CONCEPTUAL STUDY OF ROCKET-SCRAMJET HYBRID ENGINES IN A LIFTING REUSABLE SECOND STAGE

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

A simplified analysis of a prismatic-delta configuration flying a near minimum fuel consumption path from staging to orbit showed relative advantages for air-augmented propulsion systems. The trajectory had a constant product of pressure and Mach number during powered flight which extended to speeds slightly greater than orbital so that a zoom maneuver could be used to obtain final orbital altitude. For staging at Mach 10, the estimated payload fraction (payload to staging weight) for the best air-augmented case (3:1 ratio of air to rocket flow) was $\frac{3}{2}$ times better than that using pure rocket power and $7\frac{1}{2}$ times better than that using pure scramjet power.
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SUMMARY

The lifting second stage of a reusable orbital booster is analyzed employing a highly simplified approach so that major problems, trade-offs, and potential can be identified. The powerplant is a hydrogen-oxygen rocket with various degrees of air augmentation ranging from zero for the pure rocket to infinity for the pure scramjet. Additional hydrogen is injected to allow stoichiometric combustion of the air. A trajectory characterized by a constant product of pressure and Mach number is flown during powered flight to speeds slightly above orbital so that a zoom maneuver can be used to obtain the final 100-mile ($1.6 \times 10^5$-m) orbit.

The vehicle configuration was a prismatic delta (flat-top wedge with triangular cross section). A strong interdependence is shown between propulsive, aerodynamic, and structural characteristics. For a staging Mach number of 10 and the same trajectory, the estimated ratio of payload to staging weight for the best air-augmented case (3:1 ratio of air to rocket flow) was $1\frac{1}{2}$ times better than pure rocket power and $7\frac{1}{2}$ times better than that for pure scramjet power.

INTRODUCTION

Relatively little attention has been given to the study of second stages on a lifting trajectory powered by some form of air-breathing propulsion. There have been many concepts proposed for reusable first stages, ranging from rockets through a perplexing variety of air-breathing systems (see, for example, ref. 1). The second stage in accelerating from, say, 50 to 100 percent of orbital velocity must supply about 75 percent of the orbital energy and, therefore, would seem to justify further study. For either the first or upper stages, the provision of some lift capability in the form of wings or body shaping gives a controlled landing ability for easy recovery, subsequent reuse, and in-
creased flexibility of launch operations. If this added aerodynamic lift capability is
designed for use during ascent as well as recovery, then flatter ascent trajectories in the
denser, lower altitudes would be favored at the expense of increased structural weight.
For such trajectories, the advantages of increased specific impulse, but greater weight,
promised by some forms of air-breathing or air-augmented powerplants can be assessed
as well as pure rocket propulsion.

Most studies of reusable second stages have considered only rocket or scramjet
propulsion. In reference 2, the scramjet powered reusable second stage was found to
be relatively unattractive compared with a rocket-powered stage. The present study
considers a spectrum of powerplants ranging from the pure rocket at one extreme to the
scramjet at the other extreme with the accompanying air-augmentation ratio varying
from zero to infinity (air-augmentation ratio is defined as the weight of air flow, sup-
plied by an inlet, divided by the weight of rocket propellent flow). Within this spectrum
are hybrid rocket-scramjet engines with finite values of air-augmentation ratio and val-
ues of specific impulse, thrust coefficient, and engine weight intermediate between those
of the pure rocket and the pure scramjet. Proper choice of air-augmentation ratio for
the vehicle configuration and trajectory is one of the main problems of the study.

The fixed geometry hybrid engine postulated is composed of a primary rocket com-
ponent and a secondary scramjet component that includes an inlet, mixer, burner, and
nozzle. The primary rocket burns hydrogen and oxygen in a stoichiometric mixture
ratio. The rocket flow is fully mixed with the supersonic air before additional hydrogen
is added to complete the combustion of the mixture.

The propulsive, aerodynamic, and structural characteristics of the vehicle are
shown to be strongly interrelated. Within the framework of the analysis the following
important factors are related by the parameters of body volume to planform area
($V^{2/3}/S$) and span to length (b/c):

1. Lift-drag ratio affects the required thrust.
2. Skin wetted area affects the insulation weight.
3. Inlet capture area affects the air mass flow available for propulsion.

The lifting trajectory derived herein is characterized by a schedule such that the
product of atmospheric pressure and Mach number is constant. This trajectory gives
approximately minimum fuel consumption and a constant mass flow per unit area which
greatly simplifies the analysis.

In this study, the optimum payload configuration can be identified for a given staging
Mach number in terms of air-augmentation ratio, vehicle geometry, and flight path.
The payload was selected to be 25 000 pounds (11 340 kg). Staging Mach numbers be-
tween 8 and 12 are investigated. The criterion of merit postulated is the minimum gross
weight of the second stage of the orbital booster. No consideration is given in the pres-
et study to the launch vehicle that propels the second stage to its initial flight condition.
A highly simplified method of calculation based on idealized component characteristics is employed in this study. It is believed that this simple gross analysis yields useful results that are not obscured by excessive details. Moreover, any detailed analysis of reuseable second stages would be weakened by the fact that all of the required technologies currently are too poorly developed for accurate evaluation.

ANALYSIS AND RESULTS

The following parameters have been found to be useful in relating the propulsion, aerodynamic, structural, and trajectory characteristics of a reuseable second stage vehicle. A major goal of the study was to identify the optimum values:

- $V^{2/3}/S$ volume-surface parameter (it dictates the $L/D$ capability of the vehicle, the engine capture area $A_{oo}$, and the skin area $S_o$ (symbols are defined in appendix G).)
- $pM$ means of specifying the flight trajectory (it dictates the air mass flow through the engine capture area $A_{oo}$ and influences the structural weight.)
- $\bar{m}$ air-augmentation ratio (it dictates the specific impulse and the thrust of the stage.)

Engine Model

The hybrid engine incorporates both air-breathing and rocket components as shown in figure 1. Combined with the air induction system is a hydrogen-oxygen rocket operating stoichiometrically (to prevent combustion in the mixer) as opposed to the normal fuel-rich operation of the usual rocket. Both the inlet and the primary rocket are of
fixed geometry. A combustion pressure of 1000 psi \((6.89 \times 10^6 \text{ N/m}^2)\) is assumed for the rocket.

The air-breathing component of the hybrid engine, usually referred to as the secondary component, is selected by matching the inlet, mixer, and burner according to specific interface requirements, as discussed later (see Powerplant Performance, p. 13). Downstream of the rocket-inlet combination is a section to allow complete mixing of the inducted supersonic air with the primary jet. Additional fuel is supplied to complete the stoichiometric combustion of the air in the burner section. Burning occurs at supersonic speeds. A fixed-geometry nozzle is integrated into the vehicle boattail.

**Vehicle Configuration**

The vehicle configuration used in this study is a wingless lifting body. As shown in figure 2, it is idealized as a flat-top wedge with a triangular cross section and is referred to in this report as a prismatic delta. Such a configuration is easily amenable to simplified aerodynamic and structural analysis and was adopted because the calculated weight and aerodynamic characteristics were found to be comparable with those of more realistic configurations studied by others.

If this type of vehicle were to be built, it would have an upper surface shaped to aid the subsonic lift-drag ratio which should exceed a value of 5. Vertical stabilizers located at the tips of the delta planform would provide lateral stability. Auxiliary turbojet power would be used during subsonic maneuvers and to assist during the horizontal airplane-type landing. Figure 2 indicates that the usual components of the aircraft are

![Vehicle Configuration Diagram](image-url)
integrated for generating lift, providing volume, and acting as portions of the engine inlet and exhaust nozzle.

In the present study, two fundamental design constraints affected the inlet size: (1) the engine inlet (when other than the pure rocket case) is located entirely within the pressure field on the lower side of the vehicle, and (2) the capture area is arbitrarily limited to that portion of the vehicle cross section below the plane or axis that defines the zero net lift attitude (see appendix E). This constraint is based on the practical consideration that installation of a hybrid engine near the tip regions would be difficult. Some other assumption would very likely give a different combination of volume-surface area, trajectory, and air-augmentation parameters for best payload.

Trajectory

The trajectory traversed will have a strong influence on the problem of matching the powerplant and structure to the vehicle geometry for maximum payload. Inasmuch as a lifting trajectory is specified, the powered portion will terminate at an altitude considerably less than the goal of a 100-mile (1.6x10^5 m) orbit. Consequently, a velocity greater than orbital will be needed during powered flight so that a zoom maneuver can be used to attain the desired orbital altitude.

