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TERRESTRIAL HYPERSONIC FLIGHT PROPULSION

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ABSTRACT

A preliminary feasibility study is made of several new proposed propulsion concepts for sustained hypersonic propulsion using the terrestrial atmosphere. These processes are evaluated in the light of their energy production and flight energy requirements. The theoretical considerations of hypersonic flight energy requirements are indicated and magnitudes established for representative drag coefficients and fuel admixtures.

The energy released by stationary and moving detonation waves is computed and compared with propulsion requirements. It is seen that while the first process probably is limited to operational velocities no greater than 7,000 to 8,000 fps, the latter is applicable to the entire hypersonic flight corridor and will be limited only by the amount of supplemental energy available in the flight vehicle which can be used to drive and support the moving detonation. In considering this latter point, several proposed hypersonic airbreathing propulsion system applications, including electrical augmentation, are evaluated.

Within the limits of this analytical approach, an electromagnetic fuel injection afterburner combined with a fuel-lean standing wave detonation process appears the most promising. Exploratory experiments are indicated to determine the feasibility and efficacy of this concept which has a probable flight velocity growth potential to 12,000 fps at altitudes in the 20 to 50 mile range.

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I - INTRODUCTION

One of the approaches toward evaluating new propulsion concepts for cruise-type hypersonic flight vehicles in the earth's atmosphere is to compare the energy required to overcome drag with the energy produced and released by the propulsion system. The advantage of this type of analysis is that a specific flight vehicle configuration need not be defined nor engine component performance estimated in order to obtain an indication of the probable or attractive region of application of the propulsion concept.

This preliminary analytical feasibility study examines several new proposed primary chemical propulsion systems, using the terrestrial atmosphere, with particular emphasis on methods of employing electrical augmentation. Specific weight requirements of practical airborne electric power supplies will be indicated for the various types of applications considered.

Although the "continuous flight corridor" is considered appropriate for this study, an additional approximate envelope of skip-glide trajectories has been included in the potential flight regime because some aspects of electrically augmented systems appear particularly compatible with what would seem the required power plant characteristics of skip-glide operation (i.e., repeatable, high performance, bursts of thrust, of short duration).

II - PROPULSION ENERGY REQUIREMENTS

Assuming initially that all the energy released by the propulsion system is converted into exhaust gas kinetic energy, then the rate of change of energy in the working medium is:

$$E_a = \frac{dE_A}{dt} = \frac{\dot{m}_e V_e^2}{2} - \frac{\dot{m}_i V_i^2}{2} \tag{1}$$

Further, assuming that the exhaust gases are expanded to ambient static pressure and that the propulsive thrust equals the drag

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forces, then the exhaust velocity can be related to the drag coefficient (referenced to the inlet area), mass handled, and initial velocity:

$$V_e = \left(\frac{C_{D_i}}{2} + 1 \right) \frac{\dot{m}_i}{\dot{m}_e} V_i \qquad (2)$$

Thus, for steady flight, the required energy release rate into the working fluid is:

$$\frac{E_A}{m_i} = \frac{2 \times 10^{-5}}{\eta} V_i^2 \left[\frac{m_i}{m_e} \left(\frac{C_{D_i}}{2} + 1 \right)^2 - 1 \right] \qquad (3)$$

where η is the efficiency factor which accounts for incomplete conversion to kinetic energy.

Representative values of (m_i/m_e) for hydrogen-air mixtures are: 0.972 for stoichiometric; 0.987 for 1/2 stoichiometric; and 1.00 for no fuel addition within the propulsion system or if the gases are premixed. † In general the drag coefficient C_{D_i} has a far more pronounced effect on flight requirements than fuel mixture which only assumes some importance for low drag vehicles at very high velocities (Fig. 1). Thus, a refined aerodynamic shape with lifting surfaces is desirable for cruising hypersonic flight in order to minimize the primary propulsion energy requirements.

For purposes of this study it is assumed that some laminar boundary layer stabilization will be engineered into the advanced hypersonic vehicle. Therefore, a nominal value of 0.2 has been assigned to C_{D_i} throughout this study; even this low value may be considered conservative for a well designed future vehicle.

III - ENERGY PRODUCTION OF DETONATIVE PROCESSES

Standing detonation waves have been proposed and investigated for hypersonic propulsion (Ref. 1). In essence, this phenomena is a combination of a chemical bond energy release mechanism with hypersonic gasdynamics, and therefore appears fundamentally compatible with the application. The energy released is manifested as a change in total enthalpy (or static enthalpy and kinetic energy), or:

$$E_r = \sum (mh)_2 - \sum (mh)_1 + \frac{V_2^2}{2gJ} \left(\sum m_2 \right) - \frac{V_1^2}{2gJ} \left(\sum m_1 \right). \qquad (4)$$

The moving detonation wave process, characterized by the electromagnetic detonation concept (Ref. 2), has the supplemental energy of the driving force (piston work) E_p, which by acting on the product gas makes possible the translation motion of the wave. Thus, its total energy is:

$$E_T = E_r + E_p = E_r + \left(\frac{V_1^2 - 2V_1V_2}{2gJ} \right) \left(\sum m_2 \right) \qquad (5)$$

The kinetic energy change of the stationary detonation is always negative because the product gas velocity is less than the pre-wave or reactant velocity. Thus, the net energy release is derived from the static enthalpy change and must be converted by another device, e.g., a nozzle, in order to be useful for propulsion. The higher the initial (pre-wave) velocity the lower the product gas velocity, and the smaller the amount of E_r energy released; ultimately the process becomes endothermic.

