of payload can undertake a descent from a circular orbit without the assistance of braking rockets.

(ii) A low wing loading of 30 kg./sq.m. would be necessary for such an aircraft, in order to keep the skin temperatures down. As a result an all-wing design with something like a delta planform is probably most suitable; the wing thickness and aspect ratio would both have to be small, by present-day standards.

(iii) A valuable economy in weight can be brought about by limiting the indicated airspeed during descent to a value as small as reasonable. This relieves the flutter loads acting during descent, and the wing stiffness requirements.

(iv) It then appears that structure—and in particular the wing—would account for 75 per cent. of the all-up weight (of 20 tonnes) of such an aircraft.

(v) In the absence of more suitable skin materials than steel, the transport of much heavier payloads than this to Earth would seem a most difficult problem. However, lighter payloads would present less difficulty.

(vi) The reinforcement to the wing, necessary to strengthen those parts of it subjected to a high temperature, would account possibly for 10 per cent. of the weight of a 20-tonne aircraft.

(vii) It appears that the limit to the skin temperatures reached in the presence of a boundary layer at high flight speeds is determined by considering the conduction of the heat within the surface downstream from the nose.

(viii) The discussion of this paper is restricted mainly to the study of a gradual descent, at a low indicated speed. There exists a possibility that a more rapid descent, using dive brakes designed to provide a retardation of about $5g$, might be more suitable.

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III
INTERPLANETARY FLIGHT

9
Some Limiting Factors of Chemical Rocket Motors*

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Introduction

The prospect of space flight has captured the imagination and retained the interest of a great number of people, many of whom were first attracted by its technical aspects. Among the reasons for this must be included the fact that so many of its problems as yet remain unsolved. Both at present and for some time to come, these problems will provide a stimulating challenge taxing the ingenuity and resourcefulness of workers in a broad range of scientific fields. Among them, we can quite safely include the design and development of the means by which the space vehicle will derive its motive power, since the practicability, if not the possibility, of any venture beyond the atmosphere will depend most of all on the means of propulsion available.

Ten years ago, except perhaps in the minds of those with the most fertile imaginations, the chemical rocket motor was the only practical means worthy of consideration. Within the last decade, however, the release of nuclear energy has been adequately demonstrated. The opportunity has quickly been taken to speculate upon the ways and means by which this new source of power could be utilized to promote flight into space, reference 1 being a notable example. It is perhaps because of, rather than in spite of, the advent of this competitive form of prime mover, that a review of some of the possible capabilities of the chemical rocket is not out of place at this moment. That is in fact what this paper attempts to do, indicating the extent to which improvements in performance might be expected, and relating the resultant problems to those which have already been encountered and overcome.

* First published, February 1953.

To begin with, it must be made quite clear that a *complete* assessment of the chemical rocket's capabilities is too complex and detailed a study for the confines of this short paper. Consequently, only broad and general principles can be discussed and some of the more subtle implications of, for example, the use of higher chamber pressures and temperatures are left unexplored.

Again, it must be understood that no assessment of the effect of the rocket motor's performance, on the capabilities of the vehicle it propels, will give completely the right answer if the study is confined only to the rocket motor. For example, the choice of propellant combination will largely determine the exhaust velocity obtainable. But at the same time, the mean specific gravity of the propellants will also decide the volume of the tanks required, which will affect their weight regardless of the actual amount of propellant carried. Thus will the empty weight of the vehicle be affected, and this will of course critically influence its ultimate performance. The tankage weight will also depend to some extent on the particular duty which the vehicle has to perform, and this might influence, in some way, whether a "light" or "heavy" propellant combination be used. The comparison of a vehicle which is to be accelerated away from the Earth's surface, and one which is only to be accelerated out of an orbit into deeper space, is an example of this. In the former case, the tanks will be necessarily heavier in order to withstand the high acceleration required to minimize air resistance and gravitational losses. In the latter case, however, accelerations and inertial forces can be kept low and the tanks can be of comparably lighter construction.

The above comparison can also be used to illustrate another factor. Any improvement in the efficiency of the rocket motor, by the use of a higher combustion pressure or expansion ratio, will probably result in an increase in its specific weight (i.e. weight per unit of thrust generated). A higher specific weight will have a much more serious effect when the thrust of the motor is large compared with the weight of the vehicle, as in the case of take-off from the Earth's surface, because the motor weight then constitutes a more considerable fraction of the empty weight of the vehicle. In the case of acceleration away from an orbit, however, the motor weight will be a comparatively small item, so that a high specific weight could be tolerated, with the attendant benefit of improved thermodynamic performance.

These examples are cited merely to indicate that the ultimate potentialities of the chemical rocket motor cannot be completely assessed unless some parallel consideration is given to the duty and design of the particular vehicle of which it is the power plant.

Typical Missions

In order to provide some sort of yardstick by means of which the usefulness of the chemical rocket can be measured, three possible expeditions into space have been considered. They are, in effect, expeditions for which the chemical rocket might have been expected to be a suitable propulsive unit. The three cases have been chosen primarily because they represent voyages of progressively increasing difficulty, in the chronological order in which they are likely eventually to be achieved.

The cases considered are (*a*) a take-off from the Earth's surface and the acquisition of sufficient velocity to permit a stable orbit round the Earth, (*b*) acceleration away from this orbit, followed by a landing on the Moon and a return to the orbit, (*c*) acceleration away from the same orbit, leading to a landing on Mars and once again a return to the orbit. In each case the characteristic velocities assumed are those given in reference 2, while that assumed for the last journey is actually about the same as that for an Earth–Moon–Earth journey, if the effect of a small amount of atmospheric braking is included. In each case, calculations have been carried out for a range of structural factors ϵ, this parameter being as defined and used in references 3 and 4. Different numbers of separate steps were considered for each mission, constant values of ϵ being assumed for each step, resulting in optimum values of G, the payload factor. References 5 and 6 discuss the implications of such assumptions and describe the method of calculation used.

Before considering the results in more detail, let us consider some typical values of exhaust velocity and structural factor have already been obtained. Nitric acid, hydrogen peroxide and liquid oxygen burned respectively with aniline, "C stoff" and an ethanol/water mixture are fairly representative proven propellant combinations and these give exhaust velocities of 2·25, 2·2 and 2·35 km./sec. respectively. In addition to these fuels, kerosene or petrol has been also used with each of the above oxidants, the exhaust velocities being then 2·3, 2·3 and 2·5 km./sec. respectively. These exhaust velocities assume expansion ratios currently used (c. 20 : 1), average combustion and expansion losses, and have been corrected for operation at low or zero ambient pressures. With regard to structural factor, V2, WAC Corporal and Viking are probably typical of current practice and these have values of $\epsilon = 0{\cdot}25$, 0·44 and 0·21 respectively. A figure of 2·5 km./sec. thus appears to be about the best exhaust velocity presently obtainable, while a value of $\epsilon = 0{\cdot}20$ (based on existing military and research rockets) represents the present limit of structural efficiency.

Bearing in mind these values, let us consider the three missions one by one. In each case it has been assumed that they would be quite impracticable if it required an initial weight of more than 500 tons to carry 1 ton of payload, and undesirable if this were greater than 200 tons. For the earth satellite project, it appears then, that this would be marginal using chemical propellants already in general use, regardless of the structural efficiencies obtained or the number of steps used. If, however, the exhaust velocity were increased to about 4 km./sec., the project would be clearly practical if a three-step rocket were used. In fact, it would not then be worth while to use more steps than three, unless it were found impossible to obtain structural efficiencies as good as $\epsilon = 0{\cdot}2$.

For the second case, that is, the lunar journey, the use of exhaust velocities of the order of 2 to $2\frac{1}{2}$ km./sec. again leads to an impractical answer. If the exhaust velocity were increased to 4 km./sec., the operation would be feasible if a structural factor of 0·2 and five steps were used. If exhaust velocities of the order of 6 km./sec. were possible to achieve, three steps would suffice, so long as a structural factor no worse than 0·2 was obtainable.

The conclusions to be drawn from a study of the third mission (the Mars journey) are very clearly defined. Unless structural factors better than 0·1 can be obtained, the use of exhaust velocities of the order of 2·0–2·5 km./sec. admit to no solution whatsoever. An increase of exhaust velocity to 4 km./sec. hardly alters the picture unless one can tolerate an overall mass ratio in excess of 6,000 and then only if a structural factor of 0·1 and the use of as many as ten separate steps is assumed. An increase of exhaust velocity to 6 km./sec. admits a possible solution, just within the practical limits which have been arbitrarily assumed.

Summarizing very briefly, therefore, it appears that exhaust velocities of the order of 4 km./sec. are necessary to make even the least difficult mission really practical, and that structural factors better than 0·2 may even then be mandatory. Let us now examine the prospects for achieving such an advance, assuming that the chemical rocket motor is used as the method of propulsion.