The path followed during powered flight is characterized by a constant product of ambient pressure and flight Mach number (pM). This results in approximately constant air flow through the engine, and because of the assumption of stoichiometric combustion, the fuel flow rate is likewise constant.

This trajectory (pM = constant) is a close approximation to the path \( pM^{4/3} = \) constant shown in appendix A to give minimum fuel consumption over a considerable portion of the desired velocity increment. Near orbital velocity the path angle required by the optimum solution becomes excessive and finally indeterminate.

Figure 3 shows several \( pM = \) constant paths within altitudes of 50 000 to 200 000 feet (15 230 to 60 900 m) and includes one case of the more optimum \( pM^{4/3} = \) constant path (extended beyond the applicable velocity). Also, several constant dynamic pressure (pM^2 = constant) paths frequently proposed in air-breathing propulsion analyses are shown to be quite different from the other two examples. Lines of constant skin temperature 1 foot (1/3 m) aft of the leading edge of a flat plate at zero angle of attack are superimposed.

The five phases of the reusable second stage flight are as follows (see fig. 4):

(1) Separation from the launch vehicle at a given Mach number \( M_s \), a given altitude \( h \), and a path angle compatible with \( pM = \) constant flight
Figure 3. Typical flight trajectories.

Figure 4. Schematic of complete flight path.
(2) An acceleration-climb path along the $pM = \text{constant}$ trajectory culminating in a zoom velocity $V_z$ greater than the orbital velocity $V_r$.

(3) A zoom maneuver to achieve an altitude of 100 miles ($1.6 \times 10^5$ m) at the velocity $V_r$. (The zoom consists of a pullup in the atmosphere with an initial velocity $V_z$ and then a ballistic path tangential at its peak to the 100-mile ($1.6 \times 10^5$ m) orbit, where the final small impulse is applied.)

(4) Separation or unloading of the payload

(5) Retroimpulse and glide reentry of the vehicle terminated by the approach and landing with the help of turbojets.

Each of the flight paths is explored with a series of different geometry vehicles represented by the volume-surface parameter $V^2/3S$. This parameter defines the $L/D$ capability of the vehicle (see Vehicle Aerodynamics, p. 16); hence, it dictates the required magnitude of the zoom velocity. For a given $L/D$, there is a corresponding zoom velocity $V_z$ and pullup angle. For a large $L/D$, $V_z$ is moderate; for a small $L/D$, $V_z$ becomes excessive. This is shown in figure 5. (See also appendix B for the derivation of $V_z$.) Thus, a small $L/D$ vehicle will have a protracted powered flight. Each vehicle on a given $pM = \text{constant}$ path is investigated for a series of engines with different augmentation ratios $\bar{m}$.

![Figure 5. - Relation of lift-drag ratio and zoom-orbital velocity ratio necessary to attain an orbital altitude of 100 miles (160 934 m).](image-url)
Powerplant Performance

At high Mach number flight the shock envelope formed is very close to the surfaces of the vehicle. The deflection of the flow passing through the oblique shock can be calculated from the well-known equation relating wave angle $\beta$ and deflecting angle $\alpha$ (flat-plate angle of attack):

$$\frac{\tan(\beta - \alpha)}{\tan \beta} = \frac{(\gamma - 1)M^2 \sin^2 \beta + 2}{(\gamma + 1)M^2 \sin^2 \beta}$$

This can be put in the form

$$\frac{\tan(\beta - \alpha)}{\tan \beta} = \frac{(\gamma - 1) + \frac{2}{M^2 \sin^2 \beta}}{(\gamma + 1)}$$

Using small angle approximations reduces the relation to the following approximation between wave and deflection angle:

$$\beta = \left(\frac{\gamma + 1}{2}\right)\alpha \pm \sqrt{\left[\left(\frac{\gamma + 1}{2}\right)\alpha\right]^2 + \frac{4}{M^2}}$$

(1)

As $1/M^2 \to 0$,

$$\beta \to \left(\frac{\gamma + 1}{2}\right)\alpha$$

Thus, it is seen that at these very high speeds the shock wave angle is only slightly larger than the surface angle of the vehicle and is not a strong function of Mach number.

The engine inlet is located on the underside of the vehicle within the shock envelope. The size of the inlet is reduced by using the vehicle forebody as a precompression surface. The area of the captured stream tube of air does not change appreciably with angle of attack in the hypersonic region.

The powerplant performance was calculated for hydrogen fuel by the methods of references 3 and 4 assuming chemically frozen exhaust nozzle expansion. The overall stagnation pressure ratios of the combined inlet, mixer, and burner of various geometries were determined along several $pM$ = constant paths and for a range of air-augmentation ratios. The variation of engine performance with the change in vehicle angle of attack along the flight path was considered beyond the broad concepts of this re-
From equation (41) of reference 4,

$$I_s = \frac{(m + 1)V_3 - \dot{m}V_o}{g\left(1 + \frac{\dot{m}f}{a}\right)}$$

and

$$C_F = \frac{f m_1 I_s}{A_{oo} q}$$

Figure 6 shows specific impulse $I_s$ against specific thrust coefficient $C_F$ plotted for flight Mach numbers $M_o$ of 10, 17, and 25 for a number of selected augmentation ratios. The figure is plotted for an equivalence ratio of 1. The variation of the specific impulse for air-augmentation ratio of $\dot{m} = 3$ is shown in figure 7 against the equivalence ratio of the fuel in the burning duct aft of the mixing duct. There is a marked increase of the specific impulse with the increase of the equivalence ratio up to its stoichiometric value. At equivalence ratios higher than 1, the increase in specific impulse and thrust coefficient is slight. All vehicle performance calculations used engine performance data at an equivalence ratio of 1.0.

![Figure 6. Performance of the hybrid engine. Altitude, 150,000 feet (45,720 m); equivalence ratio, 1.0.](image-url)
Hybridization

The nominal propulsion system studied herein is a hybrid; that is, the rocket exhaust has been submerged in a secondary airflow in a ducted rocket arrangement. This increases specific impulse and thrust due to the exchange of thermal and kinetic energy. Further increases in thrust and impulse may be obtained by the subsequent addition of heat to the mixed flow. An alternative arrangement without the ejector effect would be the simultaneous but separate use of selected sizes of rocket and scramjet. It is shown in appendix C that the general merits of hybridization, as determined from total enthalpy considerations, are minimal in the hypersonic region (differences in engine weights are neglected). A sample comparison at a Mach number of 10, altitude of 150,000 feet (45,720 m), and an equivalence ratio of 1.0 is shown in table I.

Although these differences appear worthwhile, at higher speeds the benefit of hybridization diminishes as shown in appendix C. Hence, the rocket-scramjet hybrid considered here has similar performance to a set of a separate rocket and scramjet; the results presented here in terms of hybrids can be interpreted equally well in terms of a separate rocket and scramjet, neglecting differences in engine weights. The magnitude of the air-augmentation ratio of the single-duct concept can be translated into relative sizes of rocket and scramjet in the two-ducted concept.
TABLE I. - EXAMPLES OF HYBRID AND SEPARATE POWERPLANTS

[Flight Mach number, 10.]

<table>
<thead>
<tr>
<th>Powerplant</th>
<th>Air-augmentation ratio, ( \bar{m} )</th>
<th>Thrust coefficient, ( C_F )</th>
<th>Specific impulse, ( I_g ), sec</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hybrid</td>
<td>3</td>
<td>1.9</td>
<td>600</td>
</tr>
<tr>
<td>Separate</td>
<td>3</td>
<td>1.4</td>
<td>570</td>
</tr>
<tr>
<td>Hybrid</td>
<td>10</td>
<td>.7</td>
<td>820</td>
</tr>
<tr>
<td>Separate</td>
<td>10</td>
<td>.6</td>
<td>720</td>
</tr>
</tbody>
</table>

Vehicle Aerodynamics

The two fundamental parameters involving the shape of the configuration that affect the aerodynamic characteristics of a hypersonic vehicle are the volume-surface parameter \( V^{2/3}/S \) and the span-length ratio \( b/c \). The lift-drag ratio \( L/D \) is often shown as a function of \( V^{2/3}/S \) and \( b/c \) as described in reference 5. Similar trends of \( L/D \) for the prismatic-delta vehicle used herein are shown in figure 8 and were determined by the methods given in appendix D. These two parameters are conveniently related to the vehicle's wedge angle by the expression

![Figure 8. - Effect of vehicle geometry on lift-drag ratio.](image-url)
Thus, for large volume-surface parameters $V^{2/3}/S$, high values of $b/c$ correspond to large vertex and wedge angles which give greater nonfriction drag losses. Alternately, for the same $V^{2/3}/S$, $b/c$ must be small for thin vehicles (small $\psi$); however, the contribution from skin friction increases. Therefore, the lift-drag ratio reaches a maximum for some optimum span-length ratio for each value of volume-surface parameter.