The energy available from a moving (driven) detonation wave increases with wave velocity V_1 because of the driving force E_p. Thus, at high initial velocity (when the detonation becomes endothermic) the over-all energy E_T is still positive and the process can be exploited for propulsion. The only limitations are the amount of energy available from another source to drive the wave front, and the efficiency of energy conversion to E_p.

An analysis (Ref. 3) comparing these two processes has been made for the stoichiometric hydrogen-air mixture at velocities between

† Fuel injected into surrounding atmosphere, without drag, upstream of the propulsion system inlet.

the Chapman-Jouguet condition and 16,000 fps, and pressure altitudes in the flight corridor corresponding to the ARDC 1959 Model Atmosphere The stoichiometric air-hydrogen mixture has been chosen for comparison because it represents the best reactant combination from the standpoint of energy release, kinetics, availability, weight, storage stability, and other known properties which may be required for proper design of a hypersonic vehicle.

These results (Fig. 2) are summarized in terms of the conversion efficiency η needed to meet minimum propulsion requirements. The altitude variation of conversion efficiency shown reflects the sensitivity of the energy released by detonative processes to reactant pressure. The limits of the flight corridor and the C-J condition, likewise shown, keep the evaluation on a realistic basis. In addition to a certain inefficiency expected in the real detonation process and the utilization of its product gases for propulsion, there will be further limits imposed on the application regime of the process. Normally, thermalized gas streams are considered to exhibit half their total energy in kinetic form. By including other expected process inefficiencies, such as fuel injection inhomogeneity, wall effects, radiation, etc., a reasonable range of over-all operating efficiency may be considered between 15% and 30%. In view of the conservative drag coefficient assumed for these calculations, an over-all energy conversion efficiency η of 30% for the propulsion system may be allowed as reasonable.

On this basis, it is seen that the moving (driven) detonation process appears operational within the entire hypersonic cruise or steady flight regime while the standing detonation process is restricted to the lower range of velocity, no greater than about 7,000 to 8,000 fps, for altitudes of 158,000 and 50,000 feet, respectively. Only the moving detonation process offers a significant reserve of energy release which can be used for acceleration, increasing range (by decreasing the fuel admixture), or regenerating energy, which can then be used to provide or minimize the effective driving force needed from another source within the vehicle. A similar analysis based on a lower drag coefficient would improve the operational regime of the standing detonation process, but would not change the relative performance of the two processes as indicated in Fig. 2. The only remaining reservation one may have regarding the application of the moving detonation process is the feasible amount of energy available in a finite volume flight vehicle which can be used to support the moving detonation wave structure. This aspect is examined in the next section.

IV - METHODS OF ELECTRICAL AUGMENTATION

Electrical augmentation of hypersonic airbreathing chemical propulsion systems becomes most attractive for high altitude applications (i.e., above 100,000 feet) because the power demands on an airborne electrical source are reduced. The drag force to be overcome, and the mass rate of airflow used as the working fluid decreases as altitude increases for a flight vehicle of given dimensions.

There are several ways electricity may be used to augment hypersonic airbreathing propulsion systems which derive their primary energy source from chemical reactions. The attractive characteristics of a supported moving detonation revealed in the previous section, suggests consideration of the electromagnetic piston supported detonation concept (Ref. 2).

A second method is to introduce direct or induced electrical discharges into the gas products of standing wave detonations. A third method is to use the electromagnetic forces for injecting the fuel into a hypersonic stream. The propulsion system performance and qualitative operational features of each of these three basic types of exploitation of electrical energy, and some of their modifications, will be examined. These three types are shown schematically in Fig. 3 together with the conventional standing detonation wave arrangement.

A. Introductory Theoretical Analysis

In an airbreathing propulsion system employing a steady flow detonation process, Equation (1) can also be written as:

$$\dot{E}_A = \frac{\dot{m}_i}{2J}\left[\left(1 + \frac{.029}{\psi}\right) V_e^{\,2} - V_i^{\,2}\right]. \qquad (6)$$

Assuming complete conversion of $\dot{E}_A$ into directed kinetic energy and $\psi = 1$, the maximum available thrust per unit time (i.e., per second) is:

$$F_{max} = \frac{W_{ai}}{g}\left[1.014\left(5\times10^4 E + V_i^{\,2}\right)^{\frac{1}{2}} - V_i\right]. \qquad (7)$$

Considering the effect of centrifugal forces, a hypersonic flight vehicle with aerodynamic lift surfaces cruising at constant velocity requires a propulsive thrust of:

$$F_{RC} = \frac{W_V - F_C}{(C_L/C_D)_{max}} = \frac{W_V}{(C_L/C_D)_{max}}\left[1 - \frac{V_i^{\,2}}{gR_o}\right]. \qquad (8)$$

