Winter 2007

Propulsion Enhancement Contributions to the Performance of Space Launch Vehicles

Russell H. Edwards  
Old Dominion University

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PROPULSION ENHANCEMENT CONTRIBUTIONS

TO THE PERFORMANCE OF

SPACE LAUNCH VEHICLES

by

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B. S., Physics, 1961, Carson-Newman College
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A Dissertation Submitted to the Faculty of
Old Dominion University in Partial Fulfillment of the
Requirements for the Degree of

DOCTOR OF PHILOSOPHY

MECHANICAL ENGINEERING

OLD DOMINION UNIVERSITY
December 2007

Approved by:

Dr. Gregory Selby (Director)

Dr. Jen-Kuang Huang

Dr. Brett Newman

Dr. Billie Reed

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A research effort has been undertaken to investigate critical aspects of launch vehicle performance as affected by variations in specific launch vehicle parameters. The major portion of the study involves liquid propellant systems. However, since solid propellant systems also play a role in today's launch systems, a representative solid-propellant launch vehicle has also been analyzed. The research undertaken determined that the payload capability of a space launch vehicle, or, conversely, the vehicle total liftoff mass, is highly sensitive to the manner in which the space launch vehicle is staged. The research has led to the development and programming of a model for determining optimum staging relationships for given mission requirements. The research then utilized this optimum staging algorithm as a means of computing the sensitivities of the vehicle’s payload and liftoff mass to variations in the vehicle’s key propulsion and related performance parameters. It has been seen that significant gains in payload weight can be achieved through modest to substantial changes in specific impulse and structure factor. As an example, for a four-stage solid propellant space launch vehicle, and using the above tables, a 33% gain in payload weight can be achieved by increasing specific impulse by only 5%. As a second example, for a two-stage liquid-propellant launch vehicle, and using the above tables, a 41% gain in payload weight can be achieved through an increase in specific impulse of 10%.
ACKNOWLEDGMENTS

My deepest gratitude is extended to the faculty and staff members of the Mechanical Engineering Department at Old Dominion University for their encouragement, guidance, and support during the course of my graduate studies.

I would like to express my deep and sincere gratitude to my advisor Dr. Gregory Selby, Professor of Mechanical Engineering, for his guidance, valuable suggestions, patience, support, and encouragement. I hope that our cordial relationship in both personal and professional fields will flourish further in the future. I also wish to express my sincere thanks to Dr. Jen-Kuang Huang, Professor and Chair of Mechanical Engineering, for his assistance during the course of my graduate studies. In addition to Drs. Selby and Huang, I also wish to extend my sincere thanks and appreciation to committee member, Dr. Billie Reed, of the Engineering Management Department, for his help and encouragement. Special thanks go to Dr. Brett Newman, of the Aerospace Engineering Department, for his wise counsel and assistance throughout my studies at ODU. I would also like to express my sincere gratitude to Dr. Surendra Tiwari, Professor of Mechanical Engineering and Eminent Scholar, for his encouragement and continuing support throughout the course of my graduate studies. Thanks and appreciation go to Mr. Keegan Morrison and Mr. Ian Gullett of the Computer Science Department, Old Dominion University, for their assistance in programming into C++ computer language the optimization equations which I have derived and which appear in Chapter VII. Very special thanks go to Mrs. Diane Mitchell for her excellent assistance in putting the manuscript in final form and for her continuing excellent support to the Department of Mechanical Engineering.
I also express my deepest appreciation and regards to my former associates with the National Aeronautics and Space Administration and various agencies of the Department of Defense for the knowledge and experience gained through many years of rewarding employment with these organizations. Thanks also go to the national research and development laboratories and aerospace corporations through which I have performed research and liaison work over many years of rewarding employment. Thanks also go to the faculty and administration of the Industrial College of the Armed Forces of the National Defense University for the excellent one-year fellowship accomplished with this organization. I would also like to express the deepest appreciation to my dear wife Faye for her patience and continuing support during the course of graduate studies.
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LIST OF SYMBOLS

Latin Symbols

\( A_b \) \hspace{1cm} \text{propellant burning surface area, in}^2

\( A_e \) \hspace{1cm} \text{nozzle exit area, in}^2

\( A_t \) \hspace{1cm} \text{nozzle throat area, in}^2

\( a \) \hspace{1cm} \text{constant}

\( c \) \hspace{1cm} \text{effective exhaust velocity, ft/sec}

\( C \) \hspace{1cm} \text{exhaust velocity, ft/sec}

\( C_D \) \hspace{1cm} \text{weight flow coefficient, sec}^{-1}

\( C^* \) \hspace{1cm} \text{characteristic velocity, ft/sec}

\( C_T \) \hspace{1cm} \text{thrust coefficient}

\( E \) \hspace{1cm} \text{energy per unit mass}

\( e \) \hspace{1cm} \text{eccentricity}

\( F \) \hspace{1cm} \text{thrust, lb}

\( g_0 \) \hspace{1cm} \text{acceleration due to gravity, 32.172 ft/sec}^2

\( h \) \hspace{1cm} \text{altitude in feet}

\( i \) \hspace{1cm} \text{orbit inclination, deg}

\( I_{sp} \) \hspace{1cm} \text{specific impulse, sec}

\( I_{sp} \rho \) \hspace{1cm} \text{density impulse}

\( I_t \) \hspace{1cm} \text{total impulse, lb-sec}

\( I_v \) \hspace{1cm} \text{jet vane impulse, lb-sec}
$I_2$ vacuum impulse, lb-sec

$I_3$ vacuum impulse in lb-sec without jet vanes attached

$I_4$ vacuum impulse in lb-sec with jet vanes attached

$J$ mechanical equivalent of heat, 778 ft-lb/BTU

$K$ structure factor

$K_1$ vehicle weight in lb at stage ignition

$K_2$ consumable weight in lbs in rocket motor (propellant pulse inhibitor, etc)

$K_3$ slop of control fuel curve in, lb/sec

$K_4$ ratio of jet vane drag to vacuum thrust with vanes attached

$K_5$ ratio of jet vane drag to vacuum thrust without vanes attached

$\bar{M}$ weight of payload, lb

$M_c$ average molecular weight of combustion products

$M_g$ gross weight, lb

$M_N$ Mach number, may be tabular vs. time

$M_o$ burnout weight, lb

$P_A$ ambient atmospheric pressure in lb/ft$^2$ as computed by ARDC 1959 tables

$P_c$ pressure in combustion chamber, lb/in$^2$

$P_e$ exit pressure, lb/in$^2$

$P_t$ pressure at nozzle throat, lb/in$^2$

$\bar{q}$ dynamic pressure, lb/ft$^2$

$R$ gas constant
$R_e$  radius of earth, $2.09029 \times 10^{17}$ ft

$R_{IW}$  total impulse to weight ratio

$r$  radius of earth plus desired orbit altitude, NM

$r_a$  apogee distance from center of earth, NM

$r_b$  solid propellant burning rate, in/sec

$r_p$  perigee distance from center of earth, NM

$r_w$  propellant mixture ratio, oxidizer/fuel, by weight

$S$  effective aerodynamic area in ft$^2$

$S_1$  chamber pressure integral

$t$  time, sec

$t_b$  burn time, sec

$t_m$  upper limit of integration in seconds

$\Delta t$  time below orbital velocity, sec

$T$  thrust, lb

$T_c$  combustion temperature, °F

$T_t$  temperature at nozzle throat, °F

$T_1$  thrust in lbs at ambient conditions with jet vanes attached, lb

$T_2$  thrust in lbs at vacuum conditions with jet vanes attached, lb

$T_3$  thrust in lbs at vacuum conditions without jet vanes attached, lb

$T_4$  thrust in lbs at vacuum conditions with jet vanes attached (calculated by interior ballistic method)

$T_5$  thrust in lbs at vacuum conditions without jet vanes attached (calculated by interior ballistic method)
$V_a$ velocity of rocket in ft/sec with respect to surrounding medium (which rotates with earth), equivalent to the velocity with respect to local land based radar, tabular vs time

$V_{bo}$ burnout velocity, ft/sec

$V_c$ circular orbit velocity, ft/sec

$V_e$ escape velocity, ft/sec

$V_i$ injection velocity, ft/sec

$V_{id}$ ideal velocity, ft/sec

$V_r$ total velocity requirement, ft/sec

$V_s$ local velocity of sound in ft/sec as computed from 1959 ARDC tables

$V^*$ equivalent velocity, ft/sec

$\dot{w}$ rate of propellant flow, lb/sec

$W_{bo}$ burnout weight, lb

$W_e$ empty weight of a given stage; $W_e = W_f$ minus propellant expended at end of stage burn, lb

$W_{am}$ W minus weight of propellant expended in a given phase

$W_f$ system weight at beginning of stage burn

$W_g$ guidance weight, lb

$W_h$ hardware weight, lb

$W_i$ instrumentation weight, lb

$W_{om}$ initial weight at ignition of a given burn phase, lb

$W_p$ propellant weight, lb

$W_{pe}$ payload weight, lb

$W_1$ vehicle weight in lbs at any time $t$

$W_2$ consumable weight in lbs (propellant plus inhibitor, etc.) remaining in rocket motor at any time $t$, tabular vs time.

$W_t$ total weight, lb
Note: Subscripts 1, 2, 3, 4 indicate 1st, 2nd, 3rd and 4th stages.

Greek Symbols

$\Delta V_{loss}$  velocity loss due to gravity and aerodynamic drag, ft/sec

$\Delta V_{rot}$  velocity gain/loss due to earth's rotation, ft/sec

$\varepsilon$  expansion ratio

$\gamma$  ratio of specific heats

$\lambda$  latitude of orbit plane, deg

$\mu$  earth's gravitational constant, $1.4077 \times 10^{16}$ ft$^3$/sec$^2$

$\lambda_d$  nozzle divergence coefficient

$I_s \rho_B$  density impulse, i.e. theoretical specific impulse times bulk density.

$\eta$  combustion efficiency

$\rho$  ambient atmospheric density in slugs/ft$^3$ as computed from 1959 ARDC tables

$\rho_B$  bulk density of propellants, g/cm$^3$

$\rho_f$  density of fuel, g/cm$^3$

$\rho_o$  density of oxidizer, g/cm$^3$

$\rho_p$  solid propellant density, g/cm$^3$

$\pi_k$  solid propellant temperature sensitivity coefficient

$\psi_l$  azimuth angle measured from north in horizontal plan, deg

Subscripts

a  apogee

bo  burnout

e  empty

f  full
$L$  loss

$n$  stage number

$p$  perigee

Note:  1, 2, 3, 4 = 1$^{st}$, 2$^{nd}$, 3$^{rd}$ and 4$^{th}$ stages
CHAPTER I
INTRODUCTION

The time period for this dissertation finds the United States manned space program in recovery mode following a lengthy stand-down due to a catastrophic failure of the space shuttle in which the entire crew was lost. A high-level investigation panel subsequently made a study to determine the cause of the catastrophic failure and recommended changes to the launch vehicle before the launch program could resume. These problems placed the United States manned space program in considerable disarray for the interim when the United States relied on Soviet space launch capabilities to support launches to the International Space Station. The United States has previously played a central role in the development and operation of the International Space Station. Two recovery-mode test flights of the shuttle have since been successfully made. The author was involved in some earlier aspects of the International Space Station program.

Over the past several years, a considerable number of NASA-funded contractual studies have been accomplished with the objective of determining the physical characteristics that future United States manned space launch vehicles should display [1]. The rationale at NASA and in the overall United States space community is that the design resulting from these study efforts should constitute a highly capable follow-on successor to the space shuttle. It is a reasonable assumption that the space shuttle configuration, as we know it today, has reached such a state that its remaining years of extensive operation is somewhat limited [2].

* The numbers in brackets indicate references. The ASME and AIAA Journal format has been used in preparation of this report.
This dissertation investigates interactions between key space launch vehicle characteristics, principally those relating to propulsion, but also briefly includes structural considerations and staging relationships, and how they contribute, individually and collectively, to overall launch vehicle performance. Models are developed that are representative of two classes of vehicles: liquid propellant and solid propellant systems [3, 4, 5].

Several classes of liquid and solid rocket propellants are included. In some cases, the study utilizes parameters that are representative of higher energy propellants and high-performance structures, in order to be consistent with current national trends and those relating to futuristic space launch vehicles and concepts.

The combustion process is investigated in detail. It is shown that the efficiency of the combustion process is a key determinant of rocket-engine performance and thus overall launch-vehicle performance. Nozzle considerations are handled in a manner that minimizes performance losses for the atmospheric portion of flight. Aerodynamic and gravitational losses are included for the ascent portion of the trajectory.

