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Cruise Flight Duration of a Low Mach Number Ramjet

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The one-dimensional flow through a ramjet operating at Mach numbers between about 1.1 and 1.6 and with lean fuel-air ratios is analyzed and programmed for computation. The addition of routines utilizing empirical drag data, a model for estimating the engine mass, and a scaling technique makes it possible to calculate flight durations in cruise for a given total mass when the net thrust and cruise flight conditions are specified. The calculation of typical applications shows a limited improvement (measured in terms of flight duration) when Mach number is increased from 1.2 to 1.6. This is attributed to the use of a simple pitot intake and converging nozzle in the interest of simplicity and low cost. Increasing the altitude and using leaner fuel-air ratios, however, yield extremely good results. An increase in the flight duration by a factor of as much as five is found, assuming the very lean mixture strengths to have no detrimental effect on combustion efficiency. Since this assumption will not be valid in practice, a design which bypasses most of the excess air around the combustion chamber is considered. Comparable results are obtained.

Nomenclature

a	= constants (subscripts 0-21) in Eq. (19)
A	= area, excess air in core engine
c	= specific heat
C	= coefficient
D	= diameter, drag
f	= fuel-air ratio
F	= thrust
h	= specific enthalpy
K	= scaling factor
L	= length
$\dot{m}$	= mass flow rate
M	= Mach number
p	= pressure
R	= gas constant, radius
R_e	= Reynolds number
t	= time
T	= temperature
V	= volume
x	= dimensionless groups (subscripts 1 and 2) in Eq. (19)
y	= dimensionless group in Eq. (19)
Z	= total excess air
γ	= ratio of specific heats
Δ	= difference
η	= efficiency
ρ	= density

Subscripts

∞	= freestream
0	= stagnation conditions
1-9	= cross sections of engine
b	= bypass flow
c	= core flow
D	= diffuser, drag
$D1.96$	= diffuser with area ratio = 1.96
Df	= friction drag
Dfp	= friction drag on both sides of a flat plate
Dwn	= pressure or wave drag on nose
Dwt	= pressure or wave drag on tail
f	= fuel
$fg298$	= transition fluid to gas (vaporization) at 298 K
np	= net propulsive
p	= constant pressure
pg	= gas (fuel vapor) at constant pressure
$p0$	= stagnation pressure
r	= residence
stoich	= stoichiometric
$\Delta p0$	= stagnation pressure loss

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Introduction

WHEN it was decided to develop a small, simple, low-cost ramjet suitable for remotely piloted vehicles flying at transonic Mach numbers, it was found that the deceptively simple cycle of the ramjet needed careful study before the optimum engine parameters could be selected. Consequently it was judged worthwhile to obtain an analytical method which would determine the gas flow properties in the engine as well as the main engine dimensions for a given application. The available literature[1,2] was studied, but very little if any current research seems to be devoted to this field. Two reports[3,4] yielded some information, but they were made before the computer era and are both clumsy to use and contain unacceptable simplifications or lack of empirical information.

As a first venture a program was written[5] to calculate subsonic engine performance, yielding figures of net thrust and specific fuel consumption with an assumed constant value for combustion efficiency. Since this information required further manipulation in order to select the optimum engine design, it was decided to incorporate these additional calculations in the second program[6] which is a modification of the first to enable engines in supersonic flight to be analyzed.

The methods used to model the gasdynamic processes are described below and some typical results are discussed, including the concept of a bypass ramjet which was evolved to ensure practical exploitation of the theoretical merits of using very lean fuel-air ratios, i.e., 200% and more of excess air. But the most important result is that a computer program is now available that can calculate flight times at a constant flight condition for a given net thrust and total mass of engine and fuel. The only important shortcoming is the need to be able to express combustion efficiency as a function of especially fuel-air ratio and combustor inlet Mach number (for a given flight condition), but in general also of all other relevant factors.

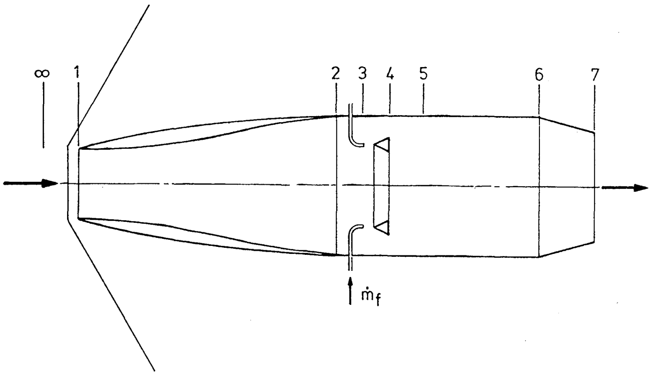

Fig. 1 Conventional ramjet.

