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Silver Spring, Maryland

APL/JHU TG-20
January 14, 1947

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NUCLEAR-POWERED FLIGHT

By

An Informal Committee

of

THE APPLIED PHYSICS LABORATORY

of

THE JOHNS HOPKINS UNIVERSITY

~~Declassified 6/24/63 by Declass Br.~~

A. E. Ruark, Chairman

A. C. Beer
E. A. Bonney
George Carlton
J. Emory Cook
George Gamow
R. B. Kershner
A. W. Lemmon
F. T. McClure
C. F. Meyer
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NUCLEAR POWERED FLIGHT

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14 January 1947

TO: L. R. Hafstad

FROM: A. E. Ruark

SUBJECT: Transmittal of Progress Report entitled "Nuclear-Powered Flight", by an Informal Committee of the Applied Physics Laboratory of the Johns Hopkins University.

In accordance with your verbal instructions of about 9 June 1946, the Committee has considered the general problem of air vehicles driven by nuclear power. Three copies of the subject report are respectfully submitted herewith. A first draft was submitted October 25, 1946. Since that time many errors have been corrected and much new material has been added. The initial distribution is indicated in the report.

Your comments and those of other interested persons will be appreciated by the Committee. Review by suitable members of APL is hereby requested.

It is believed that any further work on this subject at APL should be carried on by a small staff with fresh instructions, and that the existing large committee should be discharged in the near future.

FOR THE COMMITTEE

Arthur E. Ruark, Chairman;
Technical Supervisor
for Research Laboratory.

AER:rh

Encl. 3 -- Copies 1, 2, and 3 of subject report.

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TABLE OF CONTENTS

Page

Foreword

Abstract

I. Introduction and Summary.--By Arthur E. Ruark

Polar Map of Northern Hemisphere

II. Nuclear Reactors for Rockets and Ram-Jets.

Part A. Cylindrical Enriched Reactors.--
By Nicholas M. Smith, Jr.

Part B. Design of Cylindrical Beryllia Reactors.--
By Arthur E. Ruark

Part C. Considerations Concerning Fast-Neutron Reactors ("Amplification Reactors").--By George Gamow

Part D. Special Methods of Heat Production and Transfer.--
by R. B. Roberts

[redacted]

TABLE OF CONTENTS (Cont'd)

III. Preliminary Design of Long-Range Pilotless Airplanes.--By R. B. Roberts, E. A. Bonney, and A. C. Beer.

Part A. Aerodynamic Design.--By E. A. Bonney

IV. Preliminary Report on Nuclear Energy for Rocket Propulsion.--By F. T. McClure and R. B. Kershner.

V. Supersonic Nuclear-Powered Ram-Jets and Turbojets.

TABLE OF CONTENTS (Cont'd)

~~CONFIDENTIAL-RD~~ ~~DECLASSIFIED~~

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TABLE OF CONTENTS (Cont'd)

Appendices: Page

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CHAPTER III. PRELIMINARY DESIGN OF LONG-RANGE PILOTLESS AIRPLANES

by

R. B. Roberts, E. A. Bonney and A. C. Beer

Foreward

By R. B. Roberts

In designing jet propulsion engines using nuclear energy, three sets of design condition must be met simultaneously. These are 1) conditions under which a nuclear energy reactor will operate, 2) conditions imposed by the design of the jet engine and finally, 3) conditions arising from the considerations of properties of materials. It is easy to meet any two sets of requirements but difficult to meet all three simultaneously. For example, a nuclear energy rocket or a ram-jet would be easy if materials were available with melting points of 10,000°C. In the case of the rocket it appears that these three sets of conditions are probably not mutually exclusive but they may be for the ram-jet. In the turbo-jet, however, the design limitations of the engine appear to be relatively easy to meet.

In the report "Turbo-Jet Engines as Guided Missile Power Plants", by Lt. Edward M. Redding, the exhaust temperature of a turbo-jet engine is given as 1500°F. In addition, the performance of the turbo-jet should not be as critical to drag in the combustion chamber as is the ram-jet. These two features give much more leeway in the design of the pile and in obtaining materials which will stand the temperature.