The resulting variation of maximum lift-drag ratio with volume-surface parameter is shown in figure 9 and was the basis for subsequent results which correspond in all cases to optimum span-length ratio. The wedge angle of the vehicle $2\psi$ for optimum span-length ratio varied from about $7^\circ$ to $30^\circ$ over the range of $V^{2/3}/S$ from 0.1 to 0.6. Other hypersonic vehicle shapes such as elliptic or half cones, etc. often considered have $V^{2/3}/S$ in the 0.10 to 0.20 range, which indicates a lift-drag ratio capability between 5.5 and 4.5 (ref. 6). (By way of conceptual interpretation, for a cube, $V^{2/3}/S = 1.0$, and for a square flat plate of thickness $t_1$, $V^{2/3}/S = (t_1/c)^{2/3}$ - a very small number.) Since a small value of $V^{2/3}/S$ implies a large wetted surface and, hence, a significant weight penalty due to the heavy thermal protection required for hypersonic vehicles, the detailed design should consider the trade-offs involved at off-optimum lift-drag ratios. This was not attempted in the present study.

The relation between skin area and planform (surface) area parameters for optimum span-length ratio is shown in figure 10. For $V^{2/3}/S < 0.30$ where lift-drag ratios are increasing (fig. 9), the total outside skin area is increasing for any given volume; hence, the thermal protection penalty will be greater.

\[
\frac{V^{2/3}}{S} \left( \frac{b}{c} \right)^{1/3} = \left( \frac{2\sqrt{2}}{3} \psi \right)^{2/3}
\]
In a similar manner, figure 11 shows the dependence of capture area on planform area for optimum span-length ratios. The capture area is determined by the zero lift condition as explained in appendixes D and F. Capture area is increasing for \( V^{2/3}/S > 0.2 \) where skin area is near minimum (fig. 10) and lift-drag ratios are smaller (fig. 9). Thus, specifying the volume-surface parameters not only gives a maximum lift-drag ratio but also establishes the maximum air mass flow available for propulsion. In an alternate form of interpretation the ratio of capture to planform areas, \( A_{oo}/S \), varies from about 0.035 to 0.32 over the range of \( V^{2/3}/S \) from 0.1 to 0.6 for optimum span-length ratio.

To interpret the preceding general results in the framework of the second-stage booster problem, it must be assumed that the vehicle with a superior maximum L/D will maintain a relative advantage throughout the staging to orbit speed range. In other words, over the flight path from staging to orbit the maximum lift-drag ratio occurs at only one velocity since aerodynamic lift diminishes as centrifugal lift increases. The maximum thrust requirement occurs at staging where the lift and, consequently, the induced drag are maximum.

Structural Analysis

In contrast to the usual nonlifting rocket trajectory, the path of the air-breathing vehicle results in an environment of higher pressures and temperatures for a longer time. Therefore, the air breather will be penalized by generally higher component weights and dissipation of the large heat input is a major problem. Heat protection by radiation is assumed utilizing, as required by maximum temperature, superalloy heat shields or refractory metal heat shields with oxidation protecting coatings and special
underside insulation. Corresponding estimates of the thermal protection system weight, referred to hereinafter as skin, are shown in figure 12 as a function of panel surface temperature. The curve is based on the data of references 7 and 8. Determining a representative skin temperature for the entire vehicle is, of course, a complex and elaborate procedure far beyond the scope of this report. The unit skin weights selected for vehicles designed for three different path parameters $p_M$ are designated in the figure.

In the stringent thermal environment, a semimonocoque structure cannot be used and the aerodynamic loading will have to be supported by an internal trusswork. Figure 13 shows the variation of the weight of the trusswork per cubic foot of enclosed
volume against the flight dynamic pressure. A typical value will be around 1.5 pounds per cubic foot (24 kg/m$^3$). The preliminary structural analysis is contained in appendix E.

The weight of the tankage was taken as 1.0 pound per cubic foot (16 kg/m$^3$). The weight of the air-breathing component of the engine is assumed to be 50 pounds per square foot (244 kg/m$^2$) of capture area, and the rocket's weight is 2 percent of its thrust.

If a typical value of the weight of skin per square foot of 4 pounds (19.5 kg/m$^2$) is assumed for a $V^{2/3}/S = 0.30$ vehicle, the variation of the weight of skin per unit volume enclosed is shown in figure 14. It is seen that on a volume basis the weight of this component is comparable to that for the tanks or the truss.

Hence, the structural weight of this type of vehicle ($V^{2/3}/S = 0.30$) can be expected to be of the order of 5 pounds per cubic foot (80 kg/m$^3$) enclosed. This can be compared with the propellant densities given in table II. (Propellant density varies with air-augmentation ratio because of the fuel needed to complete the combustion of all the captured air.)

![Figure 14. Effect of vehicle volume on unit skin weight. Volume-surface parameter, $V^{2/3}/S = 0.30$; skin weight, 4 pounds per square foot (19.5 kg/m$^2$).](image-url)
TABLE II - VARIATION OF PROPELLANT DENSITY WITH AIR-AUGMENTATION RATIO

[Equivalence ratio, 1.0.]

<table>
<thead>
<tr>
<th>Air-augmentation ratio, ( \bar{m} )</th>
<th>0</th>
<th>1</th>
<th>3</th>
<th>5</th>
<th>10</th>
<th>( \infty )</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propellant density, ( \rho_f ), lb/ft(^3) (kg/m(^3))</td>
<td>26.4</td>
<td>23</td>
<td>18.7</td>
<td>16</td>
<td>12.4</td>
<td>4.4</td>
</tr>
<tr>
<td></td>
<td>(422.8)</td>
<td>(352.3)</td>
<td>(299.4)</td>
<td>(256.2)</td>
<td>(198.6)</td>
<td>(70.45)</td>
</tr>
</tbody>
</table>

Stage Performance

The overall performance of the vehicle in its function as a booster stage and as measured by the payload ratio is investigated by selecting several paths (pM = constant) and a given geometry vehicle, dictated by a specific value of \( V^{2/3}/S \). The secondary flow and its amount is dictated by the geometry and trajectory of the vehicle. The option that remains is the amount of the primary flow which is determined by the parametric choice of \( \bar{m} \). The larger this primary flow, the smaller the specific impulse and the greater the thrust. Further explanation of the calculation procedure is given in appendix F. The payload fractions presented hereinafter are primarily intended for relative comparison of the propulsion systems studied rather than absolute predictions. A variety of air-augmentation ratios have been examined resulting in figures 15 and 16 where the optimum vehicle and the path have been identified from a large number investigated. The greatest payload ratio obtained is for the vehicle \( V^{2/3}/S = 0.30 \), the path pM = 0.10 atmosphere, and it is relatively insensitive to an air-augmentation ratio between values of 3 and 6. This payload ratio \( P/W_o \) is equal to 0.11 for a staging Mach number \( M_s \) of 10, as seen from figure 15. For the same flight path, the corresponding pure rocket stage value is \( P/W_o = 0.07 \), and the corresponding scramjet stage value is 0.015. In the case of the pure rocket, the payload ratio is practically independent of the type of vehicle used. In the case of the scramjet, the only vehicle that could attain some positive performance had a high L/D with \( V^{2/3}/S = 0.15 \). Even then, the velocity reached on pure scramjet power was only 23,000 feet per second (7010 m/sec); the remaining acceleration had to be completed by rocket. In the region of maximum payload the lower value of air-augmentation ratio of 3 was chosen for further illustration. The hypothesis was that in an actual design some effects not considered herein, such as mixing and entrainment losses, would favor the smaller value of augmentation ratio.
The results in figure 15 are for only a single path, \( p_M = 0.10 \) atmosphere, which requires a powered climb between 100 000 and 130 000 feet (30 500 to 39 600 m) in altitude (see fig. 3). Figure 16 shows the performance of hybrid vehicles having an air-augmentation ratio \( \bar{m} = 3 \) for various values of the path parameter \( p_M \) and vehicle volume-surface parameters. It is seen that at higher altitudes (\( p_M < 0.06 \) atm) a \( V^{2/3}/S = 0.15 \) vehicle is better; however, over most of the paths, the \( V^{2/3}/S = 0.30 \) vehicle that combines a moderate \( L/D \) capability with low values of skin-volume \( S_{oa}/V^{2/3} \) parameter and acceptable capture area - volume parameter \( A_{oo}/V^{2/3} \) is superior. The specific component weights for the rocket, the optimum hybrid, and the scramjet stages are given in table III. The best hybrid included is \( V^{2/3}/S = 0.30, \bar{m} = 3, \) and \( p_M = 0.10 \) atmosphere. It will be observed that there is a great variation in the volume of the vehicle depending on the type of the engine. The hybrid stage is only half the size of the rocket stage and an order of magnitude smaller than the scramjet stage. To help visualize the differences, figure 17 shows the three optimum vehicles - scramjet, rocket, and hybrid (rocket-scramjet) - drawn to scale. The size of the vehicle is reduced drastically when rockets are incorporated into the propulsion system. Thus, the estimated benefit accrued by utilizing hybrid rather than rocket
TABLE III. - COMPARISON OF WEIGHT DISTRIBUTIONS AND VOLUME PARAMETERS FOR ROCKET-, HYBRID-, AND SCRAMJET-POWERED VEHICLES

[Trajectory parameter, pM = 0.1 atm.]

