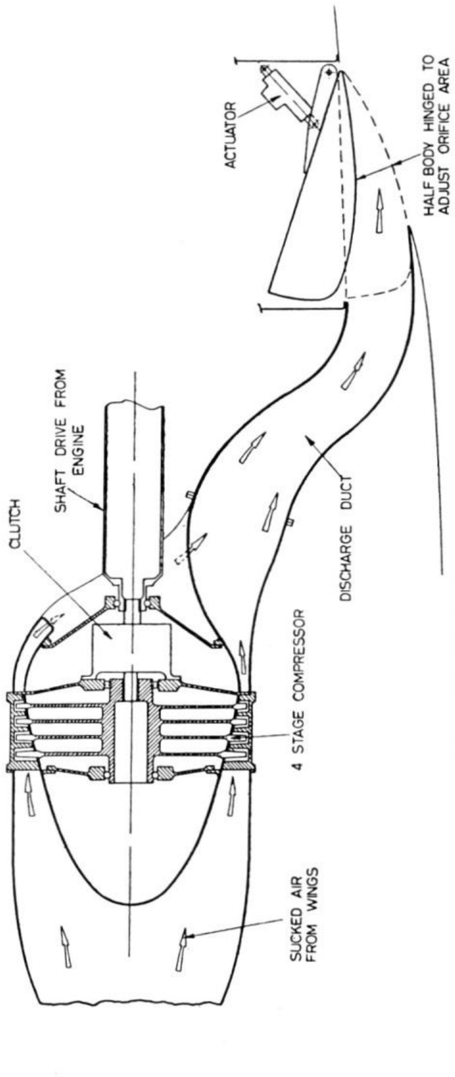


FIG. 12. Shaft-driven suction compressor unit.

In most practical cases and with efflux velocity not exceeding flight speed by much, a compression ratio of about 2·5 will suffice. The required mass flow follows by multiplying the C_Q value (order 0·0003–0·00045) with speed of flight and total sucked surface, making proper allowance for leakage and zones of intensified suction at the wing root, etc.

On multi-engined aircraft it will be desirable to have each engine drive a pump. For example, in the case of a four-engined aircraft maximum pump capacity should be 33% above normal capacity so that in case of engine or pump failure the revolutions per minute of the three remaining engines could be stepped up so that the pumps could deal with the total required mass flow.

A typical suction blower installation for a twin-engined aircraft is shown in Fig. 12. Here the pumps are driven by direct extension of the main shafts without any intermediate gearing. Greater flexibility would be obtained if a gear with variable ratio were installed between the engines and pumps. Both pumps have a clutch inserted in the shaft so that either pump can be stopped independently of its engine. Direct drive rather than a gear drive was chosen for its greater simplicity.

In addition to the pumps and their drives, the pump system includes a throttle valve upstream of the pump and also a movable half body in each of the discharge nozzles.

The purpose of the moving half body is to control the flow through the suction system because without gear the pump speed is dependent on the engine speed which cannot be chosen to suit the suction system under all conditions. If the suction flow exceeds that desired, the half bodies are moved outwards thus reducing the discharge and raising the pump peak pressure. By this means the flow is reduced to the desired rate. This method of control suffices for all cruising conditions, but care has to be taken to ensure that the operating points of the pumps do not approach the surge line. In the climb, however, surge would be encountered if this were the only method of control; therefore, additional throttling at the pump inlet is used. This throttling causes an increase in pump pressure ratio by reducing the intake pressure and hence the density of the air entering the pump. This results in a more powerful method of controlling the mass flow.

At lower altitudes reduced number of pumps can produce the desired rate of flow. In order to prevent backflow, when one pump is stopped, the discharge nozzle must be blocked completely by the half body. In addition, the throttle valves upstream of the pump can also be completely closed thus forming a double seal.

An alternative solution is shown in Fig. 13, where the pump is driven by contra-rotating free turbines arranged in the jet pipe. The pump blade rotates outside the jet pipe like a ducted fan within the annular suction duct.*

* A similar arrangement was proposed by Metro-Vickers (F/3 Turbo-jet), and recently in the General Electric CJ-805-21 a similar arrangement is used.

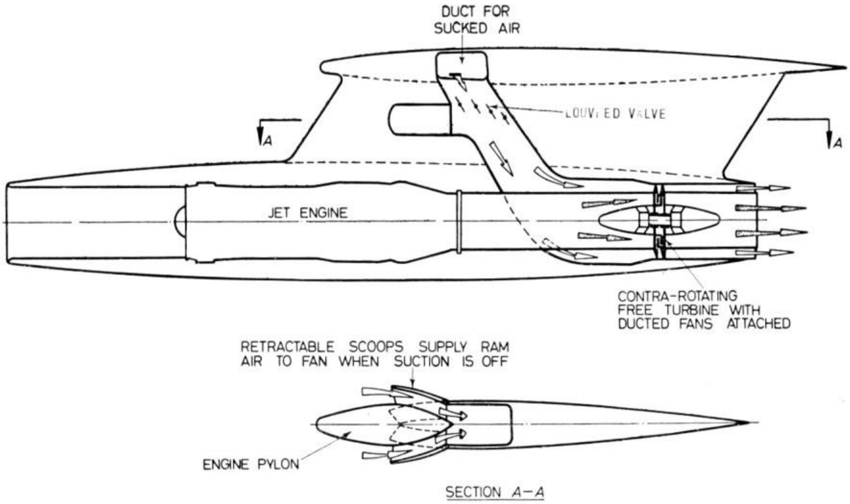


FIG. 13. Suction compressor unit driven by second stage gas turbine in jet pipe.