A final consideration in favor of the turbo-jet is that the thrust does not vary appreciably with velocity. This means that the drag of the turbo-jet engine will be smaller than the thrust below a certain velocity. In other words, the poorly designed turbo-jet with lower thrust will still fly, though at a reduced velocity. In contrast to this situation, the drag of a ram-jet increases with velocity in much the same way as does the thrust. Hence, it is conceivable that a ram-jet whose propulsive systems balk short of optimum performance due to limitations imposed by pile design, might not fly at any velocity.

PART A. AERODYNAMIC DESIGN

By E. A. Bonney

Abstract

It is desired to find the optimum design of a long-range pilotless airplane to carry a payload of 12,000 pounds a distance of at least 5,000 miles. Existing and proposed future methods of propulsion are considered with an eye to practicability as well as feasibility. It is shown that a turbo-jet design using nuclear energy from a reactor as the source of heat in place of the conventional burner will be the optimum design for the type of vehicle considered. On the basis of the assumptions used in this analysis it will be the lightest in weight and have the best performance of the three types considered.

1. Introduction

Inasmuch as the ram-jet and rocket propulsion methods have been covered in other analyses, this preliminary investigation is confined to turbo-jets and conventional propellers as the means of imparting propulsive force to the airplane. The analysis is therefore in the range of high subsonic Mach numbers.

Three plausible designs were considered as follows:

1. Turbo-jet with nuclear energy.
2. Turbo-jet with conventional fuel.
3. Air turbine driving conventional propellers with nuclear energy as the source of heat.

It is assumed that the body shape will be that of a greatly elongated tear drop with the front portion cut off to allow the air to enter and the rear cut off to allow it to exhaust through a jet. The maximum area required is taken to be that of a 6 foot circle plus the additional area necessary to allow passage of the air through the body. It develops that the necessary outside diameter is approximately 7 feet and hence the area of a 7 foot circle is the basic area on which the body drag coefficient is based. Reference 2 states that the drag coefficient for such a shape (neglecting compressibility effects) is about .120.

The optimum angle of attack for a symmetrical wing is taken as 6 degrees. (Airplane usually has incidence angle of about 3 degrees, but the angle for 0 lift is -3 degrees.) Assuming an aspect ratio of 6, the lift curve slope becomes .0758 per degree and the lift coefficient becomes .455. The profile drag coefficient is .0060. Adding 30% of this value to cover tail drag makes the total wing drag coefficient equal to .0078.

The operating altitude is arbitrarily taken as 40,000 feet. It is realized that greater speeds will be obtainable at higher altitudes with the turbo-jet design, however this altitude is a practical limitation on the propeller type and it develops that the turbo-jets will simply "dig deeper" into the region of strong compressibility effects at this altitude thereby gaining little in speed and probably considerable in control troubles.

2. Turbo-jet with Nuclear Power

With this type of design, the air will enter a diffuser, thence passing through the compressor, then through the pile heat source which replaces the conventional burner, then to the turbine and finally exhausting through the jet at the rear. The weight of the carbon pile was determined by comparison with the conventional type of turbo-jet considered in section 3, i.e. the equivalent weight necessary to produce 4000 lbs. of thrust with an expenditure of 1.08 lbs. fuel per hour per lb. of thrust. These calculations are shown in section 6 and it develops that the weight is approximately 13,500 lbs. for the given assumptions.

The structural weight is estimated by comparison with conventional airplanes of existing design. The best criterion for comparison is the ratio of total (take-off) weight to empty (without engines) weight which is of the order of 2-1/2 to 1. On this basis, the weight breakdown will be as follows:

1.	Pile source	13,500 lbs.
2.	Payload	12,000
3.	Power plant (G.E. - TG180, see ref. 1)	2,300
4.	Structure	18,200
	Total	46,000 lbs.

Determination of flight conditions is made by solving simultaneously the aerodynamic equations that lift equals weight and thrust equals drag as follows:

$$L = W = C_L \rho \frac{V_T^2}{2} A_W$$

$$T = D = C_D \rho \frac{V_T^2}{2} \cdot A_W$$

where V_T = true air speed

A_W = wing area

$C_D = C_{D_{o_B}} + C_{D_{o_W}} + C_{D_{i_W}}$

$C_{D_{o_B}} = .120 \frac{A_B}{A_W}$ (for low subsonic values, $M < .5$)

$$C_{D_{i_W}} = \frac{C_L^2}{\pi R} = \frac{.455^2}{\pi x 6} = .0110$$

$\rho = .000583$ at 40,000 ft.