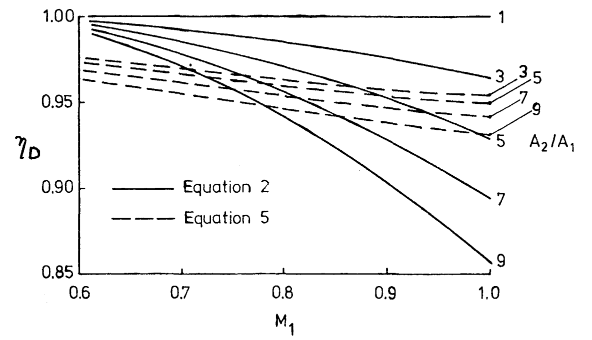

Fig. 2 Internal diffusion losses.

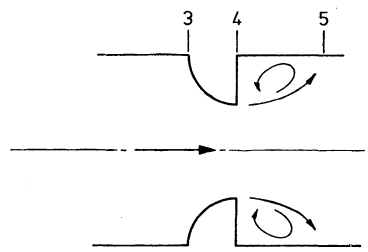

Fig. 3 Equivalent blockage of stabilizer.

Modeling of Integral Flow

The engine is modeled as shown in Fig. 1 with the various processes arranged sequentially, e.g., fuel vaporization is complete before the mixture flows around the flame stabilizers. Since the analysis is limited to low supersonic Mach numbers and simplicity and low cost are emphasized, the model has a pitot intake and a converging exhaust nozzle. There is no provision for the effects of dissociation, which again presupposes low air temperatures and weak mixtures.

The Inlet

For design point operation the inlet flow should be critical to avoid additive drag, and in the model the normal shock is placed at the inlet lip by adjusting the nozzle throat area to suit each case being calculated. The usual relations apply across the shock.

Information applicable to the internal subsonic, but compressible, flow is not readily available. Two alternative sources were considered. One[7] provided results of tests at high subsonic Mach numbers, including choked flow, but at one area ratio (1.96) only. A curve was fitted to these results and yielded the equation:

$$[\eta_D]_{1.96} = 0.993 + 0.04M_1 - 0.05M_1^2 \tag{1}$$

It was subsequently found that in most cases the optimum engine has an inlet area ratio very close to 3, although it sometimes is as high as 5/1. Consequently one possible solution was to assume a linear relationship between efficiency and area ratio and no losses when there is no diffusion.

$$\eta_D = 1 - \{1 - [\eta_D]_{1.96}\}\left\{\frac{A_2/A_1 - 1}{0.96}\right\} \tag{2}$$

These equations are presented in Fig. 2. Test results[8] covering a larger range of area ratios, but applicable to incompressible flow, were also considered.

With the loss in stagnation pressure in the diffuser between sections 1 and 2 defined in terms of a loss coefficient and the (one-dimensional) velocity at section 1

$$\Delta p_0 = C_{\Delta p0}\rho C_1^2/2 \tag{3}$$

and with

$$\rho C^2 = \gamma p M^2 = \gamma p_0 M^2 / \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\gamma/(\gamma-1)} \tag{4}$$

the diffuser efficiency can be written as

$$\begin{aligned}\eta_D &= 1 - \Delta p_0/p_{01} \\ &= 1 - \gamma C_{\Delta p0} M_1^2/2\left(1 + \frac{\gamma - 1}{2} M_1^2\right)^{\gamma/(\gamma-1)}\end{aligned} \tag{5}$$

This incompressible flow result is also plotted in Fig. 2. With flight Mach numbers of 1.1-1.6 the value of M_1 is about 0.75-0.9 and the diffuser area ratios of real interest are 3-5. These two sets of curves are only approximations and should be seen as mere attempts at modeling the real intake process. Tests at the correct M, R_e, and A_2/A_1 are required of the complete intake to obtain truly reliable information. Since the data in Ref. 7 are better documented and apply to compressible flow at an area ratio of about 2, which is not much smaller than the ratios of prime interest, it was decided to use Eqs. (1) and (2) in the program.

