CORRECTION OF RAMJET EXPERIMENTAL DATA

Thesis by

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I INTRODUCTION AND SUMMARY

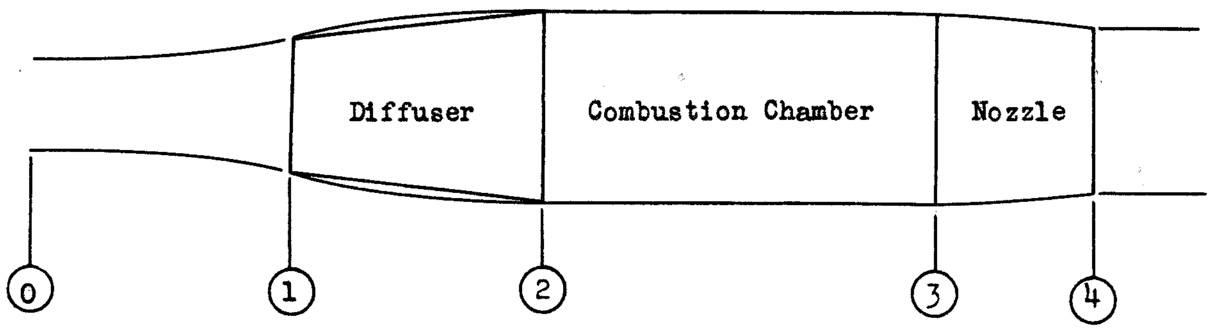
Heat energy can be converted into mechanical energy only by the expansion of a high temperature high pressure gas to a lower pressure thus changing the random energy, temperature, into a directed energy associated with a velocity. One of the simplest methods of achieving the necessary pressure for this expansion is by utilizing the high velocity of the airstream relative to a rapidly moving aircraft. The momentum of this airstream is converted into pressure by deceleration in a diffuser. The air is then heated and ejected as a high speed jet, producing thrust, hence the name given to this system is <u>ramjet</u>. Other names have also been proposed such as athodyd, which is a contraction of aero-thermodynamic duct.

The purpose of this study is to derive a simple ramjet theory and to use this theory in setting up dimensionless parameters and methods of analysing experimental data. No attempt is made to devise experimental techniques but only to give a method by which the experimental data may be correlated.

It is found that the performance of a ramjet at any altitude can be reduced to an equivalent sea level condition by the application of suitable correction factors. These factors are derived on the basis of constant flight Mach number, constant ramjet geometrical arrangement, and constant throttle setting. This latter parameter is most accurately described by the ratio of the combustion chamber temperature to the free air temperature. It is recommended that ramjet test data by corrected to this standard sea level condition for ease of correlation and analysis.

II. SIMPLE RAMJET THEORY

A typical ramjet cross section with the station numbers indicated is shown in Figure 1.



Ramjet Cross Section
Figure 1

The ramjet is considered to be at rest, and the air approaches the entrance with a velocity, in the undisturbed stream, V_0. The air may, or may not, be compressed in its passing from station 0 to station 1 depending upon the ramjet geometry. The deceleration continues to station 2 where fuel is mixed with the compressed air and combustion occurs, raising the temperature and accelerating the gases. The exhaust is then expanded to a higher velocity through the nozzle.

If the diffuser efficiency is defined as the fraction of the kinetic energy change useful for purposes of compression, then the pressure rise is given by equation (1)*

$$-dp = \eta_d \rho \, d\left(\frac{V^2}{2}\right) \qquad (1)$$

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*See table of notation for definitions of symbols.

Which leads to the well known expression

$$\frac{p_2}{p_0} = \left(\frac{T_2}{T_0}\right)^{\frac{\gamma}{\gamma-1}\eta_d} \tag{2}$$

where, from the energy equation,

$$\frac{T_2}{T_0} = \frac{1+\frac{\gamma-1}{2}M_0^2}{1+\frac{\gamma-1}{2}M_2^2} \tag{3}$$

The equation of continuity may be written as

$$A\rho V = \text{CONSTANT} \tag{4}$$

the equation of state is

$$\frac{p}{\rho} = RT \tag{5}$$

and the speed of sound is given by the expression

$$a = \sqrt{\gamma RT} \tag{6}$$

Using this set of equations, the pressure, temperature, density, and velocity of the air at station 2 can be determined from the initial condition, the ramjet geometry, and the diffuser efficiency.