Now these equations make no allowance for the large increase in drag due to compressibility at the high (subsonic) Mach numbers. If the equations are solved using the previously mentioned subsonic drag coefficients, the speed is found to be about 860 MPH. This speed is fallacious however and it becomes necessary to make an intelligent estimate of the strength of the compressibility effects in order to predict the speed more accurately. If it is assumed that the additional increment of drag due to compressibility is due principally to the body, the body drag coefficient at a flight Mach number of .9 (600 MPH) is found to be about twice that of the low subsonic value quoted above (.244 as compared to .120). Due to the high slope of the C_D vs. M curve in this region, it is probable that the flight speed will be in the neighborhood of 600 MPH. The corresponding wing area (neglecting effects of high speed on lift characteristics) will be 440 sq. ft.

3. Turbo-jet with Conventional Fuel

This is the well known system in use in the latest pursuit type of airplane today. The General Electric Model TG-180 noted in reference 1 was considered for this analysis. The engine has a rated thrust of 4000 lbs. with a fuel rate of 1.08 lbs/hr per pound of thrust. Its diameter is 38 inches.

The weight breakdown of this design is a little more difficult than for the type discussed in section 2 inasmuch as the fuel weight is a function of the range and air speed, the latter factor being in turn a function of the weight. For this calculation, a curve of body drag coefficient versus Mach number was constructed based on the low subsonic value and the value obtained for the nuclear turbo-jet. This curve while probably being incorrect in absolute value can, however, be used for purposes of comparison. By trial and error calculations, the approximate speed, take-off gross weight, available weight for fuel, and wing area can now be determined. It develops that, for ranges greater than 4000 miles, more fuel is required than can be carried. The airplane weight becomes so great that the available thrust of 4000 lbs. will not fly the airplane in level flight at the chosen altitude with the required amount of fuel for the longer ranges. A more detailed analysis would indicate the advantages gained by flying at a lower altitude or by use of greater thrust. The maximum range that can be flown at this altitude with 4000 lbs. thrust is approximately 4000 miles. For this condition, the initial speed will be around 400 MPH and the final speed 530 MPH. The wing area required will be about 1870 sq. ft.

The weight breakdown for the 4000 mile range is as follows:

1.	Fuel weight	36,700 lbs.
2.	Payload	12,000
3.	Power plant	2,300
4.	Structure	34,000
	Gross weight at take-off	85,000 lbs.
	Gross weight at end of flight	48,300

Because of the large weights, slow speeds, and difficulty in obtaining the desired ranges, it is obvious that this type of design is greatly inferior to the nuclear turbo-jet type.

4. Nuclear Air Turbine with Propeller

The use of propellers as the means of propulsion should of course be included in the investigation of subsonic pilotless airplanes. While this type does not require as much fuel for corresponding ranges and speeds as the conventional turbo-jet, the amount that is required combined with the high power plant and accessories weight still presents a major objection in the excessive weight required for long ranges and reasonably high speeds. The use of nuclear energy as the source of heat in a gas turbine driving a propeller is then probably the best solution for use with propellers.

Inasmuch as there exists a minimum critical dimension for the piles (see Table 2 of Chapter 3) which is entirely independent of the heat required, it develops that the pile can produce considerably more power than any present-day turbine or propeller could absorb. For this reason, the propeller itself becomes the limiting factor.

In the present and anticipated near-future stage of development of the propeller, it is probable that the limiting speed for reasonably efficient operation at high altitudes will be of the order of 550 MPH (M = .83). The large value of power required would necessitate the use of two contra-rotating propellers which would probably have to be mounted at the front and rear in tandem in the interests of keeping drag at a minimum. The mechanical drive difficulties of such a design, particularly in view of the size and probable shape of the payload are obvious.

The weight breakdown of such a design is as follows:

1.	Pile source	13,500
2.	Payload	12,000
3.	Power plant	6,000
4.	Propellers, gear boxes, etc.	4,000
5.	Structural weight	23,500
	Total gross weight	59,000 lbs.
	Wing area	685 sq. ft.

