THE COLLEGE OF AERONAUTICS

CRANFIELD

POST NUBES LUX

ATMOSPHERE BREATHING ENGINES IN ASTRONAUTICS

by

S. W. GREENWOOD and D. S. CARTON

NOTE NO. 88.
October, 1958.

THE COLLEGE OF AERONAUTICS

CRANFIELD

Atmosphere Breathing Engines in Astronautics

Part I. Flight in the Earth's atmosphere

- by -

S.W. Greenwood, B.Sc.(Eng.), M.Eng., A.M.I.Mech.E., A.F.R.Ae.S.

Part II. Flight in the atmospheres of other planets

- by -

Dennis S. Carton, A.F.R.Ae.S.

SUMMARY

The contents of this note fall into two sections. Part I considers the possibilities and problems involved in using ramjets as a power source for one of the stages of a satellite launching vehicle or similar project. In comparing such a system with rocket powered vehicles, consideration is given to both performance and mass of the various systems. Various trajectories are considered. This work includes a reassessment of projects that have been suggested elsewhere.

The second part examines the possibility of using forms of ramjet in the atmosphere of other planets. Because there is insufficient knowledge of these atmospheres, a study has been carried out to determine the approximate performance of a chemical ramjet in atmospheres of Methane, Ammonia, Hydrogen and Carbon Dioxide at Mach 3. The work in Part II is original, there being no previously reported papers on the subject known.

These studies, which are necessarily based on several simplifying assumptions, indicate that applications for these engines may be expected to arise in astronautics, and that this is a fruitful field for further studies.

Paper delivered to the Midlands Branch of the British Interplanetary Society on Saturday, 9th November, 1957.

CONTENTS

PART I

1.1. Introduction

The rocket engine has long been the favourite means of propulsion among those carrying out paper studies of possible space projects, and indeed it is one of the few systems capable of operating in the near-vacuum of space. As is well known, its most severe limitation is its high specific propellant consumption, and it is of interest to explore the possibility of using atmosphere-breathing engines, which are more efficient from this point of view, during periods of flight in planetary atmospheres. In this part of the paper we are concerned with flight in the Earth's atmosphere, and therefore, primarily, with entry into Earth satellite orbits.

Several writers have looked into this problem, and their views will be examined. That the scheme is worth our consideration is indicated by the practice adopted as early as 1947 in the United States of launching experimental rocket-engine powered research aircraft from conventional aircraft powered by air-breathing piston engines. If the piston engine has been found useful, surely this will prove to be the case also with air-breathing engines now being developed for flight at high supersonic speeds.

1.2. Engine Comparisons

When making paper comparisons of vehicles fitted with different types of engine it is customary to take "typical" performance data for each class of engine. This is a convenient but rather dangerous technique, for the results finally obtained are no more reliable than the original assumptions. Comparisons of total vehicle weight, for example, should be regarded as having an elastic property. Sometimes apparently relatively small changes in engine performance will have the effect of reversing a previously drawn conclusion. Having sounded a note of caution, it is nevertheless of interest to attempt such comparisons in the light of engineering data currently available.

While there are many interesting hybrids and engine combinations possible, it will be assumed for simplicity that only three main engine types will be used, and only one engine type per vehicle stage. The engines considered are the rocket, ramjet and turbojet.

The rocket operates by discharging material initially carried by the vehicle. It is thus independent of any surrounding atmosphere. The ramjet and turbojet are atmosphere-breathing engines, the thrust being obtained by discharging atmospheric gas (together with injected fuel in some cases) at a higher velocity than it was entrained. The principles of these engines are fairly well understood and will not be dealt with at length here. A ramjet for supersonic operation is shown in Fig. 1. The supersonic turbojet would look rather similar to this with a compressorturbine set interposed between the intake and exhaust nozzle.

A comparison of different engine types for flight in the Earth's atmosphere has been dealt with in a number of papers. (1, 2, 3). Air-breathing engines can only be operated within a limited range of altitudes and speeds. Fig. 2 shows in generalised form the "limitations" of ramjet operation. The top point of the range might be put at Mach 5 at 120,000 ft. The turbo-jet experiences similar restrictions, the main difference being that it produces thrust from rest. The rocket as an engine experiences no altitude-speed restriction, although at low altitudes a rocket vehicle would be subject to speed limitations due to aerodynamic heating at high speeds just as any other vehicle would.

For sustained flight in the atmosphere at a constant speed, Fig. 3 gives an approximate idea of the flight time at which a ramjet powered vehicle would be lighter than a rocket powered vehicle. (Assumptions and calculations are given in Appendix I.) At low altitudes this occurs after only a few seconds ! At high altitudes it may take a minute or more. The reason for this is that at high altitudes a ramjet engine produces less thrust for a given weight, and therefore a longer period of time has to elapse before the relatively low specific propellant consumption of the engine results in an advantage. Longer flight times are required for the turbojet than for the ramjet, as the turbojet is a heavier engine.

It is again necessary to advise caution in interpreting information such as that shown on Fig. 3. However, it does bring out one of the main features of rocket-air-breathing engine comparison.

While it is obviously desirable to aim at a low final vehicle weight, a great number of other factors have to be considered in practice. These include cost, reliability, ease of development, safety and simplicity. In this study we shall consider mainly comparisons of vehicle weight.

1.3. Influence of flight programme

For an adequate analysis, proper consideration must be given to suitable flight paths and programming. This is too long a job for a single worker, however enthusiastic, and a brief attempt has been made to deal with this by considering two alternative flight paths together with rather arbitrary accelerations. It is hoped however that this will serve as an introduction to the problem.

The simplest and most obvious path to consider first is that of vertical ascent. The only paper known to the author dealing with this is by Tsien[4]. His analysis has a number of rather severe limitations - in particular he allows a ramjet powered vehicle to produce thrust from rest, - a condition not realisable in practice !