During the fuel injection process the gas constants, gas velocity, pressure, and temperature are altered due to the addition of a mass capable of absorbing heat during evaporation and to a change in the gas composition. These effects are usually considered a part of the combustion process and ignored.

After combustion the exhaust gases occupy a considerably larger volume than before, and hence must be accelerated to a higher velocity since the combustion chamber is usually of constant cross sectional area. The only force present to accomplish this acceleration is the difference in the pressures at the ends of combustion chamber. Hence this acceleration of the gases, as they pass through the combustion chamber, is accompanied by a drop in the static pressure even if the friction and eddy losses are neglected. The impact, or stagnation, pressure is also reduced as the static pressure drop in the tube more than offsets the rise in the dynamic pressure associated with the velocity.

If the friction and eddy losses are expressed as

$$\Delta p_{losses} = (2k)\frac{\rho_2}{2}V_2^2 \tag{7}$$

then the pressure ratio becomes

$$\frac{p_3}{p_2} = 1 - \gamma M_2^2\left(\mu\frac{V_3}{V_2} - 1 + k\right) \tag{8}$$

By using the equation of continuity, written as

$$\frac{V_3}{V_2} = \mu\frac{\rho_2}{\rho_3}$$

and the equation of state as

$$\frac{\rho_2}{\rho_3} = \frac{p_2 R' T_3}{p_3 R T_2}$$

the pressure ratio, p_3/p_2 , and the velocity ratio may be shown to be funtions of the temperature ratio, T_3/T_2 , and the initial Mach number

The energy released by the combustion process goes not only into raising the gas temperature but also into increasing the gas kinetic energy. This can be seen by writing the energy equation, neglecting the temperature and velocity of the fuel injected,

$$\frac{V_2^2}{2} + \overline{C_p}\,T_2 + H = \mu\left(\frac{V_3^2}{2} + \overline{C_p}'\,T_3\right) \tag{9}$$

where H is the heat released per unit mass of air and $\overline{C_p}$ is a mean specific heat.

The flow through the nozzle is computed in the same manner as that through the diffuser except that the nozzle efficiency, η_n , is defined as the fraction of the pressure change useful in accelerating the gases.

Hence

$$\frac{p_3}{p_4} = \left(\frac{T_3}{T_4}\right)^{\frac{\gamma'}{\gamma'-1}\frac{1}{\eta_n}} \tag{10}$$

The pressure p_4 is generally close to that of the surrounding atmosphere, p_0, and hence is usually assumed equal to p_0.

The thrust of the ramjet is obtained by computing the change in the momentum of the working fluid in passing from station 0 to station 4. If the pressure p_4 is not equal to p_0 and the flow is subsonic, it is then necessary to go to a station further downstream where the pressure in the exhaust is equal to p_0. The thrust is given as

$$\text{thrust} = (m_{air} + m_{fuel})V_4 - m_{air}V_0$$

or, by defining the thrust coefficient, C_F , as

$$C_F = \frac{\text{thrust}}{\frac{\rho_0}{2}V_0^2 A_2} \tag{11}$$

it becomes

$$C_F = 2\frac{A_0}{A_2}\left(\mu\frac{V_4}{V_0} - 1\right) \tag{12}$$

The rate of fuel flow is

$$\text{fuel rate} = (\mu - 1) A_0 \rho_0 V_0 q \tag{13}$$

If the specific fuel consumption, s.f.c., is defined as the pounds of fuel per second per pound of thrust, then it becomes

$$\text{s.f.c.} = 2 \frac{A_0}{A_2} \frac{(\mu - 1) q}{C_F V_0} \tag{14}$$

or

$$\text{s.f.c.} = \frac{(\mu - 1) q}{\mu V_4 - V_0}$$

The expressions for the performance of the individual components of the ramjet are not necessarily different for supersonic and subsonic flight speeds. It is rather obvious that in the supersonic case the entrance problem is quite different even though the conditions can be expressed in the same manner as for the subsonic case. Unless a shock precedes the entrance, the ramjet will not affect the flow ahead of the entrance and no compression of the air occurs ahead of the duct. The velocity of the gas is reduced to a subsonic velocity in the diffuser before it enters the combustion chamber.