Assuming that the flight vehicle equipment includes a source of electrical energy (e.g., a nuclear reactor-turboelectric system) which will be used by the energy augmenter in the propulsion system, the vehicle weight may be considered a multiple of the electric energy source weight, or:

$$W_V = C_W W_e . \qquad (9)$$

Further, letting $(C_L/C_D)_{max} = a_L$, and $\frac{V_i^{\,2}}{gR_o} = b$, Equation (8) becomes:

$$F_{RC} = (1-b)\left(\frac{C_W}{a_L}\right) W_e . \qquad (8a)$$

The weight and output of the electrical power supply can be related by:

$$\frac{W_e}{E_e} = k_e \qquad (10)$$

If $E_T = E_c + E_e/W_{ai}$, then using the conversion factor (1 kw = .95 BTU/sec), Equation (7) takes the form:

$$F_{max} = \frac{W_{ai}}{g}\left\{1.014\left[5\times10^4\left(\frac{.95\,W_e}{k_e W_{ai}} + E_C\right) + V_i^{\,2}\right]^{\frac{1}{2}} - V_i\right\} \qquad (7a)$$

B. <u>Electromagnetic Piston-Supported Moving Detonation</u>

The energy needed to support a moving detonation (i.e., piston work) is dictated by the thermodynamics and strength of the detonation, or at the very minimum, from Equation (5):

$$E_e = E_p = \left(\frac{V_1^{\,2} - 2V_1V_2}{2gJ}\right)\sum m_2 .$$

For optimum operation it can be shown that: V_1 corresponds to V_i, and V_2 to V_e, and $\sum m_2$ indicates that the piston acts on all product gas components. Under these circumstances, F_{max} is computed directly from Equation (7), since $E = E_c + E_p$ is known; and the vehicle weight and payload may be found from Equations (8) and (9), respectively, where now $F_{max} = F_{RC}$. Furthermore, the ratio k_e, absolutely necessary for the implementation of this scheme, is:

$$k_e = \frac{W_V}{C_W E_p} = \frac{F_{max}\, a_L}{E_p(1-b)C_W} \qquad (10a)$$

$$\text{or} \quad k_e \sim \frac{a_L}{(1-b)C_W}\left\{\frac{\left[5\times10^4(E_c + E_p) + V_i^{\,2}\right]^{\frac{1}{2}} - V_i}{V_i^{\,2}}\right\} \qquad (10b)$$

From an order of magnitude analysis it is seen (Fig. 4) that the ratio k_e must become smaller as $(C_L/C_D)_{max}$, or E_c decreases and C_W or V_i increases. Further, since E_c tends to decrease with higher altitudes for the same V_i, k_e decreases slowly, inversely with altitude.

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C. Electrically Augmented Standing Detonation

Electrical energy may be added to the detonation products in order to augment the chemical energy release. It is preferable to use either direct or induced discharges that will accelerate the subsonic product gases as well as increase their static enthalpy. This may be achieved, for example, by discharging between two or more concentric ring electrodes located at either end of a divergent channel. Such shaped discharges produce a squeezing action on the gas of greater magnitude directed downstream than toward the detonation front location. The performance of such a propulsion system can be obtained by equating relations (7a) and (8a). Thus, the two thrust values will be made compatible without specifying, beforehand, the magnitude of E_e, and a value of k_e can be selected as desired or considered practical.

Then, using the relation $W_{ai} = \rho_i V_i A_i$, for a unit effective inlet area, A_i, one obtains:

$$\frac{g(1-b)\left(C_W/a_L\right)W_e}{\rho_i V_i} + V_i = \left[\frac{4.9\times 10^4 W_e}{k_e \rho_i V_i} + 5.15\times 10^4 E_c + 1.03 V_i^2\right]^{\frac{1}{2}} \qquad (11)$$

from which, letting $C_1 = \left[\frac{g(1-b)C_W}{a_L \rho_i V_i}\right]$, the quadratic in W_e is obtained:

$$W_e^2 + W_e\left(\frac{2V_i}{C_1} - \frac{4.9\times 10^4}{k_e \rho_i V_i C_1^2}\right) - \left(\frac{5.15\times 10^4 E_c + .03V_i^2}{C_1^2}\right) = 0 \qquad (11a)$$

This has the form: $W_e^2 + BW_e - C = 0$ with the practical solution:

$$W_e = -\frac{B}{2} + \frac{1}{2}\left(B^2 + 4C\right)^{\frac{1}{2}} \qquad (12)$$

Here, W_e represents the electrical power supply weight necessary to furnish the supplemental electrical energy required for cruise flight, assuming 100% energy transfer efficiency. If C is negative, or $E_c < (-.03V_i^2/5.15\times 10^4)$, W_e will be negative. This is an impossible (non-valid) practical situation, and no amount of electrical energy augmentation possible will permit steady flight. In fact, the only valid solution of Equation (11a) must yield $W_e > 0$, or the term $C > 0$; the limiting condition for the use of electrical augmentation is when $C = 0$, or $E_c = -6\times 10^{-7} V_i^2$. For $V_i \sim 10^4$ fps, this condition corresponds to a net chemical energy change in the detonation or wave process of

$E_c \approx -60$ BTU/lb air.

The aspect of this propulsion system which is difficult to assess theoretically is, whether really significant flow and wavefront instabilities will be induced by the electrical discharges, and if the kinetics of chemical reactions in the hypersonic flow will be enhanced by UV radiation, energetic particles, and pressure perturbation fronts emanating from the discharge region, as has been experimentally observed for imploding discharges into quiescent mixtures (Ref. 4).