The method selected by the author was to examine the thrust produced by a ramjet engine of given size during vertical flight assuming in turn the conditions listed on Fig. 4. The assumption of constant acceleration was made to establish the speed-altitude variation. It is clear that the resulting thrust variations therefore imply that an additional and variable thrust producing system would be necessary to give the constant acceleration assumed. However, an approximation may be made by taking the engine thrust as substantially constant over the middle part of the speed range. (Reduction in fuel weight will, in any event, tend to compensate for falling thrust in the upper part of the range.)

Conditions to which the engine would be subjected during a typical flight are shown in Fig. 5. The convention of plotting altitude vertically, though it is the independent variable, has been adopted. Conditions at a given altitude are obtained by reading across to the appropriate curve and then down to the relevant line. It will be appreciated that mechanical troubles arising from high ram temperatures will increase with altitude, although this is to some extent alleviated by the fall in ram pressure. The flight velocity progressively increases, although the flight Mach No. falls off for a while at the upper end of the range owing to the increase in atmospheric temperature above the stratosphere.

The constant thrust trajectory referred to on Fig. 4 cannot be obtained in vertical flight without a sharp increase in acceleration at medium altitudes. This may be seen from Fig. 6 where the required speed-altitude variation is given. Avoidance of a fall off in engine thrust with altitude is clearly desirable, and in practice this will call for a departure from vertical ascent and some levelling-off at medium altitudes.

The relative weights of ramjet and rocket powered vehicles over realistic ramjet speed ranges for constant payloads and for the trajectories and acceleration programmes given above are shown in Fig. 7. These quantities have the elastic properties mentioned earlier, and tend to penalise the rocket which is made to operate over a limited speed range. It does indicate, however, that ramjet powered vehicles, to be competitive with rocket powered vehicles, (a) need to be boosted to speed at low altitudes, (b) need to follow a trajectory which sustains the thrust over the speed range.

1.4. Satellite vehicles

An attempt was made to compare vehicles for establishing a fixed payload in a circular orbit near the Earth. (Note - these studies were carried out prior to the first satellite launching.) A conventional three-stage rocket was compared with a two-stage rocket launched from a ramjet powered stage that had been accelerated to operational speed by a solid propellant booster. The resulting total vehicle weights are shown in Fig. 8. The conventional rocket vehicle is lighter and simpler, having one fewer stage. One of the reasons for this is that the ramjet stage final velocity is only about 1 mile/sec., thus forcing upon each of the two stages of its "payload" a higher incremental velocity requirement than each of the three stages of the comparison rocket vehicle.

A much earlier investigation of the satellite launching problem was made by Proell[5]. His analysis has certain weaknesses in detail, but he advocates the use of a slow take-off type of space ship as the only practical type. He examined the possibility of varying speed and altitude using a winged ramjet ship - to carry the payload almost to orbital velocity ! This looks frightening at present owing to the high ram temperatures that would be experienced. Nevertheless the principle is sound, and the more the speed range of the ramjet can be stretched in future, the more interesting this proposal becomes.

Possibly one of the most interesting studies in recent years was that carried out by Sandorff[6]. He considered the problem of establishing a 500 lb. payload in a circular orbit using either a conventional three stage rocket or a two stage rocket launched by an aircraft powered by turbojets. The speed altitude variation is shown in Fig. 9. The vehicle weights are given in Fig. 10. Again the conventional rocket vehicle is seen to be lighter in weight. Some important factors are, however, emphasised by Sandorff. Firstly the weight of the expensive rocket component is halved. Secondly, the aeroplane first stage is likely to be more reliable than a rocket first stage, and the overall reliability of the system should be increased. Finally, the aeroplane stage is recoverable.

It would appear that this scheme is an attractive alternative to the conventional rocket scheme. Sandorff envisages developments in which ramjets are used to increase the final speed of the aeroplane stage.

Some proposals from M. Varvarov[7] of Russia are of interest. His speed-altitude variation for a combined turbojet-ramjet-rocket vehicle is shown in Fig. 9. Engines would be jettisoned at the end of each phase of operation. As an alternative he suggests a series of vehicles each with its own type of power plant. His selection of altitudes at which a given speed is to be reached appears to be on the high side for the development of an adequate thrust from the air-breathing engines.

1.5. Conclusion

Air breathing engines appear to offer a reasonably attractive alternative to the rocket engine for some applications of astronautical interest during flight in the Earth's atmosphere.

PART II

2.1. Introduction

Into the foreseeable future mass will continue to be a most expensive and difficult property to project any distance away from the Earth. Since, as far as a vehicle is concerned, the production of thrust is mass consuming it will always be imperative to produce it in the most efficient manner. To date, all projected studies of interplanetary voyages appear to have considered some form of rocket engine as an automatic selection for all stages of the journey outside our own atmosphere. Greenwood, in the first part of this paper, and others have shown that air breathing engines may be able to play a useful part in the initial stage of a launching programme from the Earth's surface. Looking further, in distance and time, this part investigates the possibility of using the atmospheres of other planets as a means of producing thrust more efficiently than the rocket engine. It is thought that such a possibility will be of interest to those who may study the feasibility of journeys involving either circumnavigation of, or landing and subsequent take off from, any of the bodies of the solar system possessing considerable atmospheres. See Fig. 11.

2.2. Atmospheres of the Solar System

Because of the physical difficulties involved in viewing and analysing the rest of the solar system from the bottom of our own atmosphere, the information available upon the chemical composition of the various planetary atmospheres is vague and contradictory. Unfortunately, as far as this study is concerned, opinions conflict not only upon the amounts of various constituents present, but upon the absence, or presence of a particular gas in some considerable proportion. There is in addition practically no information at all upon the temperature, pressures and densities at points within atmospheric envelopes.

Obviously the comparative methods discussed in the first part of this paper have only been possible because adequate information is available upon the chemistry and distribution of the Earth's atmosphere. Such reliable comparisons of the relative merits of atmospheric breathing and rocket engines for specific applications on other planets will only become possible when our knowledge of their atmospheres becomes as extensive as our knowledge of our own. These difficulties have forced the author to define, hypothetical atmospheres, and then to investigate the possibility of using engines in them.