At supersonic flight speeds, and with a diffuser of reasonable efficiency, the pressure at the end of the combustion chamber is sufficient to permit the acceleration of the exhaust to supersonic speeds. Unless the nozzle is operating at its design point, it is possible to have a pressure that is higher or lower than ambient. In this case it is necessary to add a pressure correction term to the ramjet thrust. The thrust is given as

$$\text{thrust} = m_{air} (\mu V_4 - V_0) + A_4 (p_4 - p_0)$$

or

$$C_F = 2\frac{A_0}{A_2}\left(\mu\frac{V_4}{V_0} - 1\right) + 2\frac{A_4}{A_2}\frac{1}{\gamma M_0^2}\left(\frac{p_4}{p_0} - 1\right) \qquad (15)$$

Again the specific fuel consumption is

$$s.f.c. = 2\frac{A_0}{A_2}\frac{(\mu - 1)g}{C_F V_0}$$

III. BASIC DIMENSIONLESS PARAMETERS

It is desirable to express the important performance items such as thrust and fuel rate in coefficient form, or as dimensionless parameters, such that their values depend only upon such independent parameters as flight Mach number, ramjet geometry, and throttle setting. This latter parameter is most easily expressed in the form of a temperature ratio, either the ratio of combustion chamber exit temperature to entrance temperature or combustion chamber exit temperature to free stream temperature.

The air mass flow is

$$m_{air} = A_2 \rho_2 V_2$$

or

$$m_{air} = A_2 p_2 M_2 \sqrt{\frac{\gamma}{RT_2}}$$

Both the pressure p_2 and T_2 are functions of the outside air conditions, p_0 and T_0, the flight Mach number, M_0, and the combustion chamber Mach number, M_2. Rewriting the expression for the air mass flow results in the expression

$$m_{air} \frac{\sqrt{T_0}}{p_0} = A_2 \frac{p_2}{p_0} M_2 \sqrt{\frac{\gamma}{R}} \sqrt{\frac{T_0}{T_2}}$$

By using the equations (2) and (3) the parameter becomes

$$m_{air} \frac{\sqrt{RT_0}}{A_2 p_0} = \sqrt{\gamma} \; M_2 \left(\frac{1 + \frac{\gamma-1}{2} M_0^2}{1 + \frac{\gamma-1}{2} M_2^2} \right)^{\frac{\gamma \eta_d}{\gamma - 1} - \frac{1}{2}} \qquad (16)$$

The diffuser efficiency η_d is a function of M_2 as a variation in this velocity will affect the external streamlines and the amount of precompression as well as the internal flow characteristics. This non-dimensional air flow parameter is completely defined at one flight Mach number by the Mach number at the end of the diffuser, and, or course, the diffuser geometry.

The primary significance of the fuel quantity is the amount of heat or energy that it releases in the combustion chamber. Its mass is small, usually less than seven per cent of the air mass, but the energy it releases is very great, being several times that associated with the entering air. Hence it appears logical that the important fuel flow parameter would be one giving the ratio of the heat released by the fuel to the energy of the entering air.

The heat added in the combustion chamber may be obtained from the energy equation

$$H = \mu \frac{V_3^2}{2} + \mu \overline{C}_p' T_3 - \frac{V_2^2}{2} - \overline{C}_p T_2$$

The total energy per unit mass of air entering the combustion chamber may also be written as

$$\frac{V_2^2}{2} + \overline{C}_p T_2 = \overline{C}_p T_{s_2}$$

where T_{s_2} is the stagnation temperature at station 2. The flow is considered to be insulated, or adiabatic, to station 2 and hence the energy at 2 must equal the energy in the free stream or

$$T_{s_2} = T_{s_0}$$

The total heat content,per unit mass of entering air, at the end of the combustion chamber can also be written

$$\mu \frac{V_3^2}{2} + \mu \bar{C}_p' T_3 = \mu \bar{C}_p' T_{s_3}$$

Hence

$$H = \mu \bar{C}_p' T_{s_3} - \bar{C}_p T_{s_0}$$

The rate of fuel flow may be written in terms of the air mass flow as

$$m_{fuel} = m_{air} \frac{H}{h}$$

where h is the heating value of the fuel per unit mass of fuel. Then

$$\frac{m_{fuel}}{m_{air}} h = \left(1 + \frac{m_{fuel}}{m_{air}}\right) \bar{C}_p' T_{s_3} - \bar{C}_p T_{s_0}$$

or

$$\frac{m_{fuel}}{m_{air}} = \frac{\dfrac{\bar{C}_p' T_{s_3}}{\bar{C}_p T_{s_0}} - 1}{\dfrac{h}{\bar{C}_p T_{s_0}} - \dfrac{\bar{C}_p' T_{s_3}}{\bar{C}_p T_{s_0}}}$$