(a) Weight distributions

<table>
<thead>
<tr>
<th>Component</th>
<th>Rocket (T)</th>
<th>Hybrid (T)</th>
<th>Scramjet</th>
<th>Rocket (m = 3)</th>
<th>Hybrid (m = 3)</th>
<th>Scramjet</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>lb</td>
<td>kg</td>
<td></td>
<td>lb</td>
<td>kg</td>
<td></td>
</tr>
<tr>
<td>Payload</td>
<td>25×10^3</td>
<td>11.34×10^3</td>
<td>11.34×10^3</td>
<td>11.34×10^3</td>
<td>11.34×10^3</td>
<td></td>
</tr>
<tr>
<td>Tanks</td>
<td>25×10^3</td>
<td>11.34×10^3</td>
<td>1.85</td>
<td>11.34×10^3</td>
<td>1.85</td>
<td></td>
</tr>
<tr>
<td>Truss</td>
<td>24</td>
<td>14.4</td>
<td>270</td>
<td>270</td>
<td>270</td>
<td></td>
</tr>
<tr>
<td>Skin</td>
<td>15.5</td>
<td>12.8</td>
<td>162</td>
<td>162</td>
<td>162</td>
<td></td>
</tr>
<tr>
<td>Engine</td>
<td>7</td>
<td>9.2</td>
<td>41</td>
<td>41</td>
<td>41</td>
<td></td>
</tr>
<tr>
<td>Landing equipment</td>
<td>12.5</td>
<td>9.2</td>
<td>131</td>
<td>131</td>
<td>131</td>
<td></td>
</tr>
<tr>
<td>(landing gear, return</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>engine and fuel, orbital</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>impulse rocket</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fuel and oxidizer</td>
<td>257</td>
<td>1660</td>
<td>162</td>
<td>162</td>
<td>162</td>
<td></td>
</tr>
<tr>
<td>Total stage weight, W_o</td>
<td>357</td>
<td>225</td>
<td>851</td>
<td>851</td>
<td>851</td>
<td></td>
</tr>
</tbody>
</table>

(b) Volume parameters

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Rocket (T)</th>
<th>Hybrid (m = 3)</th>
<th>Scramjet</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total volume, ft^3 (m^3)</td>
<td>16×10^3 (453)</td>
<td>9.6×10^3 (271.8)</td>
<td>189×10^3 (509.5)</td>
</tr>
<tr>
<td>Volume-surface parameter, v^2/3/s</td>
<td>0.3</td>
<td>0.3</td>
<td>0.15</td>
</tr>
</tbody>
</table>

propulsion is approximately 60 percent in payload fraction when the integrated aspects of vehicle geometry, trajectory, and air-augmentation ratio are considered.

It is expected that the payload ratio will vary directly with staging Mach number. This is illustrated by figure 18 which shows staging Mach numbers from 8 to 12 for the optimum configuration and trajectory. Selection of the best staging Mach number requires consideration of the first stage also; however, this is beyond the scope of the present study.
CONCLUDING REMARKS

A conceptual study was made of a lifting reusable second stage using rocket-scramjet propulsion. The hybrid engine has a hydrogen-oxygen rocket with varying degrees of air augmentation extending from the pure rocket to the pure scramjet with stoichiometric combustion in all cases. The adopted trajectory closely resembles the optimum minimum fuel consumption path where possible. The trajectory was characterized by a constant product of ambient pressure and Mach number which gave a nearly constant air mass flow rate. Powered flight extended beyond orbital velocity, depending on the maximum lift-drag capability of the vehicle, so that a zoom maneuver could be used to attain the final orbit altitude of 100 miles (1.6×10^5 m).

The vehicle geometry was assumed to be a prismatic delta (flat-top wedge with triangular cross section). A strong relation was found between the powerplant type and vehicle geometry. Span-length ratios were chosen to give maximum lift-drag ratios for each value of the volume-surface area parameter $V^{2/3}/S$. For the lower values of $V^{2/3}/S$ usually suggested for hypersonic vehicles (0.1 to 0.2), the lift-drag ratios are highest (4.5 to 5.5), but available capture area (stream tube precompressed by the bottom surface of the vehicle) is smallest and skin area (needing thermal protection) is largest. For high values of $V^{2/3}/S$ (0.5), the lift-drag ratio is low (<2.0), but capture area is highest and skin area is low. Thus, the capture area attainable affects the
powerplant selection (air-augmentation ratio) and coupled with the constraint of sufficient propellant volume determines the ability to attain orbital flight. The payload placed in orbit, however, depends on the vehicle size and the empty weight (structure, engine, and equipment).

Because of the high temperature environment, a pin-jointed trusswork with an insulating skin was selected for the structure. For the best vehicle, the weight of the truss per unit volume supported was comparable to those for the skin and propellant tanks, the sum of which was approximately 5 pounds per cubic foot (80 kg/m$^3$). This compares with propellant density requirements of 26.4 pounds per cubic foot (422.8 kg/m$^3$) for the rocket only case, 18.7 pounds per cubic foot (299.4 kg/m$^3$) for an air-augmentation ratio of 3, and 4.4 pounds per cubic foot (70.45 kg/m$^3$) for the scramjet only case (hydrogen is the only fuel considered).

The vehicle having the highest estimated payload to staging weight ratio was characterized by a volume-surface parameter $V^{2/3}/S$ of 0.3 (span-length ratio of 0.385), an air-augmentation ratio of 3, a maximum lift-drag ratio of 3.6, and a trajectory path of pressure times Mach number = 0.1 atmosphere. For a staging Mach number of 10, the estimated payload fraction for the second stage using air-augmented propulsion was $1\frac{1}{2}$ times better than using pure rocket power. For the scramjet case on the same trajectory, the best volume-surface parameter was 0.15 (higher L/D), but the payload fraction was only 14 percent of that for the air-augmentation case.

Thus, a rocket-scramjet propulsion system, whether hybrid or not, was found to promise payload capabilities superior to those attainable with a rocket or scramjet alone (on the same trajectory). Achieving this promise requires an integrated consideration of the vehicle geometry, trajectory, and powerplant. These results are, of course, dependent on the assumptions and simplified analysis used to identify some of the major problems and trade-offs involved. Also, other important factors, such as development cost, risk, availability, and complexity must, in the end, be considered.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, May 24, 1968,
789-30-01-01-22.
APPENDIX A

OPTIMUM ASCENT BY CALCULUS OF VARIATIONS

The equation of motion along the flight path (see sketch (a))

\[
T - D - W \sin \theta = \frac{W}{g} \frac{dV_o}{dt}
\]

Rearranging the equation gives

\[
\frac{\sin \theta + \frac{1}{W}}{g} \frac{dV_o}{dt} = \frac{T}{W} = \frac{1 - D}{T}
\]

However, the thrust is given by

\[
T = \frac{dW}{dt} I_s
\]

Hence, the previous equation can be put in the form

\[
\frac{dW}{W} = \frac{1}{I_s} \frac{\sin \theta \ dt + \frac{dV_o}{g}}{1 - \frac{D}{T}}
\] (A1)
As shown in sketch (b), $h = V_o \sin \theta$ or

$$\sin \theta \, dt = \frac{\dot{h}}{V_o} \, dt = \frac{dh}{V_o}$$

Substituting the previous equation into equation (A1) yields

$$\frac{dW}{W} = \frac{1}{I_s V_o} \left( 1 + \frac{1}{2g} \frac{dV_o^2}{dh} \right) \frac{dh}{T} \left( 1 - \frac{D}{T} \right)$$

$$\int_0^f \frac{dW}{W} = \ln \frac{W_f}{W_0} = \int \frac{1}{I_s V_o} \left( 1 + \frac{1}{2g} \frac{dV_o^2}{dh} \right) \frac{dh}{T} \left( 1 - \frac{D}{T} \right)$$