Each hypothetical atmosphere has, therefore, been defined as being made up entirely of one gas. The choice of these gases has been governed by the main atmospheric ingredients of the solar system excluding Earth. On this point there is a remarkable lack of disagreement amongst various authorities; in consequence, hydrogen, methane, ammonia and carbon dioxide have been chosen. The first three, all capable of producing heat when chemically reacted with oxygen, are classified as "fuel atmospheres" and are treated in similar manner.

Carbon dioxide, on the other hand, is far too stable to be used in such chemical processes and so receives separate consideration.

The next problem was the selection of suitable temperatures and pressures for these hypothetical atmospheres. To simplify the initial investigation, and to make the results comparable, it was decided to use the same temperature and pressure figures for each atmosphere. Since the larger planets, Saturn and Jupiter, have considerable atmospheres, they were used to give reasonable figures for this first tentative work. Finally, $123^{\circ}K$ ($-150^{\circ}C$) and 14.7 lb/in^2 were selected as being possible and fairly typical. Only this one spot point was considered; no work has yet been carried out on "altitude effects".

2.3. Choice of Engine for Study

It is a very interesting fact that all forms of air breathing internal combustion engine appear to be capable of redesign to enable them to operate within any of the fuel atmospheres. The main modification required would be to convert the existing fuel system so that it would meter and inject a selected oxidant. A full study of the thermodynamics and mechanism of such engines for either ground or "airborne" use will need much careful consideration at some time, but is outside the limits of this paper. It would appear that the first use to which atmosphere breathing engines may be put be as part of a "probe" programme. This may involve atmospheric entry, circumnavigation and subsequent exit, most probably without landing. Of the possible types of engine available, the ramjet appears to be an obvious choice for such a scheme. It was, therefore, decided to investigate the possibility of using such an engine in the hypothetical fuel atmospheres with one or two oxidants about which adequate information was available. Those studied so far include liquid oxygen, liquid ozone, nitric acid, and dinitrogen tetra oxide.

Engine operation within a carbon dioxide atmosphere requires even more study than the operation of engines within chemically reactive atmospheres. It is possible, once again, to envisage design changes of all air breathing engines to make them work under those conditions. This would involve the use of both tanked oxidant and fuel burning in a bi-chemical process with the carbon dioxide atmosphere used only as a heat carrier or diluent to lower the resulting temperature. Whilst this appears to make such engines an engineering possibility, the question of thermodynamic efficiency is complicated and cannot be considered here.

The ramjet was selected for the only investigation into this form of power unit for the same reasons as those in the chemically reactive atmospheres above. It is a point of incidental interest that such a system is very similar to an air breathing ducted rocket operating without secondary atmospheric combustion.

To simplify this first investigation into engines operating in other atmospheres, and to make the results directly comparable, it was decided to standardise the engine operating conditions in addition to standardising atmospheric temperatures and pressure. Since ramjet engines are susceptible to Mach Number rather than velocity, Mach 3 was taken as a reasonable operating condition. An interesting point arises out of this. The speed giving Mach 3 in the various hypothetical atmospheres varies considerably, being about 8000 ft./sec. in the case of a hydrogen atmosphere, 3200, 3000 and 2000 ft./sec. respectively in methane, ammonia and carbon dioxide. Nevertheless, such widely divergent speeds produce the same increases in pressure and temperature within the various engine intakes.

Internally, due allowance was made for losses. This lowered internal pressure from a possible 550 lb./in^2 down to less than 375 lb./in^2. Finally, as a control on the amount of combustion permitted, 2200°K was taken as the combustion temperature in all cases. This is quite a real ramjet limitation. The decision was influenced to a slight extent by the fact that it simplified the sums involved !

2.4. Performance in other Atmospheres

Calculations were first performed to determine the amount of various tanked oxidant that would be required to produce a temperature of 2200°K at constant pressure. As a comparison, one pound of air requires about .06 lb. of fuel. Results indicate that one pound of hydrogen will require something like two and a quarter pounds of liquid oxygen !! This awe-inspiring figure is beaten by a considerable margin if either nitric acid ($4\frac{1}{2}$ lb/sec.) or dinitrogen tetra oxide ($3\frac{1}{2}$ lb/sec.) are used. Ozone requires less than 2 lb. per pound of hydrogen, but in actual fact is a most improbable propellant because of other reasons. Requirement of liquid-oxygen per pound of the other two fuel atmospheres shows improvement over the hydrogen case, but is still ten times greater than the Earth comparison figure ! In the carbon dioxide case, using liquid oxygen and kerosine as a heat source, a figure of a little less than 0.4 lb. of the two propellants per pound of atmosphere is obtained.

Nevertheless, these figures give no indication of thrust producing efficiency which must be compared on a specific thrust basis. Present day rockets operating in an atmosphere at 14.7 lb/in^2 produce between 220 and 250 pounds of thrust for a total propellant flow of one pound a second. The specific thrusts of ramjets operating in fuel atmospheres are all better than this. Using liquid oxygen, a specific thrust of 360 Lb per lb/sec. is obtained in a hydrogen atmosphere whilst methane and ammonia atmospheres produce figures of 430 and 480 Lb.sec/lb. respectively. These results indicate that a strong case can be made for the use of a ramjet in the fuel atmospheres considered, but it is worth underlining the point that these comparisons have been carried out on a specific thrust basis for just one condition in each atmosphere and much more work remains to be done before realistic answers can be obtained.

In the carbon dioxide case the results are much less promising. Using the bipropellants liquid oxygen and kerosine, a specific thrust figure of

260 Lb.sec/lb is obtained which is very close to possible rocket engine performance with the same propellants. Nevertheless, further work must be carried out before the ducted rocket form of ramjet for this application is dismissed as of no interest.