Fuel Injection

The effects of the addition of fuel are calculated by assuming the specific heat of the mixture to be equal to that of air and then applying the continuity, energy, and momentum equations. The effect of friction is neglected in the latter case. These equations yield[5]:

$$T_{03} = \frac{c_p T_{02} + f[298(c_f - c_{pg}) - h_{fg298}]}{c_p(1+f) - c_{pg}f} \tag{6}$$

$$\begin{aligned}&1 - \frac{M_2}{M_3}\sqrt{T_{03}/T_{02}}\sqrt{\left(1 + \frac{\gamma-1}{2}M_2^2\right)\Big/\left(1 + \frac{\gamma-1}{2}M_3^2\right)}\,(1+f) \\ &= \gamma M_2^2\left[(1+f)\frac{M_3}{M_2}\sqrt{\left(1 + \frac{\gamma-1}{2}M_2^2\right)\Big/\left(1 + \frac{\gamma-1}{2}M_3^2\right)}\right. \\ &\left.\times\sqrt{T_{03}/T_{02}} - 1\right]\end{aligned} \tag{7}$$

Flame Stabilization

The chosen model cannot provide for conical or can-type combustion chambers. Only a V-gutter or similar stabilizer is considered with the percentage blockage of the flow passage its only characteristic deemed to be of importance. Regardless of the geometry of the stabilizer it is then "translated" into a frictionless contraction causing the same area reduction, followed by an abrupt increase in diameter as shown in Fig. 3.

The assumption of isentropic flow between sections 3 and 4 allows simple calculation of the conditions at 4. In the program the flow at 4 is tested for choking, which can occur in a design with high blockage and low diffusion. By taking p_4 to be constant across the entire section 4 and applying the flow equations, it is found that

$$1-\frac{A_4}{A_5}\frac{M_4}{M_5}\sqrt{\left(1+\frac{\gamma-1}{2}M_4^2\right)\Big/\left(1+\frac{\gamma-1}{2}M_5^2\right)}$$
$$=\gamma M_4^2\frac{A_4}{A_5}\left[\frac{M_5}{M_4}\sqrt{\left(1+\frac{\gamma-1}{2}M_4^2\right)\Big/\left(1+\frac{\gamma-1}{2}M_5^2\right)}-1\right] \tag{8}$$

$$p_{05}=p_4\frac{A_4}{A_5}\frac{M_4}{M_5}\sqrt{1+\frac{\gamma-1}{2}M_4^2}\left\{1+\frac{\gamma-1}{2}M_5^2\right\}^{(\gamma+1)/2(\gamma-1)} \tag{9}$$

Combustion

A fuel of the composition C_7H_{14} was assumed and the classic constant pressure combustion calculations were applied. The assumptions were made that complete combustion was obtained and, as mentioned previously, no dissociation occurred. The first assumption is of course completely unrealistic, but was necessitated by the lack of reliable information on combustion efficiency as a function of inlet conditions, combustion chamber volume, etc. The possibility exists to introduce this at a later stage should it become available or merely to factorize the fuel consumption by the combustion efficiency. This latter procedure is simple but somewhat inaccurate. The second assumption was justified by the low inlet air temperatures and the expected lean mixtures that would produce the optimum designs.

This calculation method was adopted because it was available "off the shelf," having been developed about a decade ago for another application. A simpler method, such as that of Fielding and Topps,[9] would have simplified the program without affecting the results noticeably.

The chosen fuel differs only slightly (14.3% H_2 content as compared to 13.92%) from that suggested[9] as convenient since it yields an average molecular mass equal to that of air for all mixture ratios. This was assumed to apply also to these calculations and the molecular mass and hence the gas constant of the combustion gases where taken to be the same as for air. The specific internal energies of O_2, N_2, CO_2, and H_2O are expressed as functions of T_{06} and an iterative solution yields the value of T_{06}, the stagnation temperature of the combustion products. The assumption is now made that the value of T_{06} obtained in such a pseudostagnant process applies to the real combustion process. This value is then used, together with the flow equations, to calculate the conditions at section 6.