If the ratio $\bar{C}_p' T_{s_3}/h$ is considered small, this may be rewritten as

$$\frac{m_{fuel}}{m_{air}} \frac{h}{\overline{C_p} T_0} = \frac{T_{s_0}}{T_0} \left(\frac{\overline{C_p}' T_{s_3}}{\overline{C_p} T_{s_0}} - 1 \right) \left(1 + \frac{\overline{C_p}' T_{s_3}}{h} + \cdots \right) \qquad (17)$$

where

$$\frac{T_{s_0}}{T_0} = 1 + \frac{\gamma - 1}{2} M_0^2$$

The heat energy represented by the term $C_p' T_{s_3}$ is of the order of 30 x 10^6 ft lbs/slug for a temperature T_{s_3} of 3600°F and the heating value of gasoline 480 x 10^6 ft lbs/slug (19,000 Btu/lb). Thus it may be seen that the correction factor $\overline{C_p}' T_{s_3}/h$ is of the order of six per cent. Since the mean specific heats $\overline{C_p}$ and $\overline{C_p}'$ are functions of temperatures T_2 and T_3 respectively, they will vary with the initial temperature and the temperature ratio. However, the variations of the specific heat and of the term $1 + \frac{\overline{C_p}' T_{s_3}}{h}$ with the initial temperature, for constant temperature ratio, are very small and the left hand side of equation (17) may be considered as a suitable fuel flow parameter. The numerator is the heat contained in the fuel and the denominator is the energy of the entering air.

An examination of the equations for the performance of the components of the ramjet, equations (1) through (10), reveals the fact that the Mach number at any station is a function of the initial Mach number, M_0 , the combustion chamber temperature ratio, T_3/T_2 , and the ramjet geometry. For each flight Mach number, the combustion chamber Mach number is uniquely determined by the ramjet geometry and the combustion chamber temperature ratio. The ratio of the pressure at any station to the free stream static pressure and the corresponding temperature ratio are also determined by the flight Mach number, combustion chamber temperature ratio, and ramjet geometry.

The combustion chamber temperature ratio is a fundamental parameter, but it can be replaced by the ratio of the final combustion chamber temperature to the free stream temperature. This latter ratio is a more useful parameter in that it makes it unnecessary to determine the temperature rise during the compression of the air on its evaluation.

Equation (15) gives the thrust of the ramjet as a thrust coefficient in the form

$$C_F = 2\frac{A_0}{A_2}\left(\mu\frac{V_4}{V_0}-1\right) + 2\frac{A_4}{A_2}\frac{1}{\gamma M_0^2}\left(\frac{p_4}{p_0}-1\right)$$

The area ratio A_0/A_2 is determined by the free stream Mach number, M_0, and the combustion chamber Mach number, M_2. Since the Mach number M_2, and hence η_d, the velocity ratio, V_4/V_0, and the pressure ratio are determined by the free stream Mach number, M_0, the temperature ratio, T_3/T_0, and the ramjet geometry, it is seen that the thrust coefficient is unaltered, except for secondary effects, by changes in free stream temperature or pressure. Thus it can be seen that the thrust coefficient is a suitable nondimensional parameter. Since

$$\frac{\rho_0}{2}V_0^2 = \frac{\gamma}{2}p_0 M_0^2$$

then

$$\text{thrust} = \frac{\gamma}{2}p_0 M_0^2 A_2 C_F$$

Thus another possible thrust parameter might be given by

$$\frac{\text{thrust}}{p_0 A_2} = \frac{\gamma}{2}M_0^2 C_F \qquad (18)$$

The fuel weight flow may be expressed as

$$W_{fuel} = q\,\frac{\overline{c}_p T_{s_0}}{h}\left[\frac{\overline{c}_p' T_{s_3}}{\overline{c}_p T_{s_0}} - 1\right]\left[1 + \frac{\overline{c}_p' T_{s_3}}{h}\right] A_0 \rho_0 V_0 \tag{19}$$