Hence, to minimize fuel consumption, the following integral has to be maximized:

$$\int \frac{1}{I_s V_o} \left( 1 + \frac{1}{2g} \frac{dV_o^2}{dh} \right) \frac{dh}{T} \left( 1 - \frac{D}{T} \right) = \int F \left( h, V_o^2, \frac{dV_o^2}{dh} \right) dh$$

The Euler-Lagrange equation is
Hence, the left side is

\[
\frac{d}{dh} \left( \frac{1}{2g} \frac{\partial F}{\partial V_o^2} \right) = \frac{\partial}{\partial h} \left[ \frac{1}{2g} \frac{1}{V_o I_s \left( 1 - \frac{D}{T} \right)} \right] + \frac{\partial}{\partial V_o^2} \left[ \frac{1}{2g} \frac{1}{V_o I_s \left( 1 - \frac{D}{T} \right)} \right] \frac{dV_o^2}{dh}
\]

and the right side is

\[
\frac{\partial}{\partial V_o^2} \left[ \frac{1}{V_o I_s \left( 1 - \frac{D}{T} \right)} \right] + \frac{\partial}{\partial V_o^2} \left[ \frac{1}{2g} \frac{dV_o^2}{dh} \right]
\]

Equating and simplifying give

\[
\frac{\partial}{\partial h} \left[ V_o I_s \left( 1 - \frac{D}{T} \right) \right] = \frac{2g \partial}{\partial V_o^2} \left[ V_o I_s \left( 1 - \frac{D}{T} \right) \right]
\] (A2)

Thus, the function to be examined can be expressed in terms of airplane and propulsion parameters

\[
\frac{V_o I_s}{T} (T - D) = \frac{V_o (T - D)}{m_1 \frac{f}{a}}
\] (A3)

for the general case of a scramjet since
\[ T = m_1 \frac{f}{a} I_s \]

The overall efficiency of the powerplant, which is relatively constant in the hypersonic region, is defined as

\[ \eta_{oa} = \frac{TV_o}{fQJ} \]

and thus

\[ \frac{TV_o}{m_1} = \eta_{oa} \frac{f}{a} JQ \]

If Newtonian flow is assumed, the aerodynamic lift coefficient is

\[ C_L \sim C_N = 2 \sin^2 \alpha \approx \frac{W(1 - \overline{V^2})}{qS} \]

from which

\[ \alpha \approx \sqrt{\frac{W(1 - \overline{V^2})}{2qS}} \]

and further, the drag due to lift coefficient is

\[ (C_D)_l = 2 \sin^3 \alpha \approx 2 \alpha^3 \approx 2 \left[ \frac{W(1 - \overline{V^2})}{2qS} \right]^{3/2} \]

So the total drag including friction is

\[ D = D_l + D_f = \left[ \frac{W(1 - \overline{V^2})}{\rho V_o^2 S} \right]^{3/2} + \frac{(C_D)_f}{2} \rho V_o^2 S \]

Since \( m_1 = \rho V_o A_{oo} \).
Differentiating the previous equation with respect to $h$ and $V_o^2$ yields

$$\frac{\partial}{\partial h} \left( \frac{(T - D)V_o}{m_1} \right) = \frac{3}{2} \frac{\left[ W(1 - \overline{V}^2) \right]^{3/2}}{\rho^{5/2}S^{1/2}V_{cA_{oo}}} \frac{d\rho}{dh} \quad (A4)$$

$$\frac{\partial}{\partial V_o^2} \left( \frac{(T - D)V_o}{m_1} \right) = \frac{3}{2} \frac{W^{3/2}(1 - \overline{V}^2)^{1/2}}{V_o^3 \rho^{3/2}S^{3/2}V_{cA_{oo}}} + \frac{1}{2} \frac{W^{3/2}(1 - \overline{V}^2)^{3/2}}{\rho^{3/2}S^{1/2}A_{oo}V_o^3} \frac{dV_o^2}{dh} \quad (A5)$$

Substituting equations (A3), (A4), and (A5) into (A2) yields

$$\frac{d\rho}{dh} = \left( \frac{2g}{V_r^2 - V_o^2} + \frac{2g}{3V_o^2} \right) \rho - \frac{2g}{3} \frac{(CD)_f V_o S^{3/2}}{[W(1 - \overline{V}^2)]^{3/2}} \rho^{5/2} \quad (A6)$$

The form of equation (A6) is

$$\frac{d\rho}{dh} = a_1 \rho - b_1 \rho^{5/2} \quad (A7)$$

Using $\rho = X^2$ and $d\rho = 2X \, dX$ gives

$$\frac{dX}{X(a_1 - b_1 X^3)} = \frac{dh}{2} \quad (A8)$$

which can be integrated directly to give

$$X^3 = \frac{a_1}{b_1 + K_1^3 \exp \left( \frac{-\frac{3}{2} a_1 h}{a_1} \right)} \quad (A9)$$
Returning to the original notation of equation (A6) gives

\[ a_1 = \frac{2g}{3v_0^2} \left( \frac{3V^2}{1 - V^2} + 1 \right) \]

\[ b_1 = \frac{2g}{3} \frac{(CD)_f S^{3/2}V_0}{W(1 - V^2)^{3/2}} \]

and equation (A9) becomes

\[ \rho^{3/2} = \frac{\left( \frac{3V^2}{1 - V^2} + 1 \right)}{(CD)_f S^{3/2}V_0^3 + \frac{K_1}{2g} 3V_0^2 \exp \left[ - \frac{3h}{2} \frac{2g}{3V_0^2} \left( \frac{3V^2}{1 - V^2} + 1 \right) \right]} \]

\[ 1 = \frac{\left( \frac{3V^2}{1 - V^2} + 1 \right)}{2D_f + \frac{3K_1}{2g} \rho^{3/2}V_0^2 \exp \left[ - \frac{hg}{V_0^2} \frac{3V^2}{1 - V^2} + 1 \right]} \]

For the initial part of the trajectory, \( D_f \ll D_i \) \( (D_i \approx 12D_f \text{ at } M_S = 10) \), and therefore

\[ \frac{3}{2g}K_2 \rho^{3/2}V_0^2 = K_3 \exp \left( \frac{hg}{V_0^2} \frac{K_3}{V_0^2} \right) \]

(A11)

where

\[ K_3 = \frac{3V^2}{1 - V^2} + 1 \]
For the region of the trajectory where \( \frac{h_g}{V_o^2} \ll 1 \) and \( \overline{V}^2 \ll 1 \),

\[
\rho^{3/2}V_o^2 = \text{Constant}
\]

or

\[
pM^{4/3} = \text{Constant}
\]

In the region near \( \overline{V} \sim 1.0 \), \( D_f \approx D_i \) and the constant increases greatly, which means the trajectory would be required to dip steeply to lower altitudes or higher pressure. Actually, it is indeterminate at the \( \overline{V} = 1.0 \) condition. The associated power requirements preclude the adoption of such a trajectory. Therefore, in the interest of simplicity, the path or trajectory defined by a constant product of atmospheric pressure and flight Mach number \( (pM = \text{Constant}) \) has been used from the staging point to the zoom velocity.

This assumption has the advantage of giving nearly constant mass flow rate \( (m_1 \propto \sqrt{t_0}, \text{ where } t_0 \text{ is the ambient temperature}) \) and fuel flow rate since the fuel-air ratio is also a constant. A comparison of paths characterized by constant \( pM \), \( pM^{4/3} \), or the commonly used constant dynamic pressure \( pM^2 \) is given in figure 3.
APPENDIX B

PULLUP MANEUVER

The flight path sequence between the attainment of zoom velocity and orbit conditions is depicted in sketch (c). The velocity at the end of the acceleration flight path is denoted by \( V_z \). This is the speed with which the pullup is initiated. During the pullup the equations of motion neglecting gravity are

\[
L = mV^2 \frac{d\theta}{dS}
\]

\[-D = m \frac{dV}{dt}\]

Dividing one equation by the other and noting that \( \frac{dS}{dt} = V \) give

\[
\frac{L}{D} \frac{dV}{V} = -d\theta
\]

Integration of this equation yields

\[
V_p = V_z \exp\left(-\frac{\theta}{L/D}\right) \tag{B1}
\]

Subsequent to the pullup, the equation of motion is

\[
h = -g
\]

Integration gives

28
\[ \dot{h} = V_z \exp\left(-\frac{\theta}{L/D}\right) \sin \theta - gt \]

and

\[ h = V_z \exp\left(-\frac{\theta}{L/D}\right)t \sin \theta - \frac{1}{2} gt^2 \]