Thermodynamics

The exhaust velocity, which is in effect a measure of the efficacy of the motor itself, is a function of a number of dependent and independent factors. If the expansion of the gases in the rocket nozzle takes place down to a pressure equal to the ambient value, the exhaust velocity is given by the equation

$$v_e = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{G\theta}{M}\left\{1 - \left(\frac{P_e}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}} \qquad (1)$$

where (in consistent units)

v_e = exhaust velocity;
γ = ratio of the specific heats of the products of combustion (averaged over the expansion cycle);
G = universal gas constant;
θ = combustion temperature;
M = mean molecular wt. of the gases (averaged over the expansion cycle);
P_e = the ambient pressure;
P_o = the pressure at which the propellants are burned.

If the gases are expanded to a pressure P_x differing from the ambient pressure, an additional term must be added to the above expression to allow for the pressure thrust which must be added to the momentum thrust, and then v_e' becomes the "effective" exhaust velocity given by the equation:

$$v_e' = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{G\theta}{M}\left\{1 - \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}} + \frac{(P_x - P_e)A_x}{dm/dt} \qquad (2)$$

where A_x = exit area of nozzle and dm/dt is the rate of propellant flow, again in consistent units.

In free space where $P_e = O$ or at extremely high altitudes when ambient pressure can be neglected, this equation can be rewritten to:

$$v_e' = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{G\theta}{M}\left\{1 - \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}} + \frac{P_x A_x}{dm/dt} \qquad (3)$$

which can be converted into the form

$$v_e' = \left[\sqrt{\frac{2\gamma}{\gamma - 1}\left\{1 - \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}} + \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}} \Big/ \sqrt{\frac{2\gamma}{\gamma - 1}\left\{1 - \left(\frac{P_x}{P_o}\right)^{\frac{\gamma-1}{\gamma}}\right\}}\right] \times \sqrt{\frac{G\theta}{M}} \qquad (4)$$

$= F_1 \times F_2$ for convenience.

It will be seen from the above that the exhaust velocity can be considered as the product of two separate functions. The first of these is dependent on γ and the expansion ratio P_o/P_x while the second is dependent only on $\frac{\theta}{M}$. It should be observed that in free space, v_e' is independent of the combustion pressure P_o and the exhaust pressure P_x considered separately, but depends only on the ratio of these quantities.

The F_1 and F_2 factors are shown plotted against $\frac{P_o}{P_x}$ and γ, and θ and M, respectively, in Figs. 1 and 2. The beneficial effect on the exhaust velocity of increasing $\frac{P_o}{P_x}$ and θ and decreasing M is apparent from these curves. Shown chain-dotted on Fig. 1 is the corresponding curve of F_1 for $\gamma = 1{\cdot}2$, assuming that expansion takes place not in a vacuum but down to the ambient pressure in each case. The difference between the two $\gamma = 1{\cdot}2$ curves indicates that between 10 and 15 per cent. improvement in effective exhaust velocity and fuel consumption can be obtained when operating *in vacuo*, compared with operation at the same pressure ratio down to an ambient pressure equal to the exhaust pressure.

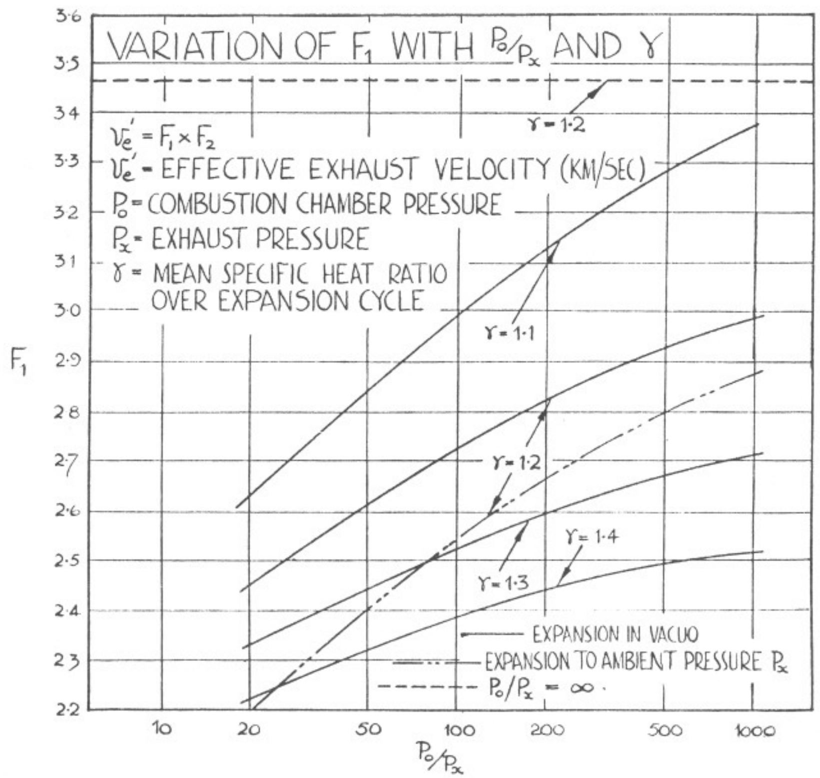


FIG. 1.

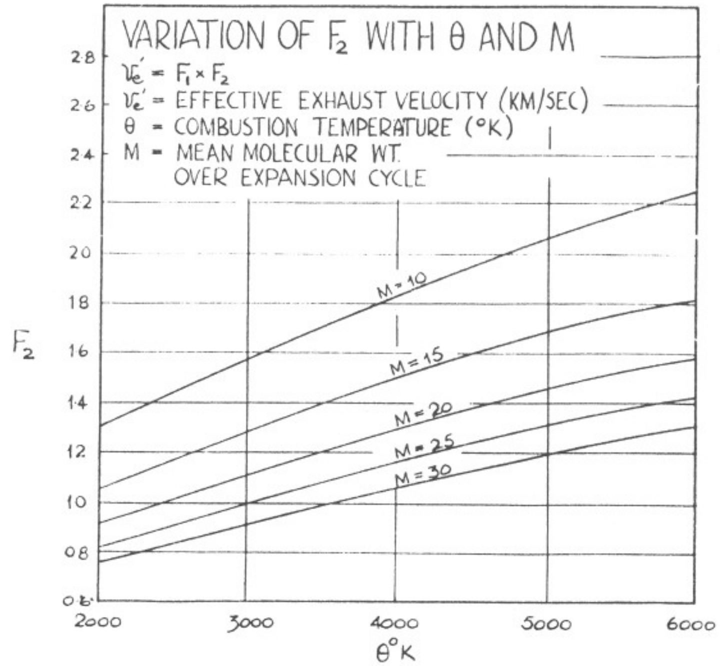


FIG. 2.

Also indicated (as a dashed line for $\gamma = 1{\cdot}2$ again) is the value of F_1 obtained by putting $\frac{P_o}{P_x} = \infty$. The value of v_e' corresponding to this value of F_1 is that which would be obtained using the simple equation $v_e' = \sqrt{2gHJ}$, where H is the calorific value of the propellant combination and J the mechanical equivalent of heat. The improvement in F_1 between this curve and that for practical expansion ratios (about 40 per cent. for $\frac{P_o}{P_x} = 20$ and as much as 15 per cent. even for $\frac{P_o}{P_x} = 1{,}000$) is a measure of the amount one can be misled when determining exhaust velocities by merely assuming that all the chemical energy can be converted into kinetic energy.

Such values of exhaust velocity cannot be realized in practice, because it can never be practicable to design for an infinite expansion ratio or in fact possible to convert from one form of energy to another at 100 per cent. efficiency.

Having established that the effective exhaust velocity is a function of γ, M, $\frac{P_o}{P_x}$ and θ, let us now consider these effects one by one and the limiting values of each which it appears we shall be able to use.

γ AND M

Both γ and M can be dealt with quite quickly. They are both physical properties of the gaseous products of combustion. Both will thus depend on the propellant combination used and can be further varied by adjusting the mixture ratio in which these are burnt. However, the effect on γ of doing this is so small, and will probably be so masked by other changes which will occur (such as one of temperature), that steps to vary γ intentionally need not be considered further.

Mean molecular weight, on the other hand, can have considerably more effect on v_e' because it can be intentionally varied more easily. Its value will vary for different propellant combinations, and those which give rise to products with low values of M show considerable advantage on this score. For example, water vapour formed when oxygen and hydrogen are burned stoichiometrically has an undissociated molecular weight of 18, whereas the mean molecular weight of the products of (say) nitric acid and kerosene (again burned stoichiometrically and again assuming no dissociation) is about 28. Therefore, everything else being equal, a 25 per cent. improvement in v_e' would be obtained by the use of the low molecular weight combination. The effect of this parameter can also be used to advantage in another way, since it is affected not only by the choice of propellant combination, but also, as stated earlier, by the ratio in which they are burned. If one of the propellants burned is appreciably lighter than the other, it may pay not to burn at the stoichiometric mixture ratio but to use the lighter component in excess, thereby decreasing the mean molecular weight of the combustion products. The reduction in molecular weight thus obtained may more than counterbalance the fall in combustion temperature and an improved performance will result. This is particularly the case with the hydrogen/oxygen and hydrogen/fluorine reactions, and curves showing v_e' values, as a function of combustion temperature while the mixture ratio is altered, are presented in Fig. 3. The fact that the highest values of v_e' do not occur at the highest temperatures is almost entirely the result of molecular weight changes. This effect is of course especially important with regard to the cooling of the motor, since it indicates a way in which the highest exhaust velocities can be obtained without having to operate at the maximum temperatures associated

