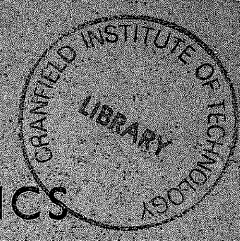


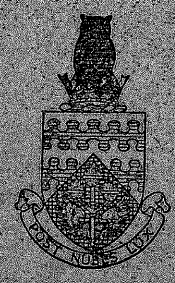
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THE COLLEGE OF AERONAUTICS
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SOME THOUGHTS ON SLENDER ALL WING
SUPERSONIC AIRLINERS

by

J. SPILLMAN



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Some thoughts on slender, all wing,
supersonic airliners

- by -

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SUMMARY

The design of slender wing, supersonic airliners has been considered from the viewpoint of obtaining maximum space utilization. A relationship between direct operating cost on trans-atlantic services and space utilization has been established, which shows, as might be expected, that the direct operating costs decrease as the utilization factor increases.

A penalty associated with a high utility factor is a high wing loading. This leads to the necessity of using auxiliary lift when high utilization factors are obtained. It is shown that a propulsive engine modified to give jet lift at landing and possibly take-off is likely to be the best means of obtaining auxiliary lift.

The optimum cruise height is less than that corresponding to maximum lift drag ratio because of the weight penalty associated with providing adequate thrust.

The integrated layout is not suitable for airliners required to carry less than a hundred passengers, but becomes extremely attractive for a very large number of passengers. In this case the central part of the wing area should be of constant depth with a cabin of side by side multi-bubble form.

The delta planform is not ideal for an integrated layout. Better space utilization can be obtained using a pointed pear shaped planform. Approximate calculations suggest that direct operating costs of a 120 seat airliner can be reduced by as much as 25% by using this type of layout.

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SYMBOLS

a	Thrust weight ratio of lifting engines	
C	Direct operating cost - pence per short ton statute mile	
C_L	Lift coefficient	
D	Total drag	lb.
D_{WW}	Wave drag of wing	lb.
D_Z	Total zero-lift drag	lb.
D_I	Drag associated with lift	lb.
F	Maximum cross-sectional area of wing	ft ²
K_E	Engine installed weight per unit thrust divided by ambient pressure ratio	
K_I	Constant associated with drag due to lift	
K_Z	Zero lift drag coefficient	
K_1	Ratio of fuel, trapped fuel and fuel system weight to useable fuel weight	
K_2	Rate of change of W_R with gross weight	
K_3	Constant in weight equations	
K_4	Constant associated with cost of lifting engines	
L	Lift	lb.
l	Overall length of aircraft	ft.
M	Mach Number	
m	Wing loading associated with the required approach speed	lb/ft ²
n	Number of decks to cabin	
P	Ratio of ambient pressure to that at sea level	
$P_{S.L.}$	Ambient pressure at sea level	lb/ft ²
$q_{S.L.}$	Dynamic pressure at sea level at appropriate Mach No.	lb/ft ²
R	Equivalent cruise range, allowing for fuel used in take-off, climb, descent and diversions	Statute miles
r	Reference value of ambient pressure ratio divided by wing loading	ft ² /lb.
S	Gross wing area	ft ²
T	Engine thrust	lb.
W_A	Equipped airframe weight	lb.

SYMBOLS Continued.

W_C	Weight of items which do not vary with aircraft size	lb.
W_E	Installed weight of propulsive engines	lb.
W_F	Useable fuel weight	lb.
W_G	Aircraft gross weight	lb.
W_L	Aircraft landing weight	lb.
W_P	Payload weight	lb.
W_{PE}	Furnishing and equipment weight which is proportional to payload	lb.
W_R	Weight of power services, undercarriage, airframe equipment and furnishings, crew	lb.
W_S	Structure weight less undercarriage weight	lb.
y	Thrust/Weight ratio at S.L.S. conditions of orthodox engine	
y_s	Thrust/Weight ratio at S.L.S. conditions of special engine	
γ	Ratio of specific heats for air	
η	Volumetric efficiency defined as ratio of plan area of cabin to gross wing area	
w	Wing loading	lb/ft ²
w_P	Weight of payload and furnishing and equipment associated with payload per unit plan area of cabin	lb/ft ²

Suffices

m.d.	Minimum drag conditions
L	Condition at landing
S.L.	Conditions at Sea Level
o	Appertaining to original layout
N	Appertaining to new layout

1. Introduction

The size of supersonic aircraft of medium or long range is invariably dictated by the necessity of obtaining sufficient volume in which to stow payload, engines and fuel. An underestimate of the capacity of a projected aircraft can lead to a very marked increase in size before adequate volume is obtained, because an increase in size results in an increase in drag and weight, leading to an increase in fuel required and consequently to an increase in volume. It follows that the maximum utilization of volume is an absolute necessity for a successful aircraft. The supersonic airliner is particularly difficult from this point of view because passenger comfort imposes severe limitations on size and upon density of loading.

A considerable amount of work has been done on the aerodynamic aspects of supersonic aircraft. This has led to the M wing, delta and gothic wing configurations with simple cross-sectional shapes. The object of this note is to tackle the problem from the viewpoint of optimum utilization of volume and minimum structure weight in order to see whether the resulting configurations can be made compatible with efficient aerodynamic characteristics.

An essential of good design is to make any one component do as many jobs as possible. The obvious example of this, in this context, is the integrated design where the wing supplies lift, and stows payload, fuel and engines. However, care must be exercised with this approach otherwise the weight of the component could exceed the sum of the corresponding single-duty components. This would appear to be the case when an integrated design is used for an aircraft which has to carry only a few passengers.

Examples have been used to indicate the quantitative effects of suggested alterations. Detail studies have not been made, but the results suggest a good basis for such studies.

2. Drag

Consider a family of aircraft of slender all-wing layout, designed to cruise at a constant Mach number of the order of 2.5.

The wing wave drag, as illustrated in figure 4 of reference 1 can be expressed as

$$D_{WW} = \text{constant} \times q_{S.L.} P \left(\frac{F}{L}\right)^2 \quad (1)$$

where $q_{S.L.} = \frac{1}{2} \gamma M^2 P_{S.L.}$ lb/ft².

M = Cruising Mach number

$P_{S.L.}$ = Ambient air pressure at sea level lb/ft²

- F = Maximum cross section area of wing ft.²
 ℓ = Centre line chord length of wing ft.
 P = $\frac{\text{ambient air pressure at cruise height}}{\text{ambient air pressure at sea level}}$

If the family have similar plan shapes and thickness/chord distributions, differing only in wing area, then since

$$\left(\frac{F}{\ell}\right)^2 \propto S$$

$$D_{WW} = \text{constant} \times q_{S.L.} P S \quad (2)$$

If the effects of Reynold's number on skin friction drag coefficient are accepted as second order effects, then the skin friction drag is closely given as

$$D_{WP} = \text{constant} \times q_{S.L.} P S \quad (3)$$

The fin size on such a family will be proportional to the wing area, if similar lateral characteristics are required. Thus the drag of the fin can be included by changing the constants in equations (2) and (3).

It follows that the total zero lift drag of any member of the family may be written as

$$D_Z = K_Z q_{S.L.} P S \quad (4)$$

Similarly the drag associated with lift at the cruising speed for any member of the family as

$$D_I = K_I q_{S.L.} P S C_L^2 \quad (5)$$

since the plan form is not a variable. Values of K_I are plotted in figure 5 of reference 1.

It follows that

$$D = D_Z + D_I = q_{S.L.} P S (K_Z + K_I C_L^2) \quad (6)$$

and consequently, at minimum drag conditions

$$\frac{1}{2} \frac{(\omega/P)_{m.d.}^2}{q_{S.L.}} = C_{L,m.d.}^2 = \frac{K_Z}{K_I} \quad (7)$$

and

$$\left(\frac{L}{D}\right)_{m.d.} = \frac{1}{2\sqrt{K_Z K_I}} \quad (8)$$

$$D_{m.d.} = 2 q_{S.L.} P S K_Z \quad (9)$$

At any other flight condition

$$\frac{L}{D} = \frac{q_{S.L.} \omega / P}{K_Z q_{S.L.}^2 + K_I \left(\frac{\omega}{P}\right)^2} \quad (10)$$

and

$$D = q_{S.L.} P S \left[K_Z + \frac{K_I}{2 q_{S.L.}} \left(\frac{\omega}{P}\right)^2 \right] \quad (11)$$

where ω is the wing loading at the conditions of flight considered.

3. Weight

The gross weight of any member of the family under consideration can be expressed as

$$W_G = W_P + W_{PE} + W_S + W_R + K_1 W_F + W_E + W_{LE} \quad (12)$$

where

W_P = Payload

W_{PE} = Furnishings and equipment weight which is proportional to the payload

W_R = Weight of power services, undercarriage, airframe equipment, furnishings and crew

W_S = Structure weight less undercarriage weight

W_F = Useable fuel weight

W_E = Installed weight of propulsive engine

W_{LE} = Installed weight of lifting engines and extra fuel required for jet lift

K_1 = Ratio of fuel, trapped fuel and fuel system weight to useable fuel weight

W_R can usually be expressed in the form

$$W_R = K_2 W_G + W_C$$

where K_2 is a constant

W_C is the weight of items which do not vary with size or weight of the aircraft, e.g. flying crew, radio.

It follows that

$$W_G = \left(\frac{1}{1 - K_2} \right) (W_P + W_{PE} + W_S + W_C + K_1 W_F + W_E + W_{LE}) \quad (13)$$

and hence the landing weight

$$W_L = \left(\frac{1}{1 - K_2} \right) (W_P + W_{PE} + W_S + W_C + (K_1 + K_2 - 1) W_F + W_E + W_{LE}).$$

$$\text{Let } (K_1 + K_2 - 1) W_F = K_3 W_L \quad (14)$$

Now $K_1 + K_2 - 1$ is usually of the order of 0.12.

