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SOLID ROCKET MOTOR CONCEPTUAL DESIGN:

THE DEVELOPMENT OF A DESIGN OPTIMIZATION
EXPERT SYSTEM WITH A
HYPERTEXT USER INTERFACE

A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville

James Brill Clegern

May, 1993

This thesis is dedicated to my parents

Mr. Robert W. Clegern

and

Mrs. Carol K. Towley

who have given me the support and encouragement throughout the years.

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I am very grateful for the Expert System and Hypertext work experience I gained while employed at ERC, Inc. in the University of Tennessee Space Institute Research Park. Under the tutorage of Mr. William Andes Hoyt, Jr. I gained much of my Hypertext and Expert Systems background which I applied in this thesis research.

ABSTRACT

Solid rocket motor (SRM) design prototypes can be rapidly formulated and evaluated by the use of advanced computer-based methodologies that apply expert system and artificial intelligence software to the SRM design optimization processes. The research program that was carried out, and is reported in this thesis, was to formulate a computer-based SRM expert system for motor design and optimization, with the assistance of a hypertext software algorithm that provides a user-friendly interface. With this interface for parameter input, the design engineer can quickly obtain rocket motor designs that satisfy the perfomance mission of the SRM, as well as meet criteria for optimized (minimum) motor mass. The computer-based software has been designated as the Solid Rocket Motor Conceptual Design Optimization System (SRM-CDOS).

SRM-CDOS is a computerized tool to facilitate and define tradeoffs between various aspects of the SRM design and optimization requirements. The main purpose of this SRM design system is to aide the SRM design engineer in making the best initial design selections and thereby reducing the overall "design cycle time" of a project.

SRM-CDOS combines a user-interfacing hypertext control shell, a fortran optimization code, and an expert system artificial intelligence database to speed up the design process and maintain a user-friendly computing environment. The utilization of the unique combination these three computer methodologies has been demonstrated with additional recommendations for future applications.

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INTRODUCTION

The basic methods for theoretical Solid Rocket Motor (SRM) design are well known throughout the aerospace community. They involve the detailed selection of structural materials, solid propellants, and geometric parameters to satisfy the performance mission specified for the rocket motor. The SRM design engineer is confronted with significant difficulties in selecting the best combinations of rocket motor materials, propellants, and geometric parameters when the rocket design must incorporate either minimum operational mass, or minimum operational cost in its design optimization. To aide the SRM design engineer, an advanced methodology was needed to help select the best design combinations while allowing the user complete control over the design process.

Current personal computer (PC) systems and programing methods offer the means to complete the user controlled selection process for the best SRM design. PC systems allow placement of extensive solid rocket design criteria into a compact and user-friendly format that facilitates a fast design turnaround time. Building the software packages to incorporate the best design methodologies, design parameters, and the most complete database is a significant challenge. Three such software packages were designed by the author. The three packages were linked together to demonstrate the advantages of a user-computer interactive SRM design process.

The combined computer software package was designated the Solid Rocket Motor Conceptual Design Optimization System (SRM-CDOS). It was developed to be a rapid prototyping tool to facilitate and define tradeoffs between various aspects of the SRM design and optimization requirements.

SRM-CDOS consists of three prototype program packages implemented on a Macintosh IIsi personal computer system:

1. Hypertext User Interface
2. SRM Fortran Optimizer Code
3. Expert System Knowledge Base

Together, these three computer packages form the basis of operation for the prototype SRM-CDOS. Figure 1 illustrates the three computer technology packages.

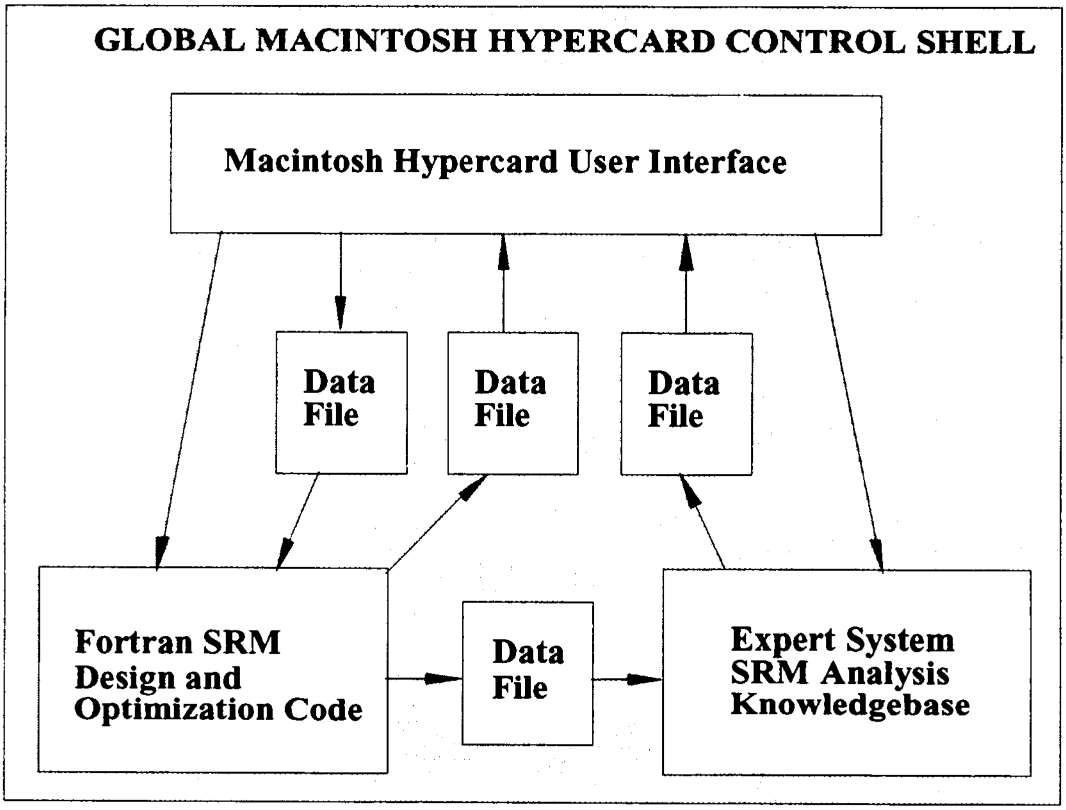

FIGURE 1 Overview of the Program Linking in the SRM-CDOS

Before describing the SRM-CDOS programs, background on the design theory and database development needs to be addressed. Chapter 1 provides an overview of current SRM design methodologies, including, design background, flow modeling, and current design optimizations. Chapter 2 provides the ways and means for developing the design selections and component databases.

Chapters 3, 4, and 5 describe the three program packages which make up the SRM-CDOS. Chapter 3 introduces the use of Apple's Macintosh HyperCard hypertext scripting language to create a computer-user interactive station controlling of the SRM design criteria into the SRM-CDOS files. The hypertext structure provides a user friendly computing environment which coaches the user for needed information and provides detailed help files and access to on-line SRM design reference material. The SRM design engineer specifies the type of optimization desired, inputs multiple ranges of the SRM design variables, selects custom costing options, chooses the solid propellant types to be examined, and picks the structural materials to be considered in the construction of the motor. The HyperCard system allows the engineer ready access to design data for changes, as well as visually enhanced, user-oriented displays of the design results. HyperCard also acts as the control shell for the entire project, directing input, output, and outside programs running in the computer background.

Chapter 4 describes the fortran design / optimization program which utilizes the user-entered design criteria from the HyperCard program. The program uses a multi-dimensional array computational search routine based on detailed SRM information, as well as five main SRM design variables input by the design engineer to generate a mass optimized SRM conceptual design. These five main SRM design variables consist of:

1) Solid Propellant Selection
2) Structural Material Selection
3) Chamber Pressure
4) Motor Diameter
5) Nozzle Half Angle

The program determines the best SRM design based on the optimum operational mass. Calculations are used to demonstrate interactions between each design variable, and to illustrate how these interactions impact the final SRM mass estimations. Sections of the design code can be modified by future SRM-CDOS users to incorporate additional, new, or advanced design methods or data for the performance predictions. Further detailed studies of each SRM component design, which are beyond the scope and purpose of the present configuration of SRM-CDOS, can be completed later by the engineering staff based on the initially optimized design.

Chapter 5 describes the SRM-CDOS expert system knowledge base. The expert system contains a limited engineering SRM design knowledge domain in a backward chaining IF-THEN rule format. The rules apply the stored knowledge in a consistent manner for each design the system analyzes. The expert system examines the optimized SRM design and recommends additional design selections involving propellant grain patterns, igniter systems, and thrust vectoring methods. The expert system used in the SRM-CDOS was created using Macintosh expert system Level5 software.

Chapters 6 and 7 detail the SRM-CDOS design runs compared to existing motor designs. Chapter six describes the design matching process comparing SRM-CDOS designs to existing designs. This process serves as a validation of both the design codes and the expert system knowledge base. The design validation shows that the SRM-CDOS can match an existing design to within a specified 5% accuracy. Additional runs are made to demonstrate the potential mass savings by having an optimized design solution. Chapter seven presents an overview of the results in chapter six and offers potential improvement areas for the future modifications.

The SRM-CDOS project provides a working example of three key emerging computer based technologies interacting within the same programing environment. The hypertext user-computer interactive interface allows extensive data entry, results display, and topic referencing in an easy-to-use, user friendly environment. The fortran-coded SRM design optimization procedure provides the best number crunching environment, while reducing the amount of data entry and computational time required for a conceptual SRM design. The expert system provides a permanent knowledge base of SRM design criteria which can be updated as design methods improve. The three systems are linked and work together to provide an indispensable design tool that will provide the best relative SRM design, based on the design criteria entered.

The SRM-CDOS design methodology was based on guidance from open literature, including the NASA SP chemical propulsion series, and existing unclassified design texts. Data selection for the testing runs was based on existing unclassified SRM designs.

CHAPTER 1
SRM DESIGN METHODOLOGIES

SRM Design Background

Solid Rocket Motors (SRMs) represent a self contained, single-use propulsion device ideal for motor and booster applications which require a single use propulsion system. The major advantage of SRMs has proven to be the large amount of energy they can store in a simple, low volume package of solid propellants. The lower volume of a SRM tends to have a higher propellant mass ratio compared with similar liquid propellant systems. The propellant mass ratio of a motor is the propellant mass / total motor mass. The high propellant mass coupled with high energy density, simpler design, lower cost, and relatively safe storage capability have made the SRM an attractive alternative to complicated liquid propellant engines. By harnessing these advantages, solid propulsion systems have played major roles in the United States space program and national defense ever since the early 1950's.

SRM technology has developed rapidly in the past forty years, offering design engineers an increasingly wide selection of structural materials, solid propellants, and geometries. New metallic alloys and composite materials allow for increased strength with a lower structural mass. New, high energy, composite solid propellant combinations offer increased performance over older propellants. To incorporate the new technology advances, new techniques for SRM design need to be created from existing physical design and thermodynamic propulsion theories.

SRM Flow Modeling

New SRM design methods began to harness the computational power of personal, mini, and super computers to do flow modeling and design optimization. By programing the computers with the theories of quasi-one-dimensional flow, with isentropic expansions, adequate estimates of the SRM performance and design parameters can be attained. These design and performance estimations are usually within 5% of actual live fire testing of the production model (Williams, Barrere, & Huang, 1969).

Increasing the complexity of the design model by incorporating the Navier-Stokes boundary layer equations, detailed aerothermochemistry, or introducing correction factors into the quasi-one-dimensional flow equations, can improve the performance prediction by a few percent. Optimizations on the SRM design can assist in the best selections of the design model and geometric parameters to provide the best operational SRM .

Quasi-One-Dimensional Flow Theory

The quasi-one-dimensional flow theory of compressible fluid flow provides a basis for evaluation and comparison of the performance of rocket motors of all kinds. The theory incorporated into quasi-one-dimensional flow is the mechanics and thermofluiddynamics involved in the conversion of the propellant generated heat in the rocket motor combustion chamber to usable kinetic energy of the propulsion fluid in the rocket motor nozzle. This conversion of heat to kinetic energy forms the basis of the traditional formulas and characteristic parameters used in the design of rocket propulsion systems.

Quasi-one-dimensional theory incorporates mass conservation, momentum conservation, and energy conservation as the basis for, and derivation of, the flow equations. Simplifications applied to one-dimensional flow theory are the assumption of isentropic flow, and that the fluid is an ideal gas. Realistically, quasi-one-dimensional flow theory describes flow in a duct or channel based on the assumption that the flow is steady in time and uniform across each section of the duct. In real applications, neither of these basic assumptions are exactly true, since viscous phenomena and aerothermochemistry affect real rocket motor exhaust flow (Williams et al., 1969).