The disadvantages of this type of design as compared with the nuclear turbo-jet are then:

1. Speed limitation.
2. Greater weight.
3. More complicated mechanical design.

5. Summary

Following is a summary of the weight breakdowns in lb., wing area and performance for the three types of designs noted in this report.

Design Type	1 Turbo-jet Nuclear Energy	2 Turbo-jet Conventional Fuel	3 Nuclear Air Turbine with Propellers
Piles or fuel weight, lbs.	13,500	36,700	13,500
Payload	12,000	12,000	12,000
Power plant	2,300	2,300	6,000
Propellers, gear boxes, etc.	- -	- -	4,000
Structure	18,200	34,000	23,500
Gross weight at take-off,	46,000	85,000	59,000
Gross weight at landing	46,000	48,300	59,000
Thrust at flight speed,	4,000	4,000	3,800
Approximate average air speed MPH	600	465	550
Wing area, sq. ft.	440	1,870	685
Maximum range	*	4,000	*

On the basis of these figures, it is concluded that the nuclear turbo-jet is the best design. This statement is based on the assumptions that a U235 or Pu reactor of the kind discussed in Part A of Chapter II, and in Part B of this chapter, can be made to operate with wall temperatures of about 4000°R (2220°K). Furthermore, cost is not considered.

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* Probably limited by material failure.

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PART B. HEAT TRANSFER CONSIDERATIONS AND REACTOR DESIGN FOR TURBO-JET

By A. C. Beer

Nomenclature.

C_p - specific heat at constant pressure of gas-stream, BTU/lb. deg. F.

d - diameter of individual tube through reactor, ft.

$\dot{m}$ - mass-flow, lb/sec.

$\dot{Q}$ - heat energy absorbed by gas-stream per second, BTU/sec.

L - length of reactor, ft.

n - number of tubes in reactor

r - radius of reactor, ft.

Γ - ratio of cross-sectional area available to gas stream to total cross-sectional area in the reactor.

γ - ratio of specific heat at constant pressure to that at constant volume.

The following symbols are usually written with subscripts to indicate location. The subscripts 1, 2, 3, ... refer to locations indicated in Fig. 3-1 and w refers to reactor wall.

A - gas stream cross-sectional area, sq. ft.

M - Mach number

p - pressure (absolute), lb/ft^2

T - static temperature, deg. Rankine

$T^{(s)}$ - stagnation temperature, deg. Rankine

V - gas stream velocity, ft/sec.

ρ - density, lb/ft^3

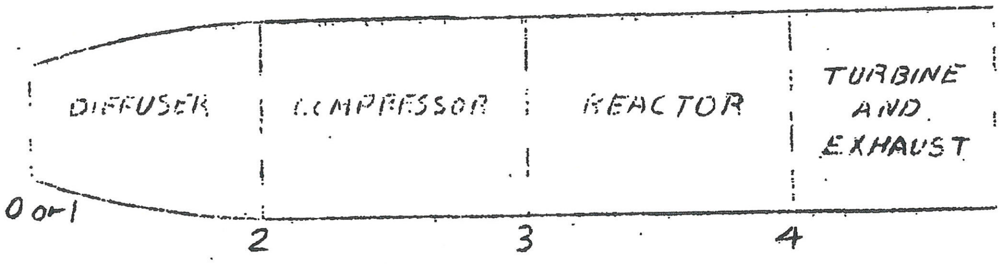

FIG 3-1

As a result of the analysis given in section 2 of Part A it is determined that the power requirements of the nuclear turbo-jet supplying the 4000 lbs. thrust necessary for flight at 600 MPH are the equivalent of approximately 4500 lbs. gasoline per hour. On the basis of 19,000 BTU per lb. as the heat of combustion of gasoline, this amounts to an energy consumption of 2.37 x 10^4 BTU per second. Assuming the gasoline turbo-jet to operate with 100% combustion efficiency (this assumption is obviously conservative to the nuclear design), the above figure represents the power which must be absorbed by the gas stream, i.e.