Consequently if approximate values for range, lift drag ratio and specific fuel consumption are used to determine $\frac{W_F}{W_L}$, it is possible to

find a value of K_3 which will be sufficiently close to the correct value as to not materially influence the final conclusions, since K_3 will be small compared to $(1 - K_2)$.

$$\text{Hence } W_L = \frac{(W_P + W_{PE} + W_S + W_C + W_E + W_{LE})}{(1 - K_2 - K_3)}$$

or

$$W_L = \left[\frac{W_P + W_{PE} + W_S + W_C}{1 - K_2 - K_3 - \frac{W_E}{W_L} - \frac{W_{LE}}{W_L}} \right] \quad (15)$$

and hence

$$a_L = \left[\frac{\frac{W_P + W_{PE}}{S} + \frac{W_S}{S} + \frac{W_C}{S}}{1 - K_2 - K_3 - \frac{W_E}{W_L} - \frac{W_{LE}}{W_L}} \right]$$

The density of packing of passengers and associated furnishings can be taken as constant for a given standard of passenger comfort. Since the height of the cabin is also a constant, then the wing area in which the payload is contained is directly proportional to the payload. Since W_{PE} is also proportional to payload, then

$$\frac{W_P + W_{PE}}{S} = n \omega_P \eta \quad (16)$$

where ω_P is weight of passengers and associated equipment per unit area of wing covering cabin

n is the number of passenger decks

η is the ratio of wing area covering cabin to gross wing area

It should be noted that η is a measure of the efficiency with which the total volume of the wing is utilized for passenger accommodation, since the cabin height required is constant.

Therefore

$$\omega_L = \left[\frac{n \omega_P \eta + \frac{W_S}{S} + \frac{W_C}{S}}{1 - K_2 - K_3 - \frac{W_E}{W_L} - \frac{W_{IE}}{W_L}} \right] \quad (17)$$

If the range specified is adjusted to include an allowance for the fuel used in normal take-off, climb, diversion and let-down, the landing weight W_L will be the same as the end of cruise weight.

Turbo-jet engines operating in the stratosphere at a fixed r.p.m. and Mach number, have the characteristic that thrust divided by relative pressure is almost constant. Thus, for a given type of engine

$$\frac{W_E}{T/P} = K_E = \text{constant}$$

$$\therefore W_E = \frac{K_E}{P} \times D \quad (18)$$

$$\therefore \frac{W_E}{W_L} = \frac{K_E}{\frac{P_L}{L_D}} \quad (19)$$

Equation (18) can also be written, using eqn.(11), as

$$W_E = K_E q_{S.L.} S \left[K_Z + \frac{K_I}{2} \left(\frac{\omega}{P} \right)_L^2 \right] \quad (20)$$

or

$$\frac{W_E}{W_L} = \frac{K_E q_{S.L.}}{\omega_L} \left[K_Z + \frac{K_I}{2} \left(\frac{\omega}{P} \right)_L^2 \right] \quad (21)$$

Substituting $\frac{W_E}{W_L}$ from eqn.(19) in eqn.(17)

$$\omega_L = \frac{\left[n \omega_P \eta + \frac{W_S}{S} + \frac{W_C}{S} \right]}{\left[1 - K_2 - K_3 - \frac{K_E}{P_L L/D} - \frac{W_{LE}}{W_L} \right]} \quad (22)$$

In order to compare aircraft of similar geometrical layout but of differing wing loading it is necessary to assume that they all fly at the same ratio of $C_L/C_{L,opt.}$ that is at the same L/D . It follows, from equation (10) that

$\frac{\omega_L}{P_L}$ must remain constant,

$$\text{or } P_L = r \omega_L \quad (23)$$

where r is the reference value of P_L/ω_L .

If lifting engines are required solely to provide sufficient lift to allow an aircraft to approach at the same speed as that of an aircraft of lower wing loading, then,

$$\frac{W_{LE}}{W_L} = a \left(1 - \frac{m}{\omega_L} \right) \quad (24)$$

where a is the lifting engine installed weight per pound of lift

m is the wing loading associated with the required approach speed.

Substituting equations (23) and (24) in equation (22)

$$\omega_L = \left[\frac{n \omega_P \eta + \frac{W_S}{S} + \frac{W_G}{S} + \frac{K_E}{I/D^r} - am}{1 - K_2 - K_3 - a} \right] \quad (25)$$

4. Cost

If it is assumed that

- (a) Cruising Mach number is 2.6
- (b) Aircraft annual utilization is 2500 hours
- (c) Fuel costs 18^d/gallon
- (d) Airframe and engine cost £20/lb. weight
- (e) Aircraft operates at 100% load factor
- (f) Aircraft operates on London - New York route

then reference 2 shows that the direct operating costs, in pence per short ton statute mile, can be estimated closely by the equation

$$C = 1.04 \frac{W_F}{W_P} + 0.96 \frac{W_A}{W_P} + 2.56 \frac{W_E}{W_P} \quad (26)$$

where W_A is the equipped airframe weight.

It is reasonable to suppose that the extra cost due to installing lifting engines can be expressed in the form $\frac{K_4 W_{LE}}{W_P}$, where K_4 will have a value somewhere between 1.04 and 2.56. A reasonable value of K_4 is 2.3 assuming that the lift fuel required is about one fifth of the lifting engine weight.

Thus eqn.(26) becomes

$$C = 1.04 \frac{W_F}{W_P} + 0.96 \frac{W_A}{W_P} + 2.56 \frac{W_E}{W_P} + 2.3 \frac{W_{LE}}{W_P} \quad (27)$$

Now $W_L = W_A + W_P + W_E + W_{LE}$.

Substituting for W_E from equation (19) and re-arranging, gives

$$W_A = W_L \left(1 - \frac{K_E}{P_L L/D} - \frac{W_{LE}}{W_L} \right) - W_P \quad (28)$$

Combining equations (27) and (28) yields

$$(C + 0.96)W_P = W_L \left[1.04 \frac{W_F}{W_L} + 0.96 + \frac{1.60 K_E}{P_L L/D} + \frac{1.34 W_{LE}}{W_L} \right]$$

Incorporating equations (23) and (24) in this equation gives

$$(C + 0.96) W_P = W_L \left[\frac{1.04 W_F}{W_L} + 0.96 + \frac{1.60 K_E}{r \omega_L L/D} + 1.34a \left(1 - \frac{m}{\omega_L} \right) \right]$$

Now the wing area is given by

$$S = \frac{W_P + W_{PE}}{n \omega_P \eta}$$

Hence

$$\frac{(C + 0.96)W_P}{W_P + W_{PE}} = \frac{1}{n \omega_P \eta} \left[\left(1.04 \frac{W_F}{W_L} + 0.96 + 1.34a \right) \omega_L + \frac{1.60 K_E}{r L/D} - 1.34 a m \right] \quad (29)$$

5. Example of the effect of volumetric efficiency, η , on landing wing loading and direct operating cost

Consider an aircraft without jet lift engines. Current design studies suggest the approximate values given below, for a trans-atlantic airliner designed to cruise at a Mach number of 2.6,

$$\omega_P = 40 \text{ lb/ft}^2 \quad (\text{single deck})$$

$$\frac{W_C}{S} = 1.0$$

$$K_1 = 1.05$$

$$K_2 = 0.075$$

$$\frac{L}{D} = 7.0$$

$$K_E = 0.08$$

$$\text{s.f.c.} = 1.7 \text{ lb/lb. hour}$$

$$R = 5000 \text{ statute miles}$$

It follows that, assuming a climb-cruise in the stratosphere,

$$\frac{W_G}{W_L} = 2.03$$

$$\text{i. e.}, \quad \frac{W_F}{W_L} = 1.03$$

$$\text{Hence } K_3 = 0.125$$

Figure 7a of reference 3 suggests that, for B.O.A.C. operations assuming an approach speed 22 knots in excess of the stalling speed in the approach configuration, the maximum value of $\frac{w_L}{C_L \text{ max.}}$ is 85 lb/ft².

For a slender delta aircraft, such as those considered in references 1, 4 and 5, the maximum useable lift coefficient, assuming a limiting incidence of 15°, is of the order of 0.5.

Thus the maximum wing loading at landing is about 42 lb/ft².

Assume that, at this wing loading, the optimum height to cruise is 67,000', e.g. $P_L = 0.05$

$$\text{then } \frac{1}{r} = 840 \text{ lb/ft}^2.$$

Current supersonic aircraft with delta wings of about 0.04 thickness-chord ratio have wing weights of the order of 8 lb/ft². The aircraft considered in this note will have wings of similar thickness chord ratio but will have a smaller aspect ratio, a larger sweep, a greater relief loading, a much larger area and will be subject to higher temperatures. By making approximate allowance for these effects, it was estimated that the value of $\frac{W_S}{S}$ would probably be between 10 lb/ft² and 14 lb/ft².

Assume that $\frac{W_S}{S} = 12 \text{ lb/ft}^2$.

Substituting the values assumed above in equation (25) gives

$$\omega_L = 50 \eta + 28.25 \quad (30)$$

This suggests that the maximum value of η that can be used, without resorting to special devices to reduce the approach speed, is 0.275.

The layouts suggested in references 1, 4 and 5 give values of η of 0.22, 0.26 and 0.18 respectively. In the first two cases the permissible cabin area is simply defined by the contour giving sufficient cabin height without any consideration of the problem of pressurization. The required differential pressure for aircraft such as these will be over 10 p.s.i. and consequently a considerable weight penalty will ensue if the cabin is not of near-cylindrical or multi-cylindrical form, (e.g. double bubble).