D. Electromagnetic Fuel Injection

One of the knotty problems associated with all hypersonic air-breathing propulsion systems is the introduction of the fuel into the airstream without deteriorating the engine performance or requiring excessive mixing lengths (Refs. 5 and 6). The third propulsion system considered here offers a solution to this problem by electromagnetically accelerating and injecting the fuel into the hypersonic stream at high velocity by means of high frequency pulses. By using this concept, each high temperature, high velocity fuel pulse will wrap around and penetrate the hypersonic air stream core within the propulsive duct as it moves past the fuel injection opening. The energetic fuel, including atomic hydrogen, will thus have a higher diffusivity and greater reactivity than from a "cold" injector, and should react with the air along a reaction front system of oblique waves. The short duration, but high frequency of the injection period will create a pulsed (quasi-steady) chemical energy release process without instabilities in the steady inlet and exit flow (Ref. 7). Enough fuel may be introduced in each pulse so as to react with more than just the mass of air passing the injection port for the pulse duration; thus, advantage can be taken even of the fuel mass transport by diffusion in the axial direction. Further, because the chemical reaction now is a transient process, its higher peak temperature requires no different a propulsion device structure than that normally designed for steady flow reaction process of lower temperature.

For purposes of preliminary evaluation, it will be assumed that the amount of electrical energy needed for electromagnetic injection is proportional, in the ratio of the masses involved, to the piston energy of E-M supported detonations, or:

$$E_{p_f} = E_p\left[\frac{f/a}{(1 + f/a)}\right] \text{ BTU/lb mixture,} \tag{13}$$

and for hydrogen fuel ($f/a = .028$), $E_{p_f} = .0272\ E_p$ BTU/1b mixture.

It will be assumed, also, that the chemical reaction energy released is E_c, and the total energy added to the working fluid for propulsion is:

$$E = E_c + E_{p_f} \quad \text{BTU/1b mixture} \quad . \tag{14}$$

The performance analysis and electrical power supply characteristics required are determined by relations similar to Equations (7), (8), (9), and (10a).

E. Partially Supported Moving Detonation

One variation of the continuous E-M piston-supported moving detonation process (see section IV-B) is to produce cyclic support of the detonation wave with electric pulses. This non-steady wave support similarly would not disturb the performance of an otherwise steady-flow inlet and exhaust, if the pulse frequency were sufficiently high and the propulsive duct were properly sized. Furthermore, a cyclic support reduces the electrical energy source requirement.

Assume that the partial support is $1/S$ of the steady flow process, where S is greater than, for example, 10. Further, because each electrical discharge pulse (electromagnetic piston formation) stops the local flow momentarily, creating transient high pressure (hammer-shock) pulses in the vicinity of the disturbance which tend to enhance the chemical energy release E_c, it may be assumed that, compared to the steady support case (or where $S = 1$),

$$F_{max_S} = \frac{1}{y}\ F_{max}, \tag{15}$$

where: $S > y$. Since F_{max} is known from Equation (7), F_{max_S} may be determined directly from Equation (15) and using Equation (8a) where now $F_{RC} = F_{max_S}$, W_{e_S} is found. Then, from $E_{e_S} = (\frac{1}{S})\ E_e$:

$$k_{e_S} = (\frac{S}{y})\ k_e, \tag{16}$$

where k_e is obtained from Equation (10a). Thus, in order to tolerate a k_{e_S} value, which is greater than the steady supported case by one order of magnitude, the net thrust must remain unchanged for the maximum degree of partial support, i.e., 1/10. This estimate appears, at first, overly optimistic. It is more likely that for 1/100 support the thrust may change to approximately 1/10 its steady state value. Thus, with this partial support scheme, only a proportionately smaller flight vehicle or a larger capture area propulsion device must be considered.

V - QUANTITATIVE EVALUATION

A. Assumptions

The determination of performance and the comparative evaluation of these various new propulsion concepts has been carried out for a pressure altitude of 158,000 feet. The flight velocity range has been considered between 8,000 and 16,000 fps, corresponsing to about Mach 7.3

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(in the ARDC 1959 Model Atmosphere) and Mach 14.5, respectively. It has been assumed that the energy conversion efficiency is 100%, the effective inlet capture area is 1 ft ^{2}, the maximum lift to drag ratio is 2.0, and the flight vehicle weight W_V (which includes structure, payload, fuel, and electrical power supply) is 1.4 times the electrical power supply weight W_e (i.e., $C_W = 1.4$).

In addition certain special assumptions were made for each of the five types of applications, as follows:

Type A (Standing Detonation) - Comparative calculations for 50% and 100% recovery of recombination energy E_R were made at $V_1 = 8,000$ fps in addition to the equilibrium condition. The latter condition becomes endothermic slightly beyond 9,000 fps, and of questionable utility for propulsion thereafter. The 100% recovery case may be considered a theoretical upper limit of performance not actually attainable in practice at this altitude.

Type B (Electromagnetic Piston Supported Detonation) - An electrical power source is contained within the flight vehicle and supplies an exceedingly large number of moving electromagnetic pistons, each of which supports a moving detonation so that, in effect, a continuous flow process exists.

Type C (Electrically Augmented Standing Detonation) - Two values of the electrical power source specific weight k_e (i.e., 1 and 5) are pre-selected for comparison.

Type D (Electromagnetic Fuel Injection) - The same exothermic chemical energy E_c as Type A is assumed, although this system may well yield a greater release of chemical energy because an oblique reaction front rather than a normal one (resulting in less pressure losses) is most likely to be produced.