One or two points that arise from the figures so far obtained are worthy of some comment. In the first place, in all the atmospheres considered, the mass of propellant added to the atmosphere is of quite considerable proportion. This added mass contributes largely to the final thrust obtained. Precise engine geometrics have not been worked out, but it would appear that intake areas will be very small compared with those at exit, this being particularly exaggerated in the hydrogen case. Finally it must be pointed out that a complete comparison between ramjet and rocket can only be carried out by taking engine specific masses into account. No work in this direction has yet been undertaken. Method of calculation and details of assumptions made are contained in Appendix II.

2.5. Conclusions

Whilst it has been shown that there is good reason to continue with this investigation, much care should be taken in using existing results as the apparent improvements in performance of atmosphere breathing engines over the rocket are at present based on a very limited investigation. Nevertheless, at the conditions laid down, in hypothetical fuel atmospheres, very considerable improvements on a tanked liquid consumption basis have been demonstrated. The investigation of the carbon dioxide atmosphere is at present inconclusive.

Acknowledgement

The author of Part II wishes to place on record the contribution made by S.W.Greenwood, who started a formal study on the use of a ramjet in a hydrogen atmosphere before the author. In particular, his suggestions were accepted with regard to the atmospheric conditions of pressure and temperature and also the external ramjet operating conditions and intake losses used throughout the study.

REFERENCES

1.	Moult, E.S.	Power Plants for Supersonic Flight, The de Havilland Gazette, August, 1955.
2.	Gardner, G.W.H.	Guided Missiles. The Chartered Mechanical Engineer, Jan.1955.
3.	Cleaver, A.V.	Supersonic Flight - Propulsion. Journal of the Royal Aeronautical Society, November, 1956.
4.	Tsien, H.S.	A Method for Comparing the Performance of Power Plants for Vertical Flight. Journal of the American Rocket Society, July-August, 1952.
5.	Proell, W.	The Design of a Practical Space Ship. The Journal of Space Flight, September, 1949 (Part 1), October, 1949 (Part 2), November, 1949 (Part 3).
6.	Sandorff, P.E.	Some New Thoughts on Space Flight. Paper presented at the American Rocket Society's Semi-Annual Meeting, Boston, June 1955.
7.	Article	Jet Propulsion News. Journal of the American Rocket Society, May 1957.
8.	Reichert, H.	Enthalpy Charts. A.R.C.Tech.Report, R. & M. No.3015. H.M.Stationery Office, 1957.
9.	Penner, S.S.	Chemistry Problems in Jet Propulsion. Pergamon Press, 1957.
10.	Rossini, F.D. and others	Selected Values of Chemical Thermodynamic Properties. National Bureau of Standards Circular 500, 1952.

APPENDIX I

Summary of assumptions made, and outline of methods of calculation used.

Note: Assumptions are based on "typical" data. Simplified methods of analysis are employed, e.g. vehicle air drag is neglected.

1. Break-even times (Fig. 3)

Flight Mach No. = 3

		Ramjet	Rocket
Thrust / Weight	S.L.	28	40
	60,000 ft.	4.5	40
Specific propellant consumption lb/hr.Lb.	S.L.	2.6	16
	60,000 ft.	2.2	13.5

e.g. for same thrust, and for break-even time at S.L., engine weight + propellant weight is the same in each case.

$$\frac{\text{Thrust}}{28} + \frac{2.6}{3600} \times \text{Thrust} \times \text{time (secs)}$$

$$= \frac{\text{Thrust}}{40} + \frac{16}{3600} \times \text{Thrust} \times \text{time (secs)}$$

$$\text{Time} = 2.9 \text{ secs.}$$

2. Conditions during trajectory (Figs. 4, 5, and 6)

N.A.C.A. Standard Atmosphere Conditions.

As an example, some details are given below for cases (c) and (d).

The specific heat ratio for the air in the intake system has been taken as $\gamma = 1.4$ for these preliminary investigations.

The intake total pressure recovery is assumed to vary linearly from 1.0 at Mach 1 to 0.6 at Mach 5.

Case (c) :

15 g boost to 2000 ft./sec. followed by 3g constant acceleration vertical trajectory.

$$\text{Separation altitude} = \frac{2000 \times 2000}{2 \times 15 \times 32.2} = 4{,}140 \text{ ft.}$$

VELOCITY ft./sec.	ALTITUDE ft.	MACH NO.	RAM TEMP. °C.	RAM PRESSURE S.L.Atmospheres
2000	4,140	1.82	190	4.54
3000	30,040	3.05	380	9.20
4000	66,340	4.12	675	6.57
5000	113,140	4.95	1100	1.74
6000	170,140	4.87	1725	0.204

Case (d) :

Approx. 15g vertical boost to Mach 2 at 4,140 ft. followed by constant thrust trajectory.

The following variation of thrust coefficient with Mach No. is assumed :-

MACH NO. :	2	3	4	5
THRUST COEFFICIENT :	1	1.5	1.2	1

The altitude necessary for constant thrust is then given by

MACH NO. :	2	3	4	5
ALTITUDE ft. :	4,140	33,300	40,700	46,000

3. Comparison of vehicle weights (Fig. 7)

As an example, consider case (c) with Mach No. range 2 - 3.5. (Fig. 4 indicates thrust roughly constant over this region - permitting constant acceleration assumption.)

	Ramjet	Rocket
$\frac{\text{Thrust}}{\text{Engine weight}}$	16	40
$\frac{\text{Structure weight}}{\text{Propellant weight}}$	0.1	0.1
Specific propellant consumption lb./hr.Lb.	2.5	14
Mass ratio	1.044	1.274

For payload of 10,000 Lb.

	Ramjet	Rocket
Thrust Lb.	56,200	53,000
Engine weight Lb.	3,510	1,320
Propellant weight Lb.	605	3,190
Structure weight Lb.	61	319
Total weight Lb. (excluding payload)	4,176	4,829

Note: Structure weight has been taken as fraction of propellant weight.

Structure weight influenced by engine weight is assumed to be incorporated in engine Thrust/weight assumption.

4. <u>Comparison of satellite vehicle weights</u> (Fig. 8)

3 stage rocket :

Vertical trajectory assumed to simplify mass ratio calculations.
Specific propellant consumption 14 lb./hr.Lb. and thrust/engine weight = 40 throughout. 3g mean acceleration. Each stage operates over a velocity interval of $1\frac{2}{3}$ miles/sec.