In all these layouts, the cabin is situated at a considerable distance forward of the aircraft centre of gravity. As a consequence a marked C.G. shift will occur with variation of load factor. This suggests that large controls and trim drag may result if loading restrictions are not to be severe.

It seems probable, therefore, that values of η in excess of 0.25 are unlikely if the simple delta planform with orthodox thickness distributions is adopted. Auxiliary lifting devices are unlikely to be required in this type of aircraft.

The use of extended wing area devices, as proposed in reference 3, could well defeat their own object, since, for efficient cruise conditions these auxiliary surfaces must be stowed in the basic wing, and almost certainly, in the region near the centre of gravity. This is likely to prevent that part of the wing from being used as a cabin and, consequently, the volumetric efficiency, and so the landing wing loading will tend to decrease. If this happens the necessity for auxiliary lifting devices is reduced.

Substituting the values previously assumed in this section, together with that for ω_L from equation (30) in equation (29) gives

$$\frac{(C + 0.96)W_P}{W_P + W_{PE}} = 2.54 + \frac{1.82}{\eta} \quad (31)$$

Since $\left(\frac{W_P + W_{PE}}{W_P}\right)$ is likely to be about 1.6 an increase in η from 0.2 to 0.275 reduces the direct operating cost from about 17.6^d to 13.7^d .

This illustrates the importance of attaining a high volumetric efficiency. The variation of cost with η is shown in Figure 1.

6. Aircraft with jet-lift devices

In view of the marked reduction in direct operating costs with increase in volumetric efficiency, η , indicated in the previous section, it appears worthwhile examining the possibility of using values of η greater than those corresponding to the simple aircraft considered above.

A larger value of η can only be achieved with the penalty of a larger value of landing wing loading. It follows that, if the approach speed is fixed, some extra lift producing device must be used. Since extending wing or flap methods appear to defeat their own objective, the extra lift must come from vertical thrust.

6.1. Aircraft with auxiliary lift engines

Equations (25) and (29) give the landing wing loading and the direct operating cost in general terms for this type of aircraft. However, it is necessary to resort to the example considered in section 5 in order to illustrate the effect of installing lifting engines.

Assuming the same values as in section 5, equation (25) gives

$$\omega_L = \left[\frac{40 \eta + 22.6 - 42a}{0.8 - a} \right] \quad (32)$$

and equation (29) yields

$$\frac{(C + 0.96)W_P}{W_P + W_{PE}} = \frac{1}{40 \eta} \left[(2.03 - 1.34a) \omega_L + 15.37 - 56.3a \right] \quad (33)$$

Now, reference 2 suggests that the installed weight of lifting engines and associated equipment is 0.19 times the lift developed. Again the fuel required for these engines is 0.04 times the lift increment.

Thus $a = 0.23$, in which case equations (32) and (33) give

$$\omega_L = 70.2 \eta + 22.7 \quad (34)$$

$$\frac{(C + 0.96)W_P}{W_P + W_{PE}} = 4.10 + \frac{1.386}{\eta} \quad (35)$$

Thus, assuming again that $\frac{W_P + W_{PE}}{W_P} = 1.6$, an increase in η

from 0.275 to 0.35 corresponds to a reduction in operating cost from 13.7^d to 11.9^d. The cost variation with η is shown in figure 1.

This system, however, has the weakness that useful volume has to be used to stow the lifting engines, so making it difficult to achieve high values of volumetric efficiency.

6.2. Slender delta aircraft with special propulsive engines

In a turbo-jet engine about one half to two thirds of the energy in the gases after combustion is used to drive the compressor. Suppose that about this fraction of the air passing through the compressor is ducted into a separate combustion chamber, turbine and jet pipe, such that the thrust from this part of the system is small, say comparable to that of a turbo-propeller engine. The remaining air from the compressor is ducted into a separate combustion chamber and nozzle, which provides the major part of the thrust from the engine. In normal operations this engine could give at least as great a thrust, and probably a greater thrust, than an orthodox turbo-jet of the same mass flow and compression ratio, since the maximum temperature of combustion in the main thrust part of the system is not limited by maximum turbine temperatures. However, the system would be heavier because of the duplication of combustion chambers, jet pipes, nozzles and cooling.

This type of engine has a particular advantage in that the compressed air which does not pass into the turbine system can be ducted into additional downward facing combustion chambers and nozzles, and so utilized to produce vertical thrust. A relatively simple valve in the compressed air duct would be sufficient to allow a rapid change from horizontal to vertical thrust from any one engine. The fact that only one third to one half of the engine mass flow, in its fully compressed state, is required to be ducted, suggests that the size of the ducting required will be relatively small and, since temperatures will not be excessive, relatively light in comparison with equivalent jet pipe weights.

It seems reasonable to suppose that such an engine could be designed so that the thrust and specific fuel consumption per unit mass flow are the same as those for an orthodox turbo-jet engine. However, the total installed weight of the engine would be greater.

If the cost per pound weight of fuel and extra engine weight required to give adequate lift is assumed to be the same as for the lifting engines and fuel considered in the previous section, it follows that the better scheme will be the one requiring the smaller additional weight to provide jet lift.

In order to compare this system with those considered previously, the same example can be used, with the same assumptions as before, including a fuel allowance of 4% of the jet lift. In addition the lift from a special engine will be taken as 80% of the S.L.S.T. that can be developed by the engine.

$$\text{Lift required} = S(\omega_L - 42)$$

$$\text{Equivalent thrust (S.L.S.)} = 1.25S(\omega_L - 42)$$

If y is the thrust weight ratio, under sea level, static conditions, of the orthodox engine and y_s that for the special engine, then the weight by which the installation with special engines exceeds that for a normal installation of the same thrust, ΔW_E is

$$\begin{aligned} & 1.25S(\omega_L - 42)\left(\frac{1}{y_s} - \frac{1}{y}\right) + 0.04S(\omega_L - 42) \\ &= S(\omega_L - 42)\left(\frac{1.25}{y_s} - \frac{1.25}{y} + 0.04\right) \end{aligned}$$

$$\text{Hence } \frac{\Delta W_E}{W_L} = \left(1 - \frac{42}{\omega_L}\right)\left(\frac{1.25}{y_s} - \frac{1.25}{y} + 0.04\right) \quad (36)$$

Since ΔW_E includes duct and fuel weights it is reasonable to cost it at the same rate as the lifting engines considered in the previous section. Thus equations (32) and (33) may be used with

$$a = \left(\frac{1.25}{y_s} - \frac{1.25}{y} + 0.04\right)$$

A typical value for y is 4, in which case

$$a = 0.31 \left(\frac{y}{y_s} - 1\right) + 0.04$$

For this method to yield a better direct operating cost than that considered in the previous section the value of a must be smaller.

$\frac{y}{y_s} = 1.61$, corresponds to the value of $a = 0.23$ used in the previous section.

This means that the installation using some special engines will give a lower direct operating cost than that using separate jet lift engines provided the special engine weighs less than 161% of the standard propulsive engine of the same thrust. It seems very probable that this will be the case, since the complete fuel system, combustion chamber, jet pipe and nozzle weight of an orthodox engine is only of the order of 25% of the total engine weight.

7. Optimum cruise condition for a slender aircraft

Consider an aircraft which has to cruise at a given Mach number. If the aircraft cruises at a height corresponding to maximum lift/drag ratio, then the thrust and consequently fuel required will be a minimum, but it does not follow that this yields the lowest direct operating costs. If the aircraft were to fly at the same Mach number at a lower altitude, it would be flying at a speed greater than the minimum drag speed and consequently at a lower lift-drag ratio. Although the thrust required would be greater than in the previous case, the engine weight to provide that thrust may be lower because of the reduction in altitude. The reduction in direct operating costs resulting from a reduction in engine size may more than offset the increase in costs resulting from greater fuel load. A general solution to this problem of finding the optimum cruising height is extremely complex, partly because of the awkward relationship between fuel required and cruise height and partly because the reduction in costs from reduction in engine size and the increase in cost of fuel and tanks are not independent parameters. They are linked because both depend upon the landing weight, W_L , which is a function of both parameters.

To illustrate the point, however, it is worthwhile calculating the effect for a particular case.

Suppose that, for the aircraft under consideration, $K_I = 0.75$ and $K_Z = 0.0060$.

$$\text{Then } C_{L_{md}} = \sqrt{\frac{K_Z}{K_I}} = 0.0895$$

$$\left(\frac{L}{D}\right)_{md} = \frac{1}{2\sqrt{K_Z K_I}} = 7.45.$$

$$\text{If } M = 2.6$$

$$\text{Then } q_{S.L.} = \frac{1}{2} \gamma M^2 P_{S.L.} = 10,000 \text{ lb/ft}^2.$$

$$\therefore \left(\frac{\omega}{P}\right)_{opt} = 895 \text{ lb/ft}^2.$$

$$\text{If } \omega_{opt.} = 42 \text{ lb/ft}^2.$$

$$\text{then } P_{L_{opt.}} = 0.0469.$$

Assuming no auxiliary lifting devices, equation (27) gives

$$W_P C = 1.04 W_F + 0.96 W_A + 2.56 W_E.$$

If height is varied, then W_A will not change except by the weight of the fuel tanks and system. Thus we may write

$$\begin{aligned} W_P C - \text{constant} &= [1.04 + 0.96 (K_1 - 1)] W_F + 2.56 W_E \\ &= \left[[1.04 + 0.96 (K_1 - 1)] \left[10 \frac{R(\text{s.f.c.})}{1520 \text{ ML/D}} - 1 \right] + \frac{2.56 K_E}{P_L \frac{L}{D}} \right] W_L \quad (36) \end{aligned}$$

If $W_P = 29,200$ lb as suggested in reference 2

$$\frac{W_P + W_{PE}}{W_P} = 1.6$$

$$n \omega_P = 40 \text{ lb/ft}^2.$$

Therefore $\eta = 0.275$

$$S = 4250 \text{ ft}^2.$$

$$W_{L_{\text{opt}}} = \omega_L S = 178,500 \text{ lb.}$$

If $K_E = 0.08$, then engine installed weight equals 40,900 lb. and the fuel weight, assuming range and s.f.c. of 5,000 statute miles and 1.7 lb/lb.hr. respectively, equals 168,000 lb.

$$\text{If } K_1 = 1.05$$

Weight of fuel tanks and system = 8,400 lb.