$$\dot{Q} = 2.37 \times 10^4 \text{ BTU/sec} \qquad (1).$$

With the flight conditions specified, namely 600 MPH at 40,000 ft., it follows that

$$\rho_1 V_1 = 17 \text{ lb/ft}^2\text{/sec} \tag{2}$$

and since we have chosen a configuration with

$$A_1/A_2 = 1/3 \tag{3}$$

we have

$$\rho_2 V_2 = \rho_1 V_1 \; A_1/A_2 = 5.667 \text{ lb/ft}^2\text{/sec} \tag{4}$$

Now the subsonic diffuser reduces the stream velocity from the initial Mach number of

$$M_o = .904 \tag{5}$$

to $M_2 = .217$ (6)

in accordance with the relation given by equation (9) of Chapter 5C. The efficiency of the diffusion process is taken to be 0.8. The equation connecting the Mach numbers across the compressor stage is (ref. 3, equation 15)

$$\frac{M_3}{M_2} = \left(\frac{p_2}{p_3}\right)^{\frac{\gamma+1}{2\gamma}} \tag{7}$$

A compression ratio of three is considered typical for turbojets and since $\gamma = 1.4$

$$M_3 = (1/3)^{6/7} M_2 = .390M_2 = .0347 \qquad (8)$$

The equation connecting stagnation temperatures across the compressor is (equation 17, <u>loc. cit.</u>):

$$\frac{T_3^{(s)}}{T_2^{(s)}} = \left(\frac{p_3}{p_2}\right)^{\frac{\gamma-1}{\gamma}} \left(\frac{1 + \frac{\gamma-1}{2} M_3^2}{1 + \frac{\gamma-1}{2} M_2^2}\right) \qquad (9)$$

$$= 3^{2/7} \frac{5 + M_3^2}{5 + M_2^2} = 1.358 \qquad (10)$$

where the stagnation temperature, which remains constant except where work is done on or heat is added to the gas stream, is the following function of static temperature and Mach number:

$$T^{(s)} = T\left(1 + \frac{\gamma-1}{2} M^2\right). \qquad (11)$$

Now since the flight temperature T_1 is 393° R and Mach number $M_o \cong 0.9$, we have:

$$T_2^{(s)} = T_1^{(s)} = 393\left(1 + \frac{.81}{5}\right) \cong 460. \qquad (12)$$

Hence, by equation (10):

$$T_3^{(s)} = 625°R. \qquad (13)$$

This is then the temperature at the input to the reactor. Since the gas temperature at the exit of the reactor must not exceed 2000°R, a constraint imposed by practical gas turbine design, we see that the temperature difference, ΔT, through which the reactor raises the gas is:

$$\Delta T = 2000 - 625 = 1375°R. \qquad (14)$$

Now in order that the gas stream in the nuclear turbo jet receive the same energy as was available in the form of heat value of the gasoline in the gasoline burning prototype, the mass flow must satisfy:

$$\dot{Q} = \dot{m} C_p \Delta T, \qquad (15)$$

that is $$\dot{m} = \frac{\dot{Q}}{C_p \Delta T} = \frac{2.37 \times 10^4}{.26 \times 1375} = 66.3 \text{ lb/sec} \qquad (16)$$

where the values of $\dot{Q}$ and ΔT are from equations 1 and 14 respectively and where an average value of C_p was taken to be .26 BTU/lb.deg.F.

Since $\dot{m} = \rho VA$, the intake area follows at once with the aid of equation 2, namely:

$$A_1 = \frac{66.3}{17} = 3.90 \text{ sq. ft.} \qquad (17)$$

And since $A_1/A_2 = 1/3$, the gas stream cross-sectional area at the diffuser exit and in the reactor must be:

$$A_2 = A_3 = 11.7 \text{ sq. ft.} \qquad (18)$$

Now heat transfer equation 5, Chapter 5B when integrated is:

$$\log \frac{T_w - T_3^{(s)}}{T_w - T_4^{(s)}} = \frac{.0604}{(\rho_3 V_3)^{0.2}} \; \frac{L}{d^{1.2}} \tag{19}$$

where L, d are the length and diameter in feet of an individual tube of the convective heater and the mass current density, ρV, is in lb/ft^2/hr. The solution for $L/d^{1.2}$ yields:

$$\frac{L}{d^{1.2}} = \frac{(20400)^{.2}}{.0604} \; \log \frac{4000 - 625}{4000 - 2000} \tag{20}$$

where a wall temperature of 4000°R in the reactor is assumed. Hence

$$\frac{L}{d^{1.2}} = 63.05 \tag{21}$$

This equation expresses the constraint imposed upon the reactor design by the heat transfer equation. Another condition is that expressed by equation 18 which gives the free gas stream cross-sectional area, namely:

$$A_3 = 11.7 \text{ sq. ft.} \tag{22}$$

A third condition may be supplied by considering optimum nuclear reactor design. As will be seen presently, these three conditions determine the reactor configuration.

Table 3 of Chapter 2 gives, for an average number of 2.1 neutrons emitted per fission, the following approximate critical sizes for optimum solid reactor design:

$$r_c = 54.\ \text{cm} = 1.78\ \text{ft.} \qquad (23)$$

$$L_c = 99.\ \text{cm} = 3.25\ \text{ft.}$$

For a reactor perforated by tubes we need to divide the above numbers by $1 - \Gamma$, hence:

$$r_c = \frac{1.78}{1-\Gamma}\ \text{ft.} \qquad (24)$$

$$L_c = \frac{3.25}{1-\Gamma}\ \text{ft.} \qquad (25)$$

where Γ is the ratio of the gas stream free cross-sectional area to the total cross-section in the reactor. But from the definition of Γ we have:

$$\Gamma \pi r^2 = A_3$$

so that the constraint expressed by equation 22 becomes:

$$r = \sqrt{\frac{11.7}{\pi\Gamma}} = \frac{1.93}{\sqrt{\Gamma}} \qquad (26)$$

The value of Γ such that both equation 24 and equation 26 yield the same value of r is given by:

$$\frac{1.78^2}{(1-\Gamma)^2} = \frac{1.93^2}{\Gamma} \qquad (27)$$

This requires that

$$\Gamma = 0.410 \ , \quad r = 3.02\ \text{ft.}$$

Hence, by equation 25,

$$L = \frac{3.25}{1-\Gamma} = 5.51 \text{ ft.}$$

The volume of reactor is therefore:

$$\text{Vol.} = (1-\Gamma)\frac{A_3}{\Gamma} L = 1.439 \times 11.7 \times 5.51 = 92.77 \text{ cu.ft.}$$

The weight data are:

	Carbon Reactor	BeO Reactor
Reactor weight, lbs.	12,800	17,500
Uranium weight, lbs.	50	44

Since L = 5.51 ft, equation 21 gives:

$$d = \left(\frac{5.59}{63}\right)^{.8333} = .131 \text{ ft.} = 1.57 \text{ in.}$$

Finally, the number of tubes necessary to accommodate the mass flow is given by

$$n = \frac{A_3}{\pi d^2/4} = 867.9 \text{ or } 868.$$

The preceding results define the reactor completely. The method of calculation, it will be recalled, was to design the reactor so as to supply the same amount of energy to the air stream as would be available from the gasoline consumption in the commercial gasoline-burning prototype turbo-jet. This is a very general treatment and one should therefore examine the validity of the major assumptions involved. One such assumption is that the thrust of the pure air stream in the nuclear vehicle and that of the air-gasoline exhaust products in the gasoline prototype are not too dissimilar. Now both models have approximately the same exhaust temperatures, due to the 1500°F. limitation at the turbine. Also, because of this limitation, turbo-jets operate at extremely lean mixtures, air-fuel ratios being of the order of 60 or more, assuring a molecular weight of the exhaust gases little different from that of air alone. Hence it is not likely that the thrusts of the two designs will be appreciably different. One other factor is the drag due to the reactor tubes. It is not believed, however, that this will be sufficiently greater than in the conventional turbo-jet burner to cause much error in the calculations. For the present reactor design with L/d ratio of 42, equation 13 of Chapter 5C yields a drag per unit cross-sectional area of approximately one dynamic head, -- a figure which is in good agreement with experimental values obtained from laboratory tests on conventional ram-jet burners.

The power added to the gas stream turns out to be approximately 33,600 H.P., while the power output of the engine (thrust times velocity) is 6400 H.P. This gives an overall efficiency of 19%.

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