Stage	3	2	1
Payload Lb.	500	4,428	39,243
Engine weight Lb.	237	2,090	18,500
Propellant weight Lb.	3,355	29,750	263,500
Structure weight Lb.	336	2,975	26,350
Total weight Lb. (excluding payload)	4,428	39,243	347,593

2 stage rocket with ramjet and booster rocket stages :

Vertical boost by solid propellant booster rocket at 15g acceleration to separation at 2000 ft./sec.

Constant thrust ramjet stage with 3g acceleration to separation at 1 mile/sec. (Trajectory no longer vertical, but assumed vertical for mass ratio calculation).

2 stage rocket with a velocity increment of 2 miles/sec. for each stage. (trajectory assumed vertical for mass ratio calculations).

2 stage rocket - assumptions as for 3 stage rocket.

Ramjet stage. Specific propellant consumption = 2.5 lb./hr.Lb. and thrust/engine weight = 16.

Booster rocket. Specific propellant consumption 17.2 lb./hr.Lb.

Total structure weight/propellant weight = $\frac{1}{10}$

	2 Stage Rocket		Ramjet	Booster Rocket
	Stage 2	Stage 1		
Payload Lb.	500	9,423	177,448	268,938
Engine weight Lb.	453	8,525	64,650	-
Propellant Weight Lb.	7,700	145,000	24,400	143,500
Structure Weight Lb.	770	14,500	2,440	14,350
Total Weight Lb. (including payload)	9,423	177,448	268,938	426,788

APPENDIX II

Methods and assumptions

Symbols, units and values

a = acoustic velocity $\left(\frac{\gamma r^{*}T}{\bar{m}}\right)^{\frac{1}{2}}$ ft./sec.

f = thrust pdls.

F = thrust Lbf.

$\dot{m}$ = mass flow rate lb./sec.

$\bar{m}$ = molecular mass

M = Mach number $\frac{V_o}{a}$

T = temperature °K.

v = velocity ft./sec.

Thermochemistry and gas dynamics

H^{*} = enthalpy chu/lb.mol.

H = enthalpy chu/lb.

h^{*} = enthalpy ft.pdl./lb.mol.

h = enthalpy ft.pdl./lb.

similarly

I^{*} = reaction enthalpy chu/lb.mol. etc.

H_f^{*} = enthalpy of formation chu/lb.mol. etc.

R^{*} = universal gas constant chu/lb.mol. °K. etc.

= 1.98

C_p^{*} = heat capacity (pressure constant) chu/lb.mol. °K. etc.

γ = $C_p/(C_p - R)$

$$\eta = 1 - \left(\frac{P_e}{P_5}\right)^{\frac{\gamma-1}{\gamma}}$$

Subscripts

0	=	free stream	3	=	intake exit	5	=	nozzle entry
e	=	nozzle exit	a	=	atmosphere	p	=	tanked propellant

To convert chu to ft.pdl. multiply by 4.5×10^4

To convert pdl. to Lbf. divide by 32.147.

1. Intake and Pressure Losses

M	P_{t}/P_{o}	P_{t3}/P_{t}	P_{t3}/P_{o}	P_{5}/P_{o}
3.0	37	0.7	25.9	25

These values have been used for all atmospheres and are based on $\gamma = 1.45$.

2. Engine Specific Performance

Assuming complete expansion in the nozzle to $P_e = P_o$

$$f = (\mathring{m}_a + \mathring{m}_p)v_e - \mathring{m}_a v_o \quad \text{pdl.} \tag{1}$$

For isentropic flow in the nozzle

$$c_{p5} T_5 + \tfrac{1}{2} v_5^2 = c_{pe} T_e + \tfrac{1}{2} v_e^2 \quad \text{ft.pdl./lb.}$$

Rewriting in terms of exit velocity

$$v_e = \left[2(c_{p5} T_5 - c_{pe} T_e) + v_5^2 \right]^{\frac{1}{2}} \quad \text{ft./sec.}$$

From this v_5^2 can reasonably be ignored as small. Taking a value c_p as approximately equal to c_{p5} and c_{pe}, i.e. assuming frozen chemical equilibrium

$$v_e = \left\{ 2 c_p T_5 \left[1 - \left(\frac{P_e}{P_5}\right)^{\frac{\gamma-1}{\gamma}} \right] \right\}^{\frac{1}{2}} \quad \text{ft./sec.}$$

From equation (1) the specific thrust in pdl. per lb/sec. of tanked propellant is then obtained N.B. $c_p T_5 = h_5$ and

$$\left[1 - \left(\frac{P_e}{P_5}\right)^{\frac{\gamma-1}{\gamma}} \right] = \eta$$

$$\frac{f}{\mathring{m}_p} = \left(\frac{\mathring{m}_a}{\mathring{m}_p} + 1\right) (2 h_5 \eta)^{\frac{1}{2}} - \frac{\mathring{m}_a}{\mathring{m}_p} V_o \quad \text{pdl.sec./lb.} \tag{2}$$

and $$\frac{F}{\mathring{m}_p} = \frac{f}{32.15\, \mathring{m}_p} \quad \text{Lb.sec./lb.}$$

3. Thermochemistry

x is defined as the number of mols. of oxidant required to raise the total products of combustion to 2200°K.

Chemical composition balance.

1 lb. mol. atmosphere + x lb. mol. oxidant → Products of combustion and excess atmosphere as functions of x.

If the water gas reaction, and dissociation are ignored, the functions of x for each simple component can be determined. This process incurs an error of unknown magnitude but is not considered too unrealistic at the comparatively low temperature and high pressure occurring in the chamber.