The effects of both friction and heat transfer from the combustor are considered negligible and ignored. Thus is obtained

$$1-\frac{M_5}{M_6}\sqrt{\gamma_5/\gamma_6}\sqrt{\left(1+\frac{\gamma_5-1}{2}M_5^2\right)\Big/\left(1+\frac{\gamma_6-1}{2}M_6^2\right)}$$
$$\times\sqrt{T_{06}/T_{05}}=\gamma_5M_5^2\left[\frac{M_6}{M_5}\sqrt{\gamma_6/\gamma_5}\right.$$
$$\left.\times\sqrt{\left(1+\frac{\gamma_5-1}{2}M_5^2\right)\Big/\left(1+\frac{\gamma_6-1}{2}M_6^2\right)}\sqrt{T_{06}/T_{05}}-1\right] \tag{10}$$

$$p_{06}=\frac{\dot m_6}{A_6M_6}\sqrt{RT_{06}/\gamma_6}\left\{1+\frac{\gamma_6-1}{2}M_6^2\right\}^{(\gamma_6+1)/2(\gamma_6-1)} \tag{11}$$

In the initial analysis a constant value of $\gamma_6=\gamma_7=1.333$ was used. This value is typical for the temperatures considered. This was subsequently improved upon by writing γ as a function of the stagnation temperature.

For C_7H_{14} the stoichiometric fuel-air ratio[5] is

$$f_{\text{stoich}}=0.0681 \tag{12}$$

With excess air present (lean mixtures) the fuel-air ratio can be written as

$$f=f_{\text{stoich}}/(A+1) \tag{13}$$

In the case of the bypass engine, which is discussed below, Z (the total excess air) is used in the place of A after mixing of the hot and cold streams have taken place.

The expression for the specific heat, based on empirical information,[5] is then

$$c_{p0}=[(953.71+940.14f)+(0.21647+3.3631f)T_0$$
$$-(3.1503\times10^{-5}+1.0818\times10^{-3}f)T_0^2]/(1+f) \tag{14}$$

and

$$\gamma_0=1/(1-R/c_{p0}) \tag{15}$$

This γ based on stagnation temperature yielded results (flight times) which differed, in both negative and positive senses, by a maximum of about 2% from the results obtained with $\gamma=1.333$. It was consequently considered not worthwhile to use γ as a function of static temperature which would have required iterative solution of a number of equations.

The Exhaust Nozzle

The exit Mach number from a nozzle providing full expansion is a function of the nozzle stagnation pressure, ambient pressure, and the value of γ in the nozzle. It can easily be shown for the type of engine being considered that the exit velocity from such a nozzle is supersonic if the flight Mach number is greater than about 1.1. Since only convergent nozzles are used in this analysis, these nozzles will be choked with an exhaust static pressure above ambient except when flight at immediately above Mach 1 is calculated. At these very low supersonic flight Mach numbers complete expansion, with a convergent nozzle, to ambient is possible.

The nozzle area is adjusted in each case being calculated to position the normal shock at the inlet lip.

The losses will be minimal in such a converging nozzle and can be expected to be a function of M_6. A high value M_6 indicates the use of a short nozzle with little loss in stagnation pressure. Applicable empirical information could not be found and the following simple relationship was assumed

$$p_{07}/p_{06}=0.99+M_6/100 \tag{16}$$

The shape, which means the cone angle, of the nozzle is determined in an optimization procedure which is described below. The outlet area and flow conditions are calculated in the usual way.

Bypass Flow

Preliminary calculations indicated the probability of achieving good results with very lean mixtures. This raised the question of whether satisfactory combustion could be achieved with such lean mixtures. It was consequently decided to investigate the possibility of bypassing part of the airflow around the combustion chamber (see Fig. 4), the two gas streams mixing before entering the exhaust nozzle.

The following assumptions were made:

1) Heat transfer to the annulus and friction in the annulus is neglected, permitting the assumption of isentropic flow in the annulus.

2) Complete mixing is achieved in the mixing section of the combustion chamber without employing mixing vanes or similar devices. The volume of this section is adjusted to aid the achievement of this ideal.

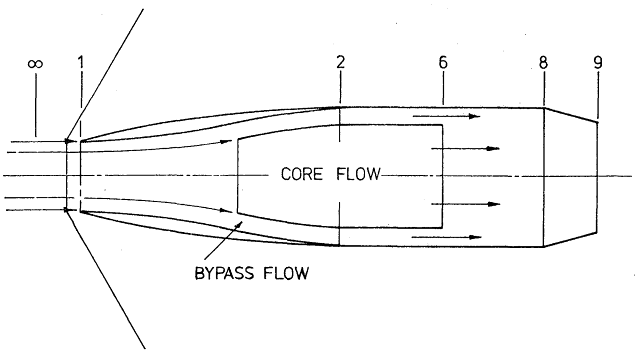

Fig. 4 Bypass engine.