In the suction duct a butterfly valve or louvres have to be provided to let ram air enter the duct when suction is inoperative. The drive of the suction pumps by means of a second stage free turbine is particularly attractive when the engines are mounted in pods.

3. POTENTIAL APPLICATIONS OF BOUNDARY LAYER CONTROL FOR LOW DRAG

Laminarization is no panacea for all aircraft. It will come into its own where the emphasis is on long range for military or civil transports and where operation at a high cruising altitude is permissible.

Reduction of direct operating costs and fares is essential in civil aviation to broaden the field from which passenger traffic can be drawn.

Both for the achievement of long range and lower direct operating costs when operating over medium long and long stage distances it is necessary to improve the lift to drag ratio and the total efficiency with which the heat content of the fuel is converted into thrust work.

The following table gives a comparison of various types of existing and potential aircraft on the basis of the range parameter

$$\frac{L}{D} \times \frac{v}{c}$$

(The term $\frac{v}{c} = \frac{\text{cruising speed}}{\text{specific consumption in lb/lb of thrust hour}}$ signifies the product of J = mechanical equivalent of heat, the heat content of 1 lb of fuel, the cycle efficiency η_{th} and the propulsive efficiency η_{ρ}.)

Type of aircraft	L/D	v Cruising speed (m.p.h.)	c Engine s.f.c. (lb/lb hr)	v/c	$L/D \times v/c$
Subsonic turbo-prop	17–18	385	0·55	700	11,900–12,600
Subsonic turbo-jet (conventional)	18–19	550	0·95	578	10,400–11,020
Subsonic turbo-jet (wings and tail unit laminarized)	35–40	550	0·95	578	20,210–23,120
Supersonic delta	7	1500	1·42	1058	7406

There are two fundamental ways of obtaining large values of lift to drag ratio in the subsonic regime. One is exemplified by the aircraft designed and built by Hurel-Dubois in France, where the emphasis is on extremely high aspect ratio, necessitating a braced wing and high cruising lift coefficients. The other method is laminarization.

The first principle is restricted to aircraft flying at relatively low subsonic Mach numbers because at high subsonic Mach numbers limitations are imposed on the magnitude of the lift coefficient in the cruise by drag rise and buffeting.

Drag reduction by boundary layer control applied to wing and tail unit and engine nacelles of aircraft operating at high subsonic Mach numbers with lift coefficients restricted to the order of 0·3–0·45, combined

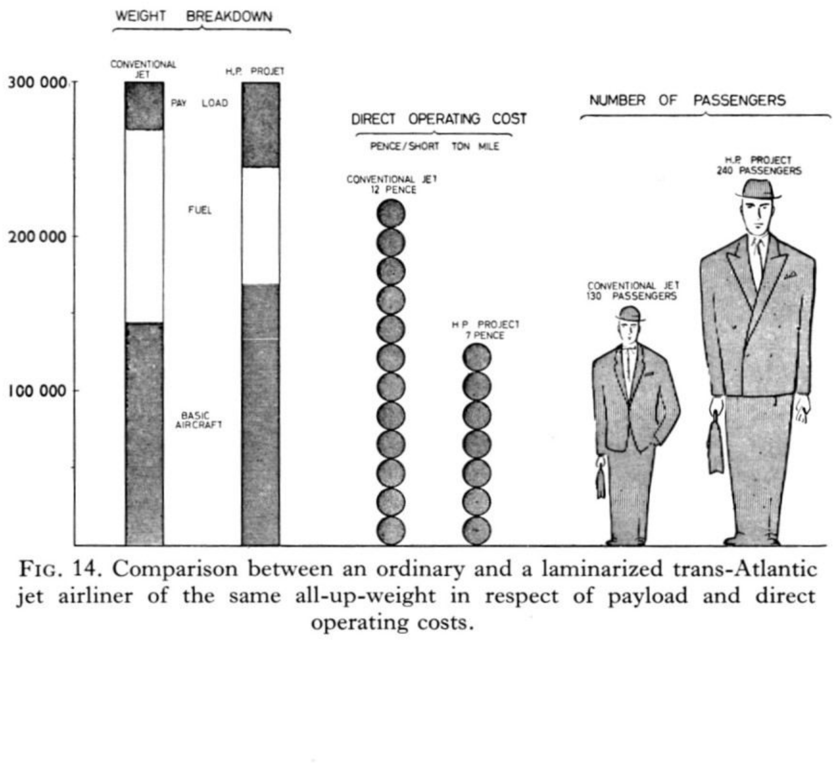


FIG. 14. Comparison between an ordinary and a laminarized trans-Atlantic jet airliner of the same all-up-weight in respect of payload and direct operating costs.

with moderately high aspect ratios of the order of 8–11, can double the effective lift-to-drag ratio of large transport aircraft of conventional design. Values for L/D of the order of 40 or more are, therefore, within the realm of possibility. (See Appendix III.)

Thus, an aircraft, for a given payload and range, can become substantially lighter.

For a given range and all-up-weight, payload can be increased and direct operating costs substantially decreased (order of magnitude for trans-Atlantic aircraft of 300,000 lb all-up-weight is about 42·5%) (Fig. 14). Reduction of direct operating costs becomes the more marked the closer the stage length approaches the ultimate range of conventional jet aircraft, that is when the conventional jet transport has to operate with its payload considerably below the cabin's volumetric capacity.

Where the emphasis is on shorter take-off and landing runs rather than direct operating costs laminarization makes it possible to employ larger wing areas resulting in much reduced wing loadings without drag penalty. In fact, there will still be a net drag reduction.