Quasi-one-dimensional flow theory is therefore an idealization of many important factors which may require a highly detailed approach to fully account for their involvement in the real rocket flow environment. The idealization of the full equations of aerothermochemistry tends to omit the effects of multi-component reacting gas flows, while the assumption of isentropic flow omits the reality of friction effects, heat transfer losses, shock losses, and the effects of laminar and turbulent regions in the flow. Correction factors applied to predicted quasi-one-dimensional flow result in reasonably accurate real flow approximations compared to experimental results (Sutton, 1985).

Navier-Stokes Boundary Layer Equations

The equations for the flow of Newtonian viscous fluids are derived from the general Navier-Stokes equations. Because of their mathematical complexity, applying simplifications and neglecting certain terms in the general equations allows for solutions to be attained for a useful range of problems. With computer systems programed with the simplified Navier-Stokes equations, the solution methodology of the equations has become known as Computational Fluid Dynamics (CFD). CFD codes have evolved in the past 20 years to provide accurate flow predictions including those of rocket motor performance (Hill & Peterson, 1970).

Using the simplified Navier-Stokes equations, CFD programs have been developed for 2-D and 3-D incompressible flow to provide quantitative descriptions of both laminar and turbulent boundary layers, with particular emphasis on skin friction, heat transfer, and flow separation within the SRM. The description of the boundary layers provides the extra details often needed for the accurate prediction of the SRM performance, but at an additional price of significantly increased computational analysis and time.

Aerothermochemistry

Real solid propellant systems deviate from the ideal behavior due to the complex nature of their propellant formulation. During combustion, the solid propellant releases many chemical species including C, H_2, N_2, O_2, H_2O, HCl, and Cl. These species have many different reaction paths, and therefore yield a highly complex reacting flow.

Combusting aluminized solid propellants will have varying amounts of liquid or particulate aluminum oxide, Al_2O_3, in the flow. This causes performance variations due to the two-phase flow effects of the particulates in the nozzle of the rocket compared to the surrounding gas flow. The final combustion combinations of the chemical species yield the mean molecular mass of the exhaust gases, as well as the flow exhaust velocity and exhaust temperature.

The complex thermochemistry of combusting solid propellants can be estimated by sampling exhaust of test fired motors or by complex thermochemistry programs. An example of such a program would be NASA's SP-273 code (Gorden, Stanford, and McBride, 1971) which is available on many mainframe computer systems. NASA's SP-273 code can provide highly accurate predictions of the final combustion products for a specific solid propellant combination. These predictions can be especially useful if the propellant is new and/or no data is available from previous solid propellant test firings.

Design Methodology Selected for the SRM-CDOS Project

In keeping with the original purpose of the SRM-CDOS (simplicity and practical application), the author opted to use the basic quasi-one-dimensional flow approximations with corrections for real rocket flows. The quasi-one-dimensional flow provides a mathematically simple means for developing a conceptual SRM design, while providing sufficiently accurate results of the flow characteristics for performance prediction of the SRM to within 5%. This accuracy is more than adequate for a rapid prototyping tool such as SRM-CDOS.

The real rocket flow correction factors are figured into the design code and can be adjustable by the user. Details of the propellant chemistry are determined from averaged experimental test data available in the open literature and will also be changeable by the program user as a design reference.

Current Design Optimizations

All SRM production firms, like United Technologies, Aerojet, Hercules, and Thiokol, in the United States of America, employ their own variations of design optimization schemes to assist in the SRM development. But, the data and methodology on these design methods are proprietary "in-house" secrets and tend to be unavailable to design engineers who are outside the original SRM firm (Roberson, 1992; Saaverdra, 1992). As a result, non-corporate or new corporate SRM design engineers must spend research time and funding to develop their own optimization methods, or they can rely on existing designs which may not fully meet the project requirements.

Current design optimization methods, presented by Trunchot (1988), Denost (1988), Hildreth (1988), and Sutton (1985) tend to center around specific areas of SRM design. Most of the published optimization information concerns motor case design, nozzles design, propellant formulation, and internal grain geometries. Very few authors have presented public domain work on multiple area design optimization.

In an initial SRM design paper presented by this author to the AIAA (Clegern, 1992), a multi-variable optimization scheme was described for a user interactive, automated computer application. The initial optimization scheme employed a varied propellant selection, structural material selection, and geometric parameter selection for the best operational mass SRM system. Details on the applied design system for this project will be described in chapter four of this thesis .

CHAPTER 2

DESIGN PARAMETERS AND DATA BASE DEVELOPMENT

In order for the SRM-CDOS computer system to be programed with the correct SRM design equations and optimization routines, the determination of the most effective SRM design parameters and critical information data bases were required. The SRM design parameters must be of significance to the overall design goals of the motor, and the information data bases must be accurate and updatable.

Current design methods and equations require a wide range of variables and parameters for a completed design. If a specific design goal is required, the field of design parameters can be narrowed. For example, for the goal of designing the SRM for the best operational mass, the author concluded after extensive study, that five main parameters emerge as dominate design criteria for a mass optimized motor:

1. Solid Propellant Selection
2. Structure Material Selection
3. Chamber Pressure
4. Motor Diameter
5. Motor Nozzle Half-Angle

Taking these five parameters, each was broken down into a range of values or component parts with which the SRM data base and optimization equations were developed. Exact details of the optimization method will be discussed in chapter four.

Solid Propellant Selection

The selection of the SRM's solid propellant plays a crucial role in the operational performance and final mass of the system. Three basic types of solid propellants are currently in use: double-base, composite, and composite-double-base (Sutton, 1985). Double-base propellants are the basic homogeneous combination of oxidizer and fuel in a mixture of nitrocellulose and nitroglycerine. Composite propellants are a modern heterogeneous mix of solid oxidizers, powdered aluminum, and a polymer binder. Composite double-base propellants are a combination of the previous two propellants with an added high energy nitramine, such as HMX (cyclotetramethilene tetranitramine).

Each propellant type has its own specific properties of density, burning temperature, burn rate, and combustion products while providing different chemical combinations and operational characteristics. Selection of the solid propellant determines not only the hazard rating of the motor (burning and deflagration vs. detonation, but also the motor's propulsion efficiency or specific impulse (Isp), the volume of propellant required, the casing structural design, and the nozzle structural design. Table 2.1 lists some of the various aerospace solid propellants and their properties used in current SRMs (NASA SP-8064, 1971; Sutton, 1985):

TABLE 2.1 Current Solid Propellants at 6.89 MPa (1000 psi) Operation

Solid Propellant	Density (kg/m^3)	Flame Temp (°C)	Pressure Exponent n	Characteristic Exhaust Velocity (m/s)	Specific Impulse (sec)
DB	1550.0	2260	0.3	1441	230
DB / AP / Al	1799.2	3600	0.4	1628	265
DB / AP-HMX / Al	1799.2	3710	0.49	1665	270
PVC / AP	1688.5	2500	0.38	1487	240
PVC / AP / Al	1771.5	3090	0.35	1631	265
PS / AP	1716.2	2600	0.43	1484	240
PS / AP / Al	1716.2	2760	0.33	1544	250
PU / AP / Al	1771.5	3300	0.15	1629	265
PBAN / AP / Al	1771.5	3200	0.33	1620	263
CTPB / AP / Al	1771.5	3200	0.4	1636	265
HTPB / AP / Al	1854.6	3200	0.4	1636	265
PBAA / AP / Al	1771.5	3300	0.35	1629	265

Structure Material Selection

The selection of the SRM's structure plays a crucial role in the safety and final mass of the system. Metal alloy and composite materials are the main contenders for use in SRM structures. Each structural material type has its own specific properties of density, strength, and failure modes while providing various construction methods and applications.

Commonly used metals are alloys of aluminum, steel, and titanium. At least one metal of each alloy type is available in preformed sheets, plates, forgings, extrusions or castings to simplify structure formation. Attachment of structural metals is by welding, adhesive bonding, or mechanical fastener systems. Metals have a long history of structural use, with well documented failure behaviors. This failure history allows very accurate predictions of a metal's operational structural life and permits lower factors of safety.

Structural non-metallics are usually formed with composite filaments of graphite, boron, Kevlar™, or glass, bonded together in an epoxy-matrix material, and more recently, wound around thin-walled steel casings in crisscross patterns. Composites can be tailored for very high stiffness and strength properties with a low density and coefficient of thermal expansion. Manufacturing composite structural materials is an expensive and elaborate process. Since the technology is new, the composite material data base is still being developed and has fewer flight proven components. With less known about the performance of composite materials, the factor of safety placed on composite designs has been higher than with structural metal counterparts. Composite parts must be carefully analyzed, inspected and tested before becoming flight certified. This certification process and the expense of the raw materials has kept composite materials from becoming the dominate aerospace structural material.

Selection of the structural material determines the motor's overall structural design and final structural mass. Table 2.2 displays a sample listing of available aerospace materials used for SRM structures (NASA SP-8025, 1970; Sutton, 1985; Wertz &Wiley, 1991; ASM Engineering Materials Handbook [EMH], 1987):

TABLE 2.2 Typical Aerospace Structural Materials

Structural Material	Density (kg/m^3)	Yield Tensile Strength (MPa)	UltimateTensile Strength (MPa)
Aluminum (7075-76)	2800	448	523
Beryllium (Extruded)	1850	413	620
Boron/Al (50%)	2600	------	1491
Boron Epoxy	2010	------	1337
Graphite Epoxy (HTS)	1490	------	1337
S-Glass	1860	------	1480
Kevlar 49	1380	------	1378
Magnesium Extrusion	1770	199	269
Steel (PH15-7 Mo)	7600	1171	1309
Titanium (T16 A1-4V)	4430	999	1103
SiC / Al	2850	------	1034
Al-Oxide Epoxy	3100	------	1380

Chamber Pressure Selection

Chamber pressure selection is one of the primary design parameters in the SRM design. It directly affects the propellant combustion by controlling the burn rate, thereby the propellant thickness, and partially determines the final motor specific impulse. Chamber pressure creates the primary load on the motor structure and therefore dictates the required thickness and design of the structure and nozzle.

Increased chamber pressures require a stronger and greater structural mass, including a larger nozzle, but tends to increase the Isp, thereby decreasing the required propellant mass. By varying the chamber pressure over an acceptable range, the best combination of trade offs between the impact of chamber pressure on the structure and propellant load can be found. Figure 2.1 illustrates the pressure acting on the SRM propellant and motor components:

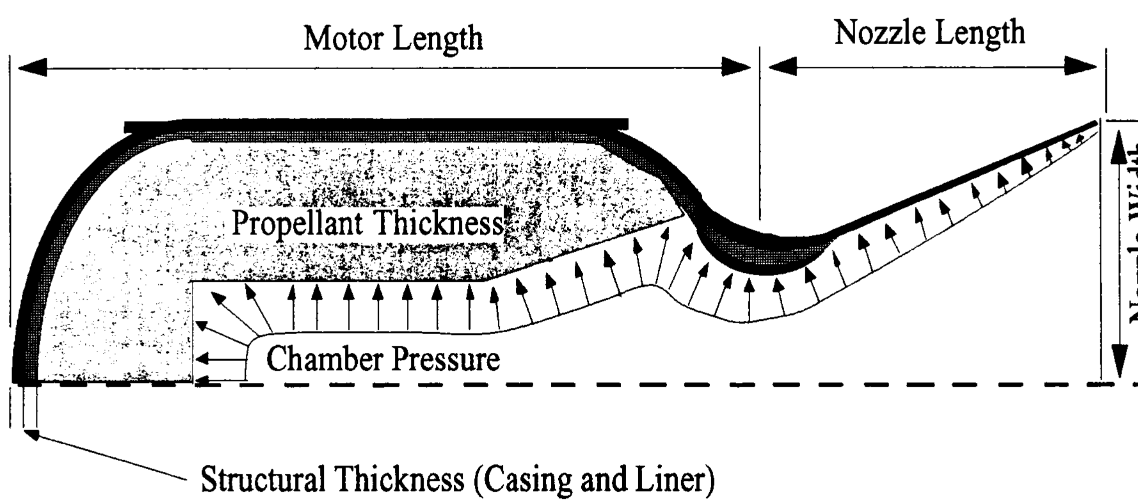

FIGURE 2.1 Chamber Pressure Acting on Motor Components

Motor Diameter Selection

The motor diameter selection assists in the determination of the overall motor length and structural thickness. Increased motor diameters require thicker motor walls while shortening the overall motor effective length (Le). The motor diameter also determines the available volume for the propellant and igniter system. By varying the motor diameter over a user defined acceptable range, the best combination of trade offs between the structure required for the motor casing can be found while maintaining the required amount of operational volume within the motor. Figure 2.2 illustrates the motor combustion chamber components affected by motor diameter selection:

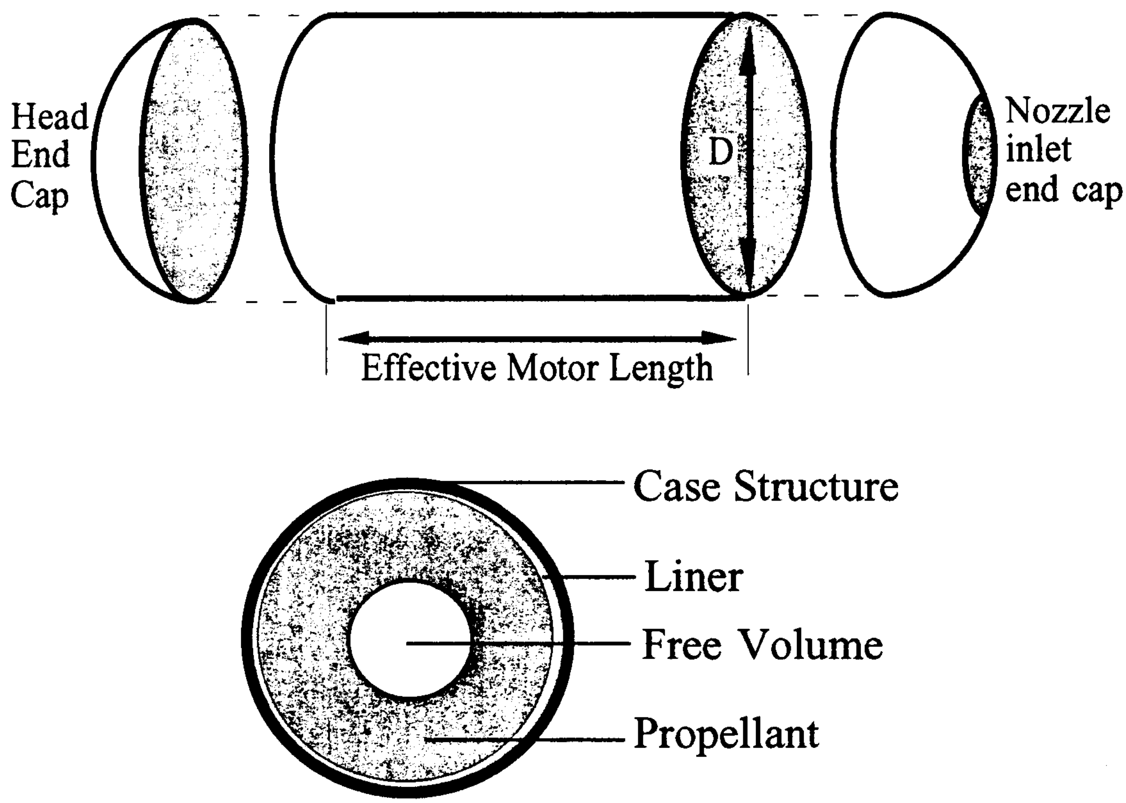

FIGURE 2.2 Motor Combustion Chamber Components

Nozzle Half Angle Selection

The nozzle half angle (α) selection determines the overall nozzle length, affects nozzle structural mass, and establishes the thrust loss of the nozzle, compared to ideal one-dimensional performance. The thrust loss (l) is due to the divergence of the real flow in the nozzle, compared to one-dimensional (totally axial) flow, and is illustrated in figure 2.3. The thrust correction factor for flow divergence is given by the equation:

$$\lambda = \frac{1}{2}\left(1 + \cos\alpha\right) \tag{2.1}$$

Where the thrust (F) is given by:

$$F = \dot{m}u_e\lambda + (P_e - P_\infty)A_e = \dot{m}C_{eff} \tag{2.2}$$

Where C_{eff} is the effective exhaust velocity of the propellant flow exiting the vehicle. Table 2.3 provides a listing of correction factors at various nozzle cone half angles.

TABLE 2.3 Nozzle Correction Factors

Nozzle Cone Half Angle, α (degrees)	Correction Factor, λ
0	1.0000
2	0.9997
4	0.9988
6	0.9972
8	0.9951
10	0.9924
12	0.9890
14	0.9851
16	0.9806
18	0.9755
20	0.9698
24	0.9636
26	0.9567
28	0.9415
30	0.9330

Increasing the nozzle half angle shortens the required nozzle and reduces the overall nozzle mass. An increased angle will also reduce thrust, compared to ideal theory, which,in actual rocket motor operation, results in the addition of more propellant to give the correct specific and total impulse delivered by the motor. By varying the nozzle half angle over an acceptable range, the best combination of trade offs between the nozzle structure and propellant mass can be found. Figure 2.3 illustrates the nozzle half angle optimization effects and velocity profile in a conic nozzle:

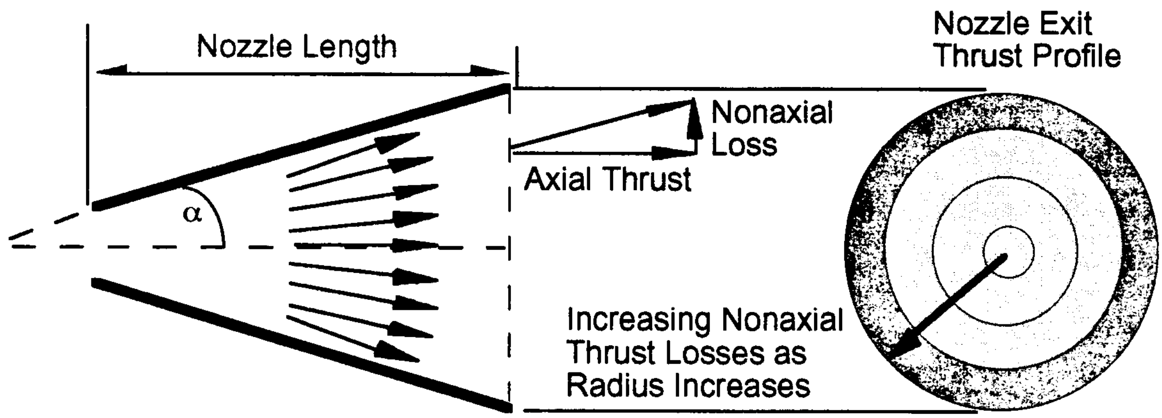

FIGURE 2.3 Nozzle Half Angle Optimization Effects

Additional Design Parameters

Other additional operational parameters such as loading forces, burn time, and operational altitudes affect the motor design directly. The parameters are dependent on the designer's knowledge of the SRM operational envelope.

Factors of safety are an important operational parameter dependent on the use and testing of the design and its failure modes. They are chosen based on engineering judgment, combined with prior experience, to obtain the desired level of reliability in the design. Table 2.4 provides an example of typical factors of safety for various situations and structural materials (NASA SP-8025, 1970; Sutton, 1985; Wertz &Wiley, 1991):

TABLE 2.4 Typical Factors of Safety

Test Option	Metalic Alloy Factors of Safety		Composite Factors of Safety
	Yield	Ultimate	Ultimate
1) Ultimate test of dedicated qualification article	1.0	1.25	1.5
2) Proof test of all flight structures	1.1	1.25	2.0
3) Proof test of one flight unit of a fleet	1.25	1.5	2.5
4) No structural test	1.6	2.0	3.0

Man rated SRM vehicles require additional safety factors to be included, usually a minimum of 1.5 ultimate (NASA SP-8025, 1970). Composite safety factors tend to be higher than their metal counterparts due to the relatively limited flight proven database. As the performance of composites becomes better known, the design engineer can justify reducing the composite materials factor of safety.

Data Base Development

Data base development for the SRM-CDOS consists of two separate systems. The first data base catalogs the details of the solid propellants and structural materials for user modification and the SRM design code. The second data base deals with the knowledge base information storage for the SRM design Expert System.

Appendix 2 provides illustrations of the cards in the main SRM-CDOS HyperCard reference stacks. Details of the Expert System knowledge base will be discussed in chapter five, with the IF-THEN rule knowledge base available in Appendix 3.

CHAPTER 3
HYPERTEXT USER INTERFACE

User Interface Background

The user interface is critical to the use and effectiveness of any computer software system. The interface affects the overall system performance by creating a coherent, nonambiguous, working environment. Several aspects of user interfacing need to be considered, including external display representation, modes of user-system interdependence, and management of uncertainty and errors (Potter, 1987). These threee aspects are summarized in figure 3.1.

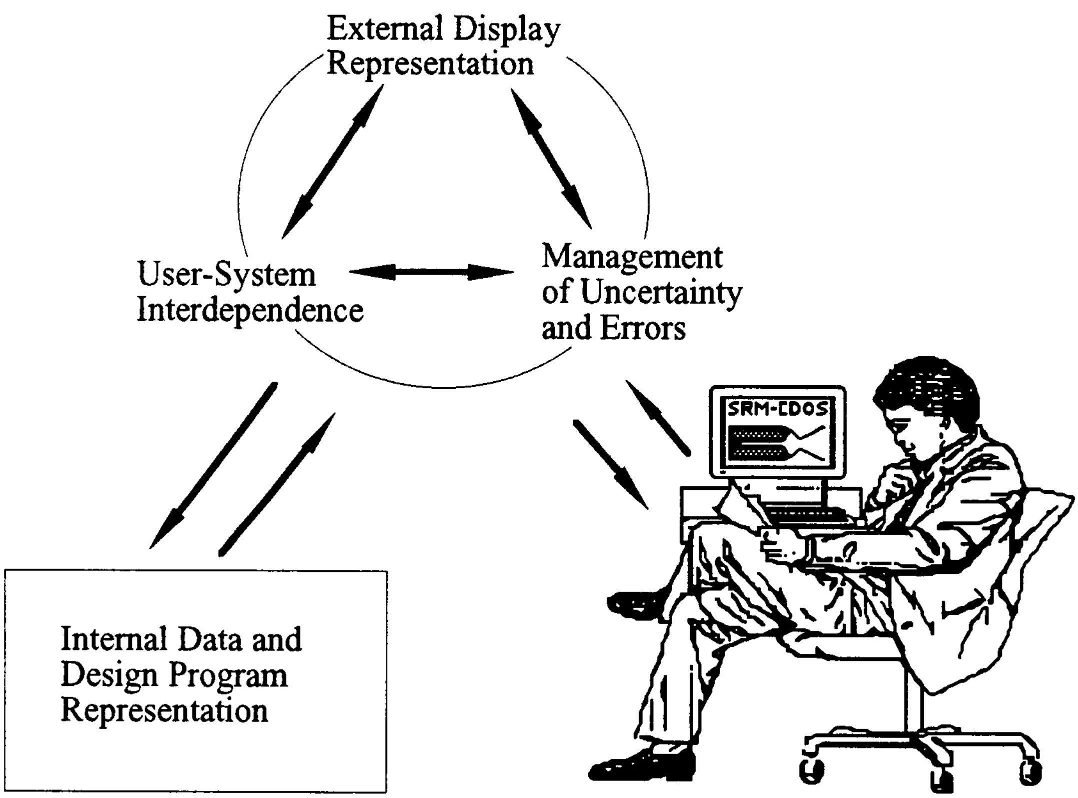

FIGURE 3.1 Interfacing Aspects of the User Interface

External display representation (EDR) is the design information provided across the user interface, and it drives the design of an user interface to make it intuitive and creditable. Intuitive software minimizes the amount of learning required for the user to use the software, and it builds on the user's previous knowledge and expectations. Interface credibility places the user at ease working with the computer software, and encourages the use and further development of the system. External displays are the main influence on the user. Time should be spent by EDR designers in the development of high quality external displays to yield visually stimulating and user friendly systems.

User-system interdependence influences the amount and complexity of information exchanged between the user and the hypertext system interface. There are three basic levels of interdependence, denoted advisory, cooperative, and autonomous systems. Advisory systems interact with the user who has little experience with the system. It provides step-by-step help and prompts for information from the user. The cooperative system provides a higher level of expertise for users familiar with the data and formatting. It allows direct user control of the data entry and provides advise only when asked. Autonomous systems are capable of selecting and executing processes without user intervention or data input. These systems read data from existing files, and the user becomes an overseer or manager who observes the program outside the operating system.

The management of data uncertainty and error plays an important part in the correct transfer of data from the user interface to the internal data files and design programs. Checks of the entered data must be done at various levels to guarantee the correct ranges and types of data have been entered. Flags should appear warning or notifying the users about missing or inappropriate data being sent to the internal programs. These self checks on the entered data help keep the internal programs from creating unaccounted for errors or global systems crashes.

Selection of HyperCard

The hypertext program, HyperCard, was selected for the SRM-CDOS project user interface. HyperCard meets all the aspects summarized in figure 3.1, while maintaining a variable level of user-system interdependence. In different parts of the SRM-CDOS, HyperCard acts as an advisory system with help cards and data prompting messages, as a cooperative system with menu driven access and limited messages, and then as an autonomous system for directing data between executing background programs.