At the peak of the free flight, \( h = 0 \); hence,

\[ t = \frac{V_z \exp\left(-\frac{\theta}{L/D}\right) \sin \theta}{g} \]

Substituting this expression for \( t \) in the expression for \( h \) yields

\[ h = \frac{1}{2} \left[ \frac{V_z \exp\left(-\frac{\theta}{L/D}\right) \sin \theta}{g} \right]^2 \]

At the peak,

\[ V_z \exp\left(-\frac{\theta}{L/D}\right) \cos \theta = V_r \quad (B2) \]

Hence,

\[ h = \frac{1}{2} \left( \frac{V_r \tan \theta}{g} \right)^2 \]

Thus,

\[ \tan^2 \theta = \frac{2gh}{V_r^2} \quad (B3) \]

The following can be substituted into equations (B1) to (B3): \( h = 100 \) miles \((1.6 \times 10^5 \) m\), \( V_r = 26000 \) ft/sec \((7925 \) m/sec\), \( \theta = 12.6^\circ \), and relations are obtained for \( V_p \) and \( V_z \) as functions of \( L/D \).
APPENDIX C

HYBRIDIZATION OF ENGINES

The hybridization of powerplants leads in principle to an advantage over their separate, though simultaneous, use. In the case of a hybrid, the total enthalpy of the mixed flow is the sum of the total enthalpies of the rocket flow and the airflow, which, per unit mass of mixed flow, becomes

\[ \frac{m(h_1^0 + Q) + h_j^0}{m + 1} \]

and the gross jet thrust obtained per unit mass of rocket flow is proportional to

\[ (m + 1) \sqrt{\frac{m(h_1^0 + Q) + h_j^0}{m + 1}} \]

In the case of a two-ducted system, the gross jet thrust of the scramjet is proportional to

\[ m_1 \sqrt{h_1^0 + Q} \]

and the gross jet thrust of the rocket is

\[ m_j \sqrt{h_j^0} \]

The total thrust of the two systems mounted together per unit mass of rocket flow is

\[ \frac{\bar{m}}{m} \sqrt{h_1^0 + Q} \quad \frac{\sqrt{h_j^0}}{m} \]

If it is assumed that the hybrid is superior, the following inequality is obtained:

\[ (m + 1) \sqrt{\frac{m(h_1^0 + Q) + h_j^0}{m + 1}} \geq \frac{\bar{m}}{m} \sqrt{h_1^0 + Q} + \sqrt{h_j^0} \]

Squaring both sides yields
\[(\bar{m} + 1)[\bar{m}(h_1^0 + Q) + h_j^0] \geq \bar{m}^2(h_1^0 + Q) + h_j^0 + 2\bar{m}\sqrt{h_1^0 + Q}\sqrt{h_j^0}\]

Simplifying gives

\[h_1^0 + Q + h_j^0 \geq 2\sqrt{h_1^0 + Q}\sqrt{h_j^0}\]

or

\[\left(\sqrt{h_1^0 + Q} - \sqrt{h_j^0}\right)^2 \geq 0\]

Since the square of a number is always positive, the inequality is always satisfied. Hence, the hybrid system is, in general, superior to the separate system. However, at very high velocities the sum \(h_1^0 + Q\), which is total enthalpy of inducted air plus heat added to it, becomes not much different in magnitude from the total enthalpy of the rocket \(h_j^0\). The inequality turns into the equality. Thus, in the hypersonic region of flight the benefits of hybridization are largely lost.
APPENDIX D

AERODYNAMIC CHARACTERISTICS OF THE VEHICLE

Expressions for the lift, drag, and lift-drag ratio are derived for a somewhat simplified model of the prismatic-delta vehicle illustrated in figure 2. A further simplification is the use of Newtonian impact theory.

The vehicle shown in sketch (d) in horizontal flight is at an angle of attack designated...
as $2\psi_0$ and has a wedge angle at the centerline of $2\psi$, both defined in the vertical plane of symmetry.

The starboard-rear view of the vehicle is shown in sketch (e). The configuration outline is shown by the bold lines, whereas the effective configuration simulated at angle of attach is shown by the light outline. Because the bottom lifting surface is skewed to the coordinate system, the usual Newtonian parameters must be converted to the coordinate axes.

Section A-A of sketch (e) taken normal to the bottom impact surface simulates a flat plate at angle of attack $\alpha$ and the usual Newtonian relations apply and are listed as follows:

$$C_N = 2 \sin^2 \alpha = C_p$$

$$C_L = 2 \sin^2 \alpha \cos \alpha$$

$$(C_D)_1 = 2 \sin^3 \alpha$$

Using the similarity of triangles at the base of the vehicle (see sketch (f))

\[ \cos \delta = \frac{d_o}{2c\psi_o} = \frac{b \psi_o}{2 \psi} = \frac{1}{\sqrt{\left(\frac{b}{2\psi}\right)^2 + (2c\psi_o)^2}} = \frac{1}{\sqrt{1 + \left(\frac{4\psi}{b/c}\right)^2}} \quad (D1) \]

But,

\[ \cos \delta = \frac{S_o}{A_o} \frac{\cos(2\psi_0 - 2\psi)}{\cos 2\psi} \quad (D2) \]
where the planform area of the effective wedge is

\[ S_0 = b \frac{\psi_o}{2} \frac{c}{\psi \cos(2\psi_o - 2\psi)} \]

The bottom surface area of the effective wedge is

\[ A_o = \frac{c}{\cos 2\psi} \sqrt{\left( \frac{b \psi_o}{2} \right)^2 + (2c\psi_o)^2} \]

Resolving the normal force coefficient to the vertical plane (see sketch (g))

\[ (C_N)_Z = C_N \cos \delta = C_N \frac{S_0}{A_o} \frac{\cos(2\psi_o - 2\psi)}{\cos 2\psi} \]

Converting to the lift coefficient in the vertical plane yields

\[ C_L = (C_N)_Z \cos 2\psi_o = C_N \frac{S_0}{A_o} \frac{\cos 2\psi_o \cos(2\psi_o - 2\psi)}{\cos 2\psi} \]

or, approximately,

\[ C_L \approx C_N \frac{S_0}{A_o} \tag{D3} \]

Now
\[
\frac{L}{q} = C_N A \frac{S_0}{A_0} = C_N A \frac{S_0}{S}
\]

and since

\[
\frac{S_0}{S} = \frac{(b)^2 \psi_0 \frac{c}{\psi \cos(2\psi_0 - 2\psi)}}{bc} = \frac{\psi_0}{\psi \cos(2\psi_0 - 2\psi)} \approx \frac{\psi_0}{\psi}
\]

and

\[
\frac{A_0}{A} = \frac{\sqrt{\left(\frac{b}{2} \frac{\psi_0}{\psi}\right)^2 + \left(2c\psi_0\right)^2 \frac{c}{\cos 2\psi}}}{\psi} \approx \frac{\psi_0}{\psi}
\]

due to lift can now be evaluated:

\[
(C_D)_l = C_L \tan 2\psi_0
\]

\[
\tan 2\psi_0 = 2\psi_0 \cos(2\psi_0 - 2\psi)
\]

Then using \(C_L\) from equation (D3) gives

\[
(C_D)_l \approx C_N \frac{S_0}{A_0} 2\psi_0
\]

and, approximately,
In order to evaluate the friction drag the outside area is found (see sketch (h)):

\[
\frac{D_i}{q} = C_N \frac{A}{A_0} \frac{S_0}{S} S 2\psi_0 \approx C_N 2\psi_0 S
\]  \hspace{1cm} (D5)

\[
c \tan \varphi = d = \frac{2\psi c (b/2)}{\sqrt{(2\psi c)^2 + (b/2)^2}}
\]

\[
\tan \varphi = \frac{\psi b}{\sqrt{(2\psi c)^2 + (b/2)^2}}
\]

\[
\frac{1}{\cos^2 \varphi} = \frac{(2\psi c)^2 + (b/2)^2 + (\psi b)^2}{(2\psi c)^2 + (b/2)^2}
\]

Hence, the under surface of the wedge is

\[
\frac{A}{2} = \frac{c}{2} \sqrt{\left(\frac{b}{2}\right)^2 + (2\psi c)^2} \sqrt{\frac{(2\psi c)^2 + (b/2)^2 + (\psi b)^2}{(2\psi c)^2 + (b/2)^2}}
\]

Since

36
\[ S = \frac{1}{2} \frac{bc}{2} \]

Therefore

\[ \frac{A}{S} = \sqrt{1 + (2\psi)^2 + \left(\frac{4\psi}{b/c}\right)^2} \]