Alternatively, the range on an aircraft for a given all-up-weight and payload can be stretched far beyond the ultimate range of a conventional aircraft where the boundary layer is turbulent. Thus, it would be possible to design aircraft which, at the all-up-weight (300,000 lb) of a conventional trans-Atlantic airliner would carry a payload of about 30,000 lb non-stop from England to Australia.

The other aspect—reduction of all-up-weight due to a reduction of drag or improvement of L/D—shows greatest promise for nuclear powered aircraft where weight is the chief problem and not range or endurance.

For nuclear-powered aircraft there is a minimum design-gross-weight at which they will just fly and carry no payload. This minimum weight is large by current aircraft standards. Its magnitude depends on reactor power-density (megawatts per cubic foot of reactor core), reactor shield weight and the lift-to-drag ratio of the aircraft.

The lift-to-drag relationship determines the total reactor power in megawatts which, with a certain total efficiency, is transformed into thrust power.

Shield weight is a function of reactor power and reactor power-density, or the heat which each cubic foot of the reactor core can produce.

With increase of reactor power, shield weight increases. It does so at a slower rate than direct proportion, rate of growth decreases as power density increases.

Robert B. Ormsby of the Lockheed Aircraft Corporation[(12)] has given figures of shield weights (with lead used against gamma rays and water against neutrons) for varying reactor powers and power densities which show that for a 600,000 lb aircraft whose lift to drag ratio is 20, a reactor power of 60 MW is required. Assuming a reactor power-density of 0·8 MW/ft^3 the shield (water and lead) weighs 292,000 lb. Estimated payload is only 8000 lb and the aircraft just exceeds the minimum size.

By doubling the lift-to-drag ratio by boundary-layer control and thereby halving reactor power, shield weight can be reduced to 222,000 lb without allowing for a reduction of reactor and powerplant weights. The cost in weight of laminarization will be about 3% (some 18,000 lb) of all-up-weight. Thus, the payload is about 60,000 lb.

Alternatively, if it were possible to double reactor density on the aircraft having a lift-to-drag ratio of 20, shield weight would be reduced to 231,000 lb and the resulting payload would rise to 69,000 lb.

The examples illustrate that an improvement of lift-to-drag ratio is of the same vital importance for the development of nuclear-powered aircraft as an increase of reactor power-density. Either improvement, or both combined, will at any value of design gross-weight improve the payload-to-weight ratio of the nuclear-powered aircraft.

In conclusion I should like to refer to a remark made recently in an American aeronautical magazine when boundary layer control was likened to a much discussed bridesmaid who never becomes a bride. Her attractions seem to have faded a little in the eyes of some, and she seems also to be considered a little difficult to manage. I hope to have succeeded in demonstrating that this bridesmaid appears to possess all the qualifications of an attractive bride. It is about time that someone should propose. There will be, of course, some bargaining about the dowry.

Acknowledgements—The author wishes to express his thanks to Handley Page Ltd., for permission to present this paper and also gratefully acknowledges the assistance he received from Messrs. J. B. Edwards, K. V. Bonney, A. W. Lindfield, H. G. Pinsent and J. H. Culhane of the Handley Page Research Department. In fact, the paper represents in many respects the essence of views and opinions which have crystallized as a result of a team effort extending over about seven years, covering wind tunnel experiments, theoretical and flight research, project and technological studies.

APPENDIX I

BOUNDARY LAYER FLIGHT RESEARCH IN GREAT BRITAIN AND IN THE U.S.A. 1952–1957

Exhaustive flight research on jet powered aircraft has been conducted between 1952 and 1957 in Great Britain by the Royal Aircraft Establishment and later by Dr. Head at Cambridge University, and by Handley Page Ltd.; in the U.S.A. by the research group at Northrop Aircraft Inc., led by Dr. Pfenninger.

The research work in Great Britain was sponsored by the Ministry of Supply.

An analysis of this combined research effort (based on published results as far as American work is concerned) leads to the following most important conclusions.

1. Full-chord laminar flow may be maintained in flight up to chord Reynolds numbers of the order of 30×10^6.
2. Suction quantities required to maintain laminar flow are sufficiently small to result in net profile drag reductions of the order of 70–80%, account being taken of the power required for suction.
3. Increase in Mach number, at least up to the critical value, has no adverse effect on the maintenance of laminar flow.
4. Consistent and repeatable results have been achieved using methods of construction which are essentially suitable for full-scale application.
5. Suction surfaces can be designed and manufactured in an engineering fashion with low weight penalties.
6. A complete solution to the problem of fly accretion is given by the use of simple discardable leading-edge covers or by spraying the wing nose with a substance which sublimates under the application of heat.
7. The certainty of maintaining laminar flow by suction has been thoroughly established by intensive flying in Great Britain and the U.S.A.
8. Distributed suction applied through a finely perforated surface is more effective than suction through isolated strips of larger perforations where crossflow occurs in the boundary layer. Suction applied through closely spaced and very narrow slits is more effective than either.
9. The tolerable degree of surface roughness and waviness increases, for a given flight Mach number, with cruising altitude, or increase of kinematic viscosity, respectively. Equally, accidental surface roughness caused by dust, flies, etc., becomes less critical with increase of altitude.
10. Clogging of the suction surface has not been found to occur in flight. In practical operation, the application of suction during the climb will be restricted to heights greater than 10,000 ft. Dust accretion chiefly occurs when the aircraft stands for long periods in a hangar or in the open; it is prevented by covering the wing. Maintenance problems will diminish on long-range aircraft which spend most of their time in the clean air of the stratosphere.
11. It has been observed that when flying at high speed and at great altitude erosion of the remains of squashed flies occurs. It is probable that at cruising altitudes of 45,000–50,000 ft the non-eroded remains may generally cause no trouble. Therefore, nose protection during take-off and climb may be less essential for aircraft operating at a high cruising altitude.