Enthalpy balance

$$I^{*}_{\text{atmosphere}} + x\, I^{*\,298}_{\text{oxidant}} = \Sigma\, I^{*\,2200}_{\text{products}}$$

$$\text{where} \quad I^{*\,298} = H^{*}_{f}$$

$$\text{and} \quad I^{*\,2200} = H^{*}_{f} + H^{*\,2200}$$

Since all I^{*} values are available from the references, x can now be determined and therefore $\frac{\dot{m}_a}{\dot{m}_p}$.

$$\text{Then } H^{*}_{5} = \Sigma\, I^{x\,2200}_{\text{products}} - \Sigma\, H^{*}_{f} \quad \text{chu/lb.mol. atmosphere.}$$

$$H_5 = \frac{H^{*}_{5}\, x}{\bar{m}_{\text{atmos}}} \quad \text{chu/lb. exhaust.}$$

$$h_5 = 4.5 \times 10^{4}\, H_5 \quad \text{ft.pdl./lb. exhaust.}$$

The only value on the r.h.s. of equation 2 now requiring consideration is η. Since the expansion ratio is known, only a value of γ must be assigned. This can be obtained from C_p tables since proportions of various components are known, and their temperature.

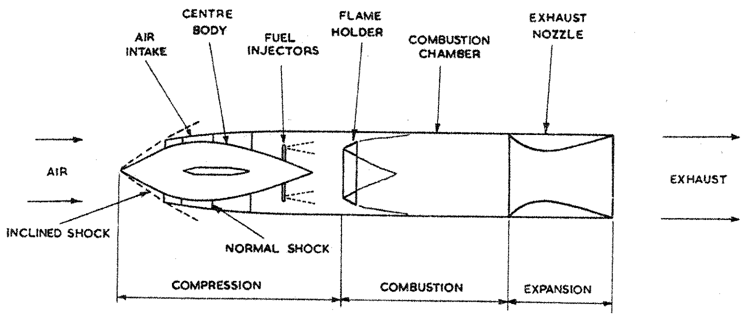


FIG. 1 DIAGRAMATIC LAYOUT OF TYPICAL SUPERSONIC RAMJET.

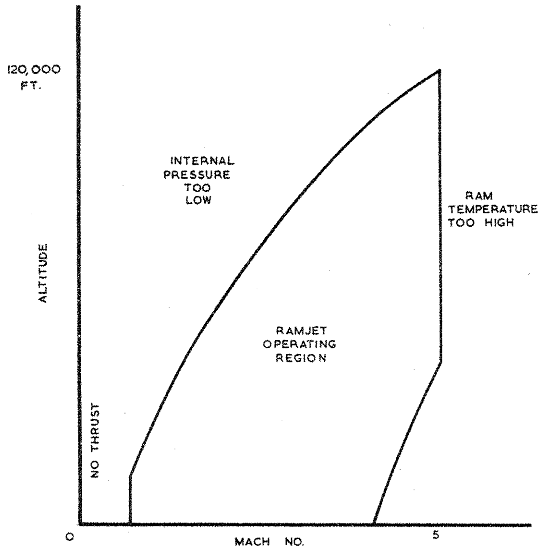


FIG. 2 OPERATING REGION OF RAMJET.

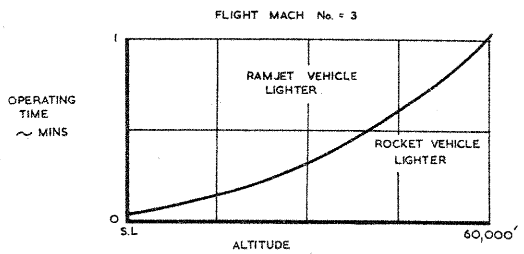


FIG. 3 BREAK-EVEN TIMES AT WHICH ENGINE + PROPELLANT WEIGHT IS THE SAME FOR RAMJET AND ROCKET VEHICLES.

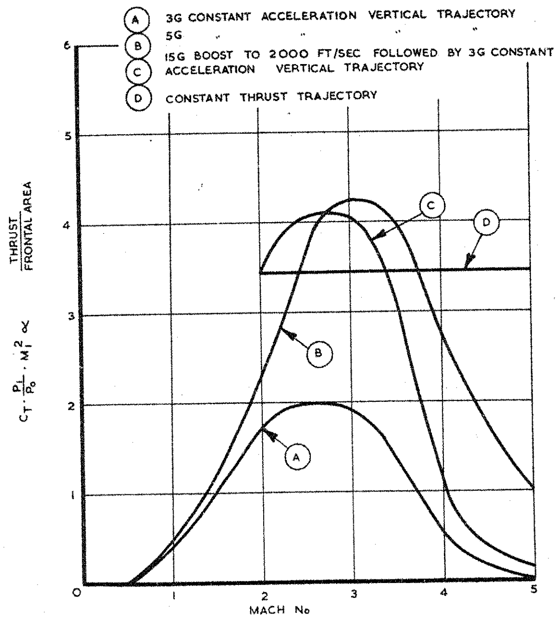


FIG. 4 VARIATION OF THRUST WITH MACH No FOR THE DIFFERENT TRAJECTORIES CONSIDERED.