Thus landing weight less engines, fuel tanks and system = 129,200 lb.

$$\text{Hence } W_L = 129,200 + \frac{0.08}{P_L \frac{L}{D}} W_L + 0.05 \left(10 \frac{2.15}{L/D} - 1 \right) W_L$$

∴

$$W_L = \frac{129,200}{1 - \frac{0.08}{P_L \frac{L}{D}} - 0.05 \left(10 \frac{2.15}{L/D} - 1 \right)}$$

Thus equation (37) can be written as

$$W_P C - \text{constant} = \left[1.098 \left(10^{\frac{(2.15)}{L/D}} - 1 \right) + \frac{0.205}{P_L L/D} \left[\frac{129,200}{1 - \frac{0.08}{P_L L/D} - 0.05 \left(10^{\frac{(2.15)}{L/D}} - 1 \right)} \right] \right] \dots (38)$$

From eqn. (10)

$$\frac{L}{D} = \frac{\omega_{L/P_L}}{60 + 0.75 \times 10^{-4} \left(\frac{\omega_L}{P_L} \right)^2} \quad (39)$$

and since $S = 4250 \text{ ft}^2$.

$$\omega_L = \left[\frac{30.4}{1 - \frac{0.08}{P_L L/D} - 0.05 \left(10^{\frac{(2.15)}{L/D}} - 1 \right)} \right] \quad (40)$$

The constant on the L.H.S. of equation (38) is equal to 0.96 times the landing weight less fuel tanks and system and payload, i.e. 96,000.

Equations (38), (39) and (40) can be solved to yield C, the direct operating cost in pence per short ton statute mile, for various values of P_L , the relative pressure at end of cruise. Figure 2 shows the results.

This example shows that a 6% reduction in direct operating costs can be achieved by cruising at a relative pressure 1.33 times that corresponding to minimum drag cruise.

This result is analogous to that obtained for the optimum cruising speed of an aircraft except that no general solution is possible.

It is worth noting that the conditions for maximum landing weight and maximum take-off weight will correspond to values of relative pressure somewhere between those for minimum drag cruise and minimum cost cruise. It follows that, if the engine size is dictated by either of these conditions, the optimum cruise height will be that corresponding to minimum weight at the critical flight phase. In the example, the total thrust at sea level of the aircraft designed for minimum cost cruise is only 75% of that for the aircraft designed to cruise at minimum drag conditions.

If the required landing speed is very low, the jet lift required could well be greater than that available from the cruise engines of the scheme suggested in the previous section.

Although it has been presumed that the only weight change which will occur as a result of changing the cruise altitude is that of engines, fuel system and fuel, based on a constant effective range, it should be realised that this is not strictly correct. The air conditioning system is likely to be less complex as a result of the lower operating altitudes and the fuel for climb and let-down will be smaller, that is a smaller effective range should be used. On the other hand, some slight increase in skin temperature is probable and the increase in equivalent air speed at cruise might require a stiffer structure but it is doubtful whether these last two effects are significant.

8. Effect of the number of passengers on the aircraft layout

Suppose that the aircraft is required to cruise at a Mach number of the order of two over a specified range. Assume that the approach condition for landing is critical on wing loading.

If the aircraft is required to carry only one or two passengers, then the cross-section of the cabin will be that giving the minimum area in which a passenger can be seated with the required standard of comfort. The compromise between wave drag, skin-friction drag and fuselage weight will yield a cross-sectional area distribution along the length of the aircraft approximating to a body revolution of a particular diameter to length ratio. (Probably of the order of 0.08). Since the body alone will yield negligible lift on the approach, wings will be required of quite small area relative to the body since the body will not be densely packed. The wing sweep and aspect ratio can be determined by a compromise between wing lift, cruise drag and weight, since the fuselage aft of the passenger cabin can be shaped to give the required overall area distribution. The resulting aircraft will be similar to that shown in figure 3a. The cabin cross-section is likely to be elliptic rather than circular, since height required will be greater than the width, because the overall saving

in size of aircraft will more than offset the weight penalty of pressurizing the small length of cabin. The section is shown in figure 4a.

As the number of passengers increases above two or three, the fuselage density will increase in spite of a slight increase in length. This will mean bigger wings and the indentation of the fuselage becomes a limitation on the number of passengers which can be accommodated. To reduce this problem, a highly swept wing or delta wing is attractive. This leads to the layout suggested in figure 3b.

Above about twelve passengers the cabin and fuselage length is likely to be such that a better overall compromise between weight and drag results if two abreast seating is used. As the passenger numbers increase so, three, four and five abreast seating becomes, in turn, the most efficient layout. The height of the cabin need not increase as the number of seats abreast increase except from the viewpoint of cabin weight in association with pressurization. In this respect three abreast seating corresponds closely to a circular section of about 9 feet diameter (see figure 4b). For higher numbers abreast, the best compromise between minimum cross-sectional area and low cabin structure weight will lead to supported arch construction of the roof and floor. This will tend to require an increase in overall cabin depth, but not as much as a circular section would give.

The greater the number of passengers sitting abreast, the higher the fuselage density of loading, since the percentage of space taken by the aisle gets smaller. However, beyond five or six abreast, the roof and floor arch spans become large and consequently the required overall cabin depth tends to increase considerably. It follows that the total cross-sectional area of two, three abreast cabins side by side, could be of the same order as one six abreast cabin, in spite of the duplication of aisles. This trend in cabin shape is illustrated in figures 4c and 4d. For six or more seats abreast the scheme shown in figure 4d obviously gives a lighter structure and a smaller cross-sectional area.

The fuselage tends to become more densely loaded as the number of passengers increases and consequently, the wing will tend to get larger relative to the body. Thus the overall length of the body will need to be increased not only to allow a longer cabin length but also to allow for the increase in cross-sectional area of the wing. One way of minimising this increase is to reduce the span of the wing near the apex and trailing edge regions, allowing the fuselage to be locally fatter and so allowing a longer cabin. This will tend to reduce the mean sweep of the wing, an effect which is not likely to be very critical because of the low aspect ratio. It appears at first sight that this modification will reduce the aspect ratio, but this is not necessarily so. The total wing area must increase as a result of the increase in weight due to added fuselage size and weight. As the fuselage length increases the maximum cross-sectional area will increase. Since the area required for

passenger accommodation at this section will remain constant, more cross-sectional area can go into the wings. Consequently the wing span at this section will tend to increase, whilst the root chord tends to be reduced. Figure 3c illustrates this development.

When only a few passengers are carried it is possible to allow the wing centre section to pass behind, above or below the cabin. However, as the wing size increases relative to the fuselage, the thickness chord ratio must be reduced in order to reduce the cabin, wing structure interference. Above a certain size this solution becomes excessively heavy and the cabin structure must carry the wing centre-section loads. This leads to the multi-spar and heavy fuselage frame construction used in some guided weapons, or to a series of load carrying bulkheads dividing the length of the cabin. Both these solutions tend to increase the size of the cabin.

If the root depth of the wing is not vastly different from the depth of the cabin, it may be worthwhile decreasing the aspect ratio of the wing, possibly by increasing the sweep of the leading edge, in order to increase its thickness sufficiently to allow integration of the top and bottom skins with the top and bottom cabin structure over the structural box region of the wing. To allow this, the root chord of the wing would have to be flattened over the region of the structural box. It should be realised, however, that decreasing the aspect ratio tends to reduce the lift coefficient obtainable on the approach and hence increase the required wing area for a given approach speed. This blended wing-fuselage layout is shown in figure 3d.

A further increase in the number of passengers requires even greater wing area. This allows the side by side cabin installation to be put inside the wing and the so called integral wing results. Examples of this are illustrated in figures 3e, 3d and 3f. The depth of the wing is kept constant over the regions of the cabin in order to allow a maximum width of cabin for a given cross-sectional area. This is consistent with the requirement for maximum volumetric efficiency considered in the previous sections. The maximum values of volumetric efficiency occur when the maximum cross-sectional area is between 0.60 and 0.65 of centre-line length from the nose.

The number of passengers at which it is worthwhile changing from one layout to the next in the sequence described above can only be ascertained by detailed layouts and calculations. It is certain however, that the fully integrated design is suitable only when a large number of passengers is required to be carried. The planforms shown in figures 3a - 3e are simply sketches to illustrate the trend and are not the result of any detailed layouts.

9. 120 Seat airliner designed to cruise at $M = 2.6$

One layout which has been suggested for a 120 seat supersonic airliner is an all wing delta configuration utilizing parabolic distribution of cross-sectional area along the length and diamond cross-sections (see ref. 5). Using a maximum pressure cabin diameter of 10 feet, the suggested aircraft has a length of almost 200 feet, and a maximum cross sectional area of 180 sq.ft. The maximum diameter region of the cabin is almost 65 feet in length. Figure 5a illustrates this layout.