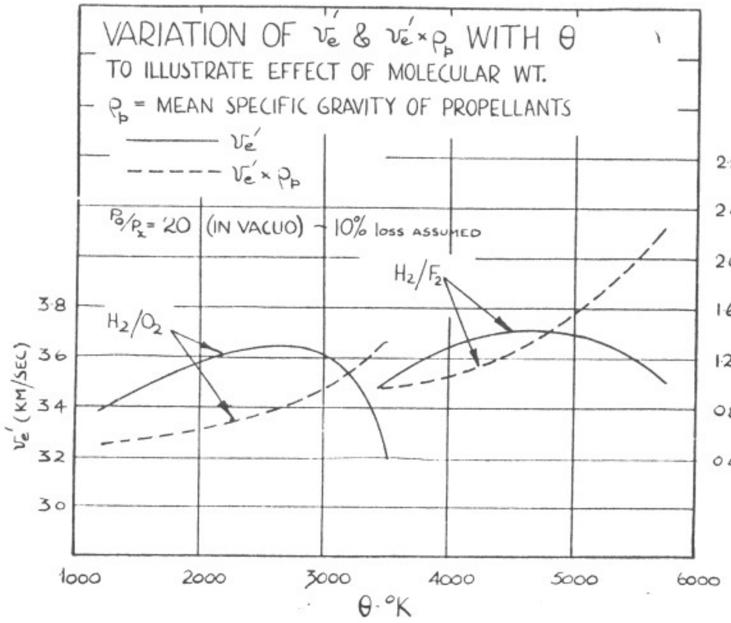


FIG. 3.

with the use of stoichiometric mixture ratios. It must be made clear, however, that such a reduction in operating temperature is not obtained without an attendant disadvantage. This arises from the fact that as the proportion of the lighter propellant is increased, the mean density of the unburnt propellants decreases, aggravating the problem of designing light tanks to accommodate them. This is shown up by the dashed curves, which show how $v_e' \times$ (mean specific gravity of propellants) varies, this factor being proportional to the volume of propellants required for a particular mission, and thus indicating the size of tanks which will be needed.

EXPANSION RATIO AND PRESSURE

Fig. 1 has indicated the way in which the exhaust velocity obtained depends on the pressure ratio over which the combustion gases are expanded. The greater this expansion ratio is, the lower the exhaust temperature of the gases will be and the greater will be the amount of thermal energy which is converted to useful kinetic energy.

It has been shown (Eqn. 4) that for operation *in vacuo*, the exhaust velocity is independent of the absolute value of the chamber pressure (apart, of course, from the secondary effect that this has on the amount of molecular dissociation which occurs and the effect which this then has on the temperature, and consequently on the exhaust velocity). Although the combustion pressure does not directly affect the exhaust velocity it must not be assumed that its choice becomes a completely free one, because it can be shown that under certain conditions, a high expansion ratio can be profitably obtained only by the associated use of a high combustion pressure.

This arises from the fact that the employment of a high expansion ratio with a low chamber pressure results in a chamber and nozzle of very large dimensions. On the other hand, an increase in chamber pressure leads to a proportional decrease in cross-sectional area of both the chamber and nozzle, and also (if the nozzle divergence angle is maintained constant) to a decrease in nozzle length.

Fig. 4 indicates, roughly, to scale, how rocket motor sizes would vary for different chamber pressures and expansion ratios, for

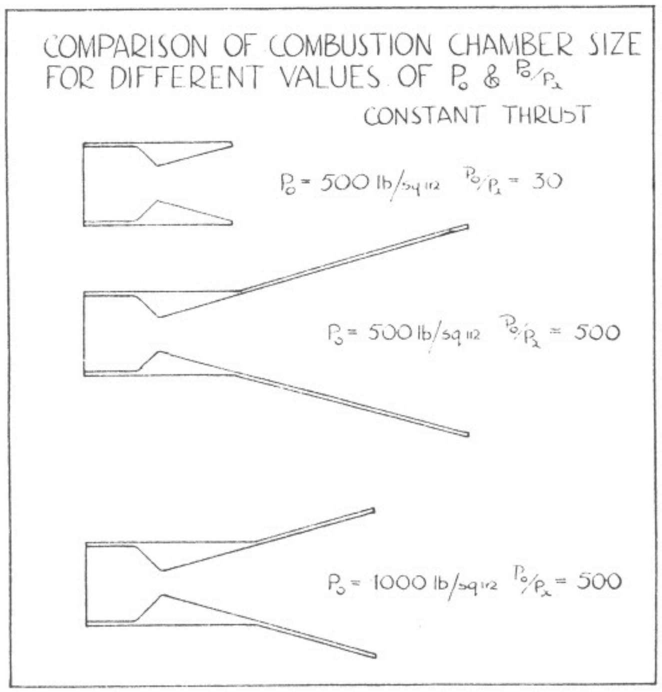


Fig. 4.

motors giving the same thrust. The first configuration represents what one might term good present-day practice, i.e. a chamber pressure of 500 p.s.i. and an expansion ratio of 30. The second indicates how the size would change if the same pressure were used, but if the expansion ratio were increased to 500. The third sketch then illustrates how the size would further vary if the expansion ratio was kept at 500 but if the chamber pressure were increased to 1,000 p.s.i. These comparisons are only intended to serve as a rough guide to indicate how much more space would be occupied by the motor if high chamber pressures are not used to obtain high expansion ratios. They give no indication, of course, of how the relative weights will vary for the different conditions. Although the higher pressure motor appears smaller in outline, it will have to be fabricated in a more robust manner to withstand the greater loads, and a complete weight analysis for the different cases proves to be both a complex and conjectural undertaking.

Such a study becomes rather speculative because it is difficult to decide on exactly what assumptions one should make comparisons, since the nature of these will depend to a great extent on the way in which rocket motor design evolves.

Very broadly, it is permissible to divide the complete motor into two main groups, the pumps, turbines, pipes and valves falling in one, and the combustion chamber in the other. The weight of the first group will fairly clearly increase in some way with pressure. A study of existing systems together with an idealized design investigation would indicate that the increase would not be quite proportional to pressure, but would vary as some lesser power of it. For the purpose of this paper it has been assumed that the weight of all the parts of the motor other than the combustion chamber will vary approximately as $P_o^{\frac{1}{2}}$.The comparison has been made, of course, on a basis of constant thrust, but the improvement in exhaust velocity at higher expansion ratios and the resultant secondary decrease in propellant flow rate has been considered sufficiently small to permit its effect on pump weight to be ignored.

With regard to the chamber weight, the overall proportions of this component are easily determined for each design. The corresponding variation in wall thickness, however, is harder to assess, and will depend on the way in which the chamber is constructed. In this assessment, it has been assumed that the chamber inner wall would not get thicker as pressures and diameters increase but would remain constant regardless of the size of the chamber and the pressure at which it is operating. This is considered to be a specific requirement for effective cooling (see later section) and could be achieved by such methods as balancing out the loads on the wall by equalizing the pressures of the gas inside and the coolant outside, or preferably, by relieving its stresses by supporting it by some means from the cool outer wall. It has also been assumed that regenerative cooling is used and that the outer wall has to withstand the combustion pressure plus a proportional injector and feed line pressure loss. It has been taken that stress levels for corresponding components are constant throughout, while no particular allowance has been made for the proportional reduction in scantling size for the bigger motors. The comparison has, for convenience, been based on a 60,000 lb. thrust motor, it being argued that a higher total thrust could probably best be obtained by a multiplicity of such motors. In any case, the actual size of the motor considered would not greatly affect the conclusions reached.

Fig. 5 records the results of this study showing how the complete

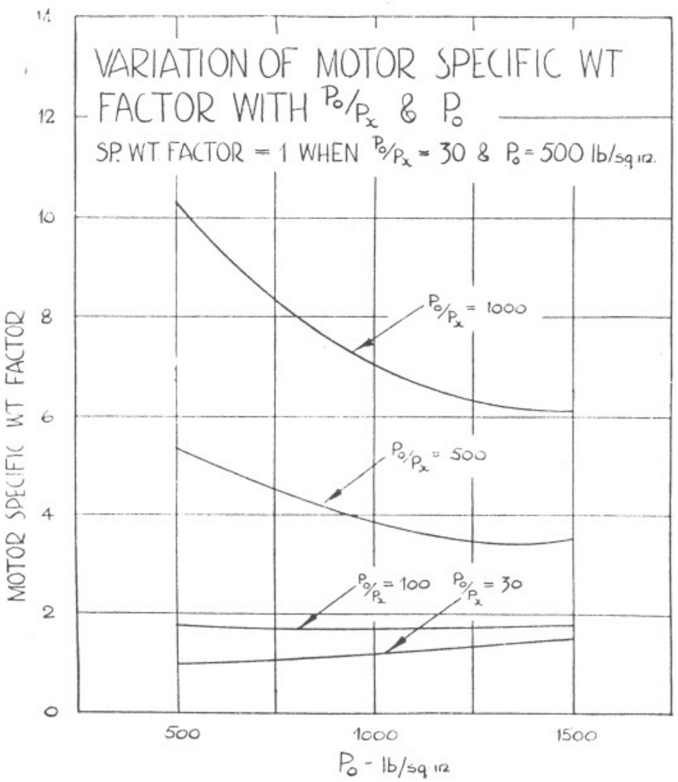


FIG. 5.

rocket motor weight might vary with chamber pressure and expansion ratio. The curves have been plotted on a comparative non-dimensional basis, all the weights being ratioed to that of a motor with a chamber pressure of 500 p.s.i. and an expansion ratio of 30.