HyperCard serves as an effective link between the user and the computer through two distinct areas. In the first area, HyperCard uses an "English" based command structure or scripts to access stored text, graphics, and data files. These scripts allows ease of use, design guidance, and data linking within the database to help the user entering SRM design data. This is the central theme of a hypertext user interface; to provide available data in a highly visual and easily accessible format for quick retrieval, modification, or data linking by the user. The more convenient the system, the more comfortable the user will be when working with it.

Secondly, HyperCard has the ability to act as a global control shell and direct all of the entered data and supporting programs. HyperCard directs incoming and outgoing data without the user having to worry about specialized commands and data pathways. Supporting programs are started and stopped as needed as they run in the computer's background memory. This direct program access and data managing provides excellent control without troubling the user for unnecessary key stokes and time.

Among the many hypertext type programs available, Apple's Macintosh HyperCard scripting language provided the best combination of hypertext and controllability. Based on the Macintosh computer, the SRM-CDOS user is introduced to a user-oriented computing environment. The HyperCard program exploits the Macintosh-user interface by providing a what-you-see-is-what-you-get graphics design, text editing, and display format capability.

The HyperCard program, version 2.1, is supplied to all current Macintosh computers with the start-up system software. This was the main reason why Macintosh's HyperCard was also the most familiar and accessible of the hypertext programs available to the author. Similar systems exist on a MS-DOS machines with hypertext development programs called Toolbook and Guide 3.0 (Lo, Shi, Wu, Hoyt, and Steinle,1992).

HyperCard User Interface Design

The HyperCard hypertext environment consists of a number of computer databases known as stacks. SRM-CDOS utilized one main control stack, three reference stacks, and a help stack. Each stack contains a number of individual cards which act as sequential pages of a book. The individual cards contain text fields, control buttons, and graphics to aide the SRM-CDOS user in understanding the SRM data to be entered.

The SRM-CDOS main control stack is the primary user interface. The control stack provides the user introduction, design background, design data entry, design program execution, and access to the reference and help stacks. Appendix 1 provides illustrations of the cards in the main SRM-CDOS user interface HyperCard stack.

The three reference stacks are secondary user interfaces which are accessible from the primary control stack. The reference stacks provide detailed information on the available aerospace solid propellants and structural materials, design correction factors, and cost factors. This information provides three data bases for the Fortran design codes during the SRM design process. These reference stacks are also capable of being modified to provided material upgrades and future design enhancements. Appendix 2 provides illustrations of the cards in the SRM-CDOS HyperCard reference stacks.

The help stack is a secondary user interface accessible from the primary control stack. The help stack provides an individual help card for each card of the primary control stack. When accessed, these help cards give the user additional information and directions for the SRM data entry and design process.

HyperCard Scripting Control

The stacks, cards, text fields, and control buttons are "programmable" through the HyperCard scripting language called HyperTalk. HyperTalk scripts allow direction and control of the stack and the information it contains. This direction and control is based on the linking and manipulation of information within the buttons, fields, cards, and stacks. (HyperCard, 1990)

Information linking allows multiple pathways for the user to access information within the user interface of the SRM-CDOS. Linking connects cards within the primary control stack as well as the reference and help stacks. Figure 3.2 illustrates the linking of cards and stacks within the user interface.

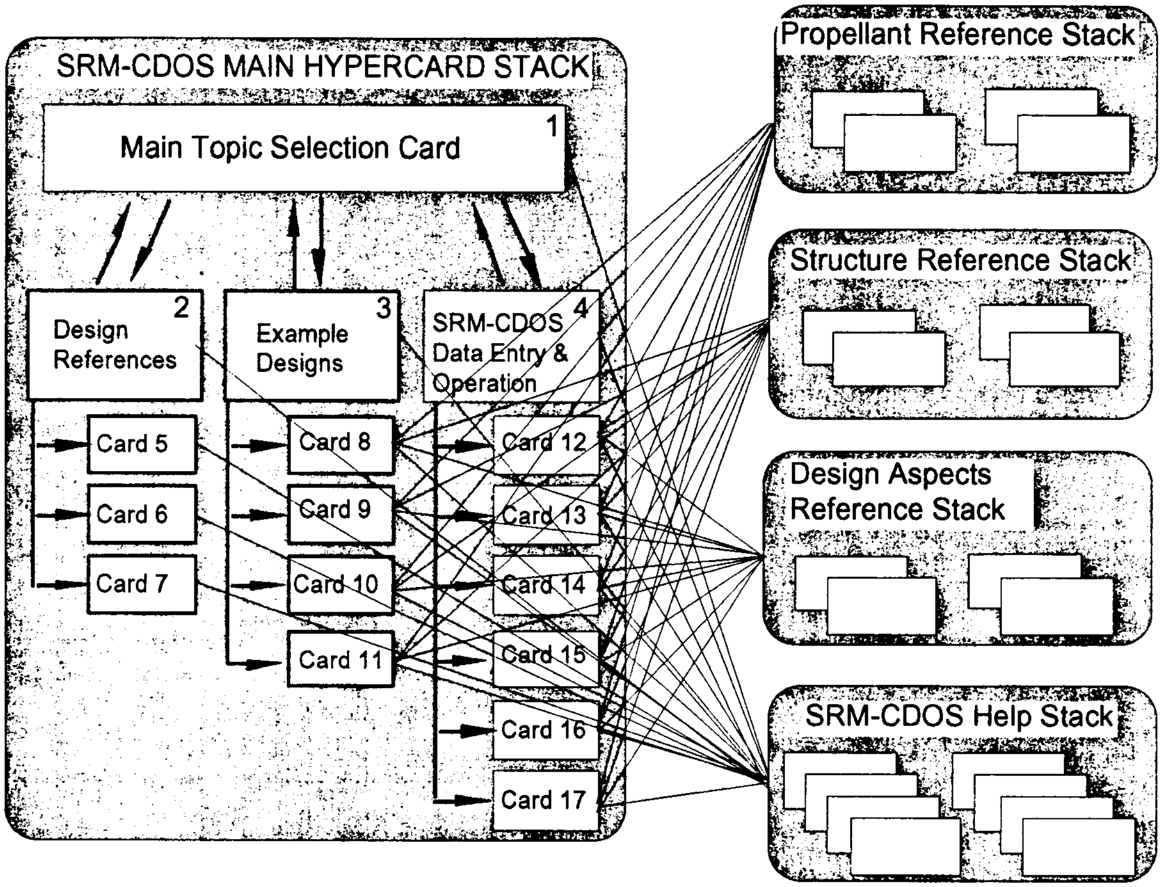

FIGURE 3.2 HyperCard User Interface Linking Pathways

The author designed each card with scripts to link relevant cards and stacks to provide a logical sequence of data entry and access. Each card has the ability to access sub and main menus, review previous cards, or access reference material and help files through the use of the linking script commands. Scripting also allows specific cards to act as data entry stations, program executing terminals, and design output locations.

The HyperCard scripting commands allowed the author to customize data manipulation within the user interface. The scripting commands follow easily understandable "English" command sequences to develop the complicated HyperCard hypertext functions and data links. With previous examples of the scripting programing style and the HyperCard references, the HyperCard set up of the hypertext interface was straight forward and fairly fast.

Command sequences are inserted directly into the global stack, each card, and most of the text fields or buttons. These sequences control the direction, and amount of data and program linking the user will be able to do. An example of the data control can be found in the scripted commands placed in the data entry text fields which act as error checkers. These error checking scripts monitor the user data entry and insure that an acceptable range of values are being entered. Figures 3.3 and 3.4 will illustrate the "English" command sequence for such an error check, for a data range of $1 < x <= 20$, and its conventional command flow chart:

```
on closefield
  put line one of card field id 19 into it
  if it <= 1 then
    put 1.1 into card field id 19
    Play "Warning"
    Put "Data out of range, Please re-enter the value or accept range limit" into Message
    Show Message
  end if
  if it > 20 then
    put 20 into card field id 19
    Play "Warning"
    Put "Data out of range, Please re-enter the value or accept range limit" into Message
    Show Message
  end if
end closefield
```

FIGURE 3.3 Scripting Error Check Command Sequence

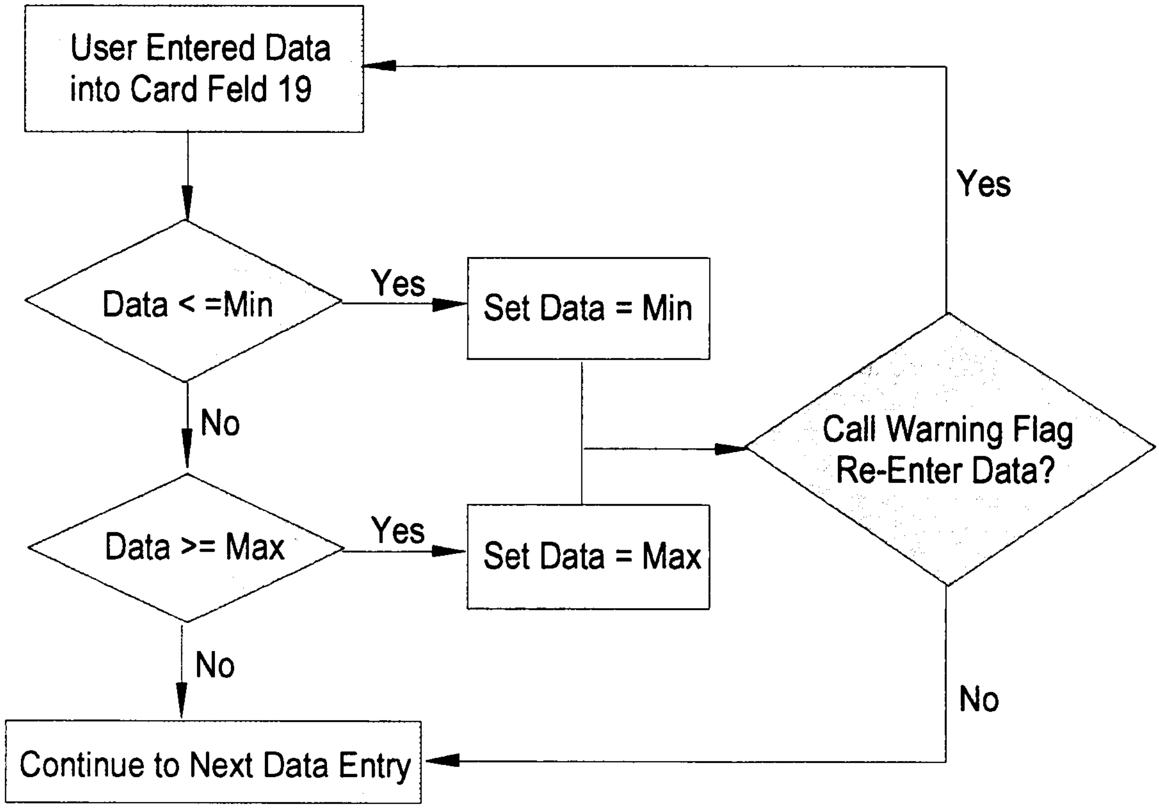

FIGURE 3.4 Scripting Error Check Flow Chart

HyperCard Control Shell Duties

With scripting, HyperCard maintains one of the best levels of user-computer interaction available and establishes a global control shell over the SRM-CDOS project. As stated in the introduction, the SRM-CDOS consists of three different programs working together. The HyperCard user interface scripting language provides the ability to manage data to and from outside text files, and control the operation of external programs.

Using the scripting commands, HyperCard became the global control shell which directed the entered data from the user interface into ASCII text files, and executed the design optimization program and design expert system. After the SRM design has been completed, HyperCard accesses the design results from the design optimization program and design expert system output text files and displays the design data for the user. Figure 3.5 illustrates the architecture of HyperCard as the global control shell.