3) Friction and heat loss are neglected in the mixing chamber.

The air entering the inlet is divided into two streams so that at section 1

$$A_{b1} = A_{c1}(Z-A)/(1+A) \quad (17)$$

The dimensioning of the combustion chamber or core engine in relation to the total engine is determined by the requirement that

$$p_{b6} = p_{c6} = p_6 \quad (18)$$

These two coaxial streams are then mixed between sections 6 and 8 (7 was reserved for the possible introduction of mixing devices). Application of the flow equations yields results similar to those obtained above. In and after the iterative solution of these equations the flow properties are controlled to determine whether further effusion is possible or the flow is choked.

The flow through the exhaust nozzle is calculated as before.

Drag Calculation

The external flow was considered to be completely free of influence by the internal flow. The empirical data[10] used provided for the separate calculation of the pressure drag of the nose and tail sections. While the latter was taken as being conical, the graphs for a parabolic nose were used.

Polynomials were fitted to these curves, using a regression routine. This yielded, as an example, for the wave drag coefficient of the nose:

$$\begin{aligned} y = {} & a_0 + a_1x_1 + a_2x_2 + a_3x_1^2 + a_4x_1^3 + a_5x_2^3 + a_6x_1^4 + a_7x_2^4 \\ & + a_8x_1x_2 + a_9/x_1x_2 + a_{10}/x_2 + a_{11}x_1/x_2 + a_{12}x_2/x_1 \\ & + a_{13}x_2^2/x_1 + a_{14}x_1/x_2^2 + a_{15}x_1^2x_2^2 + a_{16}/x_1^2x_2^2 + a_{17}x_1^2/x_2 \\ & + a_{18}x_2/x_1^2 + a_{19}x_2^2x_1^3 + a_{20}x_1^2x_2^3 + a_{21}/x_1^2x_2^3 \end{aligned} \quad (19)$$

with

$$x_1 = \frac{\sqrt{M_\infty^2 - 1}}{L_{1\text{-}2}/R_2}$$

$$x_2 = (R_1/R_2)^2$$

$$y = C_{Dwn}(L_{1\text{-}2}/R_2)^2$$

Values for the constants are given in Table 1.

To limit the number of terms to the 22 given in Eq. (19), a region of applicability was defined. The values of x_1 and x_2 are tested for this in the computer program. Extrapolation outside the region obtained by regression is obviously not possible using the same equations, but alternative methods (e.g., linear extrapolation) could be used to extend the applicability if required.

Table 1 Coefficients = $m \times 10^n$ applicable when $0.05 \le x_1 \le 1.0$ and $0.10 \le x_2 \le 0.6$

	m	n		m	n
a_0	0.586700	1	a_{11}	0.164999	0
a_1	−0.117463	2	a_{12}	−0.111060	0
a_2	−0.892893	1	a_{13}	0.311805	−1
a_3	0.152212	2	a_{14}	0.698866	−2
a_4	−0.114810	2	a_{15}	−0.185146	2
a_5	0.105913	2	a_{16}	−0.822993	−4
a_6	0.364623	1	a_{17}	−0.993277	−1
a_7	−0.858378	1	a_{18}	0.235625	−2
a_8	0.994862	1	a_{19}	0.357198	1
a_9	0.114447	−1	a_{20}	0.112606	2
a_{10}	−0.111443	0	a_{21}	0.425188	−5

The calculation of the friction drag was simplified by considering the complete engine as an equivalent cylinder moving at the freestream Mach number. Calculations showed that the transition point of the boundary layer is close to the leading edge in the typical cases to be considered. The flow was consequently taken to be turbulent over the full length of the engine.

By taking into consideration the known relationships between C_{Df} and R_e and M, the following expression was derived[6] for C_{Dfp} to fit the available empirical results:

$$C_{Dfp} = 0.08942R_e^{-0.1719} - 0.004494R_e^{-0.1005}(M-1) \quad (20)$$

This equation fits the results in Fig. 6 of the source and is applicable to a friction force acting on both sides of a flat plate. The pressure drag coefficient refers to the maximum cross-sectional area of the engine and with the same reference area

$$C_{Df} = C_{Dfp}L_{1\text{-}9}/R_2 \quad (21)$$

Combustion Chamber Friction Drag

Whereas for the calculation of the internal flow only the cross-sectional area of the combustion chamber was of importance, the drag is also dependent on the length. Consequently the volume must be known.