Type E (Partially Supported Detonation) - This application is considered a variation of Type B. The electromagnetic piston support of a moving detonation is provided 1% of the time ($1/S = .01$), and the thrust is reduced to 10% ($1/y = .10$) of the steady-support process (Type B).

B. Evaluation

The five chemical propulsion applications are compared in Fig. 5 on the basis of the ($W_v - W_e$) term per unit of effective inlet area throughout the flight velocity range of this investigation. This term is the non-electric portion of the flight vehicle's weight and allows comparison of Types B-E with Type A on what is considered a common and meaningful basis. The power supply specific weight requirements for each of the four proposed electrical augmentation devices is shown in Fig. 6 for the examined range of velocity.

Based upon near-future (i.e., 5-6 years) space power system estimates (Ref. 8) it may be assumed that practical (i.e., foreseeable within the next 10-12 years) electrical power supplies for advanced airborne vehicles will be in the 1-5 pound per Kw specific weight range; then all of Type B and Type D applications (except at flight velocities below about 9,000 fps) must be discounted as technologically impractical.

Types A, C, and E all require a means of fuel injection and mixing into the free stream. In addition, the former two are limited in operational flight velocity by the trend of the net chemical reactions to become endothermic beyond about 9,000 fps. Thus, the use of Types A and C at this or greater flight velocity depends on the amount of recovery of recombination energy possible in a practical propulsion device. Type C holds a possible advantage over Type A in that the electrical afterburner will most likely enhance chemical reactivity of the detonable mixture (Ref. 4) at very high altitudes (above approximately 100,000 ft), where the detonation wave structure will probably be quite diffuse and the reaction will not proceed to completion under ordinary circumstances. On the other hand, the practicality of Type E is wholly dependent on the S/y ratio (see Fig. 7) achievable with a working device, and requires experimental data not presently available.

Within the flight speed restrictions initially considered, Type D offers a potential means of fuel injection with high reactivity into a hypersonic airstream. A more attractive k_e ratio at higher velocities may be attained, perhaps by modifying Type D to a more discrete number of electromagnetically accelerated pulses superimposed on a steady fuel "leak" into the propulsive duct. However, as is the case for Type E, experimental data is needed to check the feasibility of this proposed application. Recovery of recombination energy also would make "practical," but just barely so, the application of Type D to higher flight velocities, although virtually complete recovery would be needed (see Fig. 6), which in the light of present experimental data is somewhat unlikely (Refs. 9 and 10).

VI - CONCLUSIONS

For the representative hypersonic flight condition examined (8,000 fps velocity and 158,000 ft altitude), the thrust and fuel specific impulse of applications Types C and D are more preferable to the other two practical applications Types A and E (see Table I). Only by virtue of permitting a heavier vehicle (i.e., 61 vs. 21 lb/ft^2) to be used, is the standing detonation process (Type A) superior under ideal conditions to the proposed electrically augmented types.

At high altitudes the fall off in chemical reactivity becomes important, showing up in performance calculations as an efficiency factor much less than 1.0. Type C appears to offer possible reactivity enhancement for the least amount of supplemental electrical power (at $k_e = 5$). However, if one recognizes that some means of fuel injection and mixing into a hypersonic stream is required with application Type C, then its more realistic performance will probably be no better than application Type D.

Previous studies (Ref. 11) show that the Chapman-Jouguet velocity for a one half stoichiometric hydrogen-air mixture is virtually the same as for the stoichiometric mixture at 158,000 ft, and that its chemical energy release is about 3/4 of the stoichiometric C-J condition. This suggests that a combination of Types C and D might prove the optimum configuration; half the fuel to be introduced in some conventional manner up-stream of the standing detonation wave as in Type C, and the balance to be electromagnetically injected in the electrical discharge afterburner as in Type D. However, the complexity of the flow dynamics of such a system almost precludes a detailed theoretical analysis beyond the approach already taken to give the results of Table I. Therefore, the next logical phase is to conduct exploratory experiments of the two-stream interaction and chemical reactivity aspects of this concept in order to determine its feasibility and efficacy.

The growth potential of this concept to higher flight velocities and altitudes is dependent largely on the ability to recover recombination energy in an exhaust duct of some kind. At this time the indications (Refs. 9 and 10) are that probably no more that about 30% recovery is practically attainable at these very high altitudes, making the probable upper limit of flight velocity just under 12,000 fps.

It may be speculated that the velocity-altitude envelope, as shown in Fig. 8, represents the potential operational area of this advanced propulsion concept. As such, it offers new areas of inquiry and analysis for future hypersonic flight systems.