where

$$m_{air} = A_0 \rho_0 V_0$$

Combining equations (18) and (19), the specific fuel consumption becomes

$$s.f.c. = \frac{q\,\dfrac{\overline{c}_p T_{s_0}}{h} A_0 \rho_0 V_0 \left[\dfrac{\overline{c}_p' T_{s_3}}{\overline{c}_p T_{s_0}} - 1\right]\left[1 + \dfrac{\overline{c}_p' T_{s_3}}{h}\right]}{\dfrac{\rho_0}{2} V_0^2 A_2 C_F}$$

or

$$s.f.c. = \frac{q\,\overline{c}_p}{\sqrt{\gamma R T_0}}\,\frac{T_0}{h}\,2\,\frac{A_0}{A_2}\,\frac{T_{s_0}}{T_0}\,\frac{\left[\dfrac{\overline{c}_p' T_{s_3}}{\overline{c}_p T_{s_0}} - 1\right]\left[1 + \dfrac{\overline{c}_p' T_{s_3}}{h}\right]}{C_F}$$

Converting to a nondimensional form

$$s.f.c.\sqrt{\frac{\gamma R}{T_0}}\,\frac{h}{q\,\overline{c}_p} = 2\,\frac{A_0}{A_2}\,\frac{T_{s_0}}{T_0}\,\frac{\left[\dfrac{\overline{c}_p' T_{s_3}}{\overline{c}_p T_{s_0}} - 1\right]\left[1 + \dfrac{\overline{c}_p' T_{s_3}}{h}\right]}{C_F} \tag{20}$$

where the right hand side, except for secondary effects, is a function only of the free flight Mach number, ramjet geometry, and throttle setting.

IV. CORRECTION OF FLIGHT TEST AND WIND TUNNEL RESULTS TO STANDARD SEA LEVEL CONDITIONS

In order to compare properly results obtained from free flight or wind tunnel tests conducted under widely varying conditions they should be put on a comparable basis. The nondimensional parameters presented in Section III could be given names, such as air flow coefficient, and used to compare results. It is perhaps better to correct all of the data to some standard atmospheric condition and leave the quantities with their same names, such as air mass flow, and with numbers that mean something to the person using these data.

The air mass flow is the first quantity considered and is given in equation (16). It is a function of the ramjet geometry, flight Mach number, M_o, and temperature ratio, T_3/T_o. The data are to be corrected to sea level standard conditions, which are $T_o = 519^0$ and $p_o = 2116$ p.s.f. The data are corrected for constant values of the independent parameters and mean that

$$m_{air} \frac{\sqrt{T_o}}{A_2 p_o} = \text{CONSTANT}$$

Hence

$$m_{air_{std.}} = m_{air_{test}} \sqrt{\frac{T_{o_{test}}}{519}} \frac{2116}{p_{o_{test}}}$$

In a similar manner the other quantities may be obtained. Thus the fuel-air ratio is

$$\left(\frac{m_{fuel}}{m_{air}}\right)_{std.} = \left(\frac{m_{fuel}}{m_{air}}\right)_{test} \frac{519}{T_{o_{test}}}$$

and the fuel mass flow is

$$m_{fuel_{std.}} = m_{fuel_{test}} \frac{2116\sqrt{519}}{p_{o_{test}}\sqrt{T_{o_{test}}}}$$

The fuel-air ratio and fuel mass flow are both corrected to give the same ratio of heat released to energy of entering air for both the test and the standard conditions. This then corresponds to the same temperature ratio T_3/T_o, or throttle setting.