Some initial experiments have been conducted by the Handley Page Research and Flight Departments where live flies were blown, prior to take-off, against the nose portion of the wing of an aircraft capable of flying at great altitude and at high subsonic Mach numbers. A special "fly gun" was used, energized by compressed air, to produce realistic fly impacts at simulated critical impact velocities, which were subsequently marked to distinguish them from impacts which might occur during landing. For easy handling the flies were anaesthetized with ether or CO_2 prior to being put into the gun.

These tests are being continued with a special envelope attached to a portion of the wing nose. This envelope can be detached after landing and after fly remains resulting from marked pre-flight impacts have been sprayed with fixative and unmarked ones removed, the envelopes will be attached to a wooden wind tunnel model whose nose portion corresponds to the actual wing profile. This model will then be tested as a two-dimensional model in a low turbulence tunnel to determine the unit Reynolds number U/ν at which transition is caused by the fly remains. Thus the critical altitude for a given Mach number can be determined above which eroded fly remains will cease to cause transition.

The author is obliged to Dr. W. S. Coleman of Blackburn and General Aircraft Ltd., for valuable advice derived from his rich and extensive experience gained by his pioneering work in the study of critical insect contamination. He also supplied a stock of live flies and "handling" instructions.

APPENDIX II

AN APPROXIMATE METHOD FOR CALCULATING CROSS-FLOW ON AN INFINITE SWEPT WING WITH ARBITRARY VELOCITY AND SUCTION DISTRIBUTIONS

THE following method was developed by A. W. Lindfield and H. G. Pinsent of the Handley Page Research Department.

Figure 15(a) shows the flow inside and Fig. 15(b) the flow at the edge of the boundary layer.

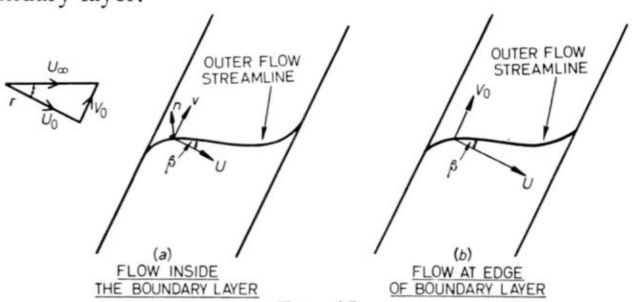


FIG. 15

From Fig. 15(a) follows:

$$n = v \cos \beta - u \sin \beta = \frac{v}{V_0} V_0 \cos \beta - \frac{u}{U} U \sin \beta$$

From Fig. 15(b):

$$U \sin \beta = V_0 \cos \beta = \frac{UV_0}{(U^2 + V_0{}^2)^{\frac{1}{2}}}.$$

Hence,

$$n = \frac{UV_0}{(U^2 + V_0{}^2)^{\frac{1}{2}}} \left(\frac{v}{V_0} - \frac{u}{U}\right) = \frac{U_0\overline{U} \tan \Gamma}{(\overline{U}^2 + \tan^2 \Gamma)^{\frac{1}{2}}} \left(\frac{v}{V_0} - \frac{u}{U}\right)$$

For an infinite wing the boundary layer equations may be separated into an independent chordwise solution and a spanwise solution which depends on the chordwise solution. Since the crossflow is proportional to $[(v/V_0)-(u/U)]$ it seemed reasonable to evaluate u/U, then v/V_0 and hence obtain n.

For the chordwise solution Truckenbrodt's[13] method was initially used but later abandoned since it predicted premature separation in the region with adverse pressure gradient.

A method due to Head[14] was found more suitable. This is based on a two-parameter system, using the momentum and energy integral equations; it gives momentum thickness to within 1 or 2% and excellent agreement with velocity profiles calculated by a more exact method. This good agreement was found to persist even in the region of the adverse pressure gradient.

For the spanwise solution an extension of Sinha's[15] method to a two-parameter system was used. The original method solves the spanwise boundary layer momentum equation by the use of one-parameter Schlichting[16] profiles.

This one-parameter method was found to be unable to cope with discontinuous changes in the velocity gradient or suction distribution. By the extension to a two-parameter system, using a spanwise energy equation as well as the momentum equation, Sinha's method was brought in line with Head's method for the chordwise solution.

It was realized, however, that this approach to obtain the crossflow from the difference of the chordwise and spanwise profiles was inadequate —except in the stagnation region—since small errors in the u/U and v/V_0 profiles were sufficient to make the error of their difference of the same magnitude as the crossflow itself.

With this in mind a method was developed which would give the crossflow directly.