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REFERENCES

1. Dugger, G. L., "Ramjets," Astronautics, Vol. 7, No. 11 (Nov. 1962) p. 138-142.
2. Foreman, K. M., "The Electromagnetic Detonation Concept," Republic Aviation Corporation Report No. PPL-TR-61-10 (AFOSR 564), (Mar. 30, 1961), also ARS preprint 1702-61, Apr. 26-28, 1961.
3. Foreman, K. M., "Chemical Propulsion for Sustained Terrestrial Hypersonic Flight," paper to be presented at the 50th National Meeting of of AIChE, Buffalo, N. Y., May 5-8, 1963.
4. Foreman, K. M. and Levy, M. E., "Preliminary Report on the Detection of H_2O Emission at 5126 Å (B. H.) from Hypersonic Wave Flow Field in a Stoichiometric Hydrogen-Oxygen Mixture," Combustion and Flame Quarterly, Vol. 6, No. 4 (Dec. 1962).
5. Weber, R. J. and MacKay, J. S., "An Analysis of Ramjet Engines Using Supersonic Combustion," NACA TN 4386 (Sep. 1958).
6. McLafferty, G. H., Jr., paper presented at the High Performance Combustion Systems II session of the XII International Astronautical Congress, Washington, D. C. (Oct 4, 1961).
7. Foa, J. V., Elements of Flight Propulsion, J. Wiley and Sons, pp. 345-346, (1960)
8. Cooley, W. C., "Spacecraft Power Generation," NASA paper N88444, presented to the Combustion and Propulsion Panel of the Advisory Group for Aeronautical Research and Development, C.I.T., Pasadena, California, Aug. 24-26, 1960.
9. Hall, J. G., and Russo, A. L., "Studies of Chemical Nonequilibrium in Hypersonic Nozzle Flows," Cornell Aeronautical Laboratory Report No. AD-1118-A-6 (AFOSR TN 59-1090), (Nov. 1959)
10. Eschenroeder, A. Q., Boyer, D., and Hall, J. G., "Exact Solutions for Nonequilibrium Expansions of Air with Coupled Chemical Reactions," Cornell Aeronautical Laboratory Report No. AF-1413-A-1 (AFOSR 622), (May 1961).
11. Foreman, K. M., and Wilson, P., "Chemical Detonations in the Upper Atmosphere," Republic Aviation Corp. Report No. PPL-TR-61-34 (AFOSR 1763), (Oct. 20, 1961).

NOMENCLATURE

A_i = effective inlet area, ft^2

A_m = maximum projected area of vehicle, ft^2

C_{D_i} = total drag coefficient (referenced to effective inlet area) = $C_{D_m} A_m/A_i$

C_{D_m} = total vehicle drag coefficient

C_L/C_D = vehicle lift to drag ratio

E_a = rate of energy addition, BTU/sec

E_A = energy added, BTU

E_c = chemical energy release, BTU/lb air-unit time

E_e = electrical energy release, BTU/lb air-unit time

E_p = piston work, BTU/lb

E_r = energy released, BTU/lb

E_R = recovered recombination energy, BTU/lb

E_s = enthalpy change, BTU/lb

E_T = total energy release, BTU/lb air-unit time

f/a = fuel to air ratio (mole basis)

F = thrust, lb

g = gravitational constant, 32 ft/ sec^2

h = static enthalpy, BTU/lb

J = joule constant = 778 ft lb/BTU

k_e = power supply specific weight, lb/Kw

m_e = exhaust gas mass flow rate, lb/sec.

m_i = mass flow rate, related to inlet, lb-sec^2/sec-ft = $\rho_i A_i V_i$

R_o = Earth's radius = 20.9×10^6 ft

V = gas velocity, fps

V_e = exhaust gas velocity, fps

V_i = flight velocity, fps

W_{ai} = air inlet flow rate, lb/sec

W_e = electrical power supply weight, lb

W_V = flight vehicle weight, lb

η = efficiency factor by which energy addition is converted to kinetic energy

ρ = ambient density, lb/ft^3

ψ = equivalence ratio = $(f/s)_{stoich}/(f/a)$

Subscripts 1 - reactants gas

Subscripts 2 - product gas

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TABLE I - PERFORMANCE COMPARISON

(C_W = 1.4, V_1 = 8000 fps, Altitude = 158,000 ft), Stoichiometric Hydrogen-Air Ratio

Propulsion Application	$\frac{F}{A_1}$ (lb/ft^2)	$I_{sp_{fuel}}$ (sec)	$\frac{W_V}{A_1}$ (lb/ft^2)	$\frac{W_V - W_e}{A_1}$ (lb/ft^2)	k_e (lb/KW)	$\frac{E_e}{A_1}$ (KW/ft^2)
Type A (Standing Detonation)						
Equilibrium	28	1525	61	61	---	---
E_R = 50%	54	3000	119	119	---	---
E_R = 100%	78	4320	171	171	---	---
Type B (Piston Supported Detonation)						
Equilibrium	116	6700	255	73	.13	1360
E_R = 100%	139	8000	305	87	.195	1120
Type C (Electrical Augmentation)						
Equilibrium	30	1735	67	20	5	9.4
	34	1960	74	21	1	53
E_R = 100%	83	4800	135	52	5	26
	91	5250	148	57	1	143
Type D (EM Fuel Injection)						
Equilibrium	28	1680	63	18	1.3	34
Type E (Partial Supported Detonation)						
S/y = 10; Equilibrium	12	670	25.5	7.3	1.4	13