Now the friction drag to dynamic pressure ratio will be

\[ \frac{D_f}{q} = C_f (A + S) = C_f \left( A + \frac{1}{S} \right) S = C_f \frac{bc}{2} \left[ 1 + \sqrt{1 + (2\psi)^2 + \left(\frac{4\psi}{b/c}\right)^2} \right] \]

The lift-drag ratio can be written

\[ \frac{L}{D} = \frac{C_N S}{C_N S 2\psi_0 + C_f S \left[ 1 + \sqrt{1 + (2\psi)^2 + \left(\frac{4\psi}{b/c}\right)^2} \right]} \]

and from previous geometry

\[ \frac{C_N}{2} = \sin^2 \alpha = \frac{1}{1 + \frac{\left(\frac{4\psi}{b/c}\right)^2}{4\psi_0^2}} \]

The lift-drag ratio in terms of \( \psi_0, \psi, b/c, \) and \( C_f \) is then, when \( \psi_0 > \psi,\)

\[ \frac{L}{D} = \frac{1}{2\psi_0 + \frac{C_f}{2} \left[ 1 + \sqrt{1 + (2\psi)^2 + \left(\frac{4\psi}{b/c}\right)^2} \right] \left[ 1 + \left[ \frac{1 + \left(\frac{4\psi}{b/c}\right)^2}{4\psi_0^2} \right] \right]} \]
For the prismatic-delta vehicle, the following relation derived from geometry is helpful in relating the basic vehicle parameters for calculations:

$$\frac{V^{2/3}}{S} \left( \frac{b}{c} \right)^{1/3} = \left( \frac{2\sqrt{2}}{3} \psi \right)^{2/3}$$

(D9)

**Condition for Zero Lift and Angles of Attack Smaller Than Wedge Angle**

Aerodynamic lift diminishes as the vehicle accelerates along the climb path and centrifugal lift increases (see fig. 4). Consequently, the angle of attack decreases as orbital speed is approached. For the condition where the angle of attack is less than the wedge angle (see sketch (i)), the negative lift and drag due to lift from the top surface must be found.

![Diagram](image)

For the top surface,

$$\left( \frac{L}{q} \right)_{\text{top}} = 2S \sin^2(\psi - \psi_o) \cos(\psi - \psi_o) = 8S \sin^2(\psi - \psi_o) \cos^2(\psi - \psi_o) \cos(\psi - \psi_o)$$

(D10)

$$\left( \frac{D_1}{q} \right)_{\text{top}} = 2S \sin^3(\psi - \psi_o) = 16S \sin^3(\psi - \psi_o) \cos^3(\psi - \psi_o)$$

(D11)

then
\[
\frac{L}{D} = \frac{L_{\text{bottom}} - L_{\text{top}}}{(D_t)_{\text{bottom}} + (D_t)_{\text{top}} + D_I}
\]

\[
\frac{L}{D} = \frac{C_N - 8 \sin^2(\psi - \psi_o) \cos^2(\psi - \psi_o) \cos 2(\psi - \psi_o)}{2\psi_o C_N \cos(\psi - \psi_o) + 16 \sin^3(\psi - \psi_o) \cos^3(\psi - \psi_o) + C_f \left[ 1 + \sqrt{1 + (2\psi)^2 + \left(\frac{4\psi}{b/c}\right)^2} \right]}
\]

\text{(D12)}

or

\[
\frac{L}{D} \approx \frac{C_N - 8 \sin^2(\psi - \psi_o)}{2\psi_o C_N + 16 \sin^3(\psi - \psi_o) + C_f (1 + A/S)}
\]

\text{(D13)}

\((C_N \text{ is found from eq. (D7).})\)

For the condition of zero net lift,

\[
C_N = 8 \sin^2(\psi - \psi_o)
\]

\text{(D14)}

This equation is solved for angle of attack which is given the special designation \(2\psi_{oo}\).

\textbf{Determination of Capture Area}

An arbitrary rule adopted for sizing the capture area was to limit the span of the inlet area at the condition for zero lift. Hence, at the effective wedge angle giving equal lift on the top and bottom, only the trailing edge of the base remaining below the horizontal was assumed to have inlet capture area (see sketch (j)). The stream tube area is equivalent to the base area below the horizontal axis. This procedure restricts the engine size but avoids the problems of hybrid engine installation near the tip regions, such as the required mixing length and the nozzle-base integration problems.
Alternate assumptions were not evaluated. However, some trade-off between propellant consumption and engine weight due to engine size selection might occur. Thus, a larger size would provide greater thrust during the acceleration phase at the expense of spillage drag (or variable geometry) near the zero lift or orbital condition.
APPENDIX E

STRUCTURAL ANALYSIS

The structural configuration chosen features as a primary element a statically determinate, pin-jointed multiple truss arrangement protected by an insulating skin system. One truss of this scheme is depicted in sketch (k).

The truss load is derived by assuming it to be the main supporting structure for the aerodynamic load occurring on a prism of $2\psi$ wedge angle and $2\epsilon$ vertex angle as shown in sketch (l). Thus, the whole vehicle includes a series of trusses joined at the vertex and separated by an angle $2\epsilon$, which may vary from truss to truss.
With a uniform pressure $p$ at the bottom surface of the prismatic wedge, the moment of the forces about a point $N$, at the distance $X$ from the vertex is

$$M = p \frac{1}{2} (2\varepsilon X)X \frac{X}{3} = p\varepsilon \frac{X^3}{3}$$

The truss consisting of two bays, as shown in sketch (m), is now considered.

![Sketch (m)](image)

Its length is approximately

$$a + 2\psi a \tan \sigma = a(1 + 2\psi \tan \sigma)$$

The length of a truss consisting of three bays (see sketch (n))

![Sketch (n)](image)

will be

$$a(1 + 2\psi \tan \sigma) + a(1 + 2\psi \tan \sigma)2\psi \tan \sigma = a(1 + 2\psi \tan \sigma)^2$$

Thus, a truss of $n$ bays will have the length

$$a(1 + 2\psi \tan \sigma)^{n-1}$$
Next is considered the \((n+1)^{th}\) bay with the loading on the truss as shown in sketch (o) where the compression member \(C\), the tension member \(T\), the diagonal member \(D\), and the strut \(S\) are identified.

Cutting a section through the bay and taking moments about the point \(N\) give

\[
p\varepsilon \frac{X^3}{3} = (X + 2\psi X \tan \sigma)2\psi c
\]

or

\[
p\varepsilon \frac{X^2}{3} = (1 + 2\psi \tan \sigma)2\psi c
\]

But \(C = A_c f_c\); hence,

\[
A_c = p\varepsilon \frac{X^2}{3} \frac{1}{(1 + 2\psi \tan \sigma)2\psi} \frac{1}{(f_c)_1}
\]

and the volume of the member \(C\) is

\[
p a^3 \varepsilon \frac{(1 + 2\psi \tan \sigma)^{3n-4}}{(f_c)_1} \tan \sigma
\]

Summing up the volume of all the elements like \(C\) gives
Considering, in turn, the member $S$ and its load gives

$$S = p \frac{1}{2} (2\varepsilon X) = p\varepsilon X^2$$

Then the volume of the member $S$ is

$$p \frac{2X^3\varepsilon\psi}{(f_c)_2}$$

The summation of the volume of all the struts $S$ is

$$p \frac{2\varepsilon\psi}{(f_c)_2} \sum X^3 = p \frac{2\varepsilon\psi}{(f_c)_2} \sum_{j=1}^{n} a^3 (1 + 2\psi \tan \sigma)^{3(j-1)}$$

$$= p \frac{2\varepsilon\psi a^3}{(f_c)_2} \frac{(1 + 2\psi \tan \sigma)^{3n} - 1}{(1 + 2\psi \tan \sigma)^3 - 1} \sim \frac{1}{3} p\varepsilon \frac{c^3}{(f_c)_2 \tan \sigma}$$

The load for the diagonal $D$ is given by

$$D = \frac{pX^2\varepsilon}{\cos \sigma}$$

Hence,

$$A_D = \frac{pX^2\varepsilon}{f_t \cos \sigma}$$

Thus, the volume of all the elements like $D$ is given by
$$a^3 \frac{p e}{f_t \cos^2 \sigma} \frac{(1 + 2\psi \tan \sigma)^{3n} - 1}{(1 + 2\psi \tan \sigma)^3 - 1} \sim p \frac{1}{3} \frac{e c^3}{f_t \cos \sigma \sin \sigma}$$

Summing the volume of all the structural material in the truss, the following is obtained:

$$\frac{p e c^3}{3} \left[ \frac{1}{3\psi f_c_1} + \frac{1}{(f_c_2 \tan \sigma)} + \frac{1}{f_t \cos \sigma \sin \sigma} \right]$$

where the volume of $T$ has been assumed to be equal to that of $C$. Volume enclosed by the prism associated with the truss is $(4/3)e \psi c^3$.