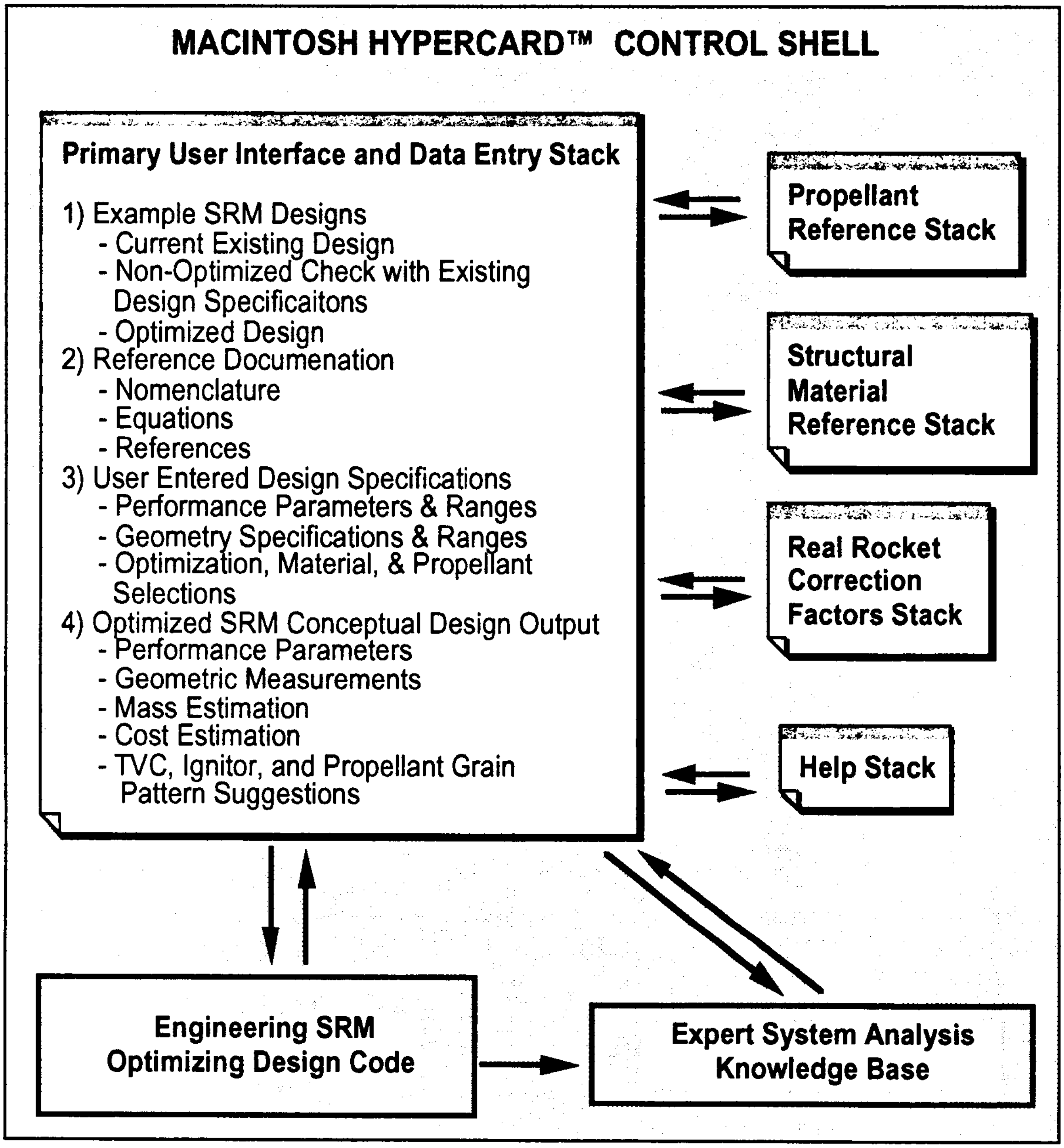

FIGURE 3.5 HyperCard Control Shell Architecture

Balancing the user interaction and guidance with the computer control of the SRM design was the overall goal of the HyperCard hypertext interface. By making the hypertext interface as easy to use and helpful as possible, the SRM-CDOS package gives its users the best combination of user interaction and global SRM design control.

CHAPTER 4

SRM FORTRAN OPTIMIZER CODE

The simplest, and perhaps the most fundamental SRM design method, is presented in Sutton's rocket propulsion text (Sutton. 1985). An engineer can apply Sutton's methodology and begin the design optimization process in many ways based on the selection of a solid propellant, structural component material, and motor geometric dimensions. Each variable selection has a place within the SRM design process to determine the final SRM mass. The set of variables resulting in the lowest final SRM mass was considered to be the optimum design selection.

The fortran SRM conceptual design optimizer code developed by the author forms the basis of the computational design work for the SRM-CDOS. The purpose of the fortran code was to provide a rapid prototyping method for SRM conceptual design with optimizations of the SRM mass or cost. The fortran code employs quasi-one-dimensional flow theory with correction factors to facilitate and define tradeoffs between different SRM designs. Optimizing the design is done through an array solution algorithm comparing multiple designs based on the user entered data ranges.

Figure 4.1 illustrates the input and output of the SRM conceptual design. It also shows the five nested design variables used in the variation of the design parameters. Additional detail of the design will be provided in the following sections.

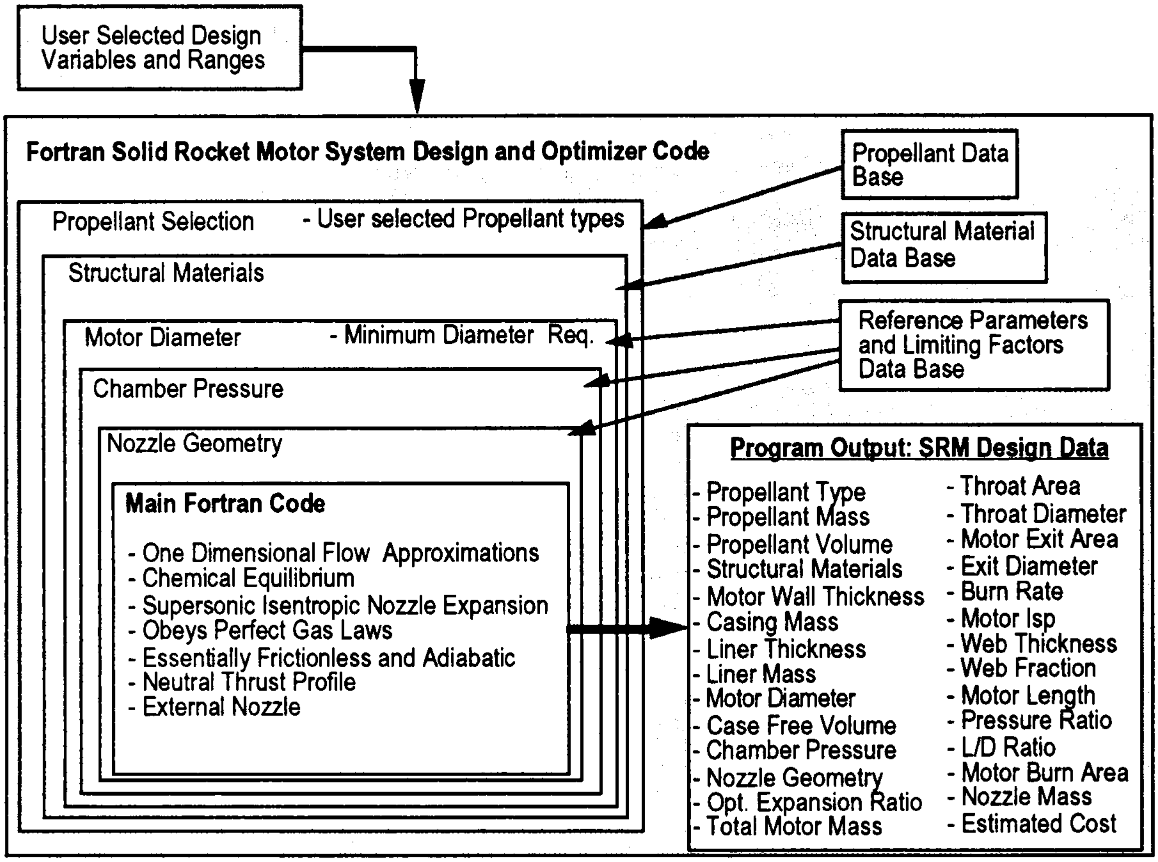

FIGURE 4.1 SRM Fortran Code Outline

Fundamental Considerations

Solid rocket motor conceptual design is a detailed and equation intensive process. In designing new SRMs, it has become accepted practice to use the ideal rocket parameters to simplify the analysis. Adding appropriate correction factors allow the ideal rocket to closely approximate a real rocket design (Sutton, 1985).

The design equations for an ideal rocket are based on the assumption of a quasi-one-dimensional flow present during the motor operation. Quasi-one-dimensional flow has the following flow assumptions:

1) Working Fluid is Homogeneous and Invariant in Composition Throughout the Combustion Chamber and Nozzle, and Obeys the Perfect Gas Laws.
2) Flow is Steady and is Chemically Frozen at the Chamber Composition with constant Cp
3) Flow is Frictionless
4) Motor Walls are Adiabatic
5) Isentropic Nozzle Expansions

Additional assumptions in the design are as follows:
6) Motor Thrust Follows a Neutral Thrust Profile
7) External Motor Nozzle Design
8) Simplified Casing Structure Design

Since the chamber temperatures are very high (2260 - 3600°C) for combusting solid propellants, the product gases are well above their respective saturation conditions and follow the perfect gas laws very closely. Assuming a steady flow with no friction, gas-dynamic shocks, or heat losses allows the use of isentropic expansion relations in the nozzle design. Isentropic expansions permit the assumption that the maximum amount of kinetic energy can be taken from the available heat energy of the exhaust, for a given nozzle design.

The wall friction losses are difficult to determine accurately, even with extensive testing and CFD work, and they tend to be very small (1 - 2% of the total energy) and can be neglected. Except for very small combustion chambers, energy transferred as heat to the walls is usually less than 2% of the total energy and can be neglected (Sutton, 1985; NASA SP-8039, 1971).

Neutral thrust profiles and steady state flow allows the designer to neglect time dependent terms in the propellant burning and motor thrust. Initial structural design is based on the motor casing subjected to the hoop stresses of the chamber pressure.

Equation nomenclature is included in Appendix 6.

SRM Design and Optimization Method

The application of the quasi-one-dimensional flow relations develop the performance equations of the SRM. Additional design methods are applied to develop equations for the motor structure and nozzle. The equations both predict the operation of the SRM, and act as a means to select important design variable needed for design optimization. By closely examining and running multiple examples through the design equations, critical design variables can be located. Later in this chapter, the critical design variables will be identified in the design equations. The equation nomenclature is provided in Appendix 6. As an overview, the critical design variables include:

1) Propellant Selection
2) Structural Material Selection
3) Chamber Pressure
4) Motor Diameter
5) Nozzle Cone Half-Angle

The optimization of the SRM design becomes an effort of varying the critical design variables to find the best combinations which yield the lowest operational SRM mass. Optimization of the SRM cost follows a similar pattern, but with the relative lowest cost of the construction and materials of the SRM as the primary optimized variable selector.

Performance Predictions

Starting from the source of the propulsive energy for a SRM design, the solid propellant selection plays a crucial role in the operational performance and final mass of the system. For the integration of the propellant selection into the SRM design, five variables need to be addressed: the average molecular weight of the propellant exhaust, adiabatic combustion temperature, exhaust specific heat ratio, motor exit pressure, and motor chamber pressure. The first three variables are dependent on the solid propellant selection, while the last two are designer selected. Each of these variables play a part in the propulsion efficiency term known as specific impulse (Isp).

Isp is defined as:

$$\mathbf{I_{sp}} = \frac{\int_0^t \mathbf{F}\,dt}{\mathbf{g_o}\int_0^t \dot{\mathbf{m}}\,dt} \tag{4.1}$$

For a propulsion system with constant mass flow ($\dot{\mathbf{m}}$), a constant thrust (F), and a negligibly short start and stop transient, the Isp can be defined as:

$$\mathbf{I_{sp}} = \frac{\mathbf{F}}{\dot{\mathbf{m}}\mathbf{g_0}} = \frac{\dot{\mathbf{m}}\mathbf{c_{eff}}}{\dot{\mathbf{m}}\mathbf{g_o}} = \frac{\mathbf{c_{eff}}}{\mathbf{g_o}} = \frac{\mathbf{c^*\,C_f}}{\mathbf{g_o}} \tag{4.2}$$

Then **c*** is defined as the propellant's characteristic exhaust velocity:

$$c^* = \frac{\sqrt{k\frac{R_o}{\mathfrak{M}}T_f}}{k\sqrt{[\frac{2}{(k-1)}]^{\frac{(k+1)}{(k-1)}}}} \tag{4.3}$$

and C_f is thrust coefficient:

$$c_f = \sqrt{[\frac{2k^2}{k-1}](\frac{2}{k+1})^{\frac{(k+1)}{(k-1)}}[1-(\frac{p_e}{p_o})^{\frac{(k-1)}{k}}]} + \frac{p_e - p_a}{p_o}(\frac{A_e}{A_t}) \tag{4.4}$$

When Pe = Pa, C_f is known as the optimum thrust coefficient.