The work is purposely limited to conventional methods of achieving launch to earth orbit. Its main intent is to show the very significant payoffs that can be achieved through future advances in propellant and rocket engine technology, and to a lesser extent, through improvement in structure technology. While the exploration and development of future space launch concepts and endeavors are to be fully encouraged, such as those funded through the NASA X-programs, it is felt that near-term gains in space launch-vehicle performance will most likely be achieved through advancements in current propulsion and, to a lesser extent, structure technologies.
The mathematical, physical, and graphical relationships utilized in this dissertation are from a broad array of sources. Some are from classical space mechanics, propulsion and specialized gas dynamic relationships, and relationships among chemical constituents of rocket propellants. In addition, a number of empirical equations, relationships, and graphical approaches are utilized that have been researched and developed by the author in connection with his work in support of this dissertation. Also included and utilized are optimization relations that the author derived for use in the analytical portions of this dissertation.

The Department of Defense is planning to develop smaller satellites that can quickly be launched into low-earth orbit by a new class of rockets that will use mobile launchers. Military planners and strategies are devoting attention and funding to developing these smaller satellites and a new class of streamlined rockets, which will require days or even a few hours of advance notice before liftoff. Today's mammoth boosters, by contrast, often entail scheduling years in advance and spending months preparing them for flight from elaborate fixed launch pads.

The current Air Force budget shows substantial funding for the development of smaller tactical launch vehicles beginning in the year 2006 and continuing through 2011. These operationally responsive space projects envision a fundamental reconsideration of how the military can use space assets in future conflicts. Local commanders, for instance, may have the option of deploying up diminutive satellites – perhaps weighing less than 1,000 pounds and therefore easily maneuvered over the battlefield – to provide targeted surveillance of nearby enemy movements over a critical period of a few weeks or months.
Instead of traditional space investments, a reshaped military must conceive that stress versatility and affordability, according to high-level officials. In addition to United States philosophy on this subject, the Franco-German European Aeronautic Defense and Space Company is proceeding to develop a small rocket that would insert a 400-pound satellite in earth orbit by 2008.

An industrial conference on the subject held in early 2005 in Los Angeles showed how far proponents have progressed in challenging traditional rocketry, satellite manufacturing, and even military tactics. It was speculated that the Air Force will ultimately established specialized squadrons, complete with their own intelligence and logistics officers, able to deploy rapidly anywhere in the world with a arsenal of small spacecraft.

The work accomplished herein concentrates on performance characteristics of smaller classes of launch vehicles such as those discussed above. The performance sensitivity coefficients that are derived herein are independent of vehicle size and are, therefore, applicable to broad categories of launch vehicles. As part of an initial baseline for comparisons and computations, certain characteristics of existing launch vehicles are shown. Some of the characteristics are representative of United States vehicles, while others are representative of foreign launch vehicles. The subsequent compare the findings of the research included in this dissertation with existing literature in the field and demonstrate the contributions made to the field.

An exhaustive search of scientific and technical literature was conducted in order to identify any existing literature relating to this dissertation. The list of references and supplementary sources that appear herein include the relevant findings of the literature
Although it was determined that a reasonable number of papers, reports, and other documents exist on the general topic of this dissertation, it was found that relatively few documents relate propulsion and structural considerations to overall launch-vehicle payload-to-orbit capabilities. Thus, this dissertation makes a detailed analysis of the effect of changes in payload-to-orbit capabilities of space launch vehicles that would result from changes in propulsion and structural parameters. A computational method is then developed for analyzing such problems and is then applied to two classes (solid and liquid propellant) of launch vehicles.

The contributions of this dissertation can be summarized as follows:

1. The identification of critical propulsion and structural parameters that most significantly influence the performance of space launch vehicles;
2. The development of a computational method for determining the magnitude of payload-to-orbit gains to be achieved when changes are made in these critical parameters; and
3. The application of the computational method for determining such performance gains achieved for two classes of space launch vehicles (solid propellant and liquid propellant) resulting from upgrading key propulsion and structural parameters.

The type of analysis described in this document provides answers in the form of gains in the ratio of payload mass-to-total launch-vehicle mass for given classes of launch vehicles. Results achieved in this manner are applicable to launch vehicles without regard to size or weight class. This feature provides for broader application of the methodology than otherwise might be achieved. The present analysis should be of high interest and
usefulness to engineers and others performing theoretical work involving initial design, upgrades, and various conceptual studies relating to the performance of launch vehicles.

The author's employment includes service with the NASA Langley Research Center, Hampton, VA; the National Air and Space Center, Wright-Patterson AFB, OH; NASA Headquarters, Washington, DC; and elements of the Department of Defense, Washington, DC. He has dealt extensively with U.S. Government laboratories and contractor organizations on topics relating to development and testing of rocket motors, hypersonic aerodynamics, trajectory optimization, etc. The author has traveled widely while representing the United States Government in foreign and diplomatic assignments. In recognition of his work, dedication, and technical expertise, he was selected over competing candidates as NASA’s representative for a one-year fellowship at the Industrial College of the Armed Forces of the National Defense University, Fort McNair, Washington, DC. Candidates for attendance at this institution are selected from senior staff members, both military and civilian, of the Department of Defense and closely-related organizations. In addition, the author served four years as an active-duty member of the U.S. Air Force, which included overseas service.
CHAPTER II

CHARACTERISTICS OF SEVERAL CURRENT LAUNCH VEHICLES

2.1 Ariane

Arianespace was established in March 1980. Under terms of an intergovernmental agreement, ESA member states transferred production, marketing, and launch responsibilities for the operational Ariane vehicle and its up-rated versions to Arianespace. The lead responsibility for Ariane's overall development is routed through ESA to CNES. Once declared operational, the vehicles are turned over to Arianespace for commercial exploitation. Participating in the creation of Arianespace were thirty-six of Europe's key aerospace and avionics manufacturers, thirteen major European banks, and CNES. Shareholding distribution among the three top nations is 59% for France, 20% for Germany, and 4% for Belgium [6-8].

A three-way coordination plan for launching the Ariane is implemented at the Guiana Space Center (CSG) in Kourou, French Guiana. CSG was established by the French Government in April 1965 and built by CNES. It became operational in April 1968 with the launch of a Veronique sounding rocket. Following the Diamant and Europa programs, CSG was selected for Ariane launch operations based on proximity to the equator, access to the ocean facilitating all inclination missions, no threat of hurricanes or earthquakes, and low population density. Currently, CSG is operated for ESA by CNES. CNES is also responsible for expanding the launch facilities to meet growing mission demands. Operation and maintenance of the launch facilities are the responsibilities of Arianespace, and CNES is reimbursed by Arianespace for the personnel, facilities, and materials used to support launch operations.
Table 2.1 contains the specific impulse and chamber pressure for each stage of the ARIANE-4.

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<th>Stage 3</th>
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<td>N2O4/UH25</td>
<td>278</td>
<td>294</td>
<td>444</td>
</tr>
<tr>
<td>LOX/LH2</td>
<td>848</td>
<td>848</td>
<td>508</td>
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*Data shown is for the AR 44L. Propulsion data for strap-ons not shown.

The gross liftoff weight for the AR 44L is 1.04 M pounds. The ratio of payload weight (100 mile orbit) to liftoff weight is 0.02 or 2%.

2.2. United States Space Shuttle

On January 5, 1972, President Nixon endorsed the Shuttle program based on the thrust-assisted orbiter concept and requested the development of the space transportation system to begin immediately. The Shuttle, he noted, would enable the United States to achieve a working presence in space by making space transportation routinely available and by reducing costs and preparation time.

After receiving this approval, NASA began acquisition and development of the Shuttle elements. Development and construction of the Space Shuttle was competitive. Separate contractors were selected for the design and manufacture of the orbiter, its main engines, the external tank, and the solid-rocket boosters. Rockwell International was selected as prime integrating contractor.

The first Shuttle launch occurred on April 12, 1981, and was spectacularly successful.
Selected propulsion data for the United States Space Shuttle are presented in Table 2.2 [6, 7].

**Table 2.2 Selected Propulsion Data* for United States Space Shuttle**

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Specific Impulse (vac)</th>
<th>Chamber Pressure (psia)</th>
</tr>
</thead>
<tbody>
<tr>
<td>LOX-LH₂</td>
<td>455</td>
<td>2,970</td>
</tr>
</tbody>
</table>

*Data for strap-ons not shown

The gross liftoff weight for the shuttle is about 4.5M lb. The ratio of payload weight (100 mile orbit) to liftoff weight is 0.012 or 1.2%.

**2.4 ZENIT**

ZENIT first appeared in 1985 in conjunction with two suborbital and two orbital qualification tests. It was the first totally new Soviet launch vehicle in twenty years. The basic ZENIT vehicle consists of two LOX/kerosene stages, with the first stage being very similar to the ENERGIA launch vehicle's strap-on boosters. In 1990-1992, ZENIT experienced three consecutive failures, including one which resulted in the destruction of a launch pad. Since then, ZENIT has had eight successful launches with no failures.

Table 2.3 shows key propulsion data for the ZENIT space launch vehicle [6-8].

**Table 2.3 Selected Propulsion Data for ZENIT**

<table>
<thead>
<tr>
<th></th>
<th>Stage 1</th>
<th>Stage 2</th>
<th>Stage 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propellant</td>
<td>LOX/Kerosene</td>
<td>LOX/Kerosene</td>
<td>LOX/Kerosene</td>
</tr>
<tr>
<td>Specific Impulse, sec (vac)</td>
<td>337</td>
<td>350</td>
<td>361</td>
</tr>
<tr>
<td>Chamber Press, psia</td>
<td>3,700</td>
<td>2,364</td>
<td>1,124</td>
</tr>
</tbody>
</table>
The gross liftoff weight for the ZENIT vehicle is 1.03M pounds. The ratio of payload weight to liftoff weight is 0.029 or 2.9%.

2.5 ENERGIA

The ENERGIA consists of a central core with four liquid hydrogen/liquid oxygen engines, and nominally four strap-on boosters (based on the first stage of the ZENIT launcher), each with a single four-chamber liquid oxygen/kerosene engine. All core and strap-on engines are ignited at liftoff. The ENERGIA is unique among expendable launch vehicles in that it does not deliver its payload directly to orbit. In order to facilitate a predictable and safe reentry of the expended core, the core engines are shut down, and the core jettisoned, before orbital velocity is reached. The final velocity increment required to achieve orbit must be provided by either the payload itself or an upper stage.

The ENERGIA vehicle design was affected by a number of operational requirements. One requirement was its ability to launch both large unmanned payloads and the Buran orbiter. This resulted in a launch vehicle payload with near-orbital capability of 230,000lb (105,000 kg) and a side-mounting payload arrangement. The ENERGIA is modular in design, with the number of strap-ons varying from two to eight.

Table 2.4 shows key propulsion information for the ENERGIA launch vehicle [6-9].

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Stage 1 (Strap-ons)</th>
<th>Stage 2 (Core)</th>
<th>EUS (Upper Stage)</th>
</tr>
</thead>
<tbody>
<tr>
<td>LOX/Kerosene</td>
<td>LOX/LH2</td>
<td>LOX/LH2</td>
<td>LOX/LH2</td>
</tr>
<tr>
<td>Specific Impulse sec (vac)</td>
<td>337</td>
<td>453</td>
<td>490</td>
</tr>
<tr>
<td>Chamber Press, psia</td>
<td>3556</td>
<td>3000</td>
<td>3000</td>
</tr>
</tbody>
</table>
The gross liftoff weight for the ENERGIA is 5.3M pounds. The ratio of payload weight (100 mile orbit) to liftoff weight is 0.037 or 3.7%.
CHAPTER III
LIQUID PROPELLANT CHARACTERISTICS AND PERFORMANCE

Theoretically, any chemical system that gives rise to an exothermic reaction can be used as a propellant. Thus, many systems are conceivable; however, there are certain qualities necessary for the proper performance of a propellant that may serve as criteria for rejecting some and considering others [9, 10].

A propellant is formed by combining an oxidizer with a fuel. The oxidizer consists mainly of atoms like those of oxygen, chlorine and fluorine, while the fuel consists mainly of such atoms as hydrogen, lithium, beryllium, boron, carbon, sodium, magnesium, aluminum, and silicon. Thus, the term propellant embraces all of the active components. From a general point of view, it is possible to compile Figure 3.1, which includes propellants composed either of one or two separate liquids, as well as those that are solid. If the oxidizer and the fuel have no chemical affinity at normal temperature and can be mixed to form a single liquid, a composite monopropellant is defined. For example, this applies to nitric acid mixed with nitrides or acetates.