Hence, the volume of the structural material per unit volume of the vehicle is

$$\frac{p}{4\psi} \left[ \frac{1}{3\psi f_c_1} + \frac{1}{(f_c_2 \tan \sigma)} + \frac{1}{f_t \cos \sigma \sin \sigma} \right]$$

Minimization of this quantity with respect to $\sigma$ reveals $\sigma \sim 45^0$.

Then, $p = 2(2\psi)^2 q$ is substituted for the condition of maximum $L/D$ during the pullup into orbit, and $\rho_{ss}$ is defined as the density of stainless steel. The trusswork weight per unit volume is

$$\rho_T = K \frac{\rho_{ss}^2 (2\psi)^2 q}{4\psi} \left[ \frac{1}{3\psi f_c_1} + \frac{1}{(f_c_2 \tan \sigma)} + \frac{2}{f_t} \right]$$

Selecting values of $(f_c)_1$, $(f_c)_2$, and $f_t$ for stainless steel at mild temperatures and an appropriate constant $K$ results in the variation of trusswork density with dynamic pressure as shown in figure 13. Instability or buckling of the thin-walled structural members was not considered in this initial analysis; however, some adjustment of the estimated trusswork density can be provided by the constant $K$. 

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APPENDIX F

CALCULATION PROCEDURE

The following dependent parameters are selected:

1. A range of values of $\frac{V^{2/3}}{S}$ with $b/c$ corresponding to maximum $L/D$ as given in figure 9 (p. 12)
2. A flight path constant such as $pM = 0.1$ atmosphere
3. A range of air-augmentation ratios $\bar{m}$
4. A range of volumes $V$ or initial weight $W_0$

The associated wedge angle $2\psi$ can be found by using equation (D9). The angle of attack for zero net lift $2\psi_{oo}$ is found from

$$L = \left[C_N - 8 \sin^2(\psi - \psi_o)\right]qS$$  \hspace{1cm} (F1)

The capture area determined at the zero lift condition as defined in appendix D is

$$A_{oo} = \frac{V^{2/3}}{S} \left(\frac{2\psi_{oo}}{2\psi}\right)^2 \frac{2\psi}{V^{2/3}}$$  \hspace{1cm} (F2)

The ratio of tail or afterbody to front or wedge volume for the $45^0$ boattail angle shown in figure 2 (p. 4) is equal to $2\psi$; hence, the total volume is

$$V_T = (1 + 2\psi)V$$  \hspace{1cm} (F3)

Now the empty stage weight $W_S$ can be found since the densities and unit weights are known for

$$W_S = K_4 \left[V_T(\rho_T + \rho_t) + A_{oo}\rho_E + 0.37 \frac{A_{oo}pM_o}{\bar{m}} + S_o\rho_s\right]$$  \hspace{1cm} (F4a)

or for the pure rocket

$$W_S = K_4 \left[V_T(\rho_T + \rho_t) + 7.9 \ m_j + S_o\rho_s\right]$$  \hspace{1cm} (F4b)
where

- $\rho_T$ weight of trusswork per unit enclosed volume (see fig. 13)
- $\rho_t$ tanks, 1.0 lb/ft$^3$ (16 kg/m$^3$)
- $\rho_E$ secondary component weight, 50 lb/ft$^2$ (244 kg/m$^2$)
- $\rho_s$ skin, function of temperature (see fig. 12) on a given path
- $K_4$ adjusting constant (landing equipment, etc.), usually 1.2

The propellant density of hydrogen and oxygen for the rocket and hydrogen for the air is combined in a single number according to the air-augmentation ratio as listed in table II. The total propellant weight is initially estimated as

$$W_F = \rho_f V \quad \text{(F5)}$$

This corresponds to filling the front prismatic wedge with propellant and having the tail or afterbody volume or its equivalent for other equipment, etc.

When the payload $P$ is specified, the stage weight can be found:

$$W_o = W_S + W_F + P \quad \text{(F6)}$$

Now, the performance along the flight path can be determined starting at the staging Mach number. The required lift coefficient (approximate normal force) is

$$C_N = 2 \cdot \sin^2 \alpha = \frac{W_o(1 - \bar{V}^2)}{qS} \quad \text{(F7)}$$

If $\psi_o \leq \psi$, equation (F1) is used.

Equation (D7) is used to find the angle of attack $2\psi_o$ and then the instantaneous L/D is found from equation (D8) when $\psi_o > \psi$ or equation (D13) if $\psi_o \leq \psi$.

The thrust can be found since the impulse $I_s$ is a function of $V_o$, $\bar{m}$, and $m_1$, and the fuel-air ratio is 0.029.

$$T = m_1 \left(1 + \frac{f}{a}\right) I_s \quad \text{(F8)}$$

The change in weight with velocity is given by equation (F9), which is numerically integrated between the desired velocity limits following the procedure outlined previously:
The final velocity is determined by the maximum $L/D$ (see fig. 5, p. 7). The total propellant weight determined by equation (F9) is compared with the estimate of equation (F5), the estimate is revised, and the calculation repeated until the iteration converges. An optimum payload to initial weight ratio $P/W_0$ can be found by calculating a range of volumes.

\[
\frac{dW}{dV_0} = -\frac{W}{g\bar{n}_s} \frac{1 + \frac{g}{V_0} \frac{dh}{dV_o}}{1 - \frac{D}{T}}
\] (F9)
APPENDIX G

SYMBOLS

A  bottom surface area of wedge
A_o  bottom surface area of simulated wedge
A_oo  capture area of engine (established at angle of attach for L/D = 0, see appendix D)
A_oo/V^2/3  capture area-volume parameter
a  length of body in truss
a_1  mathematical constant
b  vehicle span
b/c  span to length ratio
b_1  mathematical constant
(C_D)_f  friction drag coefficient
(C_D)_i  drag due to lift coefficient
C_F  specific thrust coefficient, T/qA_oo
C_f  coefficient of friction
C_L  lift coefficient
C_N  normal force coefficient
(C_N)_Z  normal force coefficient in Z-direction
c  vehicle length or chord (front wedge)
D  drag
D_f  friction drag
D_i  drag due to lift
d  length
F  function
f  fuel flow rate, lb mass/sec; kg/sec
f/a  fuel-air ratio
(f_c)_1  allowable compressive stress
(f_c)_2  allowable compressive stress
f_t  allowable tensile stress
g  acceleration due to gravity
h  altitude
h^o_j  stagnation enthalpy of rocket flow
h^o_m  stagnation enthalpy of mixed flow
h^o_l  stagnation enthalpy of air flow (secondary)
I_s  specific impulse, sec
J  mechanical equivalent of heat, 778 ft-lb/Btu; 4186 J/kg-cal
K, K_1, 2, 3, 4  mathematical constants
L  lift
L/D  lift-drag ratio
M, M_o  flight Mach numbers
M_s  staging Mach number
m  vehicle mass
(m)  air-augmentation ratio,
air flow = m_1
rocket flow = m_j

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rocket flow, lb mass/sec; kg/sec

secondary flow (air),
   lb mass/sec; kg/sec

payload

pressure

heat added, Btu/lb air;
   cal/kg

dynamic pressure, \((\gamma/2)\) \(pM^2\)

planform area of vehicle

planform area of simulated wedge

total outside area of vehicle
   (skin), \(S + A\)

skin area-volume parameter

path distance

vehicle thrust

time, sec

thickness

volume

ratio of flight to orbital velocity, \(V_o/V_r\)

flight velocity

velocity at end of pullup

orbital velocity,
   26 000 ft/sec; 7925 m/sec

total volume of vehicle

zoom velocity (at end of powered path and at beginning of pullup)

jet velocity

volume-surface parameter

instantaneous vehicle weight

propellant weight

initial vehicle weight

skin weight

empty weight

distance

equivalent flat-plate angle of attack

shock wave angle

ratio of specific heats, \(C_p/C_v\)

angle between vertical and normal to wedge lower surface

vertex angle of vehicle

flight path angle

variable

atmospheric density

weight per unit capture area of airbreathing component

propellant density

unit skin weight

density of stainless steel

trusswork density

propellant tank density

angle between vertical and diagonal truss members

ture wedge angle

wedge angle of vehicle in vertical plane (fig. 2)
$2\psi_o$ vehicle angle of attack (fig. 2)  

$2\psi_{oo}$ vehicle angle of attack for zero lift
REFERENCES