Combining the characteristic exhaust velocity and optimum thrust coefficients, the equation for Isp becomes:

$$I_{sp} = \frac{1}{g_o}\sqrt{[\frac{2kRo}{(k-1)\mathfrak{M}}]T_f[1-(\frac{p_e}{p_o})^{\frac{(k-1)}{k}}]} \tag{4.5}$$

In this form, Isp is only a function of the propellant characteristics and the two designer selected pressures. The solid propellant with the highest flame temperature and lowest average molecular weight in the exhaust flow yields the highest specific impulse, that is Isp is maximized for the desired pressure ratio.

The selected SRM exit pressure (Pe) is usually related to the atmospheric altitude pressure (Pa) conditions in which the SRM would be initially operating. The lower the operating exit pressure, or higher the altitude, the better the motor's specific impulse will become. The SRM engineer should also attempt to match the exit pressure to the ambient atmospheric pressure in order to reduce or eliminate pressure drag losses, as illustrated in equation (4.4). For space motor design criteria, when Pa = 0, the SRM-CDOS requires a non-zero exit pressure and constrains the pressure ratio, Po/Pe, to less than infinity. Ambient altitude pressure measurements can be found in literature referencing the International

Civil Aeronautical Organization (ICAO) Standard Atmosphere or the International Standard Atmosphere (ISA) (United Technologies and Pratt&Whitney, 1986).

The chamber pressure (Po) is the final design variable which has an affect on the motor specific impulse. Increasing the chamber pressure will reduce the losses at the nozzle exit and increase the specific impulse. The chamber pressure also affects the burn rate of the propellant, combustion stability, size of the expansion nozzle, and thickness of the casing materials to withstand the pressure stresses.

Propellant Mass Prediction Equations

If the average thrust required by the SRM is known, the required total impulse of the SRM system can be determined:

$$\mathbf{I}_t = \int_0^{t_b} \frac{\mathbf{F}}{\zeta_f} dt = \frac{\overline{\mathbf{F}}}{\zeta_f} t_b \tag{4.6}$$

A thrust correction factor ζ_f is included to account for deviations between the actual real rocket thrust and the theoretical ideal rocket, and is defined as:

$$\zeta_f = \zeta_v \ \zeta_d \lambda \tag{4.7}$$

The thrust correction factor, ζ_v, takes into account velocity losses in the exhaust by the square root of the energy conversion factor, ξ:

$$\xi = \frac{V_{e\ actual}^2}{V_{e\ ideal}^2} = \frac{V_{e\ actual}^2}{V_{0\ actual}^2 + C_p(T_f - T_e)} \tag{4.8}$$

$$\zeta_v = \sqrt{\xi} = \sqrt{\frac{V_e^2}{V_0^2 + \frac{kR_0}{(k-1)\mathfrak{M}}(T_f - T_e)}} \tag{4.9}$$

Past experimental results yield ζ_v with typical values ranging between 0.85 and 0.98 (Sutton 1985). The thrust correction factor, ζ_d, takes into account differences between the mass flow rate of the real rocket and that of the ideal rocket:

$$\zeta_d = \frac{\dot{m}_{actual}}{\dot{m}_{ideal}} = \frac{\dot{m}_{actual} C_{eff}}{F_{ideal}} \tag{4.10}$$

$$\zeta_d = \frac{\dot{m}\sqrt{k\frac{R_0}{\mathfrak{M}}T_f}}{A_t p_0 k \sqrt{[\frac{2}{(k-1)}]^{\frac{(k+1)}{(k-1)}}}} \tag{4.11}$$

Past experimental results have yielded ζ_d with typical values ranging between 0.98 and 1.15 (Sutton, 1985). The loss in the nozzle due to the nonaxial component of the exhaust gas velocity, λ, is noted in equation (2.1) and is tabulated in table 2.3.

The mass of the required solid propellant can be determined by using the required total impulse for the SRM.

$$M_p = \frac{I_t}{(I_{sp} g_o)} \times 1.02 \tag{4.12}$$

The value of 1.02 is a more or less standard compensation for 2% of "sliver propellant". Sliver propellant is the unburned propellant remaining at the time of web burn out that does not provide any usable thrust.

The final mass of the solid propellant is directly tied to the value of the propellants specific impulse. As pointed out earlier in equation (4.5), the specific impulse is determined by the propellant selection and the chamber pressure. Therefore, the propellant selection and chamber pressure have a direct impact on the mass of the solid propellant and can be used as optimization parameters to minimized the solid propellant mass.

Structure and Nozzle Design

After the required amount of propellant has been determined, the design of the motor structure and nozzle are developed. For the integration of the structure and nozzle into the SRM design, five variables need to be addressed: the motor diameter, the nozzle cone half angle, and the structural material strength, density, and stiffness. The first two variables are typically designer selected, the last three are dependent on the structural material selection. Each of these variables play an important part in the overall structural motor mass and design.

Primary sources of information on aerospace structural materials are from the MIL-HDBK-5E, *Metallic Material and Elements for Aerospace Vehicles*, MIL-HDBK-17A, *Advanced Composites Design Guide.*, and Engineering Material Handbooks published by ASM International. Many of the referenced authors (Sutton, 1985; Wertz & Larson, 1991) provide listings of typical aerospace structural materials.

The SRM design involves a selection of the aerospace structural materials, and application of their strength and density to the mass determination of the motor casing and nozzle structure. Simple membrane theory was used to predict the stress in the solid propellant rocket chamber case and nozzle. The resulting casing and nozzle mass for each structural material was determined to find the minimum optimum value. Denost (1988) offers a more detailed design method and criteria for SRM casings, specifically for composite cases.

For a cylindrical motor section, where the casing thickness is much less than the internal radius of curvature of the casing, the longitudinal stress σ_l is one-half of the tangential or hoop stress $\sigma_\varnothing$:

$$\sigma_\varnothing = 2\sigma_l = \frac{p_o D}{2\tau_{mc}} \tag{4.13}$$

Using the fact that the combined stress should not exceed the working strength of the casing material, the optimization method substitutes the ultimate tensile strength of the structural material with the applied stress, applies a factor of safety, motor diameter, and solves for the motor casing thickness using:

$$\tau_{mc} = f\frac{p_o D}{2\sigma_{Ult}} \tag{4.14}$$

After the casing thickness was determined, the structural mass of the motor casing was modeled assuming a cylinder with two spherical ends:

$$M_c = \pi\tau_{mc} D \rho_c L_{case} + \pi\tau_{mc} D^2 \rho_c \tag{4.15}$$

For the case of a very "squat" motor (diameter > case length), the motor end caps where modified to become elliptical caps with the minor axis 1/3 the major axis (2xMajor Axis =motor diameter). The casing mass for the elliptical caps was determined by:

$$M_c = \pi\tau_{mc} D \rho_c L_{case} + \rho_c \tau_{mc}\left(2\pi\frac{D^2}{2} + \pi\frac{\frac{D^2}{6}}{e}\log\left(\frac{1+e}{1-e}\right)\right) \tag{4.16}$$

Where "e" is the eccentricity of the ellipse determined by:

$$\mathbf{e} = \sqrt{(1-\frac{\mathbf{D}^2}{6})/\frac{\mathbf{D}^2}{2}\}} \tag{4.17}$$

The mass of the motor insulation liner was modeled in a similar fashion. The liner thickness was determined by finding a liner ablation thickness with a minimum thickness of 0.254 cm (Sutton, 1985). The local chamber mass flux of the combusting solid propellant is found by:

$$\widetilde{\mathbf{M}}_{\mathbf{prop}} = \frac{\mathbf{M_p}}{\frac{1}{4}\mathbf{t_b}\,\pi\mathbf{D}^2} \tag{4.18}$$

The local mass flux was entered into an algorithm which incorporated various types of liner materials and their characteristic ablation rates (NASA SP-8093,1976). The liner types and ablation data is illustrated in figure 4.2.

Y1 = M0+M1*x+M2*x^3+M3*x^3+M4*x^4	
M0	7.4290731e-07
M1	1.907128e-06
M2	-2.192759e-09
M3	-1.927442e-12
M4	5.895112e-15

Y2 = M0+M1*x+M2*x^3+M3*x^3+M4*x^4	
M0	-5.638728e-06
M1	2.862627e-06
M2	-1.611151e-08
M3	3.571264e-11
M4	-2.312759e-14

Y3 = M0+M1*x+M2*x^3+M3*x^3+M4*x^4	
M0	1.777168e-06
M1	7.158080e-07
M2	-4.613488e-10
M3	2.020434e-13
M4	8.478216e-16

Y4 = M0+M1*x+M2*x^3+M3*x^3+M4*x^4	
M0	9.819715e-06
M1	3.926004e-07
M2	-5.276345e-10
M3	5.713452e-13
M4	-9.796733e-18

Y5 = M0+M1*x+M2*x^3+M3*x^3+M4*x^4	
M0	4.755868e-06
M1	2.306650e-07
M2	-6.073535e-10
M3	7.953886e-13
M4	-2.444748e-16

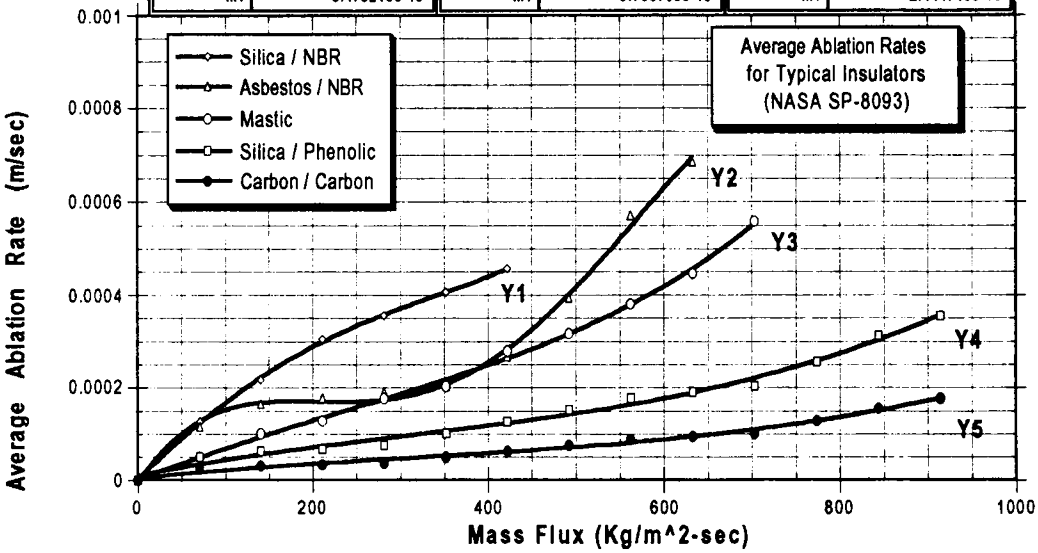

FIGURE 4.2 Liner Ablation Rates vs. Chamber Mass Flux

The resulting liner thickness, plus safety factor, were found and entered into the following equation to determine the liner mass:

$$M_l = \tau_l (D - 2\tau_{mc}) \rho_l \pi L_e + \frac{\pi}{4} \tau_l (D - 2\tau_{mc})^2 \rho_l \qquad (4.19)$$

By inspection, the motor casing mass equations are dependent on the factor of safety (f), chamber pressure (Po), motor diameter (D) and the structural material's ultimate strength. Equations (4.18) and (4.19) show that the casing thickness is also dependent on the motor diameter (D). The factor of safety is set for the entire design, and thus can not used as a good optimizer variable. The chamber pressure has already been seen be to a primary optimization variable. That leaves the motor diameter and structural material selection as two variables which can be adjusted to find the optimum combination for the lowest structural material mass.

First, in order to determine the affects of the motor diameter, the solid volume propellant is needed. The propellant volume is determined by the propellant mass and density:

$$\mathbf{V_p} = \frac{\mathbf{M_p}}{(\rho_p)} \tag{4.20}$$

The propellant thickness (b) within the motor case is denoted as the propellant web thickness and is dependent on the operational burn time of the motor (t_b)and the burn rate (r):

$$\mathbf{b} = \mathbf{r}\,\mathbf{t_b} \tag{4.21}$$

Solid propellant burn rates are determined empirically through repeated testing of propellant samples. For the sake of a conceptual SRM design, the burn rate can be expressed by empirical relations of the form:

$$\mathbf{r} = \mathbf{a}\mathbf{p}_0^{\mathbf{n}} \tag{4.22}$$

where a is an empirical constant influenced by the ambient propellant grain temperature, and n is the solid propellant burning rate pressure exponent. Values for a and n are available from solid propellant test combustion case histories. (NASA SP-8064, 1971, see also table 2.1 for n values).