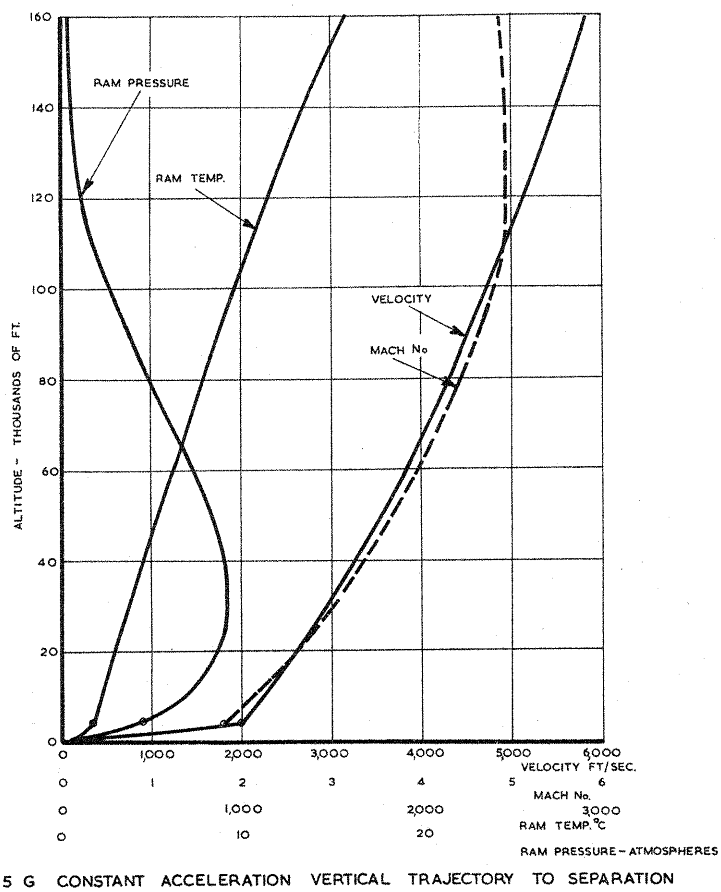


FIG.5 15 G CONSTANT ACCELERATION VERTICAL TRAJECTORY TO SEPARATION AT 2,000 FT/SEC FOLLOWED BY 3 G CONSTANT ACCELERATION VERTICAL TRAJECTORY.

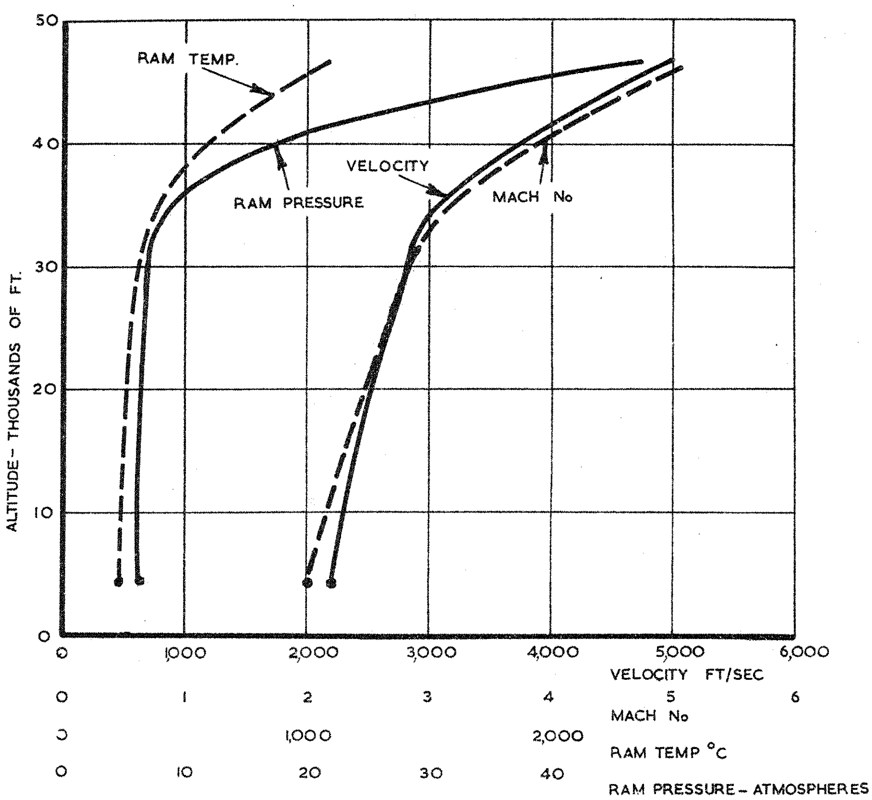


FIG. 6 CONDITIONS EXPERIENCED DURING A CONSTANT THRUST TRAJECTORY.

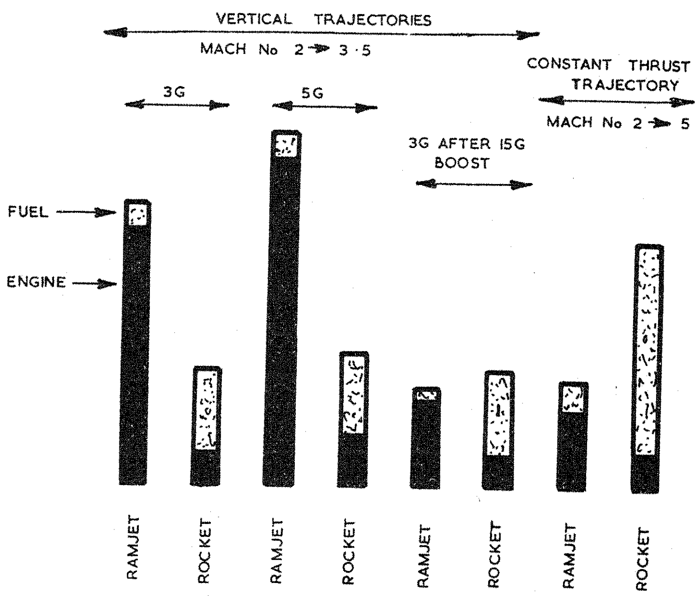
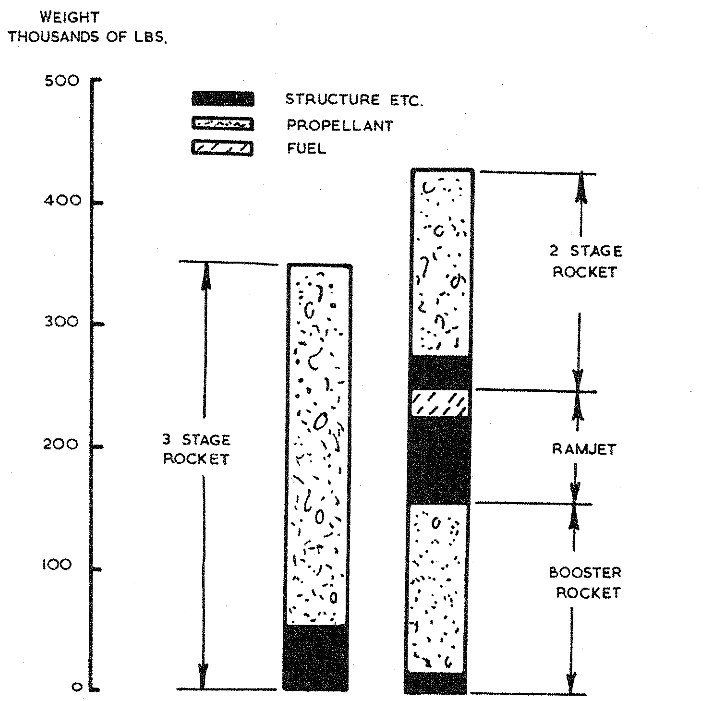