The general inference to be drawn is that for expansion ratios less than 100, very little appears to be gained, as far as weight is concerned, by employing chamber pressures greater than those in use today. For ratios of the order of 500–1,000, however, there is a considerable advantage in increasing the chamber pressure up to values far above those to which we are at present accustomed. The weight penalty incurred by the use of such high ratios is, however, considerable, regardless of the pressures used, and if pressures as low as 500 p.s.i. are employed it will be so great as almost certainly to prohibit their use.

Whether the weight increases incurred are worth while or not can be decided only by a detailed study of their effect on the performance of particular projects. In such a study exhaust velocity changes, resulting from variations in expansion ratio, must also be taken into account.

Two of the aforementioned specific cases were therefore considered in this light: (i) the orbital case and (ii) the orbit–Moon–orbit expedition. In each case the vehicle structure and tank weight was assumed to be constant and equal to 16 per cent. of the propellant weight (cf. 19 per cent. for V2), while the rocket motor was assumed for convenience to have the following characteristics:

$$\begin{aligned} \theta &= 5{,}000^\circ \text{ K.} \\ M &= 20 \\ \gamma &= 1{\cdot}2 \\ P_o &= 1{,}000 \text{ p.s.i.} \end{aligned}$$

No combustion or nozzle friction losses were included, while the specific weight of the "basic" motor of Fig. 5 (i.e. one of 500 lb./sq. in. and expansion ratio = 30) was assumed to be 0·025 lb. wt. per lb. of thrust (representing an anticipated improvement over the corresponding approximate figure of 0·036 for the V2 motor).

In Case (i), three steps and an initial acceleration of $1g = 32$ ft./sec. were assumed. The effect of varying expansion ratio on the initial weight required to carry one ton of payload is shown in Fig. 6, which indicates that an expansion ratio of only 55 would give optimum performance. The use of a chamber pressure different from the 1,000 p.s.i. assumed, would have little effect on this result.

For Case (ii), calculations have been made for two variants, one with 16 ft./sec. initial acceleration and the other with 3·2 ft./sec.

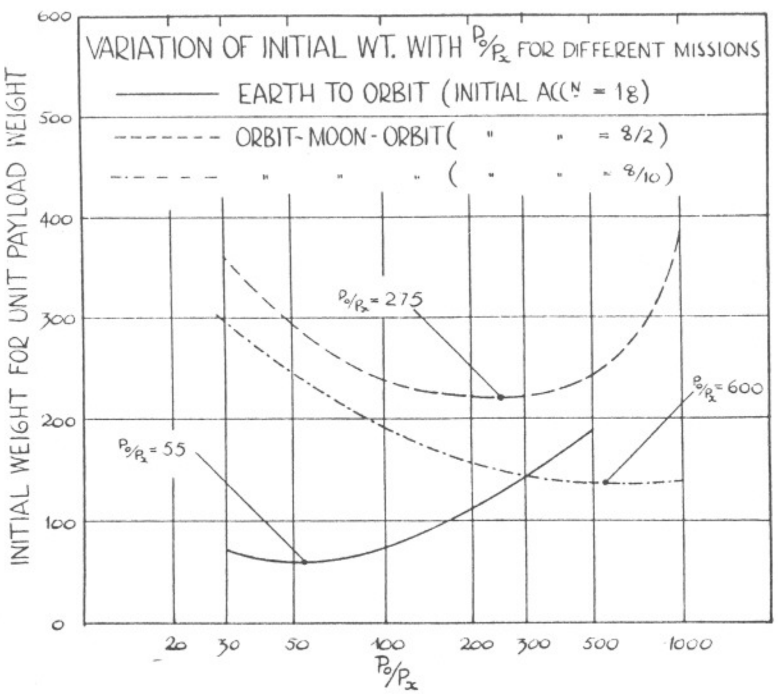


Fig. 6.

(corresponding to $\frac{g}{2}$ and $\frac{g}{10}$ respectively). Once again, 3 steps are assumed. Fig. 6 shows that for the greater acceleration the optimum expansion ratio is 275, while for the lower acceleration this increases to 600. These values would have been greater if higher chamber pressures had been assumed, but because of other considerations, which are discussed later, it is felt that chamber pressures when used in conjunction with high temperatures might, in practice, be limited to even lower values.

These curves are interesting since they show that, according to the assumptions on which they are based, the optimum expansion ratio for the orbital rocket is not far removed from present-day practice. This is particularly fortunate, since there appears to be a definite upper limit to expansion ratio for the motor of a vehicle intended for this duty, dictated by its cross-sectional area. Assuming, for example, that it was designed, because of aerodynamic considerations, to a "fineness ratio" similar to that of V2, it can be shown that the limit of expansion ratio it is possible to use is dependent on the

size of the ship, and will probably get smaller as the weight of the ship increases. If this was assumed to weigh 500 tons at take-off, and have an overall density of 30 lb./cu. ft. (about the same as V2 had), then the limit would be 200 to 1 for a chamber pressure of 1,000 lb./sq. in. If the chamber pressure was reduced to 500 lb./sq. in., then the expansion ratio would be correspondingly further restricted to about 75 to 1.

For vehicles of lower accelerations, the optimum expansion ratios are greater, since the weight of the propulsive unit becomes less compared with the total weight and a greater per cent. weight penalty can be tolerated for a given per cent. performance increase. This effect is, of course, further demonstrated by the even greater optimum expansion ratio which is obtained if the acceleration is limited to the lower value.

The actual values for optimum expansion ratios obtained will, of course, depend not only on the rocket thermodynamic conditions assumed, but also on the assumptions made to determine how the motor weights vary for different conditions. The results which have been presented, therefore, only illustrate a general trend which must exist and show that for expeditions not involving an ascent from the earth's surface expansion ratios considerably greater than those currently used may become accepted practice, if chemical rockets are to be used for this purpose.

Any considerable increase in expansion ratio or chamber pressure will inevitably create new problems or tend to aggravate those that already exist.

Cooling

For example, a higher expansion ratio in requiring a larger nozzle will increase the area which has to be kept cool. The problem of keeping the wall of the nozzle at an acceptable temperature level may not itself be especially severe, for the following reasons. The radiated heat transfer per unit area of the nozzle will be low, because of the comparatively low bulk temperature of the highly expanded gases, while the convected heat transfer rate will be correspondingly restricted by the thick protective boundary layer which will tend to build up in a long nozzle. On the other hand, the actual amount of heat which is transported *to*, and has to be carried away *by*, the coolant will increase, simply because of the greater area through which the heat can flow. The problem of having to cool with a propellant, which is probably an unsuitable coolant anyway, will thus be made more difficult.

The use of high expansion ratios will also increase the friction loss in the nozzle and mean that a greater amount of kinetic energy will be converted back to heat energy and will be lost as far as the attainment of useful exhaust velocity is concerned. While present-day values of nozzle efficiency of about 90 per cent. (based on energy) are achieved, it is possible that lower values than this will have to be accepted if higher expansion ratios are used. A decrease in nozzle efficiency with expansion ratio would, of course, influence the optimum values calculated for different missions, and would reduce them slightly in each case.

An increase in operating pressure as opposed to expansion ratio would have both its advantages and disadvantages. Firstly, most chemical reactions tend to take place more rapidly at high pressures. This might permit the use of smaller values of L^* (resulting in smaller combustion chambers), which in turn would reduce the quantity of heat transferred to the coolant. At very high chamber temperatures, when a much larger part of the heat flow is by radiation and occurs in the chamber, this might be an important consideration. In addition, shorter reaction times would result in more complete and steadier combustion, leading to higher combustion efficiencies. Whereas these at present range between 90 and 95 per cent., higher values of the order of 97–98 per cent. might be expected when increased combustion pressures are used. This especially applies to inherently energetic propellant combinations such as hydrogen and oxygen or hydrogen and fluorine.

With regard to the question of cooling the combustion chamber, any increase in operating pressure will result in an increase in heat transfer rate, which in turn would lead to a rise in wall temperature. Some estimated figures for a hypothetical chamber designed to operate at a temperature of about 4,000° C. illustrate this effect. An increase in operating pressure from 500 to 1,000 p.s.i. would result in an increase in average heat flow rate (per unit area of the chamber) of about 50 per cent. and an increase in wall temperature at the throat of about 200–300° C. However, for a given expansion ratio, the total area to be cooled would go down by about 45 per cent., so that the total amount of heat transferred to the coolant might actually be less at the higher pressure. Over and above this, if regenerative cooling was used, the coolant would have a higher boiling point, and so would be able to absorb the heat more satisfactorily. Thus, it would seem from this cursory study that, using a higher operating pressure would ease the cooling problem as far as the coolant is concerned, but would at the same time result in higher wall temperatures. It would therefore seem possible that this latter consideration might well limit the chamber pressure (and thus indirectly the expansion ratio) which could be used, and that this limitation might be more severe with high temperature propellant combinations.