Likewise, the fuel and the oxidant atoms may both be combined in the same molecule, which results in a simple monopropellant, such as propyl nitrate. Another possible means of propulsion is an exothermic reaction resulting from decomposition, such as occurs with hydrogen peroxide or hydrazine. Again, this is a monopropellant, since only a single substance is used for propulsion. Generally, however, the two liquids (the oxidizer and the fuel) are injected separately and such a system constitutes a bipropellant.
3.1 Classification of Propellants

The following diagram (Figure 3.1) shows the classification of most solid and liquid propellant types. If the two components of a bipropellant react immediately when they are in contact with one another, the propellant is described as hypergolic. This is followed by a more detailed discussion of the various liquid propellants [9, 10]. A brief discussion of solid propellant systems is presented in a separate section.

Figure 3.1 Classification of Propellants
The term “liquid propellant” is used to define both liquid oxidizers (liquid oxygen, liquid fluorine, nitric acid, etc.) and liquid fuels (RP-1, alcohol, liquid hydrogen, etc.). In some cases, additives are used (water, ferric chloride, etc.). The propellants furnish the energy and the working substance for the rocket engines. The selection of the propellants is one of the most important steps in the design of an engine. It greatly affects overall engine system performance, as well as the design criteria for each engine component. The propellant selection, subsequently, is influenced by availability, handling and storage considerations, and price [11].

3.2 Monopropellants

Liquid monopropellants may be either a mixture of oxidizer and combustible matter, or a single compound that can be decomposed with attendant heat release and gasification. A rocket monopropellant must be stable in a natural or controlled environment, yet should produce hot combustion or decomposition gases when pressurized, heated, or fed through a catalyst. A liquid monopropellant engine system usually does have the advantage of simplicity of tankage, feed plumbing, flow control, and injection. Unfortunately, most of the practical monopropellants, such as hydrogen peroxide (H₂O₂), have relatively low performance. Thus, they are mainly used as secondary power sources in rocket engine systems, such as for turbo-pump gas generators and auxiliary power drives, and for attitude and roll-control jets.

3.3 Bipropellants

In a liquid bipropellant system, two different propellants are used, usually an oxidizer and a fuel. Separate tanks hold oxidizer and fuel, which are not mixed until they reach the
combustion chamber. Contemporary liquid-propellant rocket engines use bipropellants almost exclusively, because they offer higher performance, and safer operation.

The combustion of many bipropellant combinations is initiated by ignition devices such as:

(a) chemical pyrotechnic igniters;
(b) electric spark plugs;
(c) injection of a spontaneously ignitable liquid fuel or oxidizer ("pyrophoric fluid") ahead of the propellant; or
(d) a minor combustive event wherein ignition is initiated by devices (a) or (b), in turn igniting the contents of the main chamber by the hot gas produced.

Other bipropellant combinations ignite spontaneously upon mixing. Those combinations are defined as hypergolics and permit greatly simplified ignition, but pose certain hazards. For instance, accidental mixing of the fuel and oxidizer due to tank and other hardware failures could cause a violent explosion. These hazards must be considered when designing an engine system using hypergolic propellants.

3.4 Cryogenic Propellants

Some liquid propellants are liquefied gases with a very low boiling point (−230°F to −430°F) at ambient pressure and also critical temperature (10 °F to −400°F). These propellants are defined as cryogenics. The most common cryogenic propellants for rocket applications are liquid oxygen (O₂), liquid hydrogen (H₂), liquid fluorine (F₂), and oxygen difluoride (OF₂), or mixtures of some of them. Cryogenic propellants pose storage and handling problems. Elaborate insulation must be provided in order to minimize propellant losses due to boil-off. The complexity or the insulation configuration
depends on potential storage and handling problems. Recently, novel insulating techniques have been under development that should greatly reduce these losses. Adequate venting systems are needed for the developed gases. Storage and handling equipment and their components are extremely sensitive to atmospheric or other moisture; for instance, even minute quantities may cause the malfunction of a valve. Likewise, the design criteria, including material selection for engine systems using cryogenic propellants, must consider the very low temperatures involved.

3.5 Storable Liquid Propellants

In contrast to the cryogenic propellants, certain other liquid propellants, defined as storable, are stable over a reasonable range of temperatures and pressures. These are sufficiently nonreactive with construction materials to permit storage in closed containers for periods of a year or more. Storable liquid propellants permit almost instant readiness of the rocket engine and may result in greater reliability, due to the absence of extremely low temperatures and the need to dispose of boil-off vapors. Their application to military vehicles, as well as to the upper stages of space vehicles, has increased significantly with time.

3.6 Additives for Liquid Rocket Propellants

Sometimes, additives are mixed into liquid propellants for one of the following reasons:

(a) to improve cooling characteristics;

(b) to depress freezing point;

(c) to reduce corrosive effects;
(d) to facilitate ignition; or

(e) to stabilize combustion.

3.7 Mixture Ratio

A specific ratio of oxidizer weight to fuel weight in a bipropellant combustion chamber will usually yield a maximum performance value. This is defined as the optimum mixture ratio. As a rule, the optimum mixture ratio is richer in fuel than the stoichiometric mixture ratio, at which theoretically all the fuel is completely oxidized and the flame temperature is at a maximum. This is because a gas, which is slightly richer in fuel tends to have a lower molecular weight. This results in a higher overall engine system performance. The optimum mixture ratio of some propellant combinations varies slightly with changes in chamber pressure. Also, in actual application, the mixture ratio may vary from the optimum value for one of the following reasons:

(a) lower chamber temperature to stay within the temperature limitations of chamber construction material;

(b) required coolant flow; or

(c) improved combustion stability.

3.8 Desirable Features of Liquid Propellants

When selecting a propellant or propellant combination for a specific application, it is important to realize that most propellants, in addition to their advantages, may have certain disadvantages. Thus, propellant selection usually involves some compromises. The more important and desirable propellant features are listed below. Order of importance may vary as a function of application.
(1) High-energy release per unit of propellant mass, combined with low molecular weight of the combustion or decomposition gases, for high specific impulse;
(2) Ease of ignition;
(3) Stable combustion;
(4) High density or high-density impulse to minimize the size and weight of propellant tanks and feed system;
(5) Ability to serve as an effective coolant for the thrust chamber (optimum combination of high specific heat, high thermal conductivity, and high critical temperature);
(6) Reasonably low vapor pressure at 160°F (a frequent specification value) for low tank weight and low net positive pump suction head requirement.
(7) Low freezing point (preferably less than -65 F) to facilitate engine operation at low temperature.
(8) Absence of corrosive effects (compatibility with engine construction materials);
(9) For storables: good storability, as assisted by a high boiling point (preferably above 160°F), by items 6, 7, and 8 and by the resistance to deterioration during storage;
(10) Low viscosity (preferably less than 10 cp down to -65°F) to minimize pressure drops through feed system and injector;
(11) High thermal and shock stability to minimize explosion and fire hazard;
(12) Low toxicity of raw propellants, their fumes, and their combustion products;
(13) Low cost; and

(14) Availability.

For theoretical calculations, it is generally assumed that ideal conditions exist. The prime objective of propellant performance calculations is to derive the quantities $C^*$, $C_t$, and $I_{sp}$ through evaluation of the flame temperature, $T_c$, mean gas molecular weight, $M_c$, and the specific heat ratio, $\gamma$, for given values of $P_c$, $P_e$ and $P_A$. The chamber temperature can be calculated from the heat transferred during the chemical reaction of the propellants and from the specific heat of the gases. In practice, it has been found that actual test results are usually five to twelve percent lower than the theoretical values obtained from calculations.

In addition to the assumption of specific idealized gas conditions, the performance equations assume specific singular values for the most important gas properties: $\gamma$, $M_c$, $R$, and $T_e$. Gas properties are not necessarily constant along the path of flow. Two basic approaches can be taken: 1) calculations can be based on the assumption of constant gas composition along the nozzle axis or 2) based on the assumption of variable composition. The applicable literature frequently uses the term “equilibrium” instead of “composition.”

In calculations based on constant composition, it is assumed that no further chemical reactions take place in the gases after leaving the combustion chamber and entering the nozzle. It is also assumed that the combustion products at $A_e$ are in the same relative proportion as at $A_t$. Then, the remaining principal variables are pressure and temperature at the various stations. Assuming different initial sets of mixture ratios, chamber pressures, and gas compositions, a typical set of calculations (probably involving successive approximations) may be conducted to determine optimum values of mixture
ratio, chamber length, expansion-area ratio, etc., for a given propellant combination and vehicle trajectory.

Calculations based on variable composition consider additional variations, mainly those of gas composition (as they result from incomplete combustion, dissociation, and reassociation). These calculations are an attempt to closely model the true physical processes. However, to their extreme complexity and unpredictability, the results are frequently no more reliable than those obtained from calculations assuming constant composition.

Thus, it is probably a matter of preference as to which approach should be adopted. It is noted that the theoretical data based on variable composition usually give values several percent higher than those based on a constant composition. Therefore, in presenting performance data, the assumption of the type of composition assumed must be specified.

Table 3.1 provides theoretical performance information for the main classes of liquid rocket propellants when operated under a set of specified conditions [10]. For each oxidizer-fuel combination listed in the table, the upper line of numbers correspond to equilibrium, i.e., fixed chemical composition of the gaseous products throughout expansion in the nozzle. The lower line of numbers corresponds to non-equilibrium conditions, or varying chemical composition of the gaseous products throughout expansion in the nozzle. The combustion pressure is assumed to be 1000 psia, nozzle exit pressure equals 14.7 psia, and nozzle expansion ratio (exit area/throat area) is optimum. Also, the contraction ratio (chamber area/throat area) is assumed to be infinite, the combustion is assumed to be adiabatic, isentropic expansion of ideal gas is assumed, and
compositions are expressed in percentage of weight. The density at the boiling point was for those oxidizers or fuels that boil below 68°F at one a pressure of atmosphere. Specific impulses shown in the table will increase significantly with higher combustion pressures. The following symbols and definitions apply for those parameters presented in the table.

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>$r_w$</td>
<td>Weight mixture ratio, weight flow rate of oxidizer/weight flow rate of fuel</td>
</tr>
<tr>
<td>$\rho_B$</td>
<td>Bulk density, defined as $\rho_B = \frac{\left(\frac{r_w+1}{\rho_o} - \frac{1}{\rho_f}\right)}{\rho_o}$, g/cm$^3$</td>
</tr>
<tr>
<td>$\rho_o$</td>
<td>Density of oxidizer, g/cm$^3$</td>
</tr>
<tr>
<td>$\rho_f$</td>
<td>Density of fuel, g/cm$^3$</td>
</tr>
<tr>
<td>$T_c$</td>
<td>Combustion chamber temperature, °F</td>
</tr>
<tr>
<td>$C^*$</td>
<td>Characteristic velocity, feet/second</td>
</tr>
<tr>
<td>$I_{sp}$</td>
<td>Specific impulse, sec</td>
</tr>
</tbody>
</table>
Table 3.1 Theoretical Performance of Liquid Rocket Propellant Combinations

<table>
<thead>
<tr>
<th>OXIDIZER</th>
<th>FUEL</th>
<th>$r_w$</th>
<th>$\rho_B$</th>
<th>$T_c$</th>
<th>$C^*$</th>
<th>$I_{sp}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Oxygen</td>
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<td>6269</td>
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<td>314</td>
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<td></td>
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<td>2.30</td>
<td>1.10</td>
<td>6552</td>
<td>6680</td>
<td>337</td>
</tr>
<tr>
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<td>Hydrazine</td>
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<td>1.23</td>
<td>6325</td>
<td>6640</td>
<td>321</td>
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<tr>
<td></td>
<td></td>
<td>1.50</td>
<td>1.26</td>
<td>6795</td>
<td>6845</td>
<td>345</td>
</tr>
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<td>8340</td>
<td>401</td>
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<td>1.17</td>
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<td>$\rho_S$</td>
<td>$T_c$</td>
<td>$C^*$</td>
<td>$I_{sp}$</td>
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<td>1.31</td>
<td>5300</td>
<td>5425</td>
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Table 3.2 includes a number of supplementary propellant characteristics and propellant combinations that do not appear in Table 3.1. In order to perform the propellant-related analysis, which constitute a critical part of the present research, it was necessary to utilize characteristics from both tables. Information of this type is not widely available in the literature and was assembled by the author for the present application, as it relates to space launch systems and missile systems [12].

Note that in Table 3.2, certain ideal propulsion-related parameters vary, depending on whether one is attempting to evaluate the effect of bulk density, the product of specific impulse and bulk density, or simply to maximize specific impulse. Also, note that the parameters are a function of given conditions, mainly combustion temperatures and pressures. The \( I_{sp} \) values shown in the table are lower than those used in the subsequent analysis, because of the higher combustion pressures and temperatures that are assumed in the analysis.
Table 3.2 Additional Information on Theoretical Performance of Liquid Rocket Propellant Combinations

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Table 3.2 Continued

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RP-1 is a hydrocarbon fuel in accordance with Specification MIL-F-25576 (USAF)
CHAPTER IV

COMBINED PROPULSION AND PROPELLANT RELATIONSHIPS

The purpose of this section is to present basic performance parameters relating to liquid propellant rocket engines [4, 9, 11-14].