The web thickness (b), times two, gives the minimal diameter required for the motor. But, additional room is needed for the insulative liner and case thickness, as well as interior volume for the ignition and required solid propellant burn area. Therefore the motor diameter needs to be varied to find the best value to meet these conditions. The motor casing thickness, liner thickness, outer solid propellant diameter, and inner diameter, as illustrated in figure 4.3, are all dependent on the selection of the motor diameter, D:

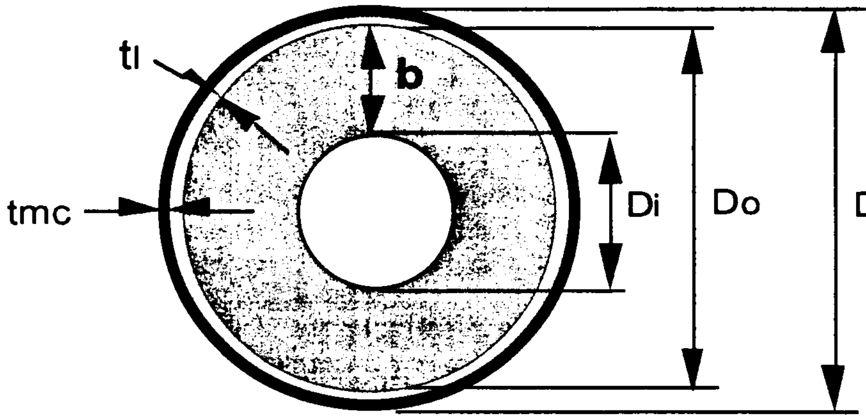

FIGURE 4.3 SRM Cross Section

Where the propellant outer and inner diameter, Do and Di, are determined by:

$$\mathbf{D_o = D\text{-}2t_{mc}\text{-}2t_l} \tag{4.23}$$

$$\mathbf{D_i = D\text{-}2t_{mc}\text{-}2t_l\text{-}2b} \tag{4.24}$$

Applying the two propellant diameters with the known solid propellant volume, the effective motor length can be determined:

$$\mathbf{L_e = V_p \frac{4}{\pi(D_o^2\text{-}D_i^2)}} \tag{4.25}$$

Then the case free volume available for the motor ignition system and solid propellant burn area is determined by:

$$\mathbf{V_{cf} = \frac{\pi}{4} L_e D_i^2} \tag{4.26}$$

Nozzle Design

The motor external nozzle design and mass estimation model was based on an axisymetrical conical nozzle, with a throat region designed for minimum erosion with a carbon/carbon throat insert. The conical shape nozzle is often used in practice for real rocket motors because of its simplicity and ease of fabrication. By varying the divergence angle, α, the designer can optimize the nozzle efficiency by optimizing the motor thrust while constraining the nozzle mass.

In the nozzle design, the chamber pressure becomes the dominate factor. Taking the thrust coefficient found in equation (4.3), the area and diameter of the nozzle throat can the determined:

$$A_t = \frac{F}{P_c C_f} \tag{4.27}$$

For a supersonic nozzle, the throat area can be related to the exit area as a function of the pressure ratio and specific heat ratio. The ratio of the exit area to the throat area is given by:

$$\frac{A_e}{A_t} = \varepsilon = \left\{ \left(\frac{k+1}{2}\right)^{\frac{1}{(k-1)}} \left(\frac{P_e}{P_o}\right)^{\frac{1}{k}} \sqrt{\frac{k+1}{k-1}\left[1-\left(\frac{P_e}{P_o}\right)^{\frac{k-1}{k}}\right]} \right\}^{-1} \tag{4.28}$$

The exit area can now be calculated using the optimal expansion ratio from

$$A_e = A_t \varepsilon \tag{4.29}$$

assuming the throat area is known.

The nozzle length can be found by using the selected nozzle cone divergence angle, and equation (4.29) to find the throat and exit diameters from the respective areas. Then,

$$L_n = \frac{\left(\frac{D_e}{2} - \frac{D_t}{2}\right)}{\tan \alpha} \tag{4.30}$$

Using the nozzle length, throat diameter, and exit diameter with the liner and case thicknesses, the mass of the nozzle can be approximated by using a set of truncated conic sections. The addition of the nozzle throat insert was also factored into the nozzle mass. Figure 4.4 illustrates the basic conical nozzle design used in the SRM design program.

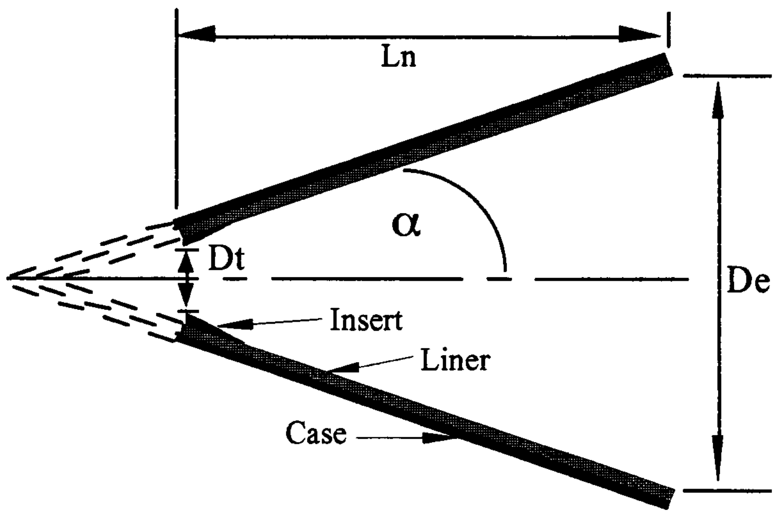

FIGURE 4.4 Axisymetric Conical Nozzle Design with Dimensions

Expert System Parameters

When the optimization process is completed, a set of variables is calculated to assist the expert system's analysis of the solid propellant grain design. Three nondimensional variables used by the expert system in the grain shape selection are calculated. These are the web fraction (b_f), length to diameter (L/D) ratio, and propellant volume to chamber volume ratio known as the volumetric fraction (V_f):

$$\mathbf{b_f} = \frac{\mathbf{2b}}{\mathbf{D}} \tag{4.31}$$

$$\frac{\mathbf{L}}{\mathbf{D}} \cong \frac{\mathbf{L_e}}{\mathbf{D}} \tag{4.32}$$

$$V_{fr} = \frac{V_p}{(V_f+V_p)} \tag{4.33}$$

The values computed with these equations are sent to the expert system, allowing it to select the best neutral thrust grain design.

General Array Solution Algorithm

After the optimization variables had been identified, they were tested on a spreadsheet based version of the design codes (Clegern, 1992). The spreadsheets were limited to optimizing one variable at a time with direct user data input. The single variable optimization did provide the general data ranges and insight into the variable interdependence and interaction. The design equations where then programed into this project's fortran code to provide an independently running design code.

The design optimization solution developed for the fortran code created a multi-dimensional array for the five design variables and their corresponding data ranges. Although not the fastest optimization solution, the array method allows for the optimization of all five variables simultaneously for the final design solution. Future versions of this optimization program will address improvements on the mathematical optimization procedure.

Figure 4.1, page 33, illustrated the "nested" optimizing design variables, while figure 4.5 provides the fortran optimization pseudocode.

```
Read in User Design Parameters From HyperCard
Read in Propellant Database
Read in Structure Database
Read in Design Correction Factors
Propellant Array Loop, Range ___ to ___
    Structural Materials Array Loop, Range ___ to ____
        Motor Diameter Array Loop, Range ___ to ___ Step ___
            Chamber Pressure Array Loop, Range ___ to ___ Step ___
                Nozzle Cone Half - Angle Array Loop, Range ___ to ___ Step ___
                    SRM Design / Performance Equations
                    SRM Design / Liner Thickness Equations
                    SRM Design / Structure Equations
                    SRM Design / Nozzle Design Equations
                    SRM Design / Mass Prediction Equations
                    SRM Design / Costing Equations
                   Select Optimized Design
                       Send Design Data to Output Files
                End Nozzle Cone Half - Angle Array Loop
            End Chamber Pressure Array Loop
        End Motor Diameter Array Loop
    End Structural Materials Array Loop
End Propellant Array Loop
END PROGRAM
```

FIGURE 4.5 Fortran Optimization Program Pseudocode

CHAPTER 5

EXPERT SYSTEM SRM KNOWLEDGE BASE

Background and Development

An expert system is a computer program that relies on knowledge and reasoning built into it's coding to perform a difficult task usually undertaken by a human expert. Expert systems are an evolving form of artificial intelligence which can provide general level reasoning on a focused, narrow domain of knowledge. From a functional point of view, an expert system can be defined as follows:

> "An expert system is a program that relies of a body of knowledge to perform a somewhat difficult task usually performed by a human expert. The principle power of an expert system is derived from the knowledge the system embodies rather than from search algorithms and specific reasoning methods. An expert system successfully deals with problems for which clear algorithmic solutions do not exist," (Parsaye and Chignell, 1988).

The two main goals of an expert system are: (1) to reduce the overall complexity of a situation, and (2) to allow the use of domain-specific knowledge. This type of domain restricted knowledge is very similar to the behavior of human experts in a specific field of study. Human experts typically know "a lot about a little," having a great deal of knowledge about a narrowly defined problem domain. Similarly, expert systems are designed to use a domain of knowledge and apply it to a narrowly defined problem or task (Parsaye and Chignell, 1988).

Human expertise is a trusted and vital portion to many work situations, but there are many reasons to make a "backup" of the knowledge in the form of the expert system. Training a human expert can take many years of academic classes and hands-on experience, while an expert system can be copied in a few seconds and transferred to a new location. Humans tend to be unpredictable, they're susceptible to irritants, sickness, retirement, and death. Expert systems will continue to work consistently and dependably all the time, assuming no host computer failure. Expert systems also are capable of being upgraded and improved as new information is developed and needs to be stored in the knowledge base for future use.

Expert systems have been enthusiastically embraced by many in the business, industrial, and professional spheres as a way of making expert knowledge routinely available. Expert systems have provided a means to retain highly valued expertise in an area once the original human experts are unavailable. This knowledge retention is especially important when replacement experts are expensive and in short supply or retraining new personnel is too costly and time consuming. Expert System examples include INTERNIST and MYCIN in the medical field, DENDRAL in chemical analysis field, and XCON in the business marketing field (Parsaye and Chignell, 1988). NASA has also explored the use of expert systems in the workplace as a means to retain aerospace engineering expertise at the NASA Ames and NASA Marshall centers.

Knowledge base expert systems have also rapidly increased in feasibility with the development of supporting commercial software. The software allows the development and deployment of expert systems for use in many automated support environments. The expert system production software designs can follow several different schemes for developing a knowledge base. One scheme would be the construction of production rules which follow conditional patterns, priorities, and resulting actions. A second scheme would be through object manipulations involving frames and semantic networks situated in network or hierarchy. Both knowledge base development schemes have been used successfully in commercial expert system development software. (Dugan, Carmody, Lennington, and Nelson, 1990).

Expert System for the SRM-CDOS Application

The design of solid rocket motors lends itself to expert systems applications which provide design advice and selections. By incorporating expert SRM design information into an expert system knowledge base, the expert system can shoulder a portion of the SRM design load. This would allow the design engineers to concentrate their energies on other portions of the SRM.

With the SRM-CDOS running on a the Macintosh computer system, a Macintosh based expert system was required. Information Builders, Inc. marketed Level5 *for the Apple Macintosh* as a rule based expert system developer. Level5 provided acceptable expert system development capabilities, as well as excellent instructions and examples. When building an expert system with Level5, the programmer specifies a goal or hierarchy of goals which are then proved or disproven by a network of interdependent rules. Expert knowledge is represented as IF...AND...OR...THEN...ELSE rules, which contain the factual information comprising the domain of the expert system. (Information Builders, Inc., 1989)

The author selected three areas of the SRM where the incorporation of a design expert system could easily assist the overall design. These three areas include the propellant grain design, the igniter selection, and the thrust vector control (TVC) system. The expert system design for the SRM-CDOS incorporates the grain design, the igniter selection, and the TVC system into one large knowledge base.

As the expert system executes, it begins at the top of the goal "tree". To meet the specific goal at each level of the tree, rules incorporating the SRM knowledge are evaluated and confidence factors are assigned to each goal. The final goal met with the highest confidence factor becomes the recommended grain pattern, igniter system, or TVC system. Details of the three areas of the expert system are provided in the following sections.

Propellant Grain Design

The propellant grain design is based on the SRM performance demands. Grain patterns dictate the available propellant burn area and the motor thrust profile. Without the correct propellant grain design, the motor operational behavior could vary wildly from the predicted performance.

There are three basic types of grain designs: progressive burning, neutral burning, and regressive burning. Progressive burning grains have a burn time where the thrust, chamber pressure, and burn area increase in time with the operation of the motor. Neutral burning grains maintain an even or constant thrust, chamber pressure, and burn area. Regressive burning grains have a decreasing motor thrust, chamber pressure, and burn area. For the SRM-CDOS the neutral thrust profile was selected to help simplify the performance calculations by keeping time dependent terms constant.