To sum up, therefore, it appears that the optimum expansion ratios differ for different missions, being not far removed from present-day practice when the weight of the propulsive unit is a large percentage of the weight of the vehicle it propels. It follows that for this case, modest values of chamber pressure are also likely to be used. When the propulsive unit is comparatively smaller, however, optimum values of expansion ratio will be greater, approaching a figure perhaps 10–20 times current practice. Higher expansion ratios will probably entail a cooling problem of greater severity due to the greater area to be cooled, and for the same reason increased nozzle losses due to friction will have to be allowed for. These high expansion ratios are likely to be obtained in conjunction with the use of higher chamber pressures, approaching 1,000 p.s.i. or else combustion chamber weights become excessive. If pressures of this order are used, the problem of keeping the chamber cool is further aggravated, but higher combustion efficiencies might be expected which will tend to counteract the increased nozzle losses.

Temperature

The temperature at which the propellants are burnt can probably have a greater influence on the exhaust velocity than any other single factor. On the other hand, any substantial increase in operating temperature will aggravate the already severe fundamental problem of cooling. The ultimate solution of this problem may depend on completely new techniques and the employment of heat-resisting materials far better than those available at present.

The problem of heat transfer is two-fold. Firstly, the temperature of the chamber wall must not at any point exceed a value at which it is likely to melt, burn away, or soften sufficiently to collapse under load. Secondly, the heat passing through the chamber wall must be safely carried away by some cooling medium which will satisfactorily absorb it and not undergo any undesirable physical or chemical change in the process.

At the present time, chamber temperatures have been limited to much less than 3,000° C. for rocket motors which have had to operate repeatedly or continuously. This limitation has been entirely a structural one, dictated by the heat-resisting properties of the materials from which the chambers have been made. It has rarely been imposed by the limiting thermo-chemical properties of the particular propellant combinations used, since these have usually been burnt at off-stoichiometric mixture ratios, merely to limit the temperatures to values which could be tolerated.

To operate at this order of temperature, combustion chambers without exception have been regeneratively cooled, passing either the oxidant or the fuel round the chamber before it is injected and burnt. The heat transfer rates (of the order of 200–300 CHU/sq. ft./sec. averaged over the whole chamber and 750–1,000 CHU/sq. ft./sec. near the throat where the rate is highest) have been low enough to permit the use of one of the propellants for this purpose, without undue risk that it would boil or decompose. By this means the temperature of the chamber wall in contact with the hot combustion gases has been kept down to about one-third of the gas temperature, even at the throat where the heating condition is most severe.

Heat is passed from the combustion gases to the chamber wall by two distinct processes, by convection within the gaseous boundary layer adjacent to the wall, and by radiation. As the combustion temperature rises the amount of heat conveyed by each process will rapidly increase, and this will, of course, result in a corresponding increase in wall temperature. Because of this it does not appear feasible that combustion temperatures greater than about 3,500° C. can be employed, using simple regenerative cooling means as described above. This limitation has been based on the assumption that maximum wall temperatures must be kept below about 1,500° C., at which temperature most of the most suitable materials in general use would begin to melt or at least soften to such an extent that they would be quickly eroded away by the scrubbing action of the expanding gases.

In order to accommodate the higher temperatures resulting from the use of more energetic propellant combinations, there are a number of detail design improvements which can be made, each of which could affect to advantage the heat transfer process through the chamber wall.

For example, the velocity of the coolant circulating through the jacket surrounding the chamber can be increased. This will have the effect of decreasing the thickness of the boundary layer on the coolant side, and of bringing the wall temperature nearer to that of the relatively cold coolant. Unfortunately, although it goes in the right direction, the resultant change in wall temperature is small (amounting to about 20° C. on the hot side of the wall for a representative case, assuming that the coolant velocity is doubled) and is furthermore accompanied by a small increase in the actual amount of heat transferred. Additionally, of course, any increase in coolant velocity will increase the pressure drop through the coolant

jacket, and a larger and heavier pumping system will be required to overcome this.

A slightly more effective way of reducing the wall temperature, again with its peculiar disadvantages, is to reduce the thickness of the chamber wall itself. For the particular example considered, reducing the wall thickness from 0·1 in. to 0·05 in. would reduce the maximum wall temperature by about 350° C., but at the same time the heat flow through the wall would be increased by nearly 30 per cent. Even if this increased heat flow could be satisfactorily absorbed by the coolant, such a thin chamber wall would not have sufficient strength, especially at its high temperature, to withstand any appreciable differential pressure loads. Steps would have to be taken, therefore, to relieve the chamber of its stresses, either by carefully regulating the coolant pressure with respect to the gas pressure inside the chamber, in order to balance out the load on the wall at its most critical regions, or else by providing support for it, by connecting it to the cool outer wall of the coolant jacket.

A much more effective method of easing the temperature problem, in all respects, is to increase artificially the thickness of the gaseous layer adjacent to the chamber wall, since this would have the effect of reducing at the same time both the wall temperature and the amount of heat transferred. The boundary layer thickness can be increased to give this result by continuously introducing through the chamber wall either a cool gas, or liquid which will then evaporate and augment the natural boundary layer. Film or sweat cooling are variants of this technique which have been successfully employed. For the representative example considered, doubling the thickness of the boundary layer would have the effect of decreasing the hot wall temperature by about 600° C. and reducing the heat transfer rate by nearly 50 per cent. Film cooling was, of course, effectively used in the V2 combustion chamber, in which alcohol was bled into the chamber through a number of small holes adjacent to the throat. A more effective method of achieving a similar result would have been to percolate coolant through a porous chamber wall, instead of using a number of discrete holes. "Sweat" cooling in this way would ensure more even distribution of the coolant, which additionally would cool the chamber wall throughout its thickness.

Up to now, the development for such a process of porous metals which would have sufficient strength and could be made in the necessary shapes and sizes, has not yet advanced sufficiently to enable it to be carried out on any large scale. Laboratory experiments, however, have already demonstrated the effectiveness of this technique, which has the added advantage that when the liquid propellant which is injected is evaporated adjacent to the chamber wall, the absorption of latent heat will further decrease the temperature of the boundary layer and therefore that of the wall itself.

Assuming either film or sweat cooling is employed, the variation of wall temperatures in the chamber and at the throat have been calculated for a range of combustion temperatures, and are shown in Fig. 7. No allowance has been made for latent heat absorption in the gaseous boundary layer, or nuclear boiling of the coolant, and on both of these counts the values obtained will be conservative. Added to this, it has had to be assumed that low temperature heat transfer laws apply throughout, although there is no experimental evidence as yet that this will be the case, and wildly extrapolated figures for viscosity, thermal conductivity, etc., have had to be assumed. The results shown therefore are only indicative of a trend and may be to some extent in error, especially at the higher temperatures. With this uncertainty in mind, therefore, it would appear that combustion temperatures, using existing materials together with provision for sweat cooling, will be limited to about 4,000° C., the criterion still being the maximum wall temperature

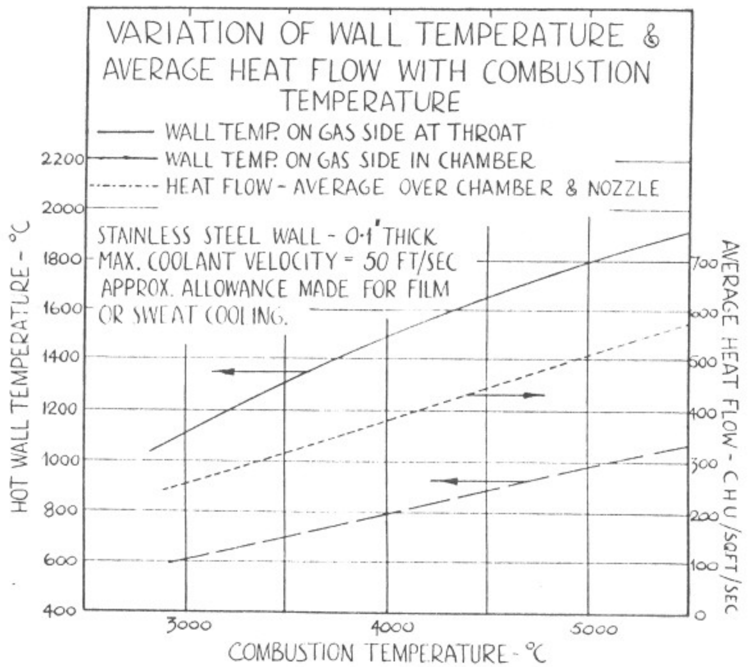


Fig. 7.

at the throat. It is interesting to note, in this respect, that even at very high combustion temperatures, the throat condition is still the most critical, in spite of the greatly increased radiated heat flow in the chamber.