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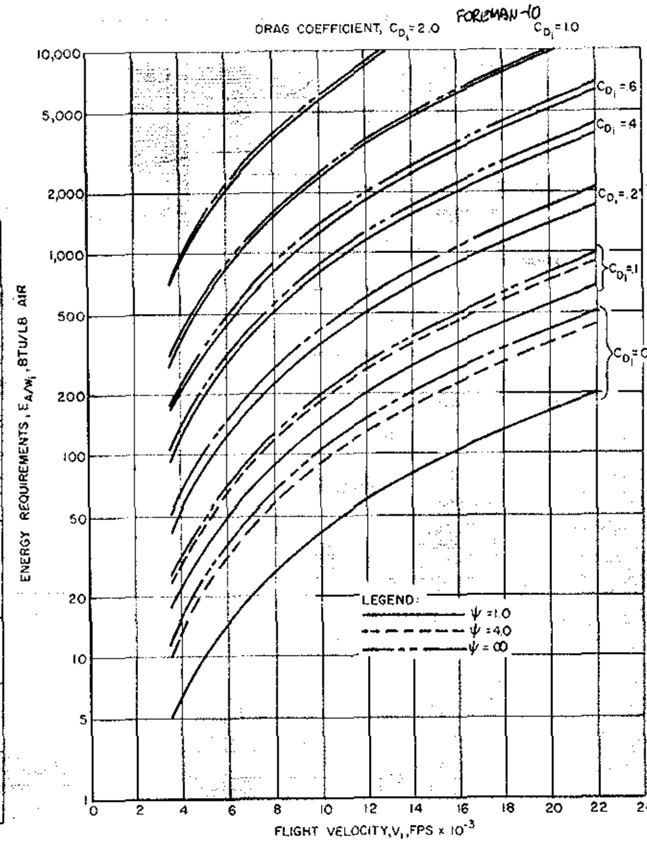

Fig. 1 Energy Requirements for Steady Hypersonic Flight

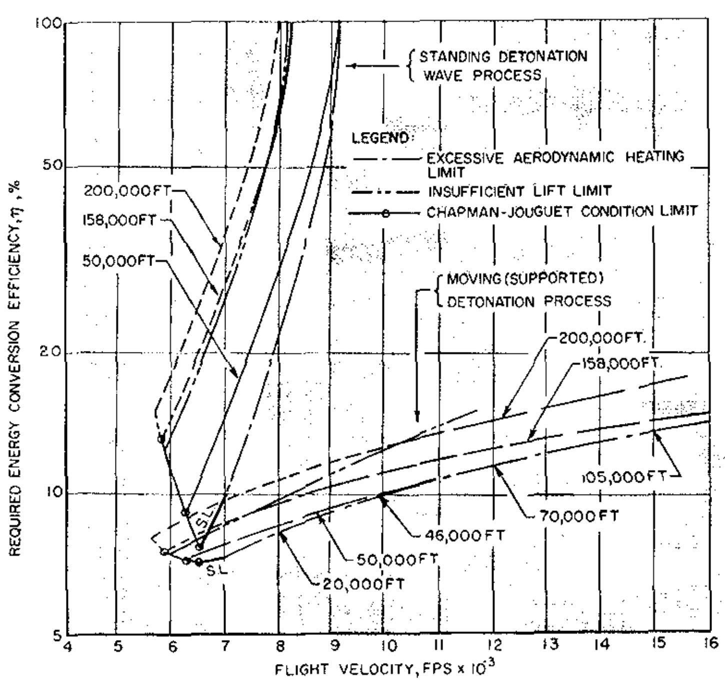

Fig. 2 Minimum Required Conversion Efficiency of Two Detonation Processes for Hypersonic Flight: Reactant Temp. = 540°R, Stoichiometric Hydrogen-Air Mixture