The assumption was made that a certain residence time is required for satisfactory combustion in a typical engine. This was found from an analysis of existing designs to be

$$t_r = 0.003 \text{ s} \quad (22)$$

The mass flow may be written

$$\dot{m} = \bar{\rho}_{2\text{-}8}A_{2,8}L_{2\text{-}8}/t_r$$

thus

$$V_{2\text{-}8} = \dot{m}t_r(RT/p)_{\text{avg}} \quad (23)$$

The average conditions were taken to be the mean of the inlet and outlet values.

It was assumed that this residence time is not affected by excess air or any other factors. It was also taken to be constant for all bypass ratios, reasoning that for a given inlet condition and fuel flow the richer mixture in the combustion zone of a bypass engine would require the same total volume (combustion plus mixing) as the lean mixture in a through-flow design. These assumptions affect only the frictional drag and mass of the engine. The former is of little consequence, but the variations in engine mass can be an important factor, especially in designs for a short flight duration.

Drag of the Outlet Nozzle

In the calculation of the drag of the complete engine, the nozzle shape (inlet and outlet diameters and length or cone angle) determines the pressure drag coefficient as a function of M_∞, while the skin friction drag is also dependent on the length of engine ahead of the nozzle. But it is obvious that a lengthened nozzle (smaller cone angle) has a lower pressure drag and higher friction drag. Although such a nozzle also has an increased mass, this effect was ignored in an optimization in the computer program which traded off external pressure drag against internal and external friction drag.

For this purpose the external friction drag was calculated as the added drag on a cylinder lengthened by the length of the nozzle. The internal drag was found by assuming the internal flow to expand isentropically to the average Mach number in the nozzle, taken as being the mean of M_8 and M_9, and then flowing through a tube as shown in Fig. 5. Friction was considered in this hypothetical tube, but not in the two nozzles at sections 8 and 9.

Scaling to the Required Net Propulsive Force

The calculations are based on an airflow of 1 kg/s into the engine and the engine has to be scaled to ensure that the full-sized item will produce the required propulsive force. It should be remembered that 100% combustion efficiency is assumed with unburned fuel the only major cause of loss of efficiency. Hence thrust is restored by increasing the fuel flow in the real engine. With a mixture near stoichiometric this would of course not be effective.

Thrust Calculation

With reference to Fig. 4 it can be seen that for an air intake of 1 kg/s the exhaust flow is

$$\dot{m}_g = \frac{A_{bI} + A_{cI}(1+f_c)}{A_{bI} + A_{cI}} \tag{24}$$

Providing for underexpanded flow in the exhaust nozzle the net thrust is then

$$F_{\text{net}} = \left[\frac{A_{bI} + A_{cI}(1+f_c)}{A_{bI} + A_{cI}}\right] M_9\sqrt{\gamma_9 R T_9} - M_\infty \sqrt{\gamma R T_\infty}$$

$$+ A_9(p_9 - p_\infty) \tag{25}$$

The R_e, and hence the friction characteristics, of the internal flow are changed by scaling of the engine dimensions, but the shock and combustion pressure losses are so much more important that the net thrust was assumed to be a linear function of the air mass flow.

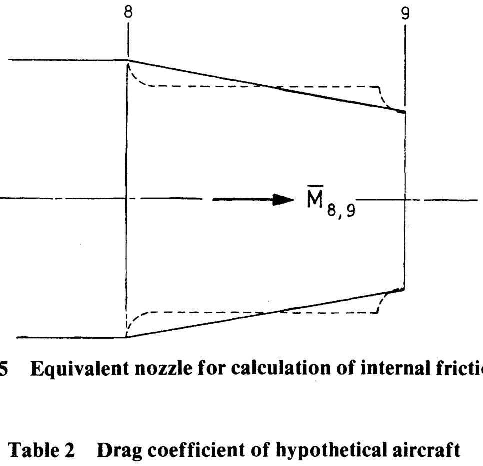

Fig. 5 Equivalent nozzle for calculation of internal friction.