Specific impulse is a very important parameter in rocket propulsion and is defined as:

\[ I_{sp} = \frac{\int_0^t Tdt}{W_p} = \frac{I_t}{W_p} \]  

(4.1)

where total impulse is defined as:

\[ I_t = \int_0^t Tdt \]  

(4.2)

Under steady-state conditions, specific impulse is further defined as [7]:

\[ I_{sp} = \frac{T}{\dot{w}} \]  

(4.3)

where the above symbols are defined as:

- \( I_{sp} \) = specific impulse, seconds
- \( I_t \) = total impulse, lb-sec
- \( T \) = thrust, lb
- \( t_b \) = burn time, sec
- \( W_p \) = propellant weight, lb
- \( \dot{w} \) = propellant flow rate, lb/sec

The following equation provides a relation for determining the theoretical value of specific impulse for given conditions:
In Equation (4.4), the first term is the most important, while the other terms are either constant or approximately constant. This shows that specific impulse is a strong function of the square root of the ratio of combustion temperature to molecular weight:

\[ I_{sp} \propto \left( \frac{T_c}{M_c} \right)^{1/2} \]  

\[(4.5)\]

The constant of proportionality for the above equation is a function of the efficiency of the rocket engine design. Of prime important in engine design is the ratio of combustion chamber volume to nozzle throat area. This ratio is a measure of the time available for combustion to take place. The longer the fuel and oxidizer mixture are in the combustion chamber, the more complete the combustion process and the larger the combustion efficiency. Engine design can sometimes be optimized to reduce the combustion chamber volume to nozzle throat area requirements.
The chamber temperature is produced by the heat transfer in the equation:

\[ I_{sp} = \sqrt{\frac{2J(H_c - H_e)}{g}} \]  

(4.6)

which also shows that \( I_{sp} \) is proportional to the enthalpy change or heat transfer per unit weight of propellant, or:

\[ I_{sp} \propto \sqrt{\Delta H} \]  

(4.7)

From the above relationships, it can be seen that specific impulse can be increased by raising the temperature of the combustion products in the chamber, by expelling lighter molecules, and to a lesser extent, by using a gas with a smaller specific heat ratio. A high gas temperature can be obtained by using a propellant mixture that produces a large quantity of heat per pound of mixture. In the final analysis, specific impulse is a function of the combined efficiency of the propellant, rocket engine, and nozzle.

In selecting nominal values of specific impulse for the present research, the latest version of the AIAA Publication, “International Reference Guide to Space Launch Systems,” has been consulted [6].

Thermodynamic nozzle theory also provides an expression for thrust as a function of nozzle throat area, combustion chamber and exhaust pressures, and the specific heat ratio, \( \gamma \), or \( \frac{C_p}{C_v} \). Thus, one obtains [8]:

\[
T = A_t P_e \left[ \frac{2 \gamma^2}{\gamma - 1} \left( \frac{2}{\gamma + 1} \right)^{\gamma + 1} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{\gamma - 1} \right] \right] + A_e (P_e - P_d) \]  

(4.8)

where:
\[ T = \text{thrust, lb} \]
\[ A_t = \text{nozzle throat area, in}^2 \]
\[ A_e = \text{nozzle exit area, in}^2 \]
\[ P_A = \text{ambient pressure, lb/in}^2 \]
\[ P_c = \text{pressure in combustion chamber, lb/in}^2 \]
\[ P_e = \text{pressure at nozzle exit plane, lb/in}^2 \]
\[ \gamma = \text{ratio of specific heats} \]

The effective exhaust velocity is defined by the relation:

\[ C = g_o \frac{T}{\dot{w}} \]  \hspace{1cm} (4.9)

or

\[ C = g_o I_{sp} \]  \hspace{1cm} (4.10)

where \( I_{sp} \) is the measured specific impulse.

As shown previously, total impulse is defined as:

\[ I_t = \int_0^t T dt \]  \hspace{1cm} (4.11)

For constant thrust total impulse is simply:

\[ I_t = T t_b \]  \hspace{1cm} (4.12)

The weight flow coefficient, \( C_D \), is defined by the following relationship:

\[ \dot{w} = C_D P_c A_t \]  \hspace{1cm} (4.13)

or

\[ C_D = \frac{\dot{w}}{P_c A_t} \]  \hspace{1cm} (4.14)
where:

\[ P_c = \text{combustion pressure, lb/in}^2 \]
\[ A_t = \text{nozzle throat area, in}^2 \]

The thrust coefficient \((C_T)\) is defined in a similar manner by the equation:

\[ T = C_T P_c A_t \]  \hspace{1cm} (4.15)

Hence,

\[ C_T = \frac{T}{P_c A_t} \]  \hspace{1cm} (4.16)

Consistent with previous equations, it is seen that:

\[ T = I_{sp} \dot{w} = C_D A_t P_c I_{sp} = C_T A_t P_c \]  \hspace{1cm} (4.17)

Hence,

\[ I_{sp} = \frac{C_T}{C_D} \]  \hspace{1cm} (4.18)

The term characteristic velocity, or \(C^*\), is a function of the propellant combination and combustion chamber design and reflects the efficiency of the combustion event occurring in the rocket engine combustion chamber. It is essentially independent of nozzle characteristics and is defined as \([18]\):

\[ C^* = \frac{P_c A_t g_0}{\dot{w}} \]  \hspace{1cm} (4.19)

where \(P_c\) is the pressure in the combustion chamber, \(A_t\) is the cross-sectional area of the nozzle throat, and \(\dot{w}\) is the rate of weight flow through the nozzle. The value of Equation (4.19) is determined from measured test values of \(P_c\), \(A_t\) and \(\dot{w}\). Another value for \(C^*\), defined as the theoretical value, is determined from the following relation \([9]\):
\[ C^* = \left[ \frac{1}{\gamma} \left( \frac{\gamma + 1}{2} \right)^{-1} \frac{RT_c}{M_c} \right]^{1/2} \]  

(4.20)

where \( \gamma \) is the specific heat ratio, \( R \) is the gas constant, \( T_c \) is the combustion temperature, and \( M_c \) is the molecular weight of the combustion products. In this equation, \( C^* \) depends mainly on conditions in the combustion chamber; i.e., on flame temperature and combustion product composition. \( C^* \) is determined from measured values of \( p_c, A_t \) and \( \dot{w} \), as in Equation (4.19), and compared to the theoretical value determined from Equation (4.20) above. The combustion efficiency is then determined from the relation:

\[ \eta = \frac{C^*(\text{measured}) \times 100\%}{C^*(\text{theoretical})} \]  

(4.21)

Propellant bulk density is a parameter that is calculated to determine the overall propellant density when combining the densities of the oxidizer and fuel at the mixture ratio utilized by the propulsion system. As shown in an earlier section, the bulk density is calculated from the following relation:

\[ \rho_B = \frac{\rho_w + 1}{\frac{r_w}{\rho_o} + \frac{1}{\rho_f}} \, \text{gm/cm}^3 \]  

(4.22)

where: \( \rho_B \) = bulk density (g/cm³)

\( r_w \) = mixture ratio as determined by the weight flow rate of the oxidizer (\( \dot{w}_o \)) divided by the weight flow rate of the fuel (\( \dot{w}_f \))

\( \rho_f \) = density of fuel, (gm/cm³)

\( \rho_o \) = density of the oxidizer, (gm/cm³)
Propellant bulk density is important as a parameter used in the overall vehicle design, modeling, and assembly process. The density at the boiling point is used for those oxidizers and fuels which boil below 80°F at one atmospheric pressure.

The term “density impulse” is defined as the product of specific impulse and bulk density \( I_{sp} \rho_B \). This parameter is used to relate propulsion system performance to the volume of the tanks required to contain the propellants. Given a choice among propellant combinations each having the same \( I_{sp} \), the combination with the highest density impulse would require smaller propellant tanks.

Another important relationship in the modeling process is a term known as equivalence ratio \( \phi^* \).

Let \( \phi \) be defined as:

\[
\phi = \frac{\text{fuel mass flow rate}}{\text{oxidizer mass flow rate}}
\]

Then, equivalence ratio, \( \phi^* = \phi / \phi_s \), where \( \phi_s \) = stoichiometric fuel/oxidizer mass flow rate. A fuel-rich mixture is indicated by the inequality:

\[
\phi^* > 1 \tag{4.23}
\]

Note that \( \phi \) and \( \phi_s \), as defined above, are inversely related to the mixture ratio, \( r_w \), as elsewhere in this document. Note also that European rocket designers define mixture ratio inversely from standard United States usage.

The total impulse-to-weight ratio is frequently used as a measure of overall rocket performance. It is denoted by \( R_{I/W} \) and is defined as:

\[
R_{I/W} = \frac{I_t}{W_h + W_p} = \frac{I_t}{W_t} \tag{4.24}
\]
Where,

\[ I_t = \text{total impulse} \]
\[ W_h = \text{hardware weight} \]
\[ W_p = \text{propellant weight} \]
\[ W_t = \text{total weight} \]

From a performance perspective, a primary objective of rocket design is to achieve the largest possible value for \( R_{j/W} \). From previous equations, it is seen that both the specific impulse and the density of the propellants have an effect on the value of \( R_{j/W} \).

General information regarding liquid propellant rocket engine design and performance is presented in several of the references listed herein. Design details regarding high performance liquid propellant rocket engines, however, is closely guarded by engine developers in basically all of those countries which design and build launch vehicles. Exactly where the upper limit of performance lies, for given propellant characteristics, is open to considerable discussion and debate. As discussed elsewhere in this document, the purpose of the present research is to determine sensitivity coefficients which are dependent on propulsion characteristics, payload capabilities, and minimum launch vehicle gross weight, for given orbit requirements.
CHAPTER V

SOLID-PROPELLANT SYSTEMS

There are two basic types of solid propellants – double-base and composite. In double-base propellants, both the oxidizer and fuel are associated with each molecule of the propellant. In composite propellants, the oxidizer and fuel are separate compounds mixed together. Typical double-base propellants are colloidal mixtures of nitrocellulose and nitroglycerin, such as ballistite and cordite. Typical composite propellants are constituted of very finely ground oxidizer crystals (perchlorates or nitrates) dispersed in a fuel matrix (polyesters, asphalt, or rubber) [4, 11, 14, 15].

The propellant in a solid-propellant rocket is referred to as the grain. The internal port area of the grain can take any one of several shapes; e.g., an internal star, cruciform, single port, etc. The internal shape of the port area defines the initial area of the burning surface. The manner in which burning surface area varies with burn-time defines the shape of the thrust-time curve. The thrust-time curve can be approximately neutral, progressive, or regressive.

In solid-propellant systems, the rate of the gas production and burning rate are defined as:

\[ \dot{w} = r_b A_b \rho_p \]  \hspace{1cm} (5.1)

and

\[ r_b = a P_c^n \]  \hspace{1cm} (5.2)

where:

\[ \dot{w} = \text{propellant weight-flow rate, lb/sec} \]

\[ A_b = \text{propellant burning surface area, in}^2 \]
\[ \rho_p = \text{propellant density, lb/in}^3 \]
\[ r_b = \text{propellant burning rate, in/sec} \]
\[ p_c = \text{combustion pressure, lb/in}^2 \]
\[ n = \text{solid propellant burning exponent} \]
\[ a = \text{constant} \]

The above variables can be combined as:

\[ \dot{w} = a \ p_c^n \ A_b \ p_p \]  \hspace{1cm} (5.3)

Figure 5.1 illustrates how \( r_b \) varies as a function of \( p_c \) for different values of initial propellant temperature, \( T_p \).

The density of solid propellants can be from 20% to 80% higher than that of liquid propellants. This advantage can partly compensate for the lower specific impulse of solid propellants. Furthermore, the combustion pressure in a solid-propellant rocket is generally higher than in liquid engines, since it is not subject to the limitations of a fuel system, thus allowing the use of higher thrust coefficients.

The thrust of a solid-propellant rocket engine is defined as:

\[ F = I_s \ A_b \ p_p \ a \ p_c^n \]  \hspace{1cm} (5.4)

Shown here are some of the important characteristics of solid rocket propellants:

1. High specific impulse, \( I_s \), which requires a high adiabatic combustion temperature, \( T_c \), and low molecular weight, \( m \);
Figure 5.1 Solid-Propellant Burning Rate Versus Chamber Pressure
2. High density, $\rho_p$, so that the required propellant quantity can be packaged in minimum volume;

3. A low burning-rate exponent, $n$, for good combustion pressure stability;

4. A reasonable burning rate, $r_b$, at the optimum pressure ($r_b$ can be varied over a wide range by modifying the oxidizer mixed with a given fuel); and

5. A low temperature sensitivity coefficient, $\pi_k$, for small changes in engine performance with propellant temperature.