In order that still higher combustion temperatures can be used, and greater values of exhaust velocities obtained, it would appear that the development of better heat-resisting materials with improved strength and scaling properties will be necessary, and that these will be used in conjunction with the detail design refinements to which reference has been made. In this category of improved materials, those which promise the greatest ultimate improvement appear to be of the refractory type. Up to now these have been used only for certain specialized applications, because their poor thermal conductivity and brittleness have resulted in inadequate thermal shock properties. The development of metal/ceramic mixtures (ceramals or ceramets) promises an improvement in these essential properties. If one can assume the eventual use of a low stressed, porous, sweat cooled, ceramet wall which could operate at face temperatures approaching 2,000° C., then it would appear possible to use combustion temperatures of the order of 5,000° C.

Up to now, the effect of higher temperatures on the coolant itself has been largely ignored. As higher temperatures are used, more heat will be transferred to the coolant, until eventually the temperature rise through the jacket will be so great that general boiling will occur, with a breakdown in the heat transfer process. The increase in average heat flow density (in CHU/sq. ft./sec.) with combustion temperature is shown in Fig. 7, from which it will be seen that as this increases from 3,000° C. to 5,000° C., the heat transfer rate for the same basic size of motor will approximately double in amount. An attendant increase in operating pressure (by weakening the extent of the gaseous boundary layer) and an increase in expansion ratio (by increasing the area to be cooled) would further increase the amount of heat transferred.

With comparatively inefficient coolants (e.g. liquid oxygen or hydrogen), it is difficult to imagine how such a quantity of heat could be absorbed satisfactorily. In such cases, it might be necessary to use a secondary coolant fluid, water being an obvious example which might be considered. The heat from this secondary coolant might, for one such possible system, be removed in the form of mechanical work, utilized for auxiliary power or perhaps to drive the propellant pumps, as was proposed in reference 7.

To recapitulate, very briefly, then, it seems probable that with the use of existing techniques and materials, combustion temperatures may well be limited to about 3,500° C. Assuming, however, the development of unstressed chamber walls made of improved heat-resistant alloys, or refractories which will resist thermal shock, then temperatures of the order of 5,000° C. may be ultimately used. Because of the greater amount of heat which will be radiated to the chamber walls, especially upstream of the throat section, film or sweat cooling will almost certainly have to be applied to the whole area to be cooled.

The use of higher chamber pressures and expansion ratios will further aggravate the cooling problem, mainly because the former will increase the heat flux and hence the wall temperature, while the latter will increase the area to be cooled. Since the high temperatures necessary for the attainment of high exhaust velocities will most likely be obtained with low-boiling-point, low-specific-heat propellants (which are also probably highly corrosive), some special cooling fluid may have to be introduced.

Miscellaneous Factors

There are a number of additional factors that affect the choice and determine the overall effectiveness of a propellant combination, other than just the attainment of a high exhaust velocity. Perhaps the most important of these is the effect of the mean propellant density on the tank weight. This consideration has particular bearing on the ultimate usefulness of the chemical rocket, since most of the chemicals which give the best thermodynamic performance tend to fall into the low-density range.

The extent to which low-density propellants are a disadvantage depends on the critical factors which dictate the tank design. To illustrate this, a factor w_t has been introduced, this being the tank specific weight, i.e. the weight of the tank divided by the propellant weight it carries (W_p). Assuming the tanks are of the same geometrical shape, it follows that if they are designed to withstand a constant internal pressure, $w_t \propto \frac{1}{\rho}$ where ρ is the mean propellant density. This relationship would probably apply if the inertia loads on the tank contents were low, as they would be in acceleration away from an orbital refuelling base. A small constant internal pressure is then not an unrealistic design assumption, since this would be necessary to suppress boiling and to keep the pumps satisfactorily primed. For such a duty a spherical tank would represent the optimum shape. If on the other hand the tanks were designed for a given large acceleration, as for example during the take-off from the Earth's surface, the inertia forces would generate greater stresses in the tank wall than those imposed by a small constant internal pressure. Under these circumstances, it can be shown that the pressure generated by the head of liquid will decide the wall thickness. It can then be shown that, assuming a given shape of tank, wall material and stress:

$$W_p \propto V_p\rho \propto d^3\rho$$

(where V_p is the volume and d a linear dimension of the tank). The pressure at any point in the tank p is then $\propto d\rho$ and the wall thickness $\propto pd$ and hence $\propto d^2\rho$.

The weight of the tank is $\propto d^2 \times d^2\rho = d^4\rho$.

The specific wt. w_t is then $\propto d^4\rho/W_p \propto d^4\rho/d^3\rho \propto d$.

But $d \propto (W_p/\rho)^{\frac{1}{3}}$ hence $w_t \propto (W_p/\rho)^{\frac{1}{3}}$.

Whichever assumption is accepted, a low propellant density always results in a heavier tank, which in turn would lead to a secondary increase in structure weight. In addition to this, a further increase would occur for the Earth take-off case, at any rate, if low-boiling-point propellants were used, due to the lagging required. For operation in space, however, this might not be such a serious problem, since external heat could only be *radiated* into the tanks, and its amount might be restricted by suitably treating their external surfaces. Means would also have to be provided, however, to prevent further heat being conducted to the propellant from the motor compartment, etc.

No specific mention has yet been made of any possible limit to the size of chemical rocket motors, and the remarks which have been made up to now have not applied to any one particular size more than another. It is quite possible, however, that although there appears to be no very hard and fast reason why size should be limited, very large thrusts will be obtained not by a single large motor but by a combination of small ones. The reasons suggested for this are as follows:

(1) Small motors will be easier and cheaper to make and develop and will require less elaborate and expensive test equipment.

(2) Different missions could be accomplished with different combinations of such small units, so that a different propulsive unit will not have to be developed for each application.

(3) A multiplicity of such units will give a greater operating reliability than a single unit.

(4) For very large motors it might become more difficult to attain such a low specific weight as for smaller motors.

(5) The cooling problem might become more difficult for large motors; due to the reduction in boundary layer width at high

Reynolds numbers. This trend particularly affects conditions in the expansion nozzle and will therefore apply with greater force when high expansion ratios are used.

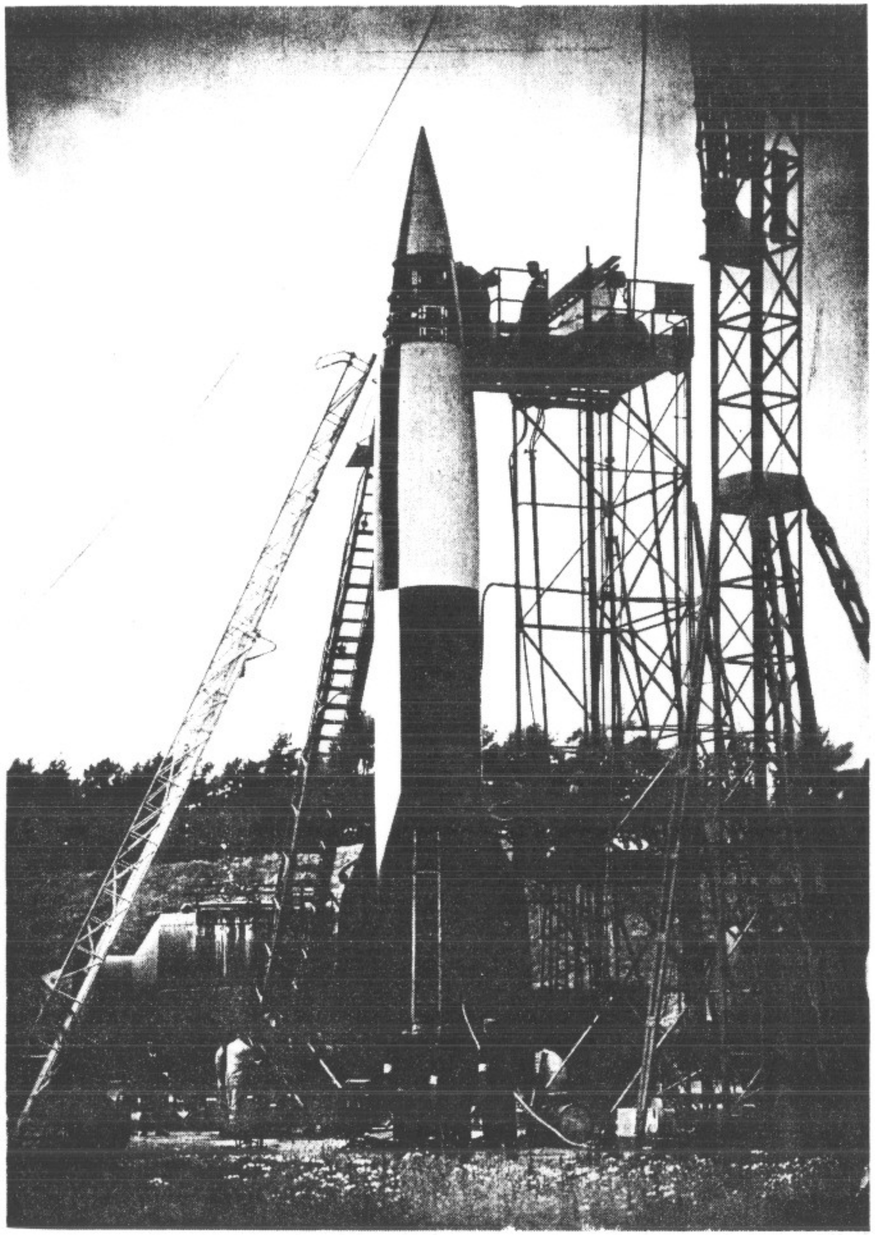
A brief mention must also be made of the means by which the propellant pumps are driven. It is essential that this system is as efficient as possible, since the per cent. flow of propellant required to drive the pumps, unless this is put to some useful propulsive effect, is equivalent to the same per cent. loss in exhaust velocity. For chamber pressures of the order of 300 to 500 p.s.i., about 1 to 5 per cent. of the propellant goes to drive the pumps and since this flow is roughly proportional to chamber pressure, the need for economy is particularly vital if higher pressures still are to be used. The problem becomes even more critical when low-density propellants are used, since the pumping power is proportional to the propellant volume rather than the propellant weight. For the missions considered a 5 per cent. loss in exhaust velocity (e.g. resulting from its equivalent in pump consumption) would result in anything up to 50 per cent. increase in initial weight for the same payload. It would therefore seem very necessary that a system be used which at least is as efficient as possible and also in which the turbine throughput is not thereafter wasted, but is put to some useful propulsive effect. Such a means was referred to in reference 8.