The thrust is

$$thrust_{std.} = thrust_{test} \frac{2116}{p_{o_{test}}}$$

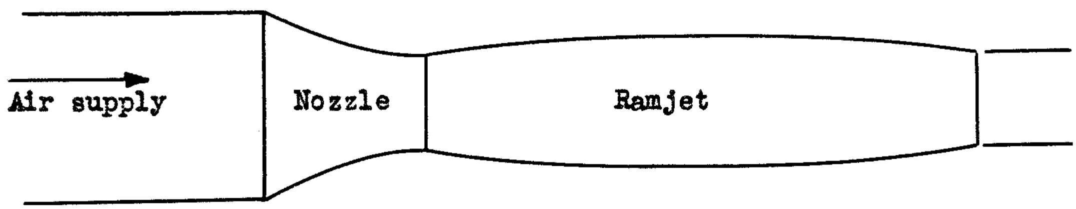
and the specific fuel consumption is

$$s.f.c._{std.} = s.f.c._{test}\sqrt{\frac{519}{T_{o_{test}}}}$$

By correcting data obtained at various altitudes and plotting on a single sheet it is possible to see directly the effect of altitude on such things as combustion efficiency. It is also useful in comparing wind tunnel and flight results obtained under different test conditions.

V. PERFORMANCE FROM CLOSED DUCT TESTS

It is frequently necessary to test ramjet units as a part of a closed duct system and to determine the corresponding flight condition. A sketch of such a set up is shown on Figure 2.



Closed Duct Testing
Figure 2

It is usually assumed that this arrangement simulates the corresponding flight conditions aft of the station at which the air is introduced into the ramjet. This is not strictly true since it would be very difficult to reproduce the same velocity profile and boundary layer. However, it is a fair approximation and is quite necessary. The corresponding flight conditions are determined from the ambient or outside pressure and the pressure and temperature of the air entering the ramjet. Usually a reasonable diffuser efficiency is assumed and the simulated flight conditions determined.

Since the ramjet thrust depends upon the momentum change of the air passing through it, a knowledge of the exhaust velocity is of considerable importance. Very simple measurements of the static and total pressure in the diffuser and at the nozzle exit, and total temperature and mass rate of flow of the entering air can yield rather useful data.

Assuming that the flow is one-dimensional in the ramjet, i.e., the conditions are assumed uniform over the entire cross-section, and using the equations of continuity and state, the temperature of the exhaust may be computed from the expression

$$T_4 = \frac{p_4 A_4 V_4}{m R'}$$

where

$$m = m_{air} + m_{fuel}$$

Since

$$a = \sqrt{\gamma R T}$$

then

$$\sqrt{T_4} = \sqrt{\gamma'} \frac{p_4 A_4 M_4}{m \sqrt{R'}}$$

or

$$T_4 = \frac{\gamma'}{R'} \frac{p_4^2 A_4^2 M_4^2}{m^2}$$

The temperature at any section is thus given if the Mach number and pressure are given since the other quantities can be easily determined from charts and the known mass flow of air and fuel. The Mach number, M_4 , can be obtained from the expression

$$M_4^2 = \frac{2}{\gamma - 1}\left\{\left(\frac{P_4}{p_4}\right)^{\frac{\gamma' - 1}{\gamma'}} - 1\right\}$$

where $M_4 < 1.0$

Thus the temperature of the gas can be determined if the mass flow, impact pressure, and static pressure are given. The fact that γ' depends on the temperature does not give difficulty since the value of γ' is not very sensitive to changes in temperature. Thus, if the temperature is computed using an assumed value of γ' and then recomputed using a corrected value based on the first calculated temperature, it will be quite accurate.

The assumption that the flow is one-dimensional is not correct and methods of obtaining mean values must be considered. The mass flow may be expressed as

$$m = \int_{A_4} \rho_4 V_4 \, dA$$

or

$$m = \int \sqrt{\frac{\gamma'}{R' T_4}} \; p_4 M_4 \, dA$$

In order to obtain a value of   from the known values of   and   and the measured impact and static pressures, mean values must be used and the expression for the mean temperature becomes

$$\overline{T}_4 = \frac{\gamma'}{R'} \; \frac{p_4^2 \, \overline{M_4^2} \, A_4}{m^2}$$