Derivation of the Crossflow Equation

The boundary layer equations for an infinite wing are:

Chordwise:
$$u \frac{\partial u}{\partial x} + w \frac{\partial u}{\partial z} = U \frac{\mathrm{d}U}{\mathrm{d}x} + \nu \frac{\partial^2 u}{\partial z^2} \tag{1}$$

Spanwise:
$$u \frac{\partial v}{\partial x} + w \frac{\partial v}{\partial z} = \nu \frac{\partial^2 v}{\partial z^2} \tag{2}$$

Condition of Continuity $$\frac{\partial u}{\partial x}+\frac{\partial w}{\partial z}=0 \tag{3}$$

Making these equations non-dimensional one obtains:

$$\overline{U}T\frac{\partial T}{\partial X}+W\frac{\partial T}{\partial Z}=(1-T^2)\frac{\mathrm{d}\overline{U}}{\mathrm{d}X}+\frac{\partial^2 T}{\partial Z^2} \tag{1a}$$

$$\overline{U}T\frac{\partial S}{\partial X}+W\frac{\partial S}{\partial Z}=\frac{\partial^2 S}{\partial Z^2} \tag{2a}$$

$$\frac{\partial(\overline{U}T)}{\partial X}+\frac{\partial W}{\partial Z}=0 \tag{3a}$$

where

$$T=\frac{u}{U}\ ;\ S=\frac{v}{V_0}\ ;\ \overline{U}=\frac{U}{U_0}$$

Subtracting equation (1a) from (2a) and writing $N=S-T$ one obtains:

$$\overline{U}T\frac{\partial N}{\partial X}+W\frac{\partial N}{\partial Z}=\frac{\partial^2 N}{\partial Z^2}-(1-T^2)\frac{\mathrm{d}\overline{U}}{\mathrm{d}X} \tag{4}$$

For step-by-step solution a difference equation was derived from (4).

Denoting a step in X by ΔX and values at the beginning of the step by a suffix "1", and values at the end of a step by a suffix "2", the following equation is finally obtained:

$$\Delta\left(\frac{\partial N}{\partial X}\right)=\frac{1}{\overline{U}_2T_2}\left[\frac{\partial^2(\Delta N)}{\partial Z^2}-W_2\frac{\partial(\Delta N)}{\partial Z}-(\Delta W)\frac{\partial N_1}{\partial Z}-\Delta(\overline{U}T)\left(\frac{\partial N}{\partial X}\right)_1-\right.$$
$$\left.-\Delta\left\{(1-T^2)\frac{\mathrm{d}\overline{U}}{\mathrm{d}X}\right\}\right] \tag{5}$$

This is the equation used to evaluate the crossflow. It should be noted that it is exact.

The abbreviated terms are written in more explicit form below:

$$\Delta N=N_2-N_1 \qquad \Delta\left(\frac{\partial N}{\partial X}\right)=\left(\frac{\partial N}{\partial X}\right)_2-\left(\frac{\partial N}{\partial X}\right)_1$$

$$\Delta(\overline{U}T)=\overline{U}_2T_2-\overline{U}_1T_1 \qquad \Delta W=W_2-W_1$$

$$\Delta\left[(1-T^2)\frac{\mathrm{d}\overline{U}}{\mathrm{d}X}\right]=(1-T_2^2)\left(\frac{\mathrm{d}\overline{U}}{\mathrm{d}X}\right)_2-(1-T_1^2)\left(\frac{\mathrm{d}\overline{U}}{\mathrm{d}X}\right)_1$$

Boundary conditions for $Z=0$ can be derived from equations (4) and (5).

The crossflow equation can be solved with a satisfactory degree of accuracy by graphical methods.

Plots of T and W vs. X with Z as parameter are required so that these functions may be determined with reasonable accuracy at any value of X required.

If a running plot of $\partial N/\partial X$ vs. X for each Z is kept it is a simple matter to compute ΔN and hence N.

Owen first suggested a Reynolds number based on the peak crossflow velocity and the thickness of the crossflow velocity profile as a criterion which was denoted by χ.

It was found that for flow in the vicinity of the leading edge the laminar flow broke down if the value of this criterion exceeded a certain value (about 125). This was accepted as a critical value.

At the National Physical Laboratory an analogy between laminar flow on a rotating disk and on a swept wing was discovered[17]. The centrifugal force of the rotating disk corresponds to the pressure gradient normal to the potential flow on a swept wing and the radial component of the boundary layer on the disk is equivalent to the crossflow profile on a swept wing. The critical value of χ was, however, found to be higher in the case of the rotating disk which suggested that stability depended on a parameter associated with the shape of the crossflow profile.

Owen later suggested a critical Reynolds number based on the distance of the inflexion point from the wall and the velocity at the inflexion point of the profile. This has the advantage of reducing the range of variation of χ_{crit} considerably.

Thus, for a profile to be stable the inflection Reynolds number should be less than the critical value which depends on a shape parameter. Theoretical work of fundamental importance has been done by another research group on this subject[18–20].

The differential method developed by Lindfield and Pinsent for solving the crossflow equation gives accurate results and can be quite rapid since steps of 5% chord may be taken over a considerable part of the wing.

In nearly all cases so far investigated comparison of the velocity profiles with exact solutions was quite favourable, the crossflow profiles being obtained accurately enough for their stability parameter to be determined.

It is felt that the method provides a simple and reasonably accurate way of calculating the laminar boundary layer for an infinite swept wing.

Simple ways of solving the boundary layer equations for a finite wing have also been investigated with promising results.

Considerable progress has been made by H. G. Pinsent and P. A. Lock towards developing a new method of obtaining crossflow profiles. This method is basically similar to Head's chordwise method in that it uses charts and is quick and easy to use.

APPENDIX III

PERFORMANCE ASPECTS

Optimum value of L/D occurs when

$$C_{D_0} = C_{D_i}$$

$$C_{D_i} = \frac{C_L^2}{\pi \cdot A_e}$$

$$\left(\frac{L}{D}\right)_{\max} = \frac{\pi \cdot A_{e\text{opt}}}{2} \sqrt{\frac{1}{C_{D_0}}}$$

where

C_{D_0} = coefficient of total wake drag

C_{D_i} = coefficient of induced drag

A = geometric aspect ratio

K = induced drag factor dependent on planform, sweep angle, etc.