Characteristics of Known Propellants and their Applications to Specific Projects

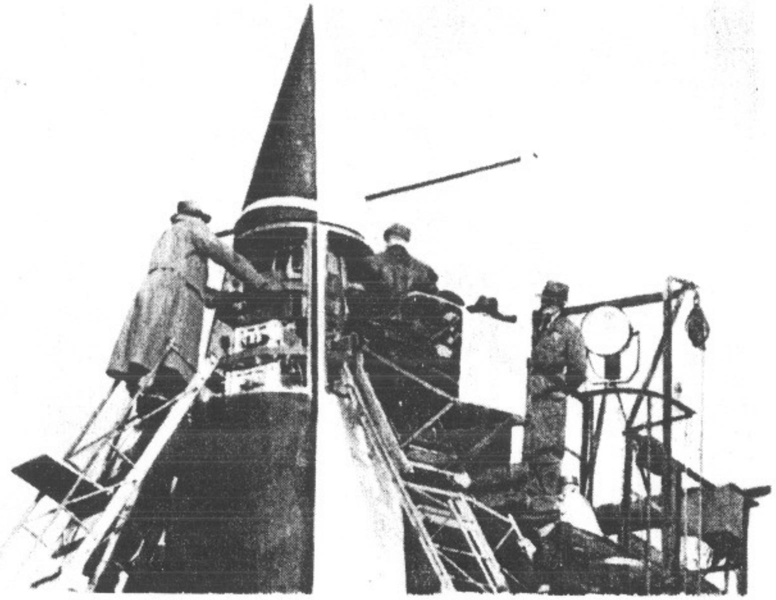
Up to now the increases in exhaust velocity which might be obtained with the chemical rocket have been discussed in the light of some of the physical difficulties which will have to be solved if they are to be achieved. Only a passing reference has been made as yet to the propellant combinations it will be necessary to use in order to make these improvements possible. A complete list of all the possible reactions which occur in nature and which might be considered for use in a rocket motor would indeed be a formidable proposition, since any energy-carrying substance, used in conjunction with another substance with which it will combine to release all or part of its energy, can be considered as a legitimate possibility.

If such a list were prepared it could be divided broadly into three parts, and perhaps fortunately for anyone who might be persuaded to undertake the task, by far the greater part would consist of reactions which are much too ineffectual to be worth considering from the point of view of rocket propulsion, since either the energy released is too small, or the products of combustion too heavy.

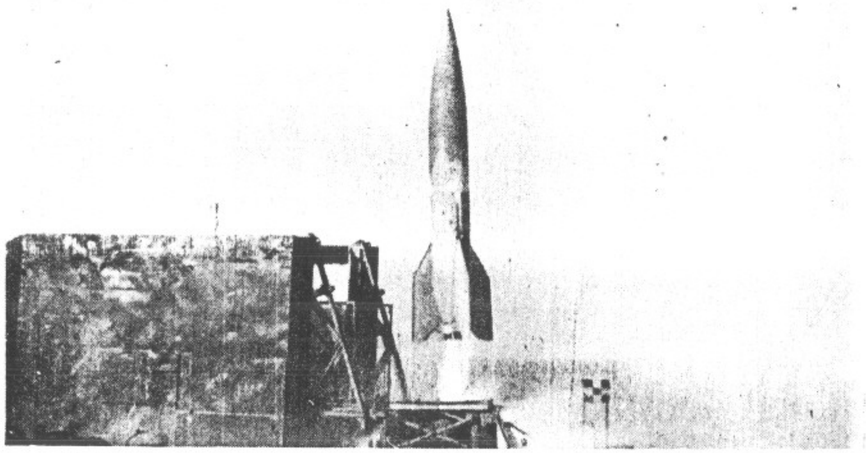
The second of the sub-divisions is shorter and of more interest



8. One of the first A4's, 1942.



9. One of the first A4's. Instrument compartment serviced with fire ladders, summer 1942.



10. A5 taking off, 1938.

since it contains a number of propellant combinations which either are already being exploited in the missile or aircraft fields, or could be with only a little further development of existing techniques. It is unfortunate that almost all these reactions appear to just fail to make the grade as far as promoting interplanetary flight is concerned. Predominant among them are liquid oxygen, hydrogen peroxide and nitric acid as oxidants and various hydrocarbons, amines and alcohols as fuels. Sea-level exhaust velocities at the expansion ratios in general use range between 2 and 2½ km./sec., combustion temperatures attain values between 1,700° C. and 3,200° C., while the mean bulk specific gravity is usually about that of water. The best fuel in this group appears to be hydrazine, which gives about the same performance when used with any one of the three usual oxidants. The performance of the hydrazine/hydrogen peroxide reaction is given in Table I as being, as it were, about the

TABLE I

PERFORMANCE OF DIFFERENT PROPELLANT COMBINATIONS

	Propellants	M.R.*	Mean sp. gravity	v_e'†	v_e'‡	Comb. temp.	
						° C.	
A	Hydrogen peroxide—hydrazine	1·4 : 1	1·24	2·75	2·85	2,400	A
B	Oxygen—hydrogen	8 : 1	0·42	3·65	3·8	3,200	B
C	,,	4 : 1	0·28	4·15	4·25	2,700	C
D	Fluorine—hydrogen	18·9 : 1	0·64	4·0	4·15	5,400	D
E	,,	9·4 : 1	0·37	4·25	4·35	4,400	E
F	,,	7 : 1	0·29	4·15	4·25	3,700	F
—	Oxygen—lithium ..	1·2 : 1	0·66	3·65	3·75	7,000	—
—	Atomic hydrogen ..	—	—	11·0	11·2	4,700	—
G	"*X*" + "*Y*"	—	1·5	5·1	5·25	4,700	G

* Mixture ratio—Oxidant to fuel by weight.
† In km./sec. for $P_o/P_x = 50$ *in vacuo*. 10 per cent. energy loss.
‡ In km./sec. for $P_o/P_x = 250$ *in vacuo*. 20 per cent. energy loss, because of greater friction loss in longer nozzle.

most attractive that the present generation of propellants can offer.

The third and by far the shortest sub-division includes a small number of reactions, which, since they give appreciably higher exhaust velocities than those of the second group, are especially interesting as possibilities for interplanetary flight. Their superior performance is, however, generally given at the expense of higher operating temperatures and lower bulk densities. The two oxidants most concerned are liquid oxygen and liquid fluorine, while the most attractive fuel is liquid hydrogen, with some of the lighter metals and metallic hydrides as second favourites. Liquid ozone might be considered as a slightly more effective but more expensive alternative propellant to liquid oxygen.

The exhaust velocities given by some of these reactions are given in Table I together with some physical data relating to them. Comparative data for some of these and numerous other propellant combinations are given in references 9, 10, 11 and 12. The velocities given here are for different expansion ratios and are effective (i.e. they assume that the operation takes place with no ambient pressure) while reasonable allowances have been made for combustion, friction and pumping losses. A greater friction loss has been assumed for the higher expansion ratios. The different values given for the oxygen/hydrogen and fluorine/hydrogen relate to different mixture ratios, the first two in each case corresponding to stoichiometric and maximum v_e' conditions respectively.

This table is only approximate, since to prepare it considerable extrapolation, interpolation and a lot of guesswork was used. Even so, it does indicate very clearly how much more effective are the reactions involving hydrogen as compared with those in present-day use. It also shows the penalties which have to be paid, e.g. the much higher temperatures of the fluorine reactions and the poor bulk specific gravities when oxygen is used.

It will be apparent from the figures that the temperatures obtained with hydrogen and oxygen are not greatly above those which have already been used, especially when an excess of hydrogen is used. There would seem, therefore, to be no fundamental difficulty in the use of these propellants together, except that arising from the fact that neither of them is very suitable as a coolant. However, if an auxiliary coolant is used, that disadvantage would be obviated. To what extent the lower bulk densities associated with the lower-temperature mixture ratios are detrimental can only be decided by a more detailed study referred to a specific requirement. The use of fluorine, on the other hand, even at its optimum condition, imposes a severe temperature problem. Because of this, the necessity to run even richer in hydrogen must be considered, and the performance when the temperature is limited to 4,000° K. is included in Table I. Also included is an estimate of the lithium/oxygen reaction, as being an example of the use of a metallic fuel. Although these give rise to higher bulk densities than liquefied gases, it appears doubtful that they will find extensive use, because of the high molecular weight of their combustion products. Consequently high exhaust velocities can be obtained only at the expense of high temperatures.