Table 2 Drag coefficient of hypothetical aircraft

M_∞	1.2	1.4	1.6
$C_{D(\text{total})}$	0.060	0.048	0.039

The scaling factor is thus defined as

$$K = F_{\text{net (full scale)}} / F_{\text{net (1 kg/s)}} \tag{26}$$

Scaling Dimensions

With the airflow for given flight conditions being a function of the intake area only and all area ratios being held constant when scaling, it applies to all diameters that

$$D_{(\text{full scale})} = D_{(1\ \text{kg/s})}\sqrt{K} \tag{27}$$

The nose and tail have fixed geometries so that the same scaling applies to $L_{1\text{-}2}$ and $L_{8\text{-}9}$. According to Eq. (23) the volume $V_{2\text{-}8}$ is directly proportional to the mass flow. Since the cross-sectional areas change proportionally to the scaling factor, the length $L_{2\text{-}8}$ is constant.

Scaling Drag

The pressure drag coefficients are independent of the size of the engine and consequently the pressure or wave drag on the nose and tail are, like the net thrust, proportional to the scaling factor.

The friction drag, however, is a function of both M and R_e based on length, [Eq. (20)]. The second term in this equation can be ignored since it is found that an increase in R_e from 10^6 to 10^7 (i.e., a tenfold increase in engine length due to scaling) at $M_\infty = 2$ leads to a change in C_{Dfp} which differs by only 3% when calculated with the complete expression from the result when using the first term only.

Since friction is only a part of the total drag and scaling affects only the lengths of part of the engine and this analysis applies to $M_\infty < 2$, this simplification is permissible.

$$\frac{C_{Df(\text{full scale})}}{C_{Df(1\ \text{kg/s})}} = \left[\frac{R_{e(\text{full scale})}}{R_{e(1\ \text{kg/s})}}\right]^{-0.1719} \tag{28}$$

$$\frac{C_{Df(\text{full scale})}}{C_{Df(1\ \text{kg/s})}} = \left[\frac{\sqrt{K}(L_{1\text{-}2} + L_{8\text{-}9}) + L_{2\text{-}8}}{L_{1\text{-}2} + L_{2\text{-}8} + L_{8\text{-}9}}\right]^{-0.1719} \tag{29}$$

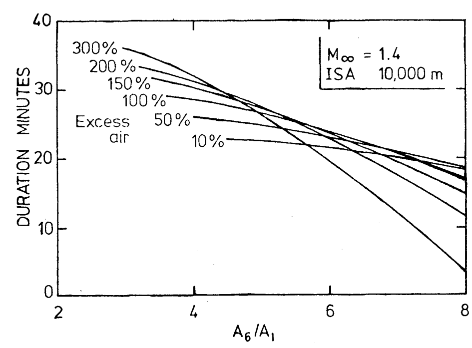

Fig. 6 Constant flight conditions.

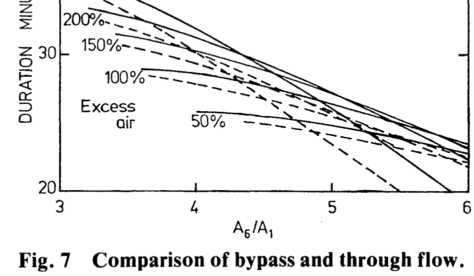

Fig. 7 Comparison of bypass and through flow.

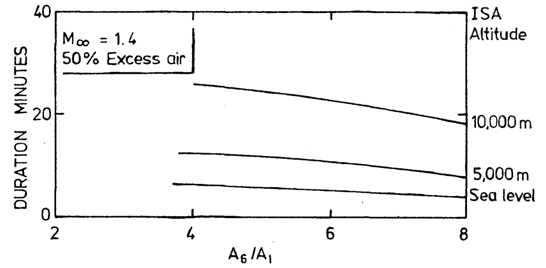

Fig. 8 Variation of altitude.

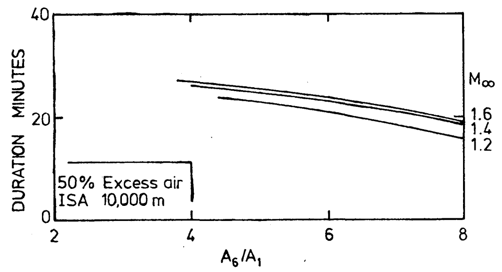

Fig. 9 Variation of Mach number.