$A_e = A/K$ = effective aspect ratio

Figure 16 illustrates the relationship between effective aspect ratio

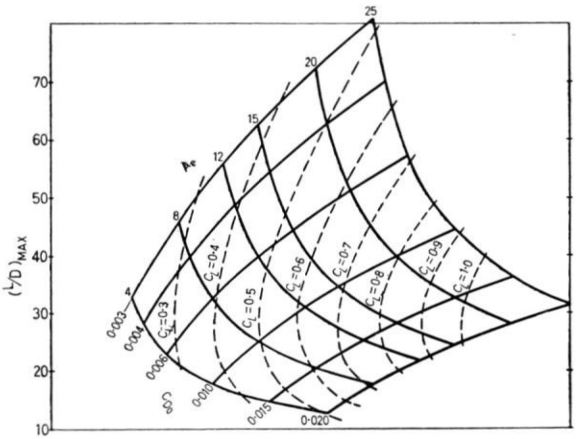


FIG. 16. Relationship between optimum values of L/D. ($C_{D_0}=C_{D_i}$) and corresponding effective aspect ratio $A_e=A/K$ for various values of C_{D_0} and C_L.

A_e, C_{D_0}, C_L and $(L/D)_{\max}$

It can be seen that very high values of $(L/D)_{\max}$ with values of C_{D_0} which are obtainable without suction ($C_{D_0} \simeq 0{\cdot}01$) are bound up with high values of C_L (unsuitable for high subsonic Mach numbers) and excessively high aspect ratios.

The effect of laminarization is threefold:

Reduction of the induced drag factor K.

Reduction of skin friction resulting in a reduced value of C_{D_0}.

Improvement of overall efficiency.

Improvement of Effective Aspect Ratio

It has been observed[21] that on swept wings there is a considerable sub-layer in the turbulent boundary layer where the flow is predominantly spanwise. This contributes to the increase of the induced drag factor K. Since a spanwise boundary layer drift will not be present on a sucked wing it is suggested that a reduction of the induced drag factor K may be assumed which will be quite substantial with greater angles of sweep.

Equivalent Values of C_{D_0}, Pump Drag and L/D

In addition to producing thrust to overcome wake and induced drag, the powerplant of an aircraft with laminar boundary layer control has to provide power to drive the suction compressors which re-accelerate the sucked air.

The necessary horsepower to drive the suction compressors will depend on:

Total sucked mass flow (including leakage).

Pump pressure ratio determined by the magnitude of static pressure acting on the wing surface against which the pumps have to operate, duct losses, etc.

Pump performance.

Engine performance and exchange rate between shaft horsepower and thrust horsepower.

Efflux speed of ejected boundary layer.

It has been explained in Section II that the loss of engine thrust horsepower is, generally, only a fraction of the shaft horsepower taken-off to drive the suction pumps. Its value depends on engine cycle characteristics and operating conditions.

This favourable exchange rate results from the comparatively greater efficiency with which energy can be extracted from the gas by the turbine than by conversion into thrust work by a straight jet.

For convenient comparison with conventional (unsucked) aircraft pump power can be expressed in terms of an equivalent pump drag so that an equivalent lift-to-drag ratio can be determined.

The equivalent pump drag $D_{p\ (\text{equ.})}$ is obtained by subtracting the actual total drag $D_{\text{tot (actual)}}$ from the equivalent total drag $D_{\text{tot (equ.)}}$.

$$D_{p\ (\text{equ.})} = D_{\text{tot (equ.)}} - D_{\text{tot (actual)}}$$

The equivalent total drag $D_{\text{tot (equ.)}}$ is defined as being equal to the thrust which the engines would develop without any power taken out of the turbines, with the aircraft flying at the same height and Mach number and with the engines consuming fuel at the same rate as when supplying thrust equalling the actual total drag plus the horsepower which is required to drive the suction compressors.

By defining equivalent total drag and equivalent pump drag in this way the improvement of overall efficiency is taken into account in the determination of the equivalent lift-to-drag ratio $(L/D)_{\text{equ}}$.

$$\left(\frac{L}{D}\right)_{\text{equ.}} = \frac{C_L}{C_{D_i} + C_{D_0\ \text{equ.}}}$$

$$C_{D_0\ \text{equ.}} = C_{D_W} + C_{D_p}$$

C_{D_W} = coefficient of total wake drag with laminar boundary layer over laminarized areas

$$C_{D_p} = \frac{D_{\text{tot (equ.)}} - D_{\text{tot (actual)}}}{S_W \cdot q}$$

S_W = wing area

q = dynamic pressure

Effect of Aircraft Size and Payload/All-up-weight Ratio on C_{D_0} and L/D

There is no fundamental reason why laminarization should not be applied to the fuselage or at least to the portion of the fuselage ahead of the trailing edge of the wings.

Certain practical difficulties would, however, arise in regard to cockpit design, windows, doors, etc.

If laminarization is restricted to wings, tail unit and engine nacelles, and if a certain payload density and payload/all-up-weight ratio are assumed, the wetted surface of the body which houses the payload will become a smaller proportion of the total wetted surface as size and weight of aircraft increase. Correspondingly, the contribution of body drag (with turbulent boundary layer) to the total wake drag and hence C_{D_0} will become smaller.

Thus, the benefit from keeping laminar flow over the surface of wings and tail unit will be more pronounced on large aircraft than on smaller ones unless in the latter case the density of payload is increased. The advantage to all-wing aircraft is obvious.