The atomic hydrogen reaction is also included mainly for the sake of completeness. It must be made quite clear that its inclusion

is an academic one, since, as far as can be foreseen at present, this reaction cannot be considered for a rocket motor because of the storage difficulty. Mon-atomic hydrogen cannot be contained in liquid form, and even in its gaseous form has an approximate life of less than a second before it reverts to its normal form. Apart from this extremely fundamental drawback, the temperature when it does react is quite high, especially compared with, say, the reaction of hydrogen and oxygen.

It is fairly unlikely that there are any natural exothermic chemical reactions as yet completely unknown which would be suitable for use in a rocket motor. The information given in Table I therefore indicates the extent to which the performance of the chemical rocket motor is limited by the chemicals at our disposal, regardless of whether we shall be able to overcome the physical problems incurred when using them. Although we cannot envisage at the present time how synthetic fuels of a greater energy content than those in existence could be manufactured, the ultimate possibility of doing this is not without interest. If by inventing, so to speak, new molecules, it might also be possible to choose them so that when they are burnt with suitable oxidants, which would also be "invented" for the purpose, then products of low molecular weight would result.

It would therefore seem reasonable to specify that such "tailor-made" propellants should have the following properties:

(*a*) A combustion temperature which it might be possible to utilize without undue difficulty.
(*b*) Products of combustion with a low molecular weight (such as might be given by a large proportion of water vapour, together with some free hydrogen).
(*c*) A mean propellant density appreciably higher than that for known high performance propellants.

We might then expect that burning such a hypothetical oxidant "*X*" with an equally hypothetical fuel "*Y*" would give the following performance:

$$\theta = 5{,}000° \text{ K.}$$
$$M = 10$$
$$\gamma = 1{\cdot}2$$
$$v_e'(P_o/P_x = 50) = 5{\cdot}1 \text{ km./sec.}$$
$$v_e'(P_o/P_x = 250) = 5{\cdot}25 \text{ km./sec.}$$
$$\rho = 1{\cdot}5$$

We shall proceed in a moment to see how such a combination would increase the field of usefulness of the chemical rocket. As stated at the beginning of this paper, the overall effectiveness of any particular propellant combination can be fairly assessed only by the study of its application to a particular project. For this reason the use of some of the combinations given in Table I have been assumed for the three missions which were postulated earlier. The results of this assessment are given in Table II, the weights given being initial values which would be required to transport one ton of payload.

TABLE II

INITIAL WEIGHTS TO CARRY UNIT WEIGHT OF PAYLOAD

	Propellants	Orbital case v_c = 10 km./sec. 3 steps				Orbit–Moon–orbit v_c = 16 km./sec. 5 steps				Orbit–Mars–orbit v_c = 30 km./sec. 5 steps			
		ε*	Wt.*	ε†	Wt.	ε†	Wt.†	ε‡	Wt.‡	ε†	Wt.†	ε‡	Wt.‡
A	Hydrogen peroxide hydrazine	0·22	1,000	0·20	550	0·16	3,450	0·12	1,470	—	—	—	—
B	Oxygen hydrogen	0·27	170	0·24	100	0·21	590	0·17	325	—	—	—	—
C	,, ,,	0·29	90	0·27	70	0·23	310	0·18	165	—	—	—	—
D	Fluorine hydrogen	0·24	60	0·22	45	0·19	230	0·15	150	—	—	—	—
E	,, ,,	0·28	60	0·25	45	0·22	230	0·17	135	0·20	87,000	0·17	9,300
F	,, ,,	0·29	90	0·27	70	0·23	310	0·18	165	—	—	—	—
G	"*X*" + "*Y*"	0·23	20	0·22	15	0·17	55	0·14	45	0·16	4,000	0·13	2,000

* These figures assume approx. V2 values for structural efficiency and tank weights, suitably modified for propellant density, and a 20 per cent. improvement in motor specific weight.

† As above, except for a 20 per cent. improvement in structural efficiency and tank weight compared with V2.

‡ As above, except for a 40 per cent. improvement in structural efficiency and tank weight compared with V2.

Since we are now considering actual propellants, more realistic estimates can be made of the tank weights and the corresponding structure weights which might be expected. For convenience it has been assumed that the tank weights vary inversely as the propellant density, which is probably true for the second two missions where low accelerations are in order, but unfair to the low-density combinations for the orbital case. For this case, a three-step programme was used as a datum, while for the other two five steps were assumed in each case. Throughout, it was assumed that at least a 20 per cent. improvement in motor specific weight compared with V2 could be realized, the data assumed for V2 being that given in reference 13. For the orbital case, calculations were carried out (*a*) assuming no improvement in structural efficiency or specific tank weight and (*b*) assuming that both of these might be improved by 20 per cent. For the other two cases, because of the lower accelerations and consequently stresses involved, it was assumed that this 20 per cent. improvement could be achieved anyway. Initial weights are therefore given on this assumption, and have also been calculated assuming that, by the use of stronger materials and advanced design, as much as 40 per cent. improvement compared with V2 could ultimately be achieved. For each case the calculated optimum values of rocket expansion

ratio were assumed, that for the Mars journey being taken to be the same as that for the Moon trip. For this project only the best hydrogen/fluorine and the hypothetical "*X*" + "*Y*" reactions were considered worthy of study.

The results of these calculations can be summarized in the following manner, which, in addition, serves as a fairly considered statement of the limited potential of the chemical rocket, based on our present knowledge but assuming reasonable improvements in structural design and motor development.

For the establishment of an orbital rocket, the hydrazine/hydrogen peroxide motor or one of about the same performance would provide a marginal answer provided great ingenuity was exercised to reduce structure weights to a minimum. Hydrogen and oxygen burnt to give a moderate temperature would make the task comparatively easy, while the use of hydrogen and fluorine would make it easier still. There seems little to be gained by burning these stoichiometrically and the lower temperatures given by a hydrogen-rich reaction can be used with advantage. If such synthetic propellants as "*X*" and "*Y*" existed, a reduction in take-off weight of nearly 80 per cent. compared with that using hydrogen and oxygen would result.

For the lunar journey, the use of such propellants as hydrazine and hydrogen peroxide is clearly out of the question, while the use of hydrogen and oxygen, even at its optimum exhaust velocity, is marginal. Hydrogen and fluorine result in a more practical solution, and again there seems little advantage in burning these propellants stoichiometrically.

For the Mars project, using an orbital base as a starting point, or for a lunar journey, taking off from the Earth's surface, there are no known propellant combinations which give sufficiently high exhaust velocities. Even using fluorine and hydrogen to give its optimum performance and the best possible estimate of structural efficiency, a quite impracticable mass ratio value is obtained. Even the assumption of a hypothetical synthetic propellant combination fails to result in practical solutions for these particular journeys.

GENERAL CONCLUSIONS

It would appear from this superficial study that to achieve the best results with chemical rockets, appreciably higher expansion ratios than have been used up to now will be needed. Associated with these it will probably be necessary to employ higher chamber pressures, in order to keep both the weight and the size of the propulsive units to acceptable values.

The engineering problems resulting from the elevated combustion temperatures of high energy propellants cannot be over-emphasized and it must be remembered that these difficulties will be accentuated by the employment of the higher expansion ratios and pressures which would appear to be necessary. It is quite possible that both of these, for rather different reasons, will be limited by considerations of cooling the combustion chamber.

It is reasonable to expect that the progressive development of existing techniques will permit the use of temperatures perhaps 1,000° C. higher than those already in use. But to exploit to the full the most effective propellants we know of, much more radical development of new methods and the use of new materials will be necessary. The more marginal the use of the chemical rocket becomes, the greater will be the need to reduce all losses to a minimum and to strive for the utmost efficiency in all aspects of design.

Even if we can overcome all the difficulties incurred by their use, the chemical combinations existing in nature which can profitably by used in the rocket motor are very limited. The tasks which can be accomplished with them are correspondingly limited. Of the propellants which have already been used, or could be without a great deal of technical difficulty, only the very best would be powerful enough for a manned venture beyond the atmosphere, and then only for the establishment of an earth satellite vehicle. The most energetic of the known propellants would make a return lunar journey possible if an orbital base were used as a starting point for this. Unless sweeping improvements in structural design were possible, to an extent which does not seem probable, this particular journey would be impracticable using contemporary propellants. There appear to be no propellants at all which would enable a return journey to be made to the Moon directly from the Earth's surface, or to Mars from an orbital base or to the more remote planets, unless extravagant mass ratios were considered acceptable. It is doubtful whether the advent of synthetic propellants which could be used without great difficulty, would extend the field of usefulness of the chemical rocket appreciably. On the other hand, if they could be made, they might make comparatively easy and practical those tasks which with known chemical propellants are only marginal.

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