The length of the 10 ft. diameter cabin can be increased to about 120 ft. by increasing the centre-line depth and reducing the span, over the regions forward and aft of the original cabin, maintaining diamond cross-sections. At the same time, the span over the region of the original cabin can be increased by reducing the centre-line depth to the minimum required to contain the pressure cabin. The resulting configuration is shown in figure 5b. The wing area is approximately half the original and consequently the structure weight will be considerably less than before. The cross-sectional area length distributions are the same, but the aspect ratio is slightly less and consequently the wave drag coefficient will be a little more than twice the original value; the skin friction drag coefficient will be practically as before. Thus the overall zero lift drag coefficient will increase by approximately 33%, since the original wave drag was about half the original skin friction drag. Consequently the L/D at cruise, assuming the same ratio of zero-lift drag to total drag, will be reduced by 15%, and the optimum C_L for cruise increased by approximately 15%. If the same end of cruise weight is presumed then

$$\frac{\text{Original end of cruise ambient pressure}}{\text{End of cruise ambient pressure}} = \frac{C_L S}{C_{L_0} S_0} = 0.58$$

If cruise thrust dictates engine size, then the engine weight required will be reduced accordingly.

The saving in landing weight of structure and engines would approximately balance the increase due to extending the length of the pressure cabin and carrying 220 passengers.

Thus 220 passengers could be carried for approximately the same direct operating cost as the original aircraft with 120 passengers. Whilst it must be agreed that this statement is optimistic in that take off would almost certainly dictate the engine size in this case, and the aircraft stability and control problems would be extremely severe, the result shows that considerable improvements can be made using this approach.

The main problem, however, is to carry 120 passengers and the limiting size of the cabin height prevents any simple scaling process from

being adopted. The number of passengers required to be carried is not sufficient to allow a fully integrated design to be efficient.

At a cruising Mach number of 2.6, the optimum exit area of a turbo-jet engine without reheat is about the same as the basic engine frontal area whilst the intake area is only about 0.3 of this area. If the engines are situated at the rear of the aircraft and all the nozzles are designed to expand the jets to ambient pressure at the exit section, then the wave drag will correspond to that of a truncated body of revolution of the same cross-sectional distribution and a base area equal to the total engine exit area less the cross-sectional area of the core of air entering the intake, measured under ambient conditions.

In the case under consideration the engine exit area will be approximately 87 sq.ft. and the intake core about 27 sq.ft. so the equivalent base area is 60 sq.ft. The maximum cross sectional area is 180 sq.ft. If we presume that the engine exit section corresponds to the position where the original parabolic body cross-sectional area was 60 sq.ft. then the distance between maximum cross-sectional area position and base section is about 65 ft. Figure 6 shows the layout developed in this manner.

The ratio of wave drag of this truncated body to that of the original parabolic body is a little less than 0.9. It is reasonable to suppose that, provided the wings remain slender and of similar aspect ratio, this ratio of wave drag will apply. Comparing the drag coefficients of this aircraft with the original delta layout, which has twice the wing area, the wave drag coefficient will be 1.8 times as much, and the skin friction drag coefficient will be about the same value. Since the original wave drag was only 0.31 of the total zero lift drag, the ratio of zero lift drag coefficients of new to original aircraft will be 1.25.

If the aircraft cruise height is adjusted such that the ratio of zero lift drag to drag associated with lift, is the same as the original configuration, then the cruise L/D will be 0.9 of the original value. For the same range the fuel weight will be 1.18 times the empty weight instead of equal to the empty weight. This is pessimistic since the cruise height will be lower and consequently the fuel used in climb and let-down will be less.

If the value of C_{DZ} is 1.25 times that of the original aircraft and the ratio of zero lift drag to total drag remains the same, the cruise lift coefficient will be 1.12 times that for the original aircraft. If the engine weight is determined by cruise conditions then, since the cruise Mach number is constant, $\frac{W_E}{T/P}$ is constant for a given type of engine,

$$\frac{W_{E_N}}{W_{E_0}} = \frac{W_{L_N}}{W_{L_0}} \cdot \frac{\left(\frac{L}{D}\right)_0}{\left(\frac{L}{D}\right)_N} \cdot \frac{P_0}{P_N}$$

where suffix N corresponds to new aircraft

suffix 0 corresponds to original aircraft.

Now

$$\frac{P_{L_0}}{P_{L_N}} = \frac{W_{L_0} S_N C_{L_N}}{W_{L_N} S_0 C_{L_0}}$$

Hence

$$\begin{aligned} \frac{W_{E_N}}{W_{E_0}} &= 1.25 \frac{S_N}{S_0} \\ &= 0.625. \end{aligned}$$

If the original wing weight is halved, the engine weight reduced to 0.625 of the original value and fin, undercarriage, power supplies, airframe equipment and reserve fuel adjusted to allow for the reduction in size and landing weight, the landing weight reduces to 0.765 of the original value. Since the original landing wing loading was just over 25 lb/ft² the new landing wing loading will be about 39 lb/ft². The aspect ratio of the revised wing is such that for ~~145 knots approach speed~~ a wing loading of up to at least 40 lb/lb² is satisfactory for 11,000 ft. runways.
 29 ft.

The take off weight will be 0.895 times the original value. This means that the C_L at take-off required in order to meet the 11,000 ft. runway requirement is about 0.8 according to the curves given in reference 5. The maximum useable C_L associated with the original delta planform was 0.53. However, the increase in aspect ratio and considerable reduction in taper of the modified planview suggests that a marked improvement in low speed lift curve slope should result, from which it appears that a 0.8 lift coefficient should be obtained with little increase in incidence above the original 15°.

It follows that the direct operating costs of the modified aircraft are only about 75% of those for the original aircraft. The wing area of the modified aircraft was arbitrarily chosen as half the original wing area, and the above calculations suggest that this is of the right size. Much more detailed calculations are required before the optimum wing area can be chosen, but the results indicate the order of the improvements which are possible.

Figure 7 shows an aircraft with the same cross-sectional area distribution but with an even smaller wing area. The propulsive engines are situated such that some of them can be used to give auxiliary lift as suggested in section 6.2., without excessive pitching moment effects. It should be noted that although the wing area is smaller than for the aircraft shown in figure 6, the number of passengers is greater.

10. Conclusions

(1) The direct operating costs of an all-wing supersonic airliner decrease as the fraction of total volume used for passenger accommodation increases. Since cabin height is effectively fixed it follows that the direct operating costs decreases as η , the fraction of cabin plan area to gross wing area, increases.

(2) For a given aspect ratio and longitudinal cross-sectional area distribution, an increase in wing loading is associated with an increase in η . Consequently there is a value of η above which auxiliary lift is required if the approach speed is limited.

It appears that the maximum wing loading at landing associated with a satisfactory approach speed is about 42 lb/ft²., and that this corresponds to a value of η of about 0.275. Current design studies have values of η somewhat below this value, which suggests that auxiliary lift is not required if normal aerodromes are used.

(3) If values of η in excess of 0.275 are achieved, the direct operating cost will decrease, although not to the same extent as before, because of the cost associated with supplying auxiliary lift.

It seems probable that the best way of obtaining small quantities of jet lift at landing is by means of a special propulsive engine in which part of the compressed air can be ducted to auxiliary combustion chambers feeding nozzles directed downwards. This method is likely to weigh less and use less space than auxiliary lifting engines.

(4) The best cruising height depends upon the flight condition which determines the engine size. If the engine size is dictated by the thrust required for cruise, then the best operating height is considerably below that corresponding to maximum lift-drag ratio. If thrust at take-off or landing is critical, it is probable that the best operating altitude is below that for maximum L/D .

(5) An all wing or integrated layout is attractive only when the size is sufficiently large to necessitate a merging of wing and body. It would appear that, for transatlantic aircraft, the change from a wing, body and tail configuration to an integrated layout occurs when the number of passengers to be carried is between 100 and 150.

(6) As the number of passengers increases above 150, the integrated layout becomes increasingly efficient. The best configuration of cabin is a side by side, multi-bubble arrangement leading to a region of constant depth over the central area of the wing.

(7) The delta planform is not ideal because the cross-sectional area over the rear part of the aircraft cannot be utilized for accommodation. A pointed pear-shaped planform appears to be much better especially when used in conjunction with engines at the rear and maximum cross-sectional area positions behind the mid-length position. Approximate calculations indicate that the direct operating costs of a 120 passenger trans-atlantic airliner can be reduced by about 25% if this planform is adopted.

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TABLE I

Comparison of assumptions and costs with those of other studies

Quantity	Ref.1	Ref.2	Ref.4	Ref.5	Sect.9	Sects.5,6,7.
Ideal Wing Wave Drag ft^2	12.0	-	12.0	9.7	8.7	12.0
$c_{SL} P. (\frac{F}{l})^2$						
Zero lift drag coefficient	-	-	-	0.0074	0.00925	0.006
Drag associated with lift C_L^2	0.75	-	0.52	0.51	0.51	0.75
Cruise L/D	6.0 approx.	-	7.1	7.46	6.67	7.0
($\frac{\text{Wing loading}}{C_{L \text{ max.}}}$) approach lb/ft^2	-	-	78	48.5	85	85
C_L at 15° incidence	0.45 approx.	-	0.4	0.53	0.75-0.8	0.5
Cruising Mach number	2.5	2.6	2.15	1.8	1.8	2.6
Gross range S.A.S.M.	5460	5000	5750	5300	5300	5000
K_E	0.094	0.08	-	0.087	0.087	0.08
s.f.c.	1.63	1.7	-	1.27	1.27	1.7
Sea level static thrust Installed Engine Weight	3.0	-	3.0	3.77	3.77	4.0
Payload lb.	26,400	29,200	26,400	26,400	26,400	29,200
$\frac{WP + WPE}{S^2}$ lb/ft^2	-	-	20 approx	39	39	40
Structure less undercarriage weight Gross Wing Area lb/ft^2	-	-	9.3 ^x	6.8	6.8	12.0
W_R	-	$0.075W_G \dagger$ + 4250	$0.071W_G$ + 4250	$0.0715W_G$ + 4250	$0.0715W_G$ + 4250	$0.075W_G$ + 4250
K_1	1.05	-	1.05 ^x	1.04	1.04	1.05
$\frac{WPE}{WP}$	-	0.6	0.6 ^x	0.58	0.58	0.6
Fuel cost pence/I.G.	13.5	18	-	18	18	18
Annual utilization hours	2250	2500	-	3000	3000	2500
Airframe and engine initial cost $\text{£}/\text{lb.}$	£20	£20	-	£20	£20	£20
Cost method	As given in ref.1	As given in ref.2	B.E.A./ B.O.A.C.	B.E.A./ B.O.A.C.	B.E.A./ B.O.A.C.	As given in ref.2
Cost Pence/short ton statute mile	26	-	20.6	16.67	12.6 approx.	13.7
Wing loading at Take off lb/ft^2	86	-	68	51.3	84	86