In scaling the drag the ratio of the wetted areas must also be introduced.

$$\frac{D_{f(\text{full scale})}}{D_{f(1\ \text{kg/s})}}=\sqrt{K}\left[\frac{\sqrt{K}(L_{1\text{-}2}+L_{8\text{-}9})+L_{2\text{-}8}}{L_{1\text{-}2}+L_{2\text{-}8}+L_{8\text{-}9}}\right]^{0.8281} \tag{30}$$

Thus the net propulsive force of the scaled engine is

$$F_{np(\text{full scale})}=K[F_{\text{net}(1\ \text{kg/s})}-D_{wn(1\ \text{kg/s})}-D_{wt(1\ \text{kg/s})}]$$

$$-\sqrt{K}\left[\frac{\sqrt{K}(L_{1\text{-}2}+L_{8\text{-}9})+L_{2\text{-}8}}{L_{1\text{-}2}+L_{2\text{-}8}+L_{8\text{-}9}}\right]^{0.8281}D_{f(1\ \text{kg/s})} \tag{31}$$

With the required full-scale net propulsive force known it is now possible to solve for K.

Results of the Analysis

It was thus possible to calculate the dimensions and performance characteristics of an engine for a given application, defined as a net propulsive force at a given flight condition. Adding to this a simplified model to determine the mass of the engine, it was possible to calculate flight time in cruise at such a given flight condition for a total available mass of engine plus fuel. As an example a hypothetical aircraft was considered which has a drag of 3000 N when flying at Mach 1.4 at sea level. The drag coefficient was assumed to be a function of Mach number only, neglecting the effect of changes in R_e. It was assumed that the combined mass of the engine (excluding accessories such as the fuel pump or fuel tank) and fuel was 100 kg. (See Table 2.)

Typical results are shown in Figs. 6-9. Figure 6 shows that lean mixtures are very desirable at the flight condition chosen. For most of the curves the left-hand cutoff points are the limits of applicability of the drag data, but these are very close to the internal flow limits (most frequently flow choking in combustion chamber) which would make operation impossible in any case. With the peaks of the curves so close to these cutoff points, it is found that M_8 for the optimum design is close to unity. This means that there is very little expansion in the nozzle and consequently D_9 is not much smaller than D_8. The importance of boattail drag is a major contributing factor to this result. In Fig. 7 the bypass design is compared with the conventional engine. There is a small difference in favor of the conventional engine (although at other flight conditions the reverse has been found), but the important point is that bypass operation should be easier at the very lean mixtures in which the designer would be interested. Duct burning could be possible in a bypass engine for thrust augmentation.

Figure 8 was calculated by again considering the same aircraft at a constant Mach number at different altitudes and shows the strong preference of the ramjet for high-altitude flight. In Fig. 9 the Mach number is varied at a constant altitude. An increase in M_∞ is beneficial, but a more sophisticated intake and exhaust would have enabled better use to have been made of an increase in flight velocity.

Conclusions

In a search for possible errors, which are expected to exist only, if at all, in the programming, the general trends of the calculated internal flow have been compared with both simplified analyses (one of which[4] was found to be an early protagonist of the bypass design) and published test results. The drag coefficients found by calculating and adding the drag of the various components have been compared with published data applicable to similar configurations. No indication of major discrepancies has been found.

Further investigation is sought by reprogramming to remove a certain untidiness which appeared in the program as additions were made to the original and by a series of directly connected tests of a ramjet in specific comparison with the theoretical results of the internal flow calculations. The following improvements and additions are planned once the reprogramming is completed. These changes will then be performed by merely changing data or inserting new subroutines.

1) The parabolic nose, which is a relic of the subsonic origin of the program, will be replaced by a conical nose.

2) Internal friction in the combustion chamber and bypass duct and mixing chamber will be taken into account.

3) Provision will be made for the effects of using vanes to promote mixing in the mixing chamber.

4) The present tests may provide some guide toward formulating an equation for the combustion efficiency, taking cognizance of the general turbulent combustion laws as well as published data (e.g., Ref. 2).

5) More reliable and generally applicable information on the losses in both the inlet and outlet nozzles is desirable and may warrant an experimental program.

6) The next evolutionary step could be the addition of a convergent-divergent nozzle and a multishock inlet as well as the provision for dissociation in the combustion process to extend the Mach number limit of applicability.

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