Specific solid-propellant rocket system parameters chosen for use in the calculations and simulations will be presented in another section.

The temperature at which a solid-propellant grain is soaked prior to launch has an effect on the linear burning rate, $r_b$. Figure 5.1 shows how the burning rate varies with temperature for a typical solid-propellant rocket motor [10]. (Note that this graph is logarithmic.) From the equations previously shown, a change in $r_b$, or linear burning rate, has an effect on propellant flow rate and thrust level, but does not significantly change the total impulse produced. In some critical space launches (and also in some missile launches), the temperature of the solid-propellant motor may be monitored and controlled prior to launch.
CHAPTER VI
STRUCTURE AND VELOCITY MODELLING

Figure 6.1 shows structure factor as a function of propellant weight for individual stages of space launch vehicles. The chart is useful as a modeling aid when analyzing the effects of changes in structure factor on launch vehicle performance. The three curves shown in this figure were constructed using data derived from a large number of United States and foreign space launch vehicles. The upper and lower lines in the figure are constructed to form boundaries of structure factors from known systems and are intended to be used as upper and lower limits for modeling purposes. The following is a list of definitions and symbols used in deriving Figure 6.1:

\[ K = \frac{w_e}{w_p} \quad \text{structure factor} \quad (6.1) \]

\[ w_p = \text{propellant weight} \]

\[ w_e = \text{empty weight of a given stage} = w_{bo} - (w_{pe} + w_g + w_i) \]

\[ w_{bo} = \text{burnout weight} \]

\[ w_{pe} = \text{payload weight} \]

\[ w_g = \text{guidance weight} \]

\[ w_i = \text{instrumentation weight} \]

The lines in Figure 6.1 are based on the equation [6]:

\[ c = \frac{b}{w_p}, \text{ where } b \text{ is a constant} \quad (6.2) \]
Figure 6.1 Propellant Weight (lb) Versus Structure Factor
Data from the center line conforms to the equation:

\[ c = \frac{26}{w_p^a} \]  \hspace{1cm} (6.3)

Information from these curves is useful in assembling sets of data for use in overall modeling efforts.

In the process that follows, an empirical approach is used for estimating the velocity losses due to gravity and aerodynamic drag. This technique consists of correlating velocity loss data obtained from the numerical integration of trajectories with the energy per unit mass at burnout as the correlating parameter.

Thus with respect to velocity losses, the modeling relations become [12, 17, 18]:

\[ \Delta V_{id} = \sum \ln I_{SP} \frac{W_{en}}{W_{en}} \]  \hspace{1cm} (6.4)

where:

\[ \Delta V_{loss} = \Delta V_{id} - (V^* - \Delta V_{rot}) \]  \hspace{1cm} (6.5)

\[ E = \frac{1}{2} V_{bo}^2 + gh = \frac{1}{2} V_{bo}^2 + g_o \left( \frac{R_e}{R_e + h} \right)^2 h \]  \hspace{1cm} (6.6)

\[ V^* = \sqrt{2E} = \sqrt{V_{bo}^2 + 2g_o \left( \frac{R_e}{R_e + h} \right)^2 h} \]  \hspace{1cm} (6.7)

Figure 6.2 shows velocity loss as a function of time spent below circular orbit velocity [12].

Other useful relations include the following:

\[ V_c = \left( \frac{\mu}{r} \right)^{1/2} = \sqrt{\frac{\mu}{r}} \]  \hspace{1cm} (6.8)
Figure 6.2 Velocity Loss Versus Time Below Circular Orbit Velocity

Time (Δt) below circular orbit velocity, sec × 10²
\[ r_a = \frac{\left( \frac{V_i}{V_c} \right)^2}{2 - \left( \frac{V_i}{V_c} \right)^2} \]  

(6.9)

\[ e = \frac{r_a - r_p}{r_a + r_p} \]  

(6.10)

\[ \cos i = \cos \lambda \sin \psi_i \]  

(6.11)

\[ \sin \psi_i = \frac{\cos i}{\cos \lambda} \]  

(6.12)

Since this study assumes a circular orbit with an inclination of 90°, the velocity loss or gain due to earth's rotation is zero. Thus the last three equations become zero and can, therefore, be neglected.

Thus, the ideal velocity as described above, minus the velocity losses due to aerodynamic drag and gravity, must equal the circular orbit velocity that corresponds to the orbit injection altitude, or

\[ V_c = \Delta V_{id} - \Delta V_{loss} \]  

(6.13)

The solutions for \( \Delta V_{loss} \) have been obtained from Figure 6.2 as a function of burn time.
CHAPTER VII

DEVELOPMENT OF COMPUTATIONAL ALGORITHM

When examining the performance of a multistage rocket system, certain staging relationships are important. For the purpose of establishing appropriate staging relationships, the following symbols and equation are used:

\[ M_g = \text{gross weight} \]
\[ M_o = \text{burnout weight} \]
\[ \overline{M} = \text{weight of payload} \]
\[ C = I_{sp} \dot{g}_o = \text{exhaust velocity} \]
\[ K = \text{structural factor} \]
\[ = \frac{\text{empty weight of a stage}}{\text{filled weight of the same stage}} \]
\[ V_r = \text{total velocity requirement (orbit injection velocity plus velocity losses due to gravity and aerodynamic drag)} \]

Subscripts

1, 2, 3, 4 indicate 1\textsuperscript{st}, 2\textsuperscript{nd}, 3\textsuperscript{rd}, and 4\textsuperscript{th} stages respectively.

Figure 7.1 Typical Launch Vehicle Staging Arrangement

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From velocity requirements and the rocket velocity equation, staging relationships can be written as follows:

Velocity required = velocity available, or:

$$V_r = C_1 \ln \frac{M_{g1}}{M_{01}} + C_2 \ln \frac{M_{g2}}{M_{02}} + C_3 \ln \frac{M_{g3}}{M_{03}} + C_4 \ln \frac{M_{g4}}{M_{04}}$$  \hspace{1cm} (7.1)$$

Expressions are next written for the burnout weight at the end of each stage.

$$M_{04} = \bar{M} + K_4 (M_{g4} - \bar{M}) = K_4 M_{g4} + (1 - K_4)\bar{M}$$  \hspace{1cm} (7.2a)$$

$$M_{03} = K_3 M_{g3} + (1 - K_3)M$$  \hspace{1cm} (7.2b)$$

$$M_{02} = K_2 M_{g2} + (1 - K_2)M_{g3}$$  \hspace{1cm} (7.2c)$$

$$M_{01} = K_1 M_{g1} + (1 - K_1)M_{g2}$$  \hspace{1cm} (7.2d)$$

Substituting Equations (7.2a-7.2d) into Equation (7.1) provides the following:

$$V_r = C_1 \ln \frac{M_{g1}}{K_1 M_{g1} + (1 - K_1)M_{g2}} + C_2 \ln \frac{M_{g2}}{K_2 M_{g2} + (1 - K_2)M_{g3}} + C_3 \ln \frac{M_{g3}}{K_3 M_{g3} + (1 - K_3)M_{g4}} + C_4 \ln \frac{M_{g4}}{K_4 M_{g4} + (1 - K_4)\bar{M}}$$  \hspace{1cm} (7.3)$$

Since \( \ln \frac{a}{f} = \ln a - \ln f \), Equation (7.3) is rewritten as:

$$C_1 \ln M_{g1} - C_1 \ln [K_1 M_{g1} + (1 - K_1)M_{g2}] + C_2 \ln M_{g2} - C_2 \ln [K_2 M_{g2} + (1 - K_2)M_{g3}] + C_3 \ln M_{g3} - C_3 \ln [K_3 M_{g3} + (1 - K_3)M_{g4}] + C_4 \ln M_{g4} - C_4 \ln [K_4 M_{g4} + (1 - K_4)\bar{M}] - V_r = 0$$  \hspace{1cm} (7.4)$$

It is desired to minimize \( M_{g1} \) for a given velocity requirement. Therefore, in Equation (7.4), \( M_{g2}, M_{g3} \) and \( M_{g4} \) become the variables and the following must be satisfied for minimum initial weight:
\[
\frac{\partial M_{g1}}{\partial M_{g2}} = 0 \quad \frac{\partial M_{g1}}{\partial M_{g3}} = 0 \quad \frac{\partial M_{g1}}{\partial M_{g4}} = 0
\]  
(7.5)

The left side of Equation (7.5) is written implicitly as:

\[
\frac{\partial f}{\partial M_{g2}} = \frac{\partial f}{\partial M_{g1}} \quad \frac{\partial f}{\partial M_{g3}} = \frac{\partial f}{\partial M_{g1}} \quad \frac{\partial f}{\partial M_{g4}} = \frac{\partial f}{\partial M_{g1}}
\]  
(7.6)

Equations (7.5) and (7.6) are equivalent if:

\[
\frac{\partial f}{\partial M_{g2}} = 0, \quad \frac{\partial f}{\partial M_{g1}} \neq 0, \quad \frac{\partial f}{\partial M_{g3}} = 0, \quad \frac{\partial f}{\partial M_{g4}} \neq 0, \quad \frac{\partial f}{\partial M_{g1}} = 0, \quad \frac{\partial f}{\partial M_{g1}} \neq 0
\]  
(7.7)

Minimizing \( M_{g1} \) for a given velocity is achieved by differentiating Equation (7.4) with respect to \( M_{g2} \), \( M_{g3} \) and \( M_{g4} \), or:

\[
\frac{\partial f}{\partial M_{g2}} = -\frac{C_1(1-K_1)}{K_1 M_{g1} + (1-K_1)M_{g2}} + \frac{C_2}{K_2 M_{g2} + (1-K_2)M_{g3}} - \frac{C_3 K_4}{K_3 M_{g3} + (1-K_3)M_{g4}}
\]  
(7.8a)

\[
\frac{\partial f}{\partial M_{g3}} = -\frac{C_2(1-K_2)}{K_2 M_{g2} + (1-K_2)M_{g3}} + \frac{C_3}{K_3 M_{g3} + (1-K_3)M_{g4}} - \frac{C_4 K_4}{K_4 M_{g4} + (1-K_4)M}
\]  
(7.8b)

\[
\frac{\partial f}{\partial M_{g4}} = -\frac{C_3(1-K_3)}{K_3 M_{g3} + (1-K_3)M_{g4}} + \frac{C_4}{K_4 M_{g4} + (1-K_4)M}
\]  
(7.8c)

Setting Equations (7.8a), (7.8b) and (7.8c) equal to zero to satisfy Equation (7.7), the following relations are achieved through algebraic manipulation:

\[
\frac{M_{g3}}{M_{g2}} = \frac{C_1 K_2 (1-K_1)}{(1-K_1)(1-K_2)(C_2 - C_1) + \frac{C_2 K_1 (1-K_2)}{M_{g2}}}\frac{M_{g2}}{M_{g1}}
\]  
(7.9a)
Equation (7.3) can be rewritten as:

\[
\frac{M_{g4}}{M_{g3}} = \frac{C_2 K_3 (1 - K_2)}{(1 - K_2)(1 - K_3)(C_3 - C_2) + \frac{C_3 K_2 (1 - K_3)}{M_{g3}}}
\]  

(7.9b)

\[
\frac{\bar{M}}{M_{g4}} = \frac{C_3 K_4 (1 - K_3)}{(1 - K_3)(1 - K_4)(C_4 - C_3) + \frac{C_4 K_3 (1 - K_4)}{M_{g4}}}
\]  

(7.9c)

An iterative process, programmed on a digital computer, is utilized to simplify the calculations. First, numerical value for \( \frac{M_{g2}}{M_{g1}} \) is selected and substituted into Equation (7.9a). Then, the value found for \( \frac{M_{g3}}{M_{g2}} \) is substituted into Equation (7.9b). Finally, the value found for \( \frac{M_{g4}}{M_{g3}} \) is substituted into Equation (7.9c). The ratios thus determined are substituted into Equation (7.10). If Equation (7.10) is not satisfied, the iterative process is repeated.