^x Weights of these items grouped together in reference 4
[†] Assuming U/C weight = $0.045 W_G$ and excluding crew

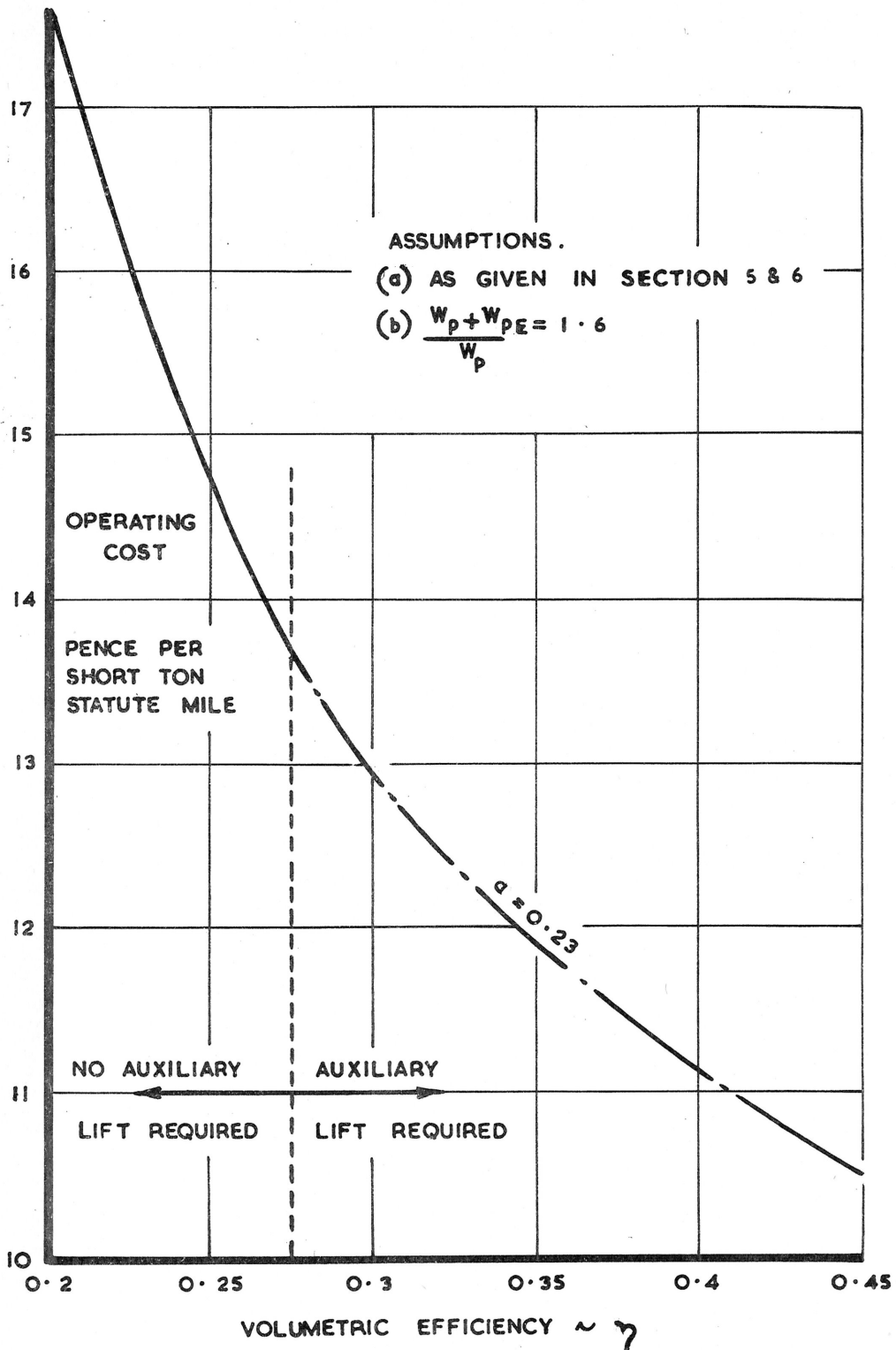


FIG. 1. VARIATION OF DIRECT OPERATING COST WITH VOLUMETRIC EFFICIENCY FOR EXAMPLES USED IN SECTIONS 5 & 6

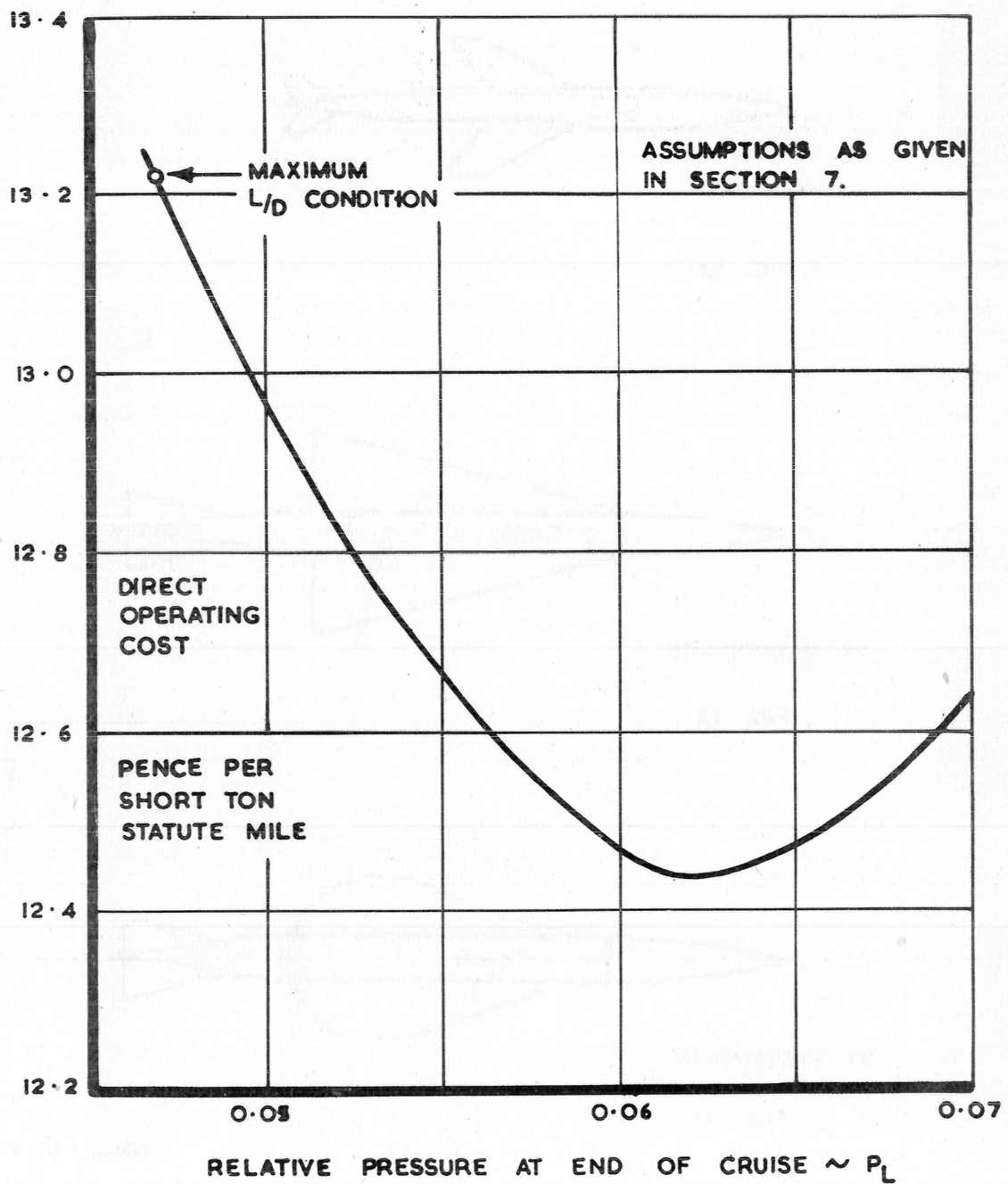
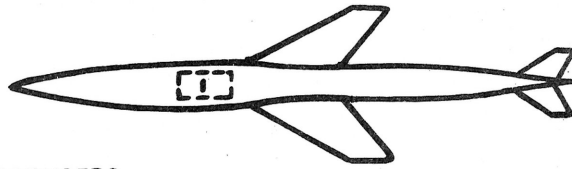
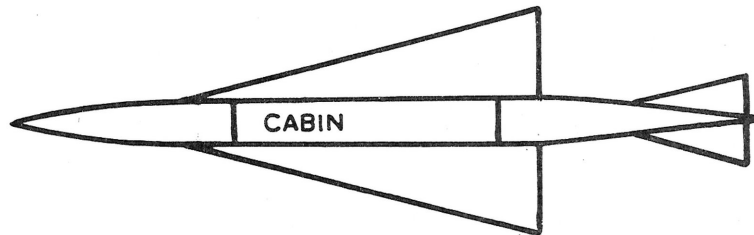


FIG. 2. VARIATION OF DIRECT OPERATING COST WITH RELATIVE PRESSURE AT END OF CRUISE FOR EXAMPLE OF SECTION 7.