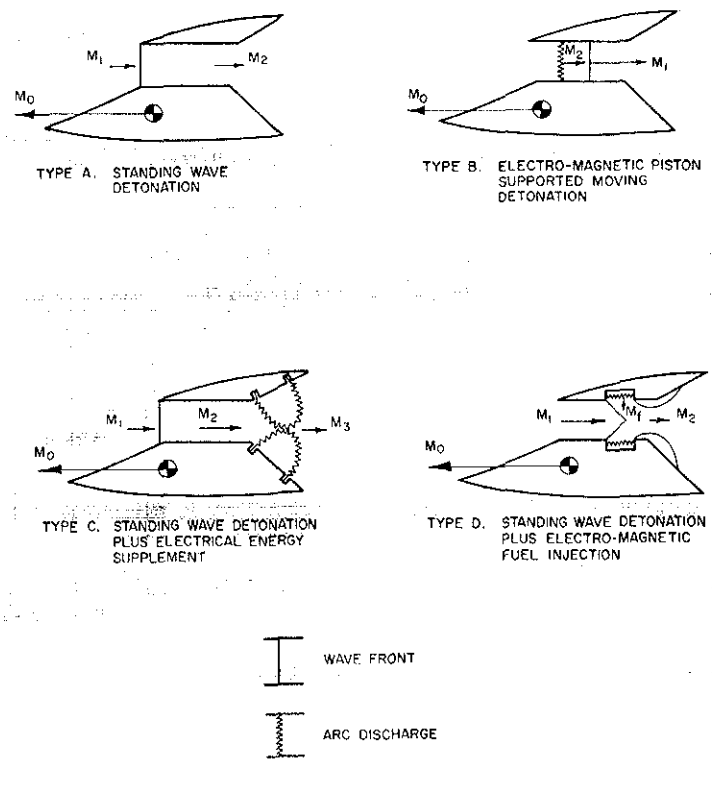

Fig. 3 Schematics of Four Types of Hypersonic Airbreathing Propulsion Systems

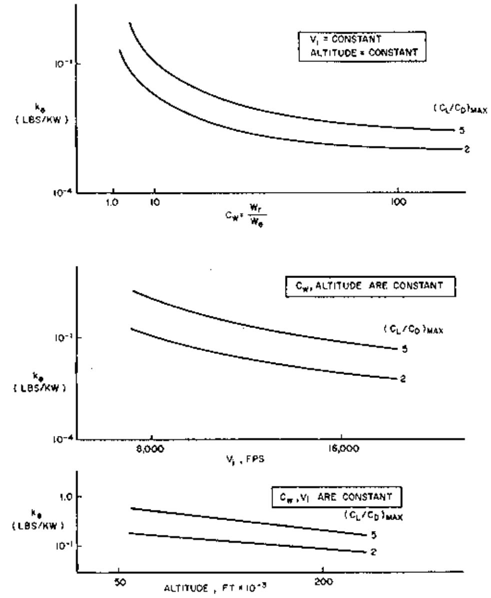

Fig. 4 Typical Variation of k_e With Operation Parameters for Electro-Magnetic Piston Supported Moving Detonations of a Stoichiometric Hydrogen-Air Mixture

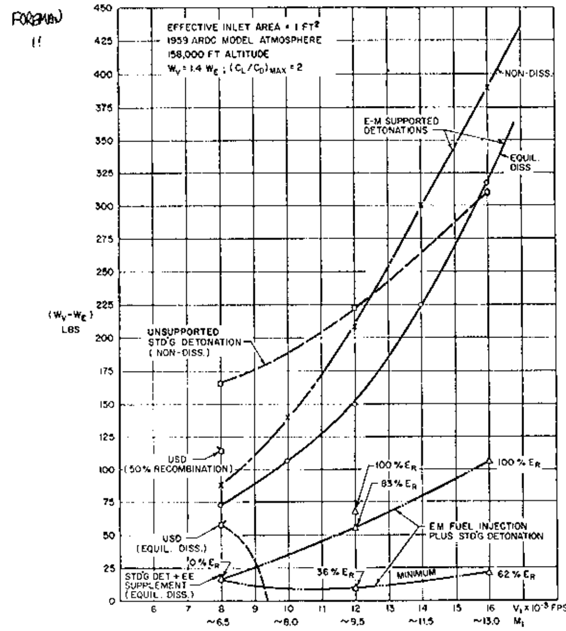

Fig. 5 Vehicle Weight Characteristics of Several Hypersonic Airbreathing Chemical Propulsion Systems

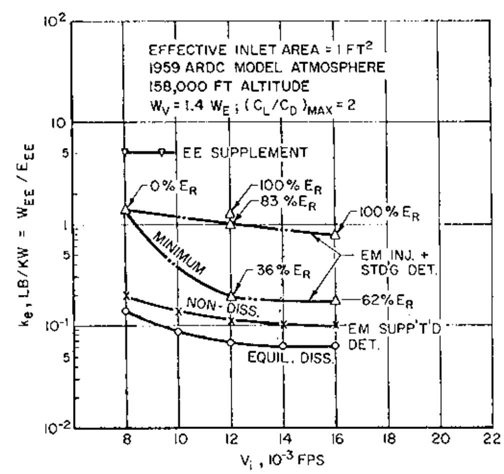

Fig. 6 Primary Power Supply Characteristics of Several Hypersonic Airbreathing Chemical Propulsion Systems

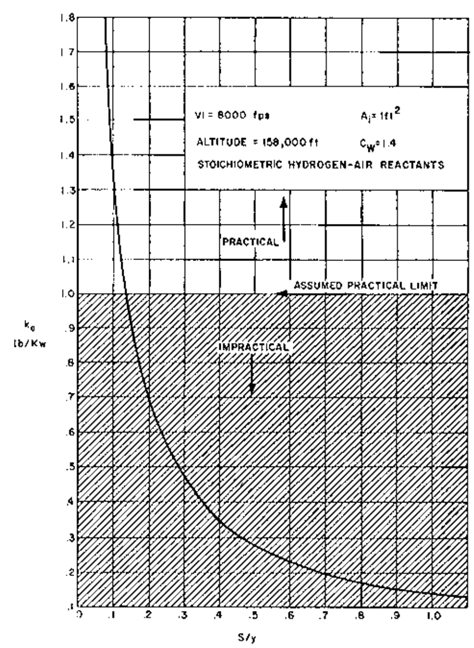

Fig. 7 Variation of Electric Power Supply Specific Weight with S/y Ratio

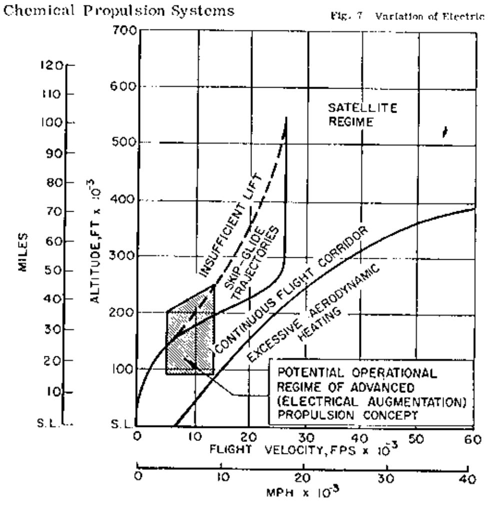

Fig. 8 The Flight Corridor