After Equation (7.10) is satisfied, the maximum ratio of payload weight to vehicle gross weight is found from the relationship:
\[
\frac{\overline{M}}{M_{g1}} = \frac{M_{g2}}{M_{g1}} \times \frac{M_{g3}}{M_{g2}} \times \frac{M_{g4}}{M_{g3}} \times \frac{\overline{M}}{M_{g4}}
\] (7.11)

Figure 7.2 provides a flowchart describing the computational steps to obtain the final results. Basic inputs into the program, for each case being solved, consist of the following for each stage: specific impulse \((I_{sp})\), structure factor \((K)\), velocity required \((V_r)\), plus the number of stages. The computational process begins with the manual input of an initial estimate for the value of \(\frac{M_{g2}}{M_{g1}}\). (See page 45, including Figure 7.1, for a definition of symbols appearing in the flow chart). As shown in the flowchart, for a given set of input values \((I_{sp}, K, V_r, \# \text{ stages})\) an iterative process takes place, using Equations (7.9a), (7.9b) and (7.9c), to find a set of terms for \(\frac{M_{g3}}{M_{g2}}, \frac{M_{g4}}{M_{g3}}\), and \(\frac{\overline{M}}{M_{g4}}\). The intermediate values for these latter terms are then substituted into Equation (7.10). Subsequent iterations take place until Equation (7.10) is satisfied for the specified value of \(V_r\). The values for \(\frac{M_{g2}}{M_{g1}}, \frac{M_{g3}}{M_{g4}}, \frac{M_{g4}}{M_{g3}}\) and \(\frac{\overline{M}}{M_{g4}}\) that satisfy Equation (7.10) are substituted into Equation (7.11) which then defines the ratio of payload mass to total launch vehicle mass, i.e. \(\frac{\overline{M}}{M_{g1}}\), for the given set of input values. The analysis executed in subsequent sections of the present report corresponds to a value for \(V_r\) of 30, 400 ft/sec. This value corresponds to the velocity requirement for a 100 NM circular orbit; i.e. 25,600 ft/sec (see Equation 6.8), plus combined gravity and aerodynamic drag losses of 4,800 ft/sec (see Figure 6.2).
INPUT
$I_{sp}, K, V_r, \#STAGES$

CALL COMPUTATIONAL ALGORITHM

INITIAL ESTIMATE
\[ \frac{M_{g2}}{M_{g1}} \]

COMPUTE VALUES
FOR \( \frac{M_{g3}}{M_{g2}}, \frac{M_{g4}}{M_{g3}}, \bar{M} \)

IF \( V_{rc} < V_r \)
INCREASE \( \frac{M_{g2}}{M_{g1}} \)

COMPARE \( V_{rc} \)

WHEN \( V_{rc} = V_r \), STOP

OPTIMUM VALUE
\[ \frac{\bar{M}}{M_{g1}} = \frac{M_{g2}}{M_{g1}} \frac{M_{g3}}{M_{g2}} \frac{M_{g4}}{M_{g3}} \]

IF \( V_{rc} > V_r \)
REDUCE \( \frac{M_{g2}}{M_{g1}} \)

Figure 7.2 Flowchart for Computer Program
CHAPTER VIII
DERIVATIONS AND ANALYSIS

Results obtained employing the previously described approach to determine changes in the ratio of payload to total launch vehicle weight, when changes are made to key launch-vehicle design parameters, are presented in the present chapter. Some of the previous work herein utilized results of integrated equations of motion in order to determine velocity losses attributed to gravity and aerodynamic drag. The results in this section utilizes an iterative solution to equations based upon specified velocity requirements for a given orbital mission coupled together with key launch vehicle design parameters.

Equations (7.9a), (7.9b), (7.9c) and (7.10), when employed together with velocity requirements, specific impulse and structure factor, can be solved to define optimum staging ratios for launch vehicles. These ratios can then be combined, as in Equation (7.11), to define the optimum ratio of payload mass to total launch vehicle mass for given space missions having specified velocity (or energy) requirements. The equations can only be solved through an iterative process, with the mathematical relationships being programmed for solution by a digital computer. Comments on the computational algorithm, together with a flowchart, were previously provided.

Table 8.1 shows selected parameters corresponding to a four-stage solid propellant launch vehicle. These parameters consist of actual data for the four-stage Scout-G1 Vehicle. This vehicle is capable of placing a payload of 460-pounds into a circular orbit of 100 NM altitude [6].
Table 8.1 Key Characteristics for the Scout-G1 Four-Stage Solid Propellant Vehicle

<table>
<thead>
<tr>
<th></th>
<th>Stage 1</th>
<th>Stage 2</th>
<th>Stage 3</th>
<th>Stage 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>$I_{SP}$</td>
<td>244</td>
<td>280</td>
<td>295</td>
<td>288</td>
</tr>
<tr>
<td>$P_C$</td>
<td>450</td>
<td>700</td>
<td>700</td>
<td>670</td>
</tr>
<tr>
<td>Propellant</td>
<td>PBAN</td>
<td>CTPB</td>
<td>HTPB</td>
<td>CTPB</td>
</tr>
<tr>
<td>$\varepsilon$</td>
<td>6.5:1</td>
<td>21.2:1</td>
<td>58.8:1</td>
<td>50.5:1</td>
</tr>
<tr>
<td>$K$</td>
<td>0.160</td>
<td>0.229</td>
<td>0.215</td>
<td>0.151</td>
</tr>
</tbody>
</table>

Table 8.2 presents variations about the nominal values for structure factor and specific impulse. These nominal and variational values are used in the computation process leading to the results provided in Table 8.3.

Table 8.2 Variational Values for Parametric Study Based on Structure Factor ($K$) and Specific Impulse ($I_{SP}$) for the Scout-G1 Four-Stage Solid-Propellant Vehicle

<table>
<thead>
<tr>
<th>$K$ Excursions</th>
<th>Stage 1</th>
<th>Stage 2</th>
<th>Stage 3</th>
<th>Stage 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nominal</td>
<td>0.160</td>
<td>0.229</td>
<td>0.215</td>
<td>0.151</td>
</tr>
<tr>
<td>10% Reduction</td>
<td>0.140</td>
<td>0.206</td>
<td>0.194</td>
<td>0.136</td>
</tr>
<tr>
<td>20% Reduction</td>
<td>0.128</td>
<td>0.183</td>
<td>0.172</td>
<td>0.121</td>
</tr>
<tr>
<td>30% Reduction</td>
<td>0.112</td>
<td>0.160</td>
<td>0.151</td>
<td>0.106</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Variation in $I_{SP}$</th>
<th>Stage 1</th>
<th>Stage 2</th>
<th>Stage 3</th>
<th>Stage 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nominal</td>
<td>244</td>
<td>280</td>
<td>295</td>
<td>288</td>
</tr>
<tr>
<td>5% Increase</td>
<td>256</td>
<td>294</td>
<td>310</td>
<td>302</td>
</tr>
<tr>
<td>10% Increase</td>
<td>268</td>
<td>308</td>
<td>325</td>
<td>317</td>
</tr>
<tr>
<td>15% Increase</td>
<td>281</td>
<td>322</td>
<td>339</td>
<td>331</td>
</tr>
</tbody>
</table>

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The following table shows stage ratios and values for the ratio of payload mass to total launch vehicle mass ($\bar{M} / M_{g1}$) corresponding to the input combinations described in the notes at the end of this table.

**Table 8.3 Payload to Launch Vehicle Ratios and Other Stage Ratios
(*$V_r = 30,400$ ft/sec) for the Scout-G1 Four-Stage Solid-Propellant Vehicle**

<table>
<thead>
<tr>
<th>Case #</th>
<th>$M_{g2} / M_{g1}$</th>
<th>$M_{g3} / M_{g2}$</th>
<th>$M_{g4} / M_{g3}$</th>
<th>$\bar{M} / M_{g4}$</th>
<th>$\bar{M} / M_{g3}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.331</td>
<td>0.366</td>
<td>0.303</td>
<td>0.207</td>
<td>0.008</td>
</tr>
<tr>
<td>2</td>
<td>0.356</td>
<td>0.386</td>
<td>0.316</td>
<td>0.218</td>
<td>0.009</td>
</tr>
<tr>
<td>3</td>
<td>0.402</td>
<td>0.395</td>
<td>0.319</td>
<td>0.225</td>
<td>0.011</td>
</tr>
<tr>
<td>4</td>
<td>0.448</td>
<td>0.405</td>
<td>0.324</td>
<td>0.231</td>
<td>0.014</td>
</tr>
<tr>
<td>5</td>
<td>0.361</td>
<td>0.393</td>
<td>0.322</td>
<td>0.221</td>
<td>0.010</td>
</tr>
<tr>
<td>6</td>
<td>0.391</td>
<td>0.419</td>
<td>0.341</td>
<td>0.234</td>
<td>0.013</td>
</tr>
<tr>
<td>7</td>
<td>0.415</td>
<td>0.442</td>
<td>0.360</td>
<td>0.247</td>
<td>0.016</td>
</tr>
<tr>
<td>8</td>
<td>0.555</td>
<td>0.469</td>
<td>0.370</td>
<td>0.266</td>
<td>0.026</td>
</tr>
</tbody>
</table>

Case #1: nominal values for $K$ and $I_{sp}$

- #2: 10% reduction in $K$; nominal value for $I_{sp}$
- #3: 20% reduction in $K$; nominal value for $I_{sp}$
- #4: 30% reduction in $K$; nominal value for $I_{sp}$
- #5: 5% increase in $I_{sp}$; nominal value for $K$
- #6: 10% increase in $I_{sp}$; nominal value for $K$
- #7: 15% increase in $I_{sp}$; nominal value for $K$
- #8: 15% increase in $I_{sp}$; 30% reduction in $K$

*$V_r$ represents orbital velocity corresponding to a 100 NM circular orbit plus velocity losses due to gravity and aerodynamic drag.
Variations in the ratio of payload mass to total launch vehicle mass \( \frac{\tilde{M}}{M_{g_1}} \), for different values of structure factor and specific impulse, are evident from an examination of the data in the far right column of Table 8.3. These ratios vary from 0.8% to 2.6% for the range of values chosen for \( I_{sp} \) and \( K \). These values demonstrate just how important it is, in terms of payload mass delivered to a given orbit, to obtain optimum values for \( I_{sp} \) and \( K \), as well as optimum staging in overall launch vehicle design. Methods have thus been provided to measure the relevance of these parameters.

Table 8.4 below lists selected parameters corresponding to a two-stage liquid propellant launch vehicle. These parameters consist of actual data for the Delta IV-M Launch Vehicle. This vehicle is capable of placing a payload of 15,150 pounds into a circular orbit of 100 NM altitude [6].

**Table 8.4 Key Characteristics of a Typical Two-Stage Liquid Propellant Vehicle**

<table>
<thead>
<tr>
<th></th>
<th>Stage 1</th>
<th>Stage 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>( I_{sp} ) (sec)</td>
<td>383</td>
<td>462</td>
</tr>
<tr>
<td>( P_c ) (psi)</td>
<td>1410</td>
<td>465</td>
</tr>
<tr>
<td>Prop.</td>
<td>LOX/LH2</td>
<td>LOX/LH2</td>
</tr>
<tr>
<td>( \varepsilon )</td>
<td>21.5:1</td>
<td>285:1</td>
</tr>
<tr>
<td>( K )</td>
<td>0.12</td>
<td>0.16</td>
</tr>
</tbody>
</table>

Table 8.5 lists variations about the nominal values for structure factor and specific impulse for a liquid-propellant launch vehicle. The nominal and variational values are used in the computation process leading to the results provided in the subsequent Table 8.6.
Table 8.5 Variational Values for Parametrics Study Based on Structure Factor \((K)\) and Specific Impulse \((I_{SP})\) for a Two-Stage Liquid-Propellant Vehicle

<table>
<thead>
<tr>
<th>Variation in (K)</th>
<th>Stage 1</th>
<th>Stage 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nominal</td>
<td>0.120</td>
<td>0.160</td>
</tr>
<tr>
<td>10% Reduction</td>
<td>0.108</td>
<td>0.144</td>
</tr>
<tr>
<td>20% Reduction</td>
<td>0.096</td>
<td>0.128</td>
</tr>
<tr>
<td>30% Reduction</td>
<td>0.084</td>
<td>0.112</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Variation of (I_{SP})</th>
<th>Stage 1</th>
<th>Stage 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nominal</td>
<td>383</td>
<td>462</td>
</tr>
<tr>
<td>5% Increase</td>
<td>402</td>
<td>485</td>
</tr>
<tr>
<td>10% Increase</td>
<td>421</td>
<td>508</td>
</tr>
<tr>
<td>15% Increase</td>
<td>440</td>
<td>531</td>
</tr>
</tbody>
</table>

Table 8.6 shows the computed stage ratios and values for the ratio of payload mass to total launch vehicle mass \((\overline{M} / M_{gl})\) corresponding to the input combinations described in the notes at the end of the table.
Table 8.6 Two-Stage Liquid Propellant Vehicle Values for the Ratio of Payload Mass to Vehicle Mass (*$V_r = 30,400$ ft/sec)

<table>
<thead>
<tr>
<th>Case #</th>
<th>$M_{g2}/M_{g1}$</th>
<th>$\overline{M}/M_{g2}$</th>
<th>$\overline{M}/M_{g1}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.228</td>
<td>0.205</td>
<td>0.047</td>
</tr>
<tr>
<td>2</td>
<td>0.248</td>
<td>0.211</td>
<td>0.052</td>
</tr>
<tr>
<td>3</td>
<td>0.270</td>
<td>0.216</td>
<td>0.058</td>
</tr>
<tr>
<td>4</td>
<td>0.296</td>
<td>0.217</td>
<td>0.064</td>
</tr>
<tr>
<td>5</td>
<td>0.253</td>
<td>0.222</td>
<td>0.056</td>
</tr>
<tr>
<td>6</td>
<td>0.277</td>
<td>0.238</td>
<td>0.066</td>
</tr>
<tr>
<td>7</td>
<td>0.301</td>
<td>0.253</td>
<td>0.076</td>
</tr>
<tr>
<td>8</td>
<td>0.381</td>
<td>0.254</td>
<td>0.097</td>
</tr>
</tbody>
</table>

Case # 1: nominal values for $K$ and $I_{SP}$

# 2: 10% reduction in $K$; nominal value for $I_{SP}$

# 3: 20% reduction in $K$; nominal value for $I_{SP}$

# 4: 30% reduction in $K$; nominal value for $I_{SP}$

# 5: 5% increase in $I_{SP}$; nominal value for $K$

# 6: 10% increase in $I_{SP}$; nominal value for $K$

# 7: 15% increase in $I_{SP}$; nominal value for $K$

# 8: 15% increase in $I_{SP}$; 30% reduction in $K$

*$V_r$ represents orbital velocity corresponding to a 100 NM circular orbit plus velocity losses due to gravity and aerodynamic drag.
The tables and figures in Chapters III through VIII provide definitions and data of the general type used as inputs in the modeling and computational process, the results of which are shown in Chapter IX. Information is provided in Chapters III through VIII on structure factor, specific impulse, velocity losses, propellant characteristics, chamber pressure and other vehicle characteristics. The data shown are representative of both United States and international launch vehicles.