FIG. 7 RELATIVE WEIGHTS FOR DIFFERENT TRAJECTORIES FOR CONSTANT PAYLOAD.



COMPARISON OF VEHICLES TO PLACE 500 LB PAYLOAD IN CIRCULAR ORBIT AROUND EARTH.

FIG. 8

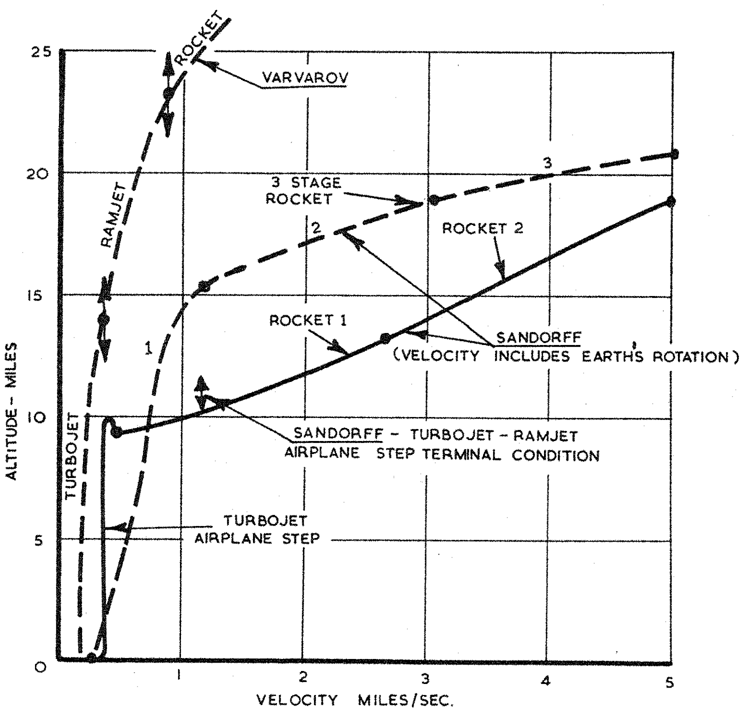
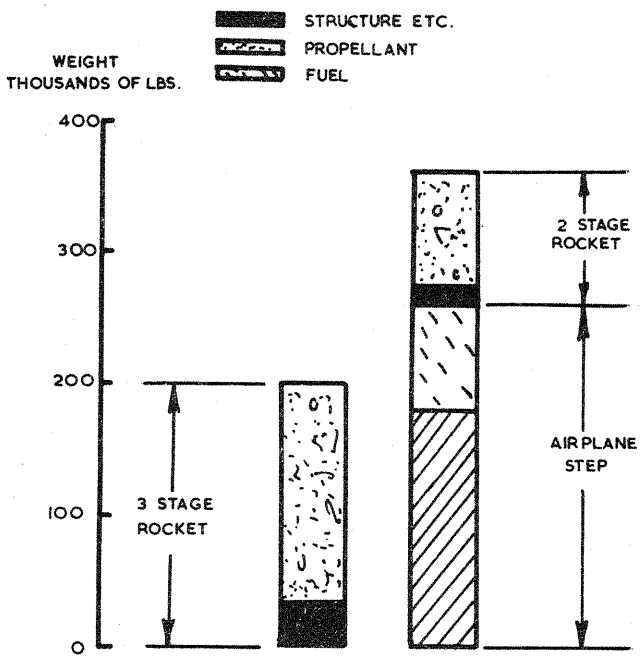


FIG. 9 PROPOSALS INCORPORATING AIR-BREATHING ENGINES



COMPARISON OF VEHICLES TO PLACE 500LB. PAYLOAD IN CIRCULAR ORBIT AROUND EARTH (SANDORFF)

FIG. 10

Body	Distance from Sun (10^6 kms.)	Radius —(kms.)	Surface Area (Earth=1)	Gravity (Earth=1)	Day	Escape velocity (km./Sec.)	Atmosphere
Mercury	58	2,400	0.14	0.26	88d	3.5	None
Venus	108	6,100	0.91	0.90	?	10	CO_2+?
Earth	150	6,370	1	1	24h	11.2	N_2+O_2
Moon		1,740	0.07	0.16	27d	2.3	None
Mars	228	3.400	0.28	0.38	25h	5.0	CO_2+?
Phobos		10					None
Deimos		5					None
Jupiter	779	70,000	120	2.65	10h	60	CH_4+NH_3
Io		1,700	0.07	0.1	?	2.3	None
Europa		1,500	0.06	0.1	?	2.0	None
Ganymede		2,600	0.17	0.2	?	2.9	None
Callisto		2,500	0.15	0.2	?	2.2	None
+7 others							
Saturn	1,430	60,000	84	1.14	10h	35	CH_4
Titan		2,800	0.2	0.2	?	3.0	CH_4
Rhea		900				0.7	None
+7 others							None
Uranus	2,870	25,000	15	1	11h	22	CH_4
+5 moons							
Neptune	4,500	26,000	17	1	16h	23	CH_4
Trition		2,500	0.15	0.2	?	3.0	None?
+1 other							
Pluto	5,900	3,000	0.2	?	?	?	?

FIG. 11. BODIES OF THE SOLAR SYSTEM