It is a usual practice to measure the impact pressure of the exhaust gases at the end of the exhaust nozzle and assume that the static pressure is that of the surrounding atmosphere. If this condition is not correct and there is a further contraction of the flow after it has passed through the nozzles, the calculated temperature must be corrected. Since this flow must be nearly isentropic, the total pressure measured at the nozzle exit is unchanged as one follows the streamline. Hence, since the pressure is now atmospheric, the calculated mean Mach number is correct and the area is the only quantity that must be corrected. Thus,

$$T_{4'} = \frac{\gamma'}{R'} \frac{p_{atm}^2}{m^2} A_{4'}^2 M_{4'}^2$$

where $M_{4'}$ is the previously calculated mean Mach number based on the total head reading P_4 and the atmospheric pressure. It can readily be seen that a five per cent contraction in the jet area would correspond to a ten per cent lower total temperature. Since

$$a_4' = \sqrt{\gamma' R' T_{4'}}$$

then

$$V_{4'} = \overline{M}_{4'}^2 \gamma' \frac{p_{atm}}{m} A_{4'}$$

The simulated flight conditions are determined from the pressure at the end of the diffuser and an assumed diffuser efficiency, preferably from other test results. The thrust coefficient becomes

$$C_F = 2 \frac{A_0}{A_2}\left(\mu \frac{V_{4'}}{V_0} - 1\right)$$

The specific fuel consumption is

$$s.f.c. = \frac{f(\mu-1)g}{V_0\left(\mu \frac{V_{4'}}{V_0} - 1\right)}$$

These results are then to be corrected to the sea level standard values by the methods of section IV.

If the exhaust velocity is supersonic, the static pressure in the jet is of greater importance and is necessary if the pressure method is to be used in determining the thrust. Corrections must be applied to the pressures to account for the entropy changes in the shock waves preceding such measuring devices. Another reliable method of obtaining the ramjet performance is by the use of flexible ducting and thrust measuring devices. This is quite reliable if the momentum of the air passing through the flexible joint is considered.

The actual heat released in the combustion chamber can be computed by subtracting the heat content of the entering air from the heat content of the exhaust gases. Dissociation products may carry a considerable proportion of the potential heating value of the fuel, but the exact amount is difficult to determine. The heat released in the combustion chamber is given, as before,

$$H = \mu \left(\frac{V_3^2}{2} + \bar{c}_p' T_3 \right) - \left(\frac{V_2^2}{2} + \bar{c}_p T_2 \right)$$

All of the other quantities such as air mass flow, fuel mass flow, and station pressures can be corrected to standard conditions to aid in the interpretation of the data.

V. CONCLUSIONS

It is seen that the most important parameters defining the operation of a ramjet are the flight Mach number, M_0 , the ramjet geometry, and the throttle setting. The flight Mach number and ramjet geometry are easily defined and specified, but the throttle setting must be defined as an energy ratio. This ratio is most easily defined and specified as the energy contained in the fuel introduced into the combustion chamber divided by the energy of the air entering the ramjet. The ratio of the combustion chamber temperature to the free stream temperature is essentially the same parameter.

Secondary effects of variations in altitude may affect the performance results in that there may be corresponding changes in the combustion efficiency or in the combustion chamber gas constants.

NOMENCLATURE

A_x Cross-sectional area, sq ft, (subscript indicates station)

C_F Thrust coefficient (thrust/qA_2)

$\bar{C}_p$ Mean specific heat at constant pressure for air ahead of the combustion chamber, ft lbs/°R slug

$\bar{C}_p'$ Mean specific heat at constant pressure for the combustion products, ft lbs/°R slug

H Heat released in combustion chamber per unit mass of entering air, ft lbs/slug

M_x Local Mach number

P_x Impact pressure, psf

R Gas constant for air ahead of the combustion chamber, ft lbs/°R slug

R' Gas constant for the combustion products, ft lbs/°R slug

T_x Absolute temperature,

V_x Velocity, ft/sec

a_x Local speed of sound, ft/sec

s.f.c. Specific fuel consumption, lbs of fuel per sec per lb of thrust

h Heating value of the fuel, ft lbs/slug

k Combustion chamber friction and eddy loss coefficient

m_{air} Air mass flow, slugs/sec

m_{fuel} Fuel mass flow, slugs/sec

p_x Static pressure, psf

W_{fuel} Fuel weight flow, lbs/sec

γ Ratio of specific heat at constant pressure to specific heat at constant volume for air ahead of combustion chamber

γ' Ratio of specific heat at constant pressure to specific heat at constant volume for combustion products

η_d Diffuser efficiency

η_n Nozzle efficiency

μ Ratio mass flow of combustion products to mass of entering air ($\mu = 1 + \frac{m_{fuel}}{m_{air}}$)

ρ Gas density, slugs/cu ft