For modeling the solid-propellant vehicle (Scout-G1), the type of propellant utilized, as shown in Figure 3.1, is a composite employing a mixture of separate molecules of oxidizer and fuel. Table 8.1 lists nominal characteristics used as a starting point in modeling this solid-propellant launch vehicle. Table 8.2 shows variational values (about the nominal) that are used in the computations.

The model of the liquid propellant vehicle (Delta IV-M) used in the computational process for the present research employed the type of propellant shown in Figure 3.1 as a bipropellant, wherein oxidizer and fuel are injected separately. Detailed characteristics of propellants for the liquid propellant vehicle; i.e. liquid oxygen and hydrogen, are shown in Tables 3.1 and 3.2. These include variables such as molecular weight, mixture ratio, bulk density, etc. Table 8.4 lists numerical values for the nominal variables. The values used are representative of high-performance liquid-propellant space launch vehicles. Table 8.5 shows variational values (about the nominal) used in the computational process.
CHAPTER IX

RESULTS AND DISCUSSION

Tables 9.1 and 9.2 and Figures. 9.1-9.4 show the changes in the magnitude of payload weight to be realized when specified changes are made in propulsion; i.e., specific impulse and/or structure factor. Examples provided are obtained through performance calculations based on variations in specific impulse and structure factor, as compared to the nominal characteristics of a two-stage liquid-propulsion launch vehicle and a four-stage solid-propellant launch vehicle [5].

From the foregoing, it has been seen that significant gains in payload weight can be achieved through modest to substantial changes in specific impulse and structure factor. As an example, for a four-stage solid propellant space launch vehicle, and using the above tables, a 33% gain in payload weight can be achieved by increasing specific impulse by only 5%. As a second example, for a two-stage liquid-propellant launch vehicle, and using the above tables, a 41% gain in payload weight can be achieved through an increase in specific impulse of 10%.

The following relations thus become apparent:

\[
\frac{\partial P_l}{\partial I_{sp}} = 3 \left( \frac{\partial P_l}{\partial K} \right) \quad \text{(four-stage solid propellant)}
\]

\[
\frac{\partial P_l}{\partial I_{sp}} = 3.4 \left( \frac{\partial P_l}{\partial K} \right) \quad \text{(two-stage liquid propellant)}
\]

That is, for a typical four-stage solid propellant launch vehicle, in terms of potential increases in payload weight, improvements in specific impulse are more effective than structure factor by a factor of three. For a typical two-stage liquid-propellant launch vehicle, the same comparison yields a factor of 3.4.
Table 9.1 Payload Gain as Function of Changes in Structure Factor and Specific Impulse for a Two-Stage Liquid-Propellant Launch Vehicle

<table>
<thead>
<tr>
<th>Change in $K$</th>
<th>Gain in Payload Wt.</th>
</tr>
</thead>
<tbody>
<tr>
<td>- 10%</td>
<td>+ 12%</td>
</tr>
<tr>
<td>- 20%</td>
<td>+ 25%</td>
</tr>
<tr>
<td>- 30%</td>
<td>+ 38%</td>
</tr>
<tr>
<td>Change in $I_{sp}$</td>
<td></td>
</tr>
<tr>
<td>+ 5%</td>
<td>+ 20%</td>
</tr>
<tr>
<td>+ 10%</td>
<td>+ 41%</td>
</tr>
<tr>
<td>+ 15%</td>
<td>+ 63%</td>
</tr>
<tr>
<td>Change in $I_{sp}$ and $K$</td>
<td></td>
</tr>
<tr>
<td>+ 15% in $I_{sp}$ and</td>
<td>+ 107%</td>
</tr>
<tr>
<td>- 30% in $K$</td>
<td></td>
</tr>
</tbody>
</table>

Table 9.2 Payload Gain as Function of Changes in Structure Factor and Specific Impulse for a Four-Stage Solid Propellant Launch Vehicle

<table>
<thead>
<tr>
<th>Change in $K$</th>
<th>Gain in Payload Wt.</th>
</tr>
</thead>
<tbody>
<tr>
<td>- 10%</td>
<td>+ 24%</td>
</tr>
<tr>
<td>- 20%</td>
<td>+ 50%</td>
</tr>
<tr>
<td>- 30%</td>
<td>+ 79%</td>
</tr>
<tr>
<td>Change in $I_{sp}$</td>
<td></td>
</tr>
<tr>
<td>+ 5%</td>
<td>+ 33%</td>
</tr>
<tr>
<td>+ 10%</td>
<td>+ 72%</td>
</tr>
<tr>
<td>+ 15%</td>
<td>+ 115%</td>
</tr>
<tr>
<td>Change in $I_{sp}$ and $K$</td>
<td></td>
</tr>
<tr>
<td>+ 15% in $I_{sp}$ and</td>
<td></td>
</tr>
<tr>
<td>- 30% in $K$</td>
<td>+ 236%</td>
</tr>
</tbody>
</table>
Figure 9.1 Gain in P/L Wt. for a Four-Stage Launch Vehicle (Solid Propellant)
\[ \Delta P/L = f(I_{sp}, k) \]
Figure 9.2 Gain in $P/L$ Wt. for a Two-Stage Launch Vehicle (Liquid Propellant)

$$\Delta P/L = f(I_{sp}, k)$$

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Figure 9.3 Gain in $P/L$ Wt. for a Four-Stage Launch Vehicle (Solid Propellant)

$\Delta P/L = f(\dot{k})$
It should be noted that in the previous results, the parameters of specific impulse and structure factor were assumed to be increased by given amounts in all stages simultaneously. Through additional analysis, as outlined in this report, these parameters can be varied in each stage separately, so as to observe the gains to be achieved when a change might be made in any given stage or stages, leaving other stages unchanged. Using the methods outlined in this report, a designer can utilize the work to evaluate payload gains when upgrades are made to a given vehicle either one stage at a time or while using any combination of these parameters and stages.

In addition to the above analysis, the approach given in this report can be applied to original launch vehicle design. In so doing, it may be desirable to factor in values from the earlier tables that provide characteristics of propellants and utilizing given equations for combining fuel and oxidizer in the optimum or near-stoichiometric quantities.

Using the same relationships, the approach can also be used to study the effects of changes in specific impulse and structure factor with regard to reducing launch vehicle size and mass for specified orbital mission requirements while maintaining constant payload mass.
CHAPTER X

CONCLUDING REMARKS

A study has been carried out to research and establish a method for defining those characteristics of a space launch vehicle that are critical in terms of a vehicle’s ability to place payloads of maximum mass (or weight) into an orbit or trajectory of specified characteristics. A second objective of the research was to establish a method for determining the effect of incremental changes in these characteristics on the vehicle’s capability for placing payloads of increased mass (or weight) into specified orbits. A third objective was to determine if the method that was researched and established could also be used to determine minimum launch vehicle size or mass that could place a payload of given mass into a specified orbit. Similarly, a method was researched and established to determine the effect of incremental changes in the vehicle’s characteristics on changes in the launch vehicle’s total mass (or weight).

The characteristics of a launch vehicle, which are critical to the research objectives of this report, have been investigated in a systematic manner. First, an approach was researched that will allow the vehicle’s total composition to be modeled. The complexity of the model is increased when considering that launch vehicles must be multi-stage in order to provide the required energy to achieve orbital velocities. Secondly, relationships are researched to determine the energy (or velocity) requirements that correspond to the class of orbits utilized in the research. Also, a method is researched and established that allows velocity losses due to gravity and aerodynamic drag to be modeled. Orbital
velocities and velocity losses due to gravity plus losses due to aerodynamic drag are summed so as to establish the total velocity requirement for specified missions.

The research undertaken determined that the payload capability of a space launch vehicle, or, conversely, the vehicle total liftoff mass, is highly sensitive to the manner in which the space launch vehicle is staged. The research has led to the development and programming of a model for determining optimum staging relationships for given mission requirements. The research then utilized this optimum staging algorithm as a means of computing the sensitivities of the vehicle’s payload and liftoff mass to variations in the vehicle’s key propulsion and related performance parameters.

Recommendations for future research into the field might include research into optimum types of vehicles and their characteristics that could be utilized to achieve a wide array of future space missions. This could include, but would not necessarily be limited to, future lunar missions and flights to the inner planets (Mercury, Venus, and Mars). This research could involve types of propulsion systems well beyond the chemical propulsion systems that are researched in this report, e.g. solar, nuclear, electric, ionic, etc. Long term interplanetary flight, particularly as it relates to manned space flight, raises demanding issues, many of which have not yet been fully addressed within the scientific community.
REFERENCES


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APPENDIX A

FLOWCHART FOR COMPUTER PROGRAM

The symbols appearing in Figure A.1 are defined in the List of Symbols appearing on pages x through xv and page 45 of the report. A small computer program was designed by the author to solve the equations. The solution is arrived at by making an initial estimate for the ratio \( \frac{M_{g2}}{M_{g1}} \) in Equation (7.9a). The value thus computed for \( \frac{M_{g3}}{M_{g2}} \) is then substituted in Equation (7.9b). The value found for \( \frac{M_{g4}}{M_{g3}} \) is substituted into Equation (7.9c). The ratios for \( \frac{M_{g2}}{M_{g1}} \), \( \frac{M_{g3}}{M_{g2}} \), \( \frac{M_{g4}}{M_{g3}} \), and \( \frac{M}{M_{g4}} \) are substituted into Equation (7.10). If the value thus computed for the right side of Equation (7.10) does not match the mission velocity requirement \( V_r \) (i.e. the left side of the equation), a reiteration is accomplished. This iterative process is continued until the right side of Equation (7.10) matches the mission requirement velocity \( V_r \) (left side of the equation).