Equally well, with all-up-weight fixed, a better L/D will result for an aircraft designed to cover a very long stage distance with a conventional payload/all-up-weight ratio than in the case of a laminarized aircraft designed for trans-Atlantic range and capable of carrying a greater payload than its conventional counterpart. In the latter case the fuselage will have to be enlarged to accommodate the greater payload; hence its weight will go up and its wetted surface and drag will be greater than in the case of the long-range aircraft with smaller payload.

In Fig. 17 estimated values of $C_{D_0\,\text{equ.}}$ and cruising $(L/D)_{\text{equ.}}$ are plotted against all-up-weight.

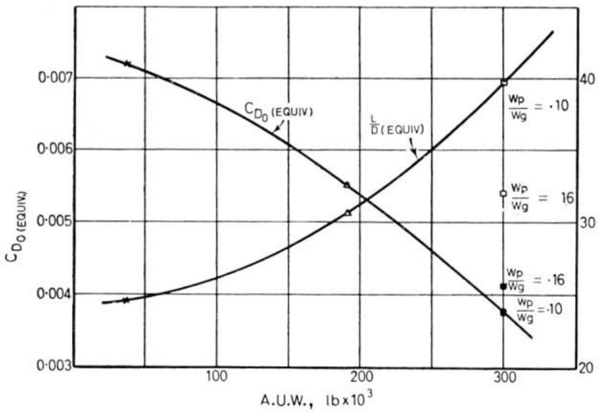


FIG. 17. Variation of $C_{D_0\,\text{equ.}}$ and $L/D_{\text{equ.}}$ with all-up-weight. (Fuselage not laminarized.)

These estimates refer to a number of detailed design studies made for aircraft of various sizes and stage lengths.

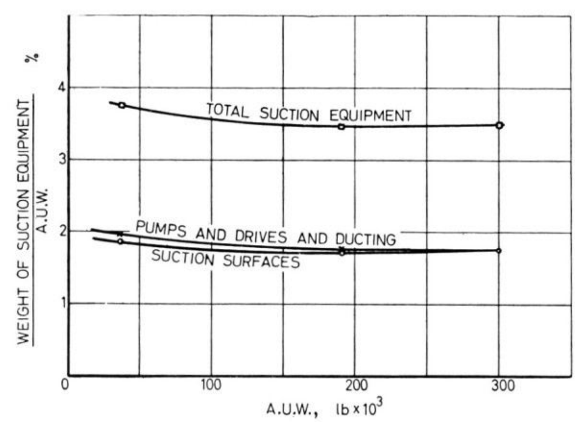


FIG. 18. Variation of percentage weight of suction equipment with all-up-weight. (Fuselage not laminarized.)

Weight of Suction Equipment

Additional weight for suction equipment will result from:

Suction compressors and drives

Suction surfaces (additional to structural skin)

Suction ducting

Estimated values for the percentage values of the component weights in terms of all-up-weight are plotted in Fig. 18 for aircraft of different weights.

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DISCUSSION

H. WITTENBERG*: For the estimation of the equivalent drag coefficient—symbol $C_{D_0\,\text{equ.}}$ in Dr. Lachmann's paper—one has to know the suction quantity as well as the pressure losses in the ducting system of the aircraft.

In literature a lot of information is found on the necessary suction quantities, normally expressed in the suction coefficient C_Q. Much less figures are known on the pressure losses, which of course depend heavily on the design of the ducting system.

I should like to ask Dr. Lachmann what values of the pressure coefficient C_p are to be used for the calculation of $C_{D_0\,\text{equ.}}$ for the kind of ducting system shown in Fig. 10 of his paper.

G. V. LACHMANN: The ducting system shown in Fig. 10 is designed for a duct Mach number of 0·2. Estimated duct losses amount to five times the dynamic pressure of the duct flow.

The value of C_p is $-0{\cdot}88$. Multiplying this value with $\rho/_2 v^2$ the difference between ambient pressure and pressure at the pump entry is obtained. This pressure difference comprises the pressure differential due to the supervelocities at the wing profile plus duct losses.

The compression ratio of the suction blower is 2·3, assuming that the air sucked from the boundary layer is ejected with flight velocity.

E. W. C. WILKINS†: Dr. Lachmann has pointed in his lecture to the potential application of boundary layer control to drag reduction for long-range civil transports, and has shown that by maintaining laminar flow the L/D ratio for the complete aircraft—in the subsonic régime—can be just about doubled. In supersonic flight, however, we still talk of L/D ratios of only 5, 6, or possibly 7. These figures are too low for really good economic operation. Even a small increase in L/D, however, might make all the difference to satisfactorily economic operation. I would, therefore, like to ask Dr. Lachmann what, in his view, are the future possibilities of obtaining larger lift over drag ratios at high supersonic speeds, and to what extent, if any (because in the supersonic régime the resistance is, as we know, mostly due to wave drag), boundary layer control may help.

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† Lockheed Aircraft Corp., U.S.A.

G. V. LACHMANN: In considering potential application of B.L.C. (for low drag) for aircraft operating at high supersonic Mach numbers the following facts must be borne in mind:

(1) The suction quantities are proportional to flight speed, and pump power is approximately proportional to $\rho \, . \, v^3$. Hence larger and heavier pumps will be required than at high subsonic speeds.

(2) The exchange rate $\dfrac{\text{loss of thrust power}}{\text{pump power}}$ becomes less favourable as flying speed increases in view of the improvement of propulsive efficiency of the jet with increase of flight speed.