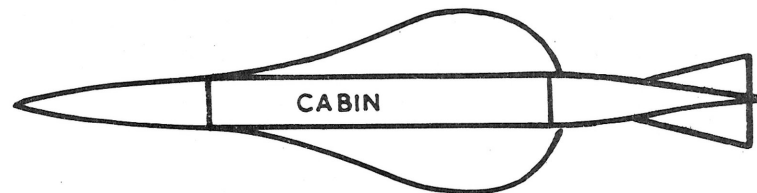
2 PASSENGERS

FIG. 3a.



25 PASSENGERS

FIG. 3b.

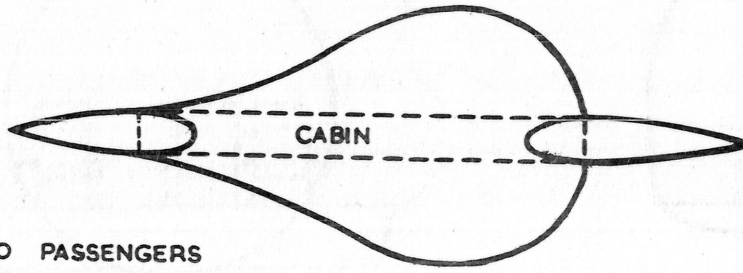


27 PASSENGERS

FIG. 3c.

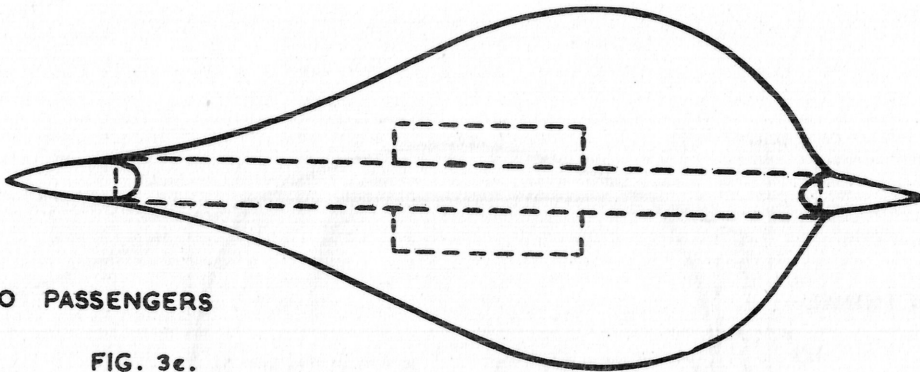
SCALE 1" = 25'

FIGS. 3a. 3b. & 3c. VARIATION OF PLANFORM TO SUIT
VARIATION IN NUMBER OF PASSENGERS



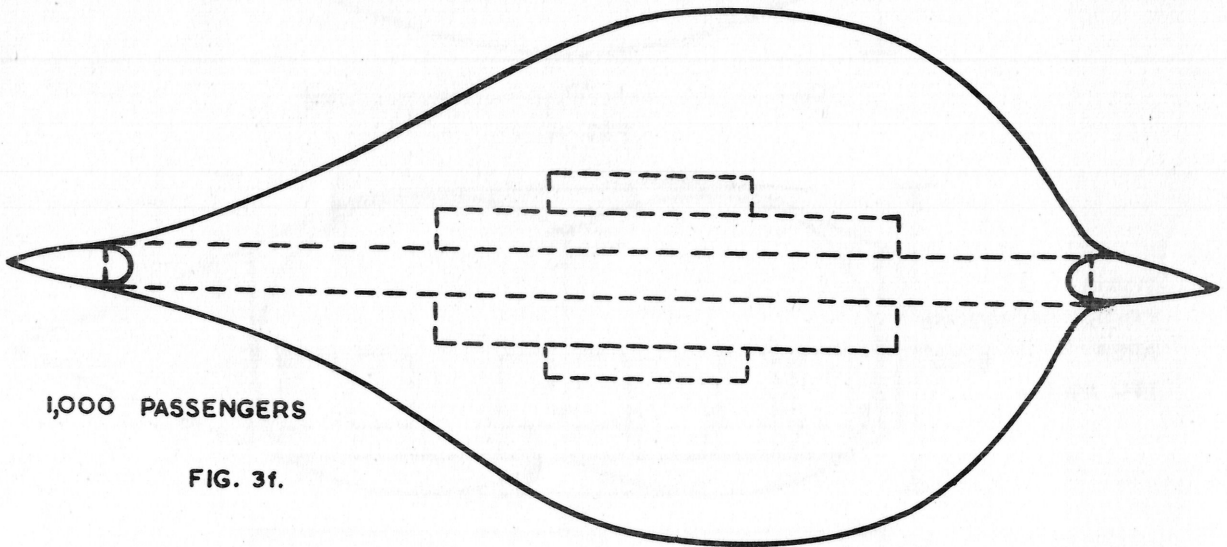
180 PASSENGERS

FIG. 3d.



450 PASSENGERS

FIG. 3e.



1,000 PASSENGERS

FIG. 3f.

SCALE 1" = 50'

FIGS. 3d. 3e. & 3f. VARIATION OF PLANFORM TO SUIT VARIATION IN NUMBER OF PASSENGERS.

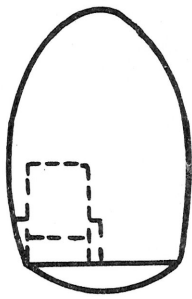


FIG. 4a.

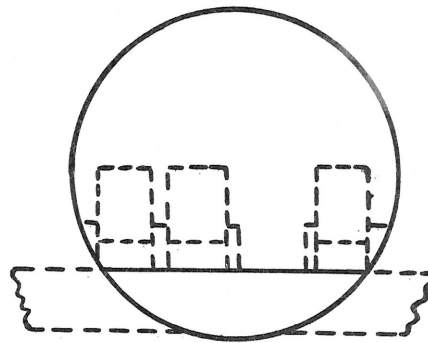
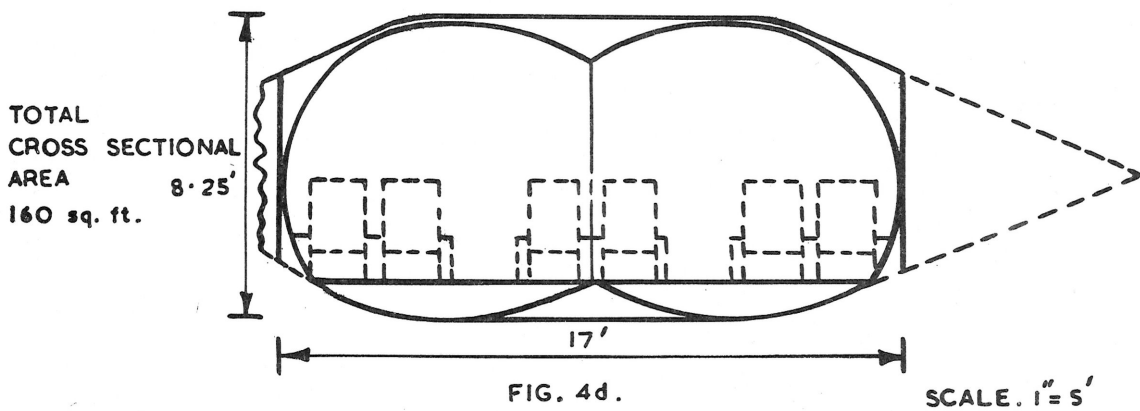
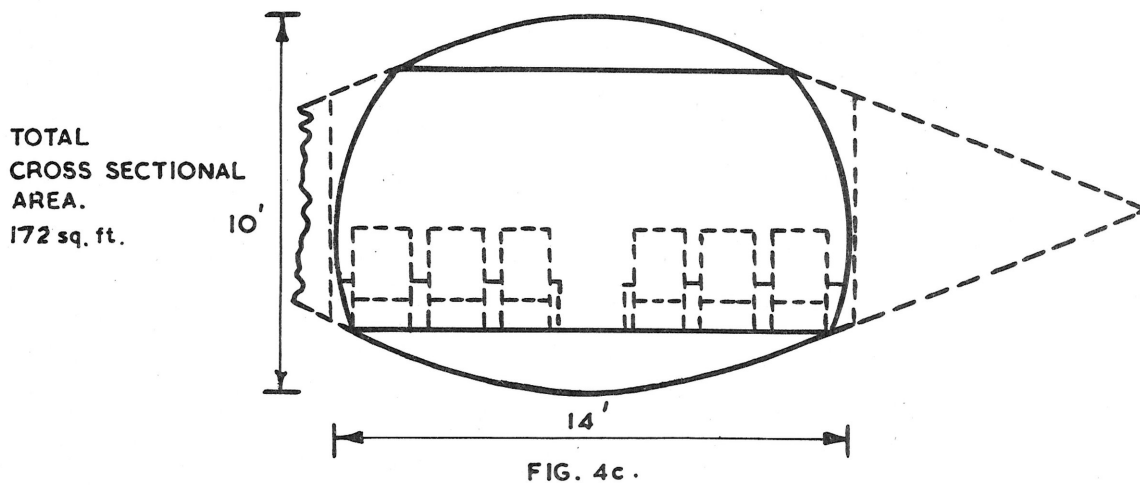


FIG. 4b.



FIGS 4a. b. c. & d. CROSS SECTIONAL VARIATION WITH
NUMBER OF SEATS ABREAST.