When this match is achieved, the solved values for \( \frac{M_{g2}}{M_{g1}} \), \( \frac{M_{g3}}{M_{g2}} \), \( \frac{M_{g4}}{M_{g3}} \) and \( \frac{M}{M_{g4}} \) are multiplied together to give the solved value of \( \frac{M}{M_{g1}} \). This last term is the value that we are looking for, i.e. the ratio of launch vehicle payload weight to launch vehicle liftoff weight.
The equations in Chapter VII were developed by the author through a mathematical process, fully explained in the chapter, so that the value of $\frac{\bar{M}}{M_{g1}}$ corresponds to a typical space launch vehicle. A test is simply conducted, as shown in the flow chart, to determine if the right side of Equation (7.10) matches the assigned value for $V_r$ on the left side of Equation (7.10). The iterations are simply continued until a match is achieved. The program is set up to achieve answers out to the tenth decimal place. The author has verified the correctness of the computations. A printout of the C++ program coding is included immediately after the flowchart.
IF $V_{rc} > V_r$

**COMPUTE** $V_{rc}$

WHEN $V_{rc} = V_r$, **STOP**

**INPUT**
- $I_{sp}$, $K$, $V_r$, #STAGES

**CALL COMPUTATIONAL ALGORITHM**

**INITIAL ESTIMATE**
- $\frac{M_{g2}}{M_{g1}}$

**COMPUTE VALUES**
- FOR $\frac{M_{g3}}{M_{g2}}$, $\frac{M_{g4}}{M_{g3}}$, $\frac{M}{M_{g1}}$

IF $V_{rc} < V_r$

**INCREMENT** $\frac{M_{g2}}{M_{g1}}$

**OPTIMUM VALUE**
- $\frac{M}{M_{g1}} = \frac{M_{g2}}{M_{g1}} \cdot \frac{M_{g3}}{M_{g2}} \cdot \frac{M_{g4}}{M_{g3}} \cdot \frac{M}{M_{g4}}$

**Figure A.1 Flowchart for Computer Program**

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#include<iostream>
#include<iomanip>
#include<cmath>
#include<sstream>
#include<fstream>
#include<string>
#include<vector>
using namespace std;

const double GRAVITY = 32.1739;

double Mg[5], K[4] = {0.1, 0.1, 0.1, 0.1}, C[4], ISP[4] = {260.0, 290.0, 300.0, 300.0}, ratio[5];
double VRESTIMATE = 50400.0000, Vr = 0.000000, offset = 0.0000001;

double mcalc(int i)
{
    double top, bottom, spurge;
    spurge = (C[i+1] * K[i]) * (1 - K[i+1]) / ratio[i];
    top = C[i] * K[i+1] * (1 - K[i]);
    bottom = (1 - K[i]) * (1 - K[i+1]) * (C[i+1] - C[i]) + spurge;
    return top / bottom;
}

double vcalc(int i)
{
    double right, left;
    right = 1 / (K[i] + C[i] * ratio[i]);
    left = C[i] * log(right);
    return left;
}

double calculations()
{
    C[0] = ISP[0] * GRAVITY;
    ratio[1] = mcalc(0);
    ratio[2] = mcalc(1);
    ratio[3] = mcalc(2);
    for(int i = 0; i < 4; ++i)
        Vr += vcalc(i);
    return Vr;
}

bool check(const string & in)
{
    if(in[0] != '#')
        return true;
    else
        return false;
}

double str_to_db1(const string & in)

Figure A.2 Source Code in C++ Language
rocket.cpp
{
    double returned;
    cout << in;
    ofstream out("tmp_rocket.txt");
    if(!out.is_open())
    {
        cout << "Could not open the temp file tmp_rocket.txt for writing\n";
        exit(1);
    }
    out << in;
    out.close();
    ifstream infile("tmp_rocket.txt");
    if(!infile.is_open())
    {
        cout << "Could not open the temp file tmp_rocket.txt for reading\n";
        exit(1);
    }
    infile >> returned;
    infile.close();
system("del tmp_rocket.txt");
    return returned;
}

void read_input()
{
    string file = "input.txt";
    string input;
    vector<double> vars;
    ifstream InFile("input.txt");
    if(!InFile.is_open())
    {
        cout << "Could not find the input file input.txt\n";
        exit(1);
    }

    //todo: fix this! it only gets the last item in the file! bah!
    while(getline(InFile, input))
    {
        cout << input << endl;
        if(check(input))
            vars.push_back(str_to_dbl(input));
    }
    InFile.close();
    K[0] = vars[0];
    K[3] = vars[1];
    K[2] = vars[2];
    K[5] = vars[3];
    ISP[0] = vars[4];
    ISP[1] = vars[5];
    ISP[2] = vars[6];
    ISP[3] = vars[7];
    VRESTIMATE = vars[8];
}

Figure A.2 Continued
int main()
{
    //todo: make ratio[0] more accurate, user input, mbar/ml
    double VrReturned = 0.0000000, VrLast = 0.0000000;
    //double offset = 0.0000001;
    double diff = 2, one_minus_diff;
    read_input();
    //ratio[0] = 0.43;
    cout << "Ratio: " << ratio[0] << endl;
    VrReturned = calculations();
    one_minus_diff = 1 - diff;
    while(one_minus_diff != 0)
    {
        //cout << "Init: " "ratio[0] " "\t" "VrRet: ";
        diff = VrReturned / VRESTIMATE;
        one_minus_diff = 1 - diff;
        //cout << VrReturned << "\t" << diff << "\t" << diff << "\t" << diff << "\t"
        if(one_minus_diff > 0 && (one_minus_diff < offset))
            break;
        if((one_minus_diff < 0 && (one_minus_diff > -offset))
            break;

        if(diff > 1)
            ratio[0] += offset;
        else
            ratio[0] -= offset;

        VrLast = VrReturned;
        VrReturned = calculations();
        Vr = 0;
    }
    cout << "Init: " "ratio[0] " "\t" "VrRet: " "VrReturned " "\t" "VrLast:" " VrLast << "\tdiff: " "diff " "endl;
    cout << "The end result: " "(VrLast + VrReturned) / 2 " "endl;
    cout << "The value for MBar is: " "ratio[4] " "endl;
    return 0;
}

Figure A.2 Continued
APPENDIX B

ROCKET MOTOR PERFORMANCE DETERMINATION FROM TELEMETERED FLIGHT DATA

The following describes a computer program designed by the author for the purpose of determining rocket motor performance from telemetered flight data. It is included here to illustrate how rocket propulsion parameters affect overall launch vehicle performance.

Vehicle weight, consumable weight remaining, chamber pressure integral, dynamic pressure, aerodynamic drag, aerodynamic drag impulse, jet vane drag, vane drag impulse, thrust, including a conversion to vacuum conditions, and total impulse, are calculated as a function of time.

Various options in the program provide maximum flexibility. The usual method of determining performance utilized linear acceleration. When this method is used, a weight-time history is calculated internally by a tabular input of chamber pressure. When chamber pressure is not available, an option in the program makes it possible to insert a tabular weight-time history. Should chamber pressure be available, but linear acceleration not available, another option makes it possible to calculate motor performance from nozzle throat area, and nozzle divergence, discharge, and thrust coefficients.

In calculating motor performance where aerodynamic drag is significant, the program utilizes a tabular velocity-time input and a tabular input of $C_D S$ as a function of Mach number. Atmospheric properties are calculated by a subroutine based on information in the 1959 ARDC model atmosphere. A tabular altitude-time history serves as an input table for this subroutine.
Symbols used are as follows:

- $A_e$ Nozzle exit area in ft$^2$
- $A_L$ Linear accelerometer reading in “g” units tabular vs time
- $A_t$ Nozzle throat area in (inches)$^2$, tabular vs time
- $C_D$ Drag coefficient, tabular vs Mach number
- $C_N$ Nozzle discharge coefficient
- $C_T$ Vacuum thrust coefficient
- $D_A$ Aerodynamic drag in lbs
- $D_V$ Jet van drag in lbs
- $H$ Altitude in feet, tabular vs time
- $I_v$ Jet vane impulse in lb-sec
- $I_v$ Vacuum impulse in lb-sec
- $I_3$ Vacuum impulse in lb-sec without jet vanes attached
- $I_4$ Vacuum impulse in lb-sec with jet vanes attached
- $K_1$ Vehicle weight in lbs in rocket motor (propellant plus inhibitor, etc.)
- $K_3$ Slope of control fuel curve in lb/sec
- $K_4$ Ratio of jet vane drag to vacuum thrust with vanes attached
- $K_5$ Ratio of jet vane drag to vacuum thrust without vanes attached
- $M$ Mach number, may be tabular vs time
- $t_n$ Upper limit of integration in seconds
- $P_A$ Ambient atmospheric pressure in lb/ft$^2$ as computed by ARDC 1959 tables
- $P_C$ Chamber pressure in lbs/in$^2$, tabular vs time
- $q$ Dynamic pressure in lb/ft$^2$
$S$ Effective aerodynamic area in ft$^2$

$S_1$ Chamber pressure integral in lb-sec/in$^2$

$T_1$ Thrust in lbs at ambient conditions with jet vanes attached

$T_2$ Thrust in lbs at vacuum conditions with jet vanes attached

$T_3$ Thrust in lbs at vacuum conditions without jet vanes attached

$T_4$ Thrust in lbs at vacuum conditions with jet vanes attached (calculated by interior ballistic method)

$T_5$ Thrust in lbs at vacuum conditions without jet vanes attached (calculated by interior ballistic method)

$V_A$ Velocity of rocket in ft/sec with respect to surrounding medium (which rotates with earth).

$V_S$ Local velocity of sound in ft/sec as a computed from 1959 ARDC tables

$W_1$ Vehicle weight in lbs at any time $t$

$W_2$ Consumable weight in lbs (propellant plus inhibitor, etc.) remaining in rocket motor at any time $t$. Tabular vs time

$\rho$ Ambient atmospheric density in slugs/ft$^3$ as computed from 1959 ARDC tables

$\lambda$ Nozzle divergence coefficient

Note that some of the symbols used in this appendix may differ, or be supplementary to, those used in the main text.

Equations:

\[ \int_0^t P_c \, dt \] \hspace{1cm} (B.1)

\[ S_1 = \int_0^t P_c \, dt \] \hspace{1cm} (B.2)

\[ W_2 = K_2 \left[ 1 - \frac{\int_0^t P_c \, dt}{\int_0^t P_c \, dt} \right] \] \hspace{1cm} (B.3)
\[ W_i = K_1 - K_2 + W_2 - K_3 t \quad \text{(B.4)} \]
\[ \bar{q} = 1/2 \; \rho V_A^2 \quad \text{(B.5)} \]
\[ M = \frac{V_A}{V_S} \quad \text{(B.6)} \]
\[ \bar{q} = 1/2 (MV_S)^2 \quad \text{(B.7)} \]
\[ D_A = \bar{q} C_D S \quad \text{(B.8)} \]
\[ T_A = \int_0^t D_A \, dt \quad \text{(B.9)} \]
\[ T_1 = W_i A_L + D_A \quad \text{(B.10)} \]
\[ T_2 = T_1 + P_A A_E \quad \text{(B.11)} \]
\[ I_2 = \int_0^t T_2 \, dt \quad \text{(B.12)} \]
\[ D_V = K_4 T_2 \quad \text{(B.13)} \]
\[ I_V = \int_0^t D_V \, dt \quad \text{(B.14)} \]
\[ T_3 = T_2 + D_V \quad \text{(B.15)} \]
\[ I_3 = \int_0^t T_3 \, dt \quad \text{(B.16)} \]
\[ T_5 = P_C A_t \lambda C_N C_T \quad \text{(B.17)} \]
\[ I_5 = \lambda C_N C_T \int_0^t P_C A_t \, dt \quad \text{(B.18)} \]
\[ T_4 = T_5 (1 - K_5) \quad \text{(B.19)} \]
\[ I_4 = \int_0^t T_4 \, dt \quad \text{(B.20)} \]
Options:

(1) $P_C$ will be tabular input against time, and $W_1$ will be calculated (and printed out) from Equations (B.1), (B.2), and (B.4).

(2) $W_2$ will be a tabular input against time. $W_1$ will then be calculated from Equation (B.4) alone. $P_C$ will not appear in input.

(3) When option (1) is in use, fix $S_1$ so that it may or may not be shown in print-out, as optionally desired.

(4) If an altitude ($H$) vs time table is not included in input, and $V_S$ should not be picked up from ARDC tables, and $\bar{q}$ from Equation (B.5) or (B.6) should be printed out as zero. $P_a$ not to be picked up from ARDC tables (making the $P_aA_e$ term in Equation (B.1) = 0).

(5) In tabular input of $H$ vs time, $H$ can be in ft or nautical miles. If $H$ is in nautical miles it is converted to feet from the relationship $H_{ft} = (6076.1033) H_{nm}$.

(6) A tabular input of $V_A$ vs time will be included as input and $\bar{q}$ calculated from Equation (B.5).

(7) A tabular input of $M$ vs time in input and $\bar{q}$ calculated from Equation (B.6). When this option is in use, $V_S$ may either be picked up from ARDC tables or specified as a constant.
VITA

RUSSELL H. EDWARDS

EDUCATION

Doctor of Philosophy, Old Dominion University, Norfolk, Virginia, December 2007

Master of Science, University of Dayton, Dayton, Ohio, 1975

Bachelor of Science, Carson Newman College, Jefferson City, Tennessee, 1961

Graduate, Industrial College of the Armed Forces, National Defense University, Washington, DC, 1982

EXPERIENCE

Employed for 30 plus years as an Aerospace Engineer and Physical Scientist with NASA and organizations in the Department of Defense. Assignments include service with the Air Force Systems Command, Wright-Patterson AFB, OH; NASA Langley Research Center, Hampton, VA; NASA Headquarters, Washington, DC; and other assignments with the Department of Defense, Washington, DC.

Assignments include broad background and experience in detailed aspects of missile and space systems design, flight-testing, and operation. Authored a number of official DOD reports, technical briefs, and working papers dealing with scientific and engineering details of missiles and space systems. Served four years in military service with U.S. Air Force in U.S. and overseas assignments.

PROFESSIONAL MEMBERSHIPS

American Institute of Aeronautics and Astronautics

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