(3) Skin friction becomes a smaller proportion of total drag as wave drag predominates.

The combined result of these three effects is that laminarization through suction becomes less beneficial at high supersonic Mach numbers than at high subsonic Mach numbers or low supersonic Mach numbers when shock-free flow on the wings is still possible.

A rough estimate for a supersonic aircraft (slim delta) of $M = 1{\cdot}8$–2, assuming that laminarization by suction were possible, shows that about the following improvements of L/D in terms of a factor

$$K = \frac{\text{(friction drag + pump drag) with laminar flow}}{\text{friction drag with turbulent flow}}$$

are possible.

K	1	0·5	0·25	0
L/D	7·4	9·02	10·24	12·2
$\dfrac{(L/D)\text{ laminar}}{(L/D)\text{ turbulent}}$	1	1·21	1·37	1·63

The friction drag of the aircraft with turbulent boundary layer amounts to about 35% of the total drag.

The following assumptions were made:

(1) The laminarized aircraft flies at the same Mach number but at a higher value of q, i.e., at a lower height, so that the ratio $\dfrac{\text{lift dependent drag}}{\text{total zero lift drag}}$ remains the same.

(2) Engine size and nacelle drag are reduced for the laminarized aircraft in proportion to the reduction of total drag.

Whether a net gain can be achieved depends very critically on the difference between the weight saving in respect of fuel and engines and the extra weight of suction plant, ducts and suction surfaces. A reliable answer would require a very detailed investigation. Additional technical problems will arise from the fact that the hot air from the lower strata of the boundary layer will be sucked into the wing ducts. It is possible that at Mach numbers >2 stabilization of laminar flow by cooling might offer better prospects.

A. RASPET*: The author makes the statement that the reduction of wing and tail area compatible with landing and take-off performance has been reached. This discussor in an earlier discussion of Schlichting's paper given at this Congress, showed experimental results obtained in flight of maximum power-off, trimmed airplane lift coefficients around 4·4. Since the $C_{L\,\max}$ of swept-wing airplanes even with flaps is around 1·5, we must admit that boundary layer control for high lift must be a fundamental consideration if by using high lift B.L.C. we can at least think of reducing the wing area by a factor of three.

* Head, Aerophysics Dept., Mississippi State University, U.S.A.

However, the more inspiring thought on designing boundary layer controlled aircraft is that offered by a compatible integration in a single system of both boundary and high lift gains. This compatible marriage appears possible for the distributed suction B.L.C. when momentum conservation is used as a design principle.

G. V. LACHMANN: The statement quoted by Dr. Raspet from the paper leaves out the words "on modern airliners". It is important that we both should think in terms of the same type and size of aircraft.

Even if it were possible to increase by means of distributed suction the maximum lift coefficient of swept wings at acceptable ground angles in the ratio of 3 : 1 compared with present-day values does Dr. Raspet seriously suggest that the wing area of an airliner could then be reduced to one third of present-day size ?

After all, internal volume for fuel stowage is an essential design consideration and so is span. If we keep the span unchanged the wing with reduced area would have its aspect ratio trebled.

Furthermore, there is such a thing as cruising lift coefficient whose value is limited by considerations of critical Mach number, buffeting limit, etc.

A characteristic consequence of boundary layer control for low drag is that the drag penalty resulting from wing area is much less than for a conventional aircraft with turbulent boundary layer flow over the wing.

This has two beneficial consequences: lower take-off and approach speeds result from increased wing area (or lower wing loadings, respectively) and a large span is made possible with only a very small friction drag penalty.

Since

$$d_0 = \frac{D_0}{q} = \frac{\text{zero lift drag}}{\text{dynamic pressure}}$$

is almost invariant with area it can be shown that the maximum lift to drag ratio of an aircraft with laminarized wings and tail unit is proportional to the span and cruising C_L inversely proportional to the mean chord, C_{mean}.

$$\frac{L}{D}_{\max} \sim \frac{b}{2} \cdot \sqrt{\frac{\pi}{K \, . \, d_0}}$$

$$(C_L) \text{ for } (L/D)_{\max} \sim \frac{1}{C_{\text{mean}}} \cdot \sqrt{\frac{\pi d_0}{K}}$$

As with all aircraft the optimum wing span is a compromise between structural and aerodynamic requirements, but, in general, the optimum wing span for a laminarized aircraft is greater than for a turbulent one of corresponding weight and cruising speed.

Reduction of landing and approach speeds would certainly be desirable for an airliner, but it should be borne in mind that the optimum wing loading of a laminarized aircraft is at least 25% less than that of a conventional type.

Besides, on a long range airliner the landing weight is very much reduced, compared with take-off weight, since most of the fuel which represents a large proportion of the all-up-weight has been used up. The highest lift coefficient which can be used for the shortest take-off depends very decisively on thrust/weight ratio.

Whilst it is not disputed that benefits would result, especially for smaller aircraft, from a successful marriage between boundary layer control for low drag and boundary layer control for high lift I should like to emphasize a different possibility of utilizing the suction blowers of a laminarized aircraft.

By feeding the suction pumps with ram air on take-off and using them as propulsive ducted fans substantial increase of take-off thrust can be obtained, perhaps up to 30% with by-pass engines and 50% or more with ordinary jets, if the ratio of pump shaft power to basic engine thrust power were increased.

The use of such fully integrated engine-pump units of relatively low mean jet velocities will greatly help to reduce the noise made by large jet airliners at take-off.

On an airliner I would, therefore, give preference to using suction blowers in this fashion than in the manner suggested by Dr. Raspet, i.e., for increasing the maximum lift coefficient.