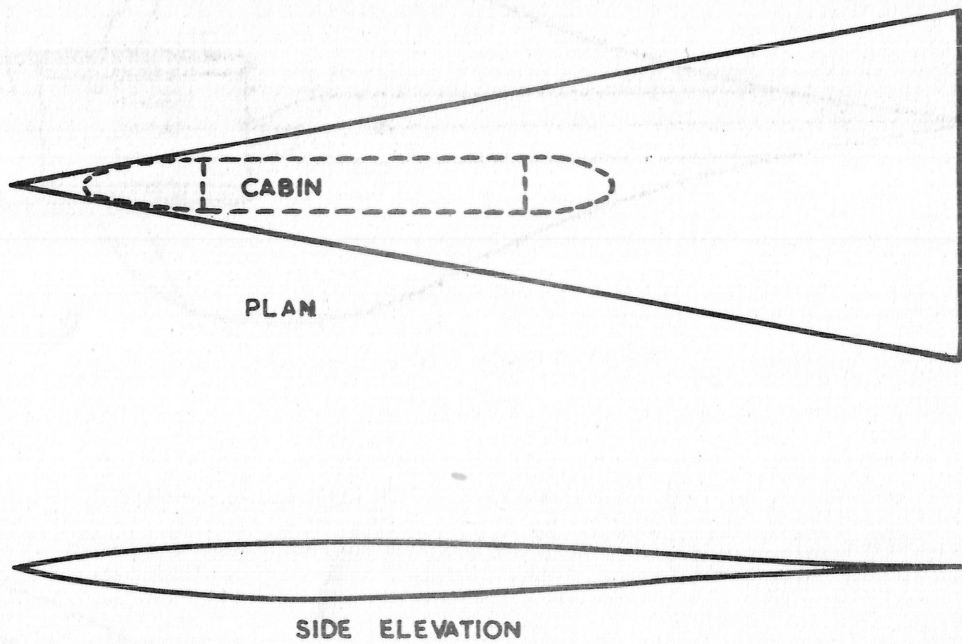


FIG. 5a. DELTA AIRLINER WITH 120 PASSENGERS

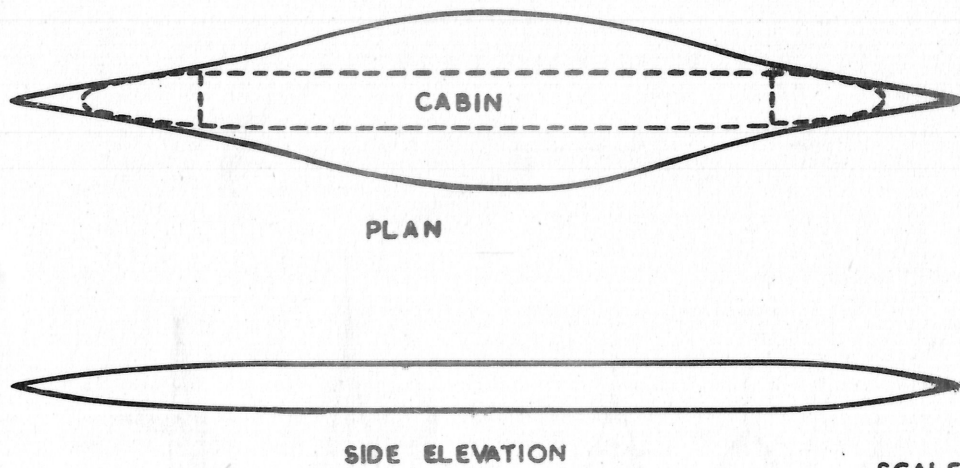
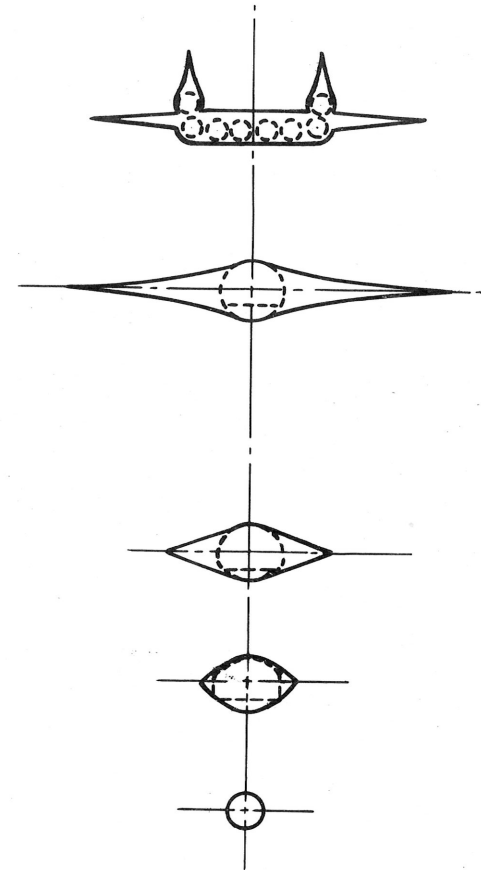
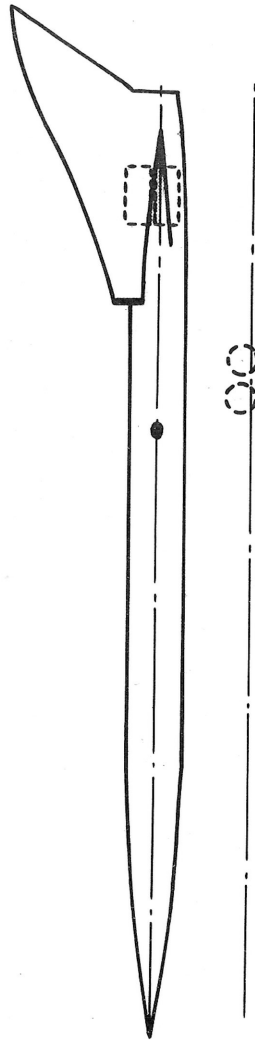
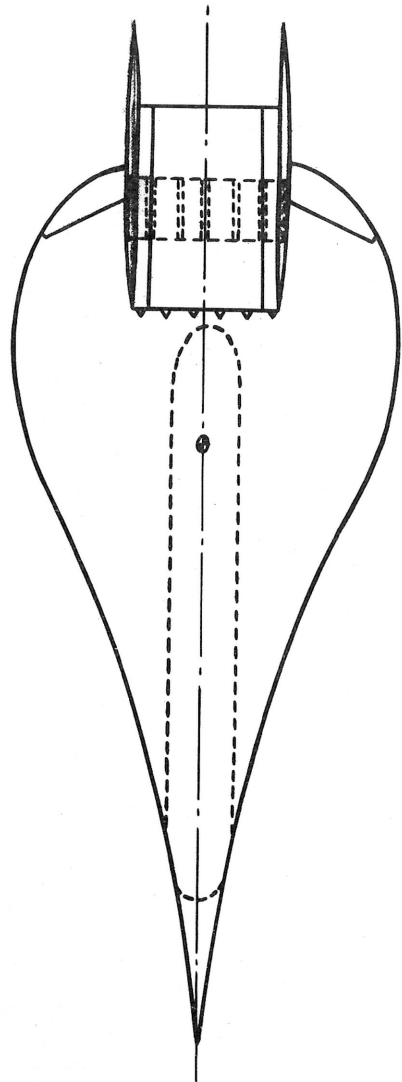
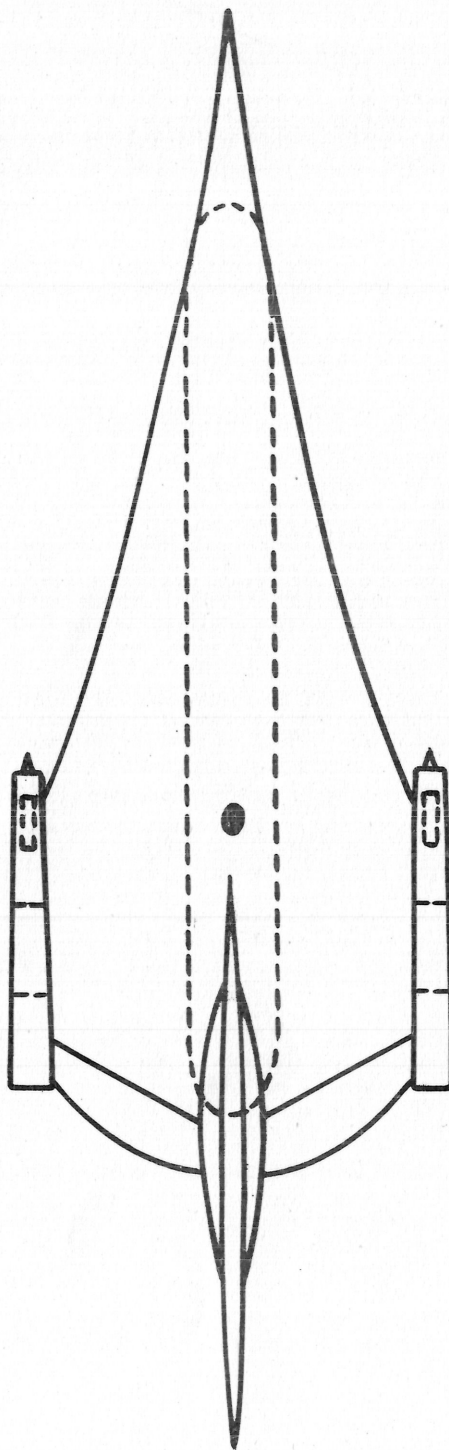


FIG. 5b. AIRLINER OF SAME CROSS-SECTIONAL SHAPE & AREA AS DELTA AIRLINER WITH ACCOMMODATION FOR 220 PASSENGERS



SCALE 1" = 60'

FIG. 6. SUGGESTED LAYOUT FOR 120 PASSENGER SUPERSONIC AIRLINER.



SCALE 1" = 40'

**FIG. 7. SUGGESTED LAYOUT FOR 144 PASSENGER
SUPERSONIC AIRLINER WITH SOME LIFT FROM ENGINES**