Current grain designs concentrate on a small number of classical configurations which are varied to meet the specific needs of the SRM design. The design trend has been to stay away from configurations which yield a weak grain structure susceptible to cracking, production of a high sliver rate, yield a low volumetric loading fraction, or are simply too expensive to manufacture (Sutton, 1985).

Propellant grain configurations can be classified according to their web fraction (b_f), length to diameter (L/D) ratio, and volumetric fraction (V_f). These three characteristic variables are provided to the expert system by the SRM optimization design code. Some propellant grain designs have an overlap of the characteristics, as given in table 5.1.

TABLE 5.1 Propellant Grain Design Characteristics (Sutton, 1985)

Configuration	Typical web fraction	Typical L/D Ratio	Volumetric Fraction
• End Burner	>1.0	N.A.	0.90-0.95
• Internal-external burning tube	0.3-0.5	N.A.	0.75-0.85
• Internal burning tube	0.5-0.9	<2	0.85-0.95
• Segmented tube	0.5-0.9	>2	0.80-0.95
• Rod and tube	0.3-0.5	N.A.	0.60-0.85
• Star	0.3-0.6	N.A.	0.75-0.85
• Wagon wheel	0.2-0.3	N.A.	0.65-0.70
• Dogbone	0.2-0.3	N.A.	0.70-0.80
• Slotted tube	0.5-0.9	>3	0.85-0.95
• Conocyl	0.5-0.9	2 to 4	0.85-0.95
• Finocyl	0.6-0.9	1 to 2	0.85-0.95
• Spherical grain	0.2-0.5	N.A.	0.90-0.95

Additional information on the propellant grain designs can be referenced from Sutton's *Rocket Propulsion Elements*, (1995) in figures 10-8 and 10-9.

The propellant grain section of the expert system incorporated the propellant grain design knowledge listed in table 5.1 into a set of IF-THEN-ELSE rules. These rules guide the expert system goal matching and provide the confidence factors for the grain designs. The confidence factor will be dependent on which grain designs meet the design criteria and the practicality of a grain configuration.

Igniter Selection

The SRM igniter system generates the heat and gas required for motor ignition. A wide variety of igniter "types" have been developed. Major igniter systems include pyrotechnic and pyrogen types, while minor types include hypergolic, jelly roll, and film. (NASA SP-8051,1971) However, with a few exceptions, current operational SRM use pyrogen or pyrotechnic igniters. These are the only two ignition systems considered in the expert system.

The pyrogen igniter is basically a small SRM used to ignite a larger motor. The pyrogen uses conventional rocket motor grain formulations and design technology, but is not designed to produce any thrust in the system. Heat transfer from the pyrogen to the solid propellant grain is highly convective, with the hot gases emitted from the combusting pyrogen motor. The hot gases impinge upon the surface of the SRM grain, producing motor ignition. Small motors typically have a forward mounted pyrogen. Larger SRMs tend to mount the pyrogen igniter on the launch pad pointing up into the large motor nozzle. In this case, the igniter become a piece of ground support equipment. Figure 5.1 illustrates a typical pyrogen igniter.

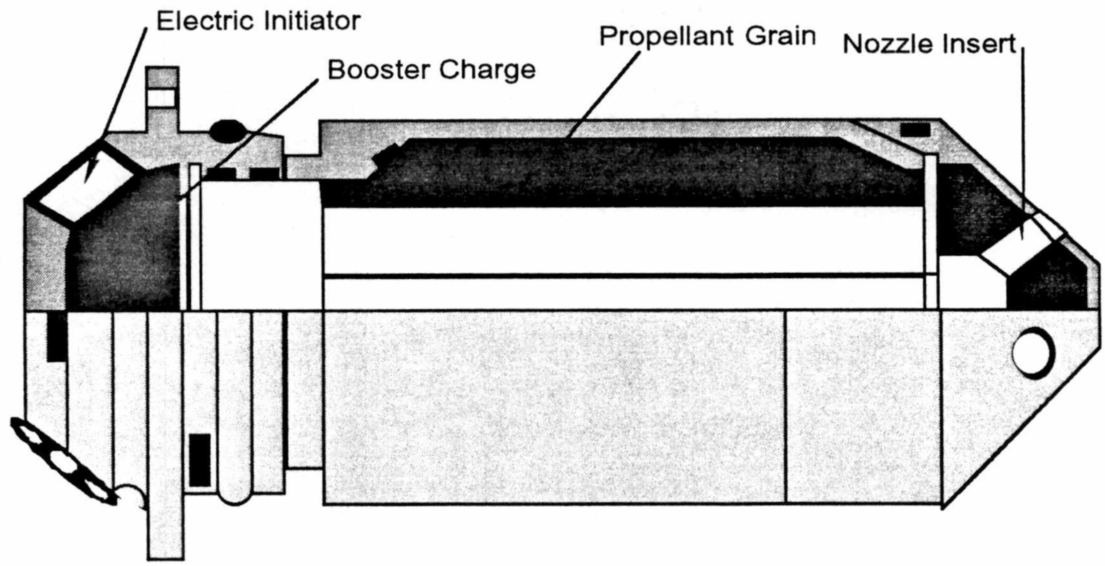

FIGURE 5.1 Typical Pyrogen Igniter (NASA SP-8051, 1971)

The pyrotechnic igniter is defined as using solid explosives or energetic propellant type formulations as heat producing materials. This describes a wide variety of igniter types. For the purposes of the expert system, the common pellet basket design will be examined. During ignition, the pellets are ignited in a wire basket where controlled burning takes place. The energy transfer to the solid propellant grain is largely radiative (Sutton,1985). Pyrotechnic igniters are typically mounted at the front of the motor. Smaller motors with high volumetric fractions (0.85-0.95) can use a surface mounted or grain mounted pyrotechnic igniter. The advantages of a surface mounted igniter include lower mass and volume with a simple design. Figure 5.2 illustrates the common basket type pyrotechnic igniter.

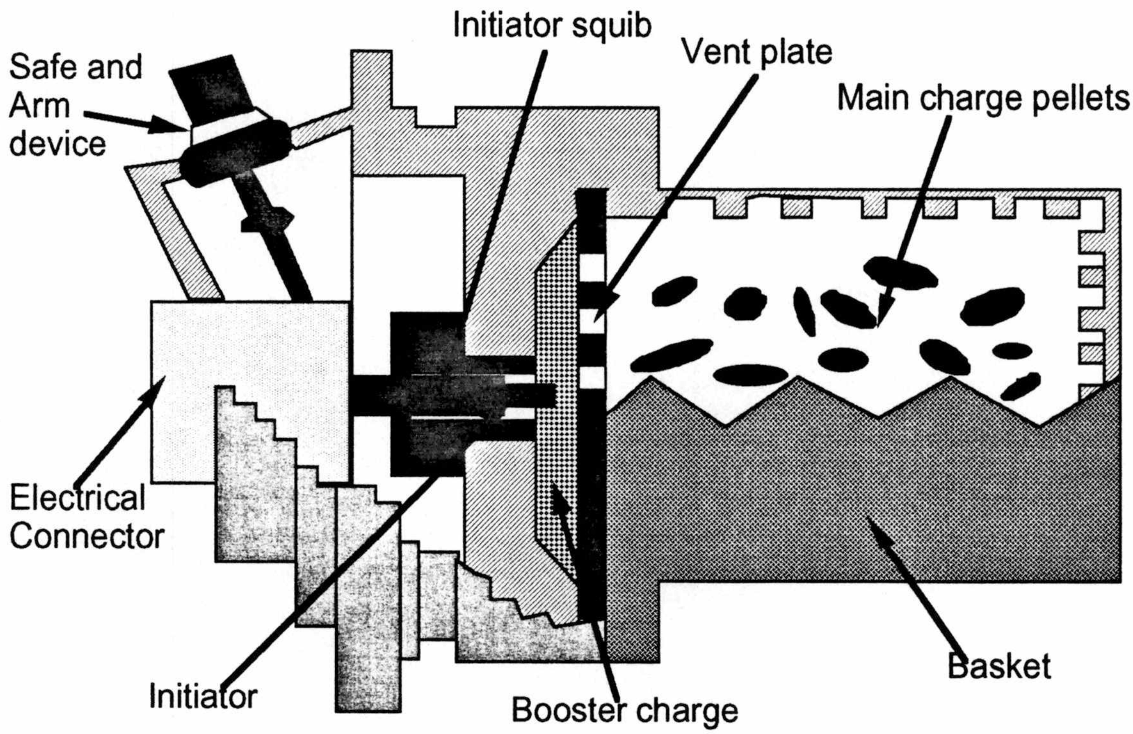

FIGURE 5.1 Typical Basket Type Pyrotechnic Igniter (Sutton, 1985)

Analysis and design of SRM igniter systems, regardless of type, depend heavily on prior experimental results of full scale motors. Using prior igniter results, an estimation of the igniter main charge mass (in kilograms) for various sized motors can made from a motor's available internal free volume. From this data the following empirical formula was developed (Sutton, 1985):

$$\mathbf{M_{ic}} = 1.019(\mathbf{V_{cf}})^{0.7} \tag{5.1}$$

Based on this formula, the expert system estimates the selected igniter mass. Each igniter type was given a base structural mass, comparable to existing igniter systems, and additional mass for the igniter main charge. An additional factor was included as a weight for efficiency (NASA SP-8051, 1971).

For a pyrogen igniter, the mass estimation equation was (in kg):

$$\mathbf{M_{Pyrogen}} = 6 + (1.019(\mathbf{V_{cf}})^{0.7})\times 1.1 \tag{5.2}$$

For a pyrotechnic igniter, the mass estimation equation was (in kg):

$$\mathbf{M_{Pyrotechnic}} = 4 + (1.019(\mathbf{V_{cf}})^{0.7})\times 1.3 \tag{5.3}$$

For a pyrotechnic sheet igniter, the mass estimation equation was (in kg):

$$\mathbf{M_{Pyrotechnic\ Sheet}} = 1.1 + (1.019(\mathbf{V_{cf}})^{0.7}) \tag{5.4}$$

Thrust Vector Control System Selection

Thrust vector control (TVC) systems provide the guidance or steering to ensure that the required flight trajectory is achieved. The steering is needed to compensate for flight disturbances (e.g., wind and weather), course corrections (e.g., maneuvering), and for vehicle imperfections (e.g., misalignment of thrust and center of gravity). Various mechanical and aerodynamical methods have been employed to redirect the motor thrust and provide steering forces.

The criteria governing the selection of a TVC design is based on the vehicle needs, including steering-force moments, force rates of change, reliability, dimensional and mass limitations, and cost. These needs translate into such factors as duty cycle, deflection angle, angle slew rate, and required power. The expert system analyzes the user entered data as well as the predicted SRM design to determine the best TVC system for the SRM (Sutton, 1985).

The SRM-CDOS expert system examines TVC methods which create a side force in the nozzle to create a moment about the motor center of gravity which is used to redirect the motor thrust. Various types of thrust vector controls have been developed, and they can be grouped into two broad categories: (1) mechanical thrust deflection methods, and (2) secondary fluid injection methods. The expert system contains three of the main TVC systems in current use: gimbaled movable nozzle, flex-seal movable nozzle, and a Liquid Injected TVC (LITVC) system (NASA SP 8114, 1974). The expert system calculates the TVC system mass, and modifies the estimated nozzle mass based on the thrust vectoring method chosen.

Table 5.2 provides advantages, disadvantages, and current status of the three types of TVC examined by the expert system.

TABLE 5.2 Advantages, Disadvantages, and Current TVC Status

TVC System	Advantages	Disadvantages	Technology Status
Liquid Injection	Liquid injection thrust adds to motor thrust and almost offsets weight penalty. High slew rate. Easy to adapt to various motors. Fast response capability.	Limited thrust deflection. High system weight. Toxic liquids required for best performance. Subject to maintenance and repeated checkouts.	Operational, flight proven technology. Typical exhaust flow deflection angle +/- 5°.
Flexible Bearing	No sliding hot parts. No gimbal ring. Reliable gas seal. Negligible thrust loss. Lighter than a gimbaled nozzle. Predictable and uniform actuation power	Higher actuation power than for gimbaled nozzle. Complex assembly. Joint requires thermal protection. Slight alignment problems at off design pressure.	Operational, flight proven technology. Typical deflection angle +/- 10°.
Gimbaled Nozzle	Flexible duty cycle. Negligible thrust loss. Large deflection capability. Thrust Vector angle linear with nozzle movement.	Hot parts require thermal protection. Heavier than flexible bearing nozzle. High actuation power required.	Operational, flight proven technology. Typical deflection angle +/- 14°

Of all the mechanical deflection types, the movable nozzle is the most efficient and most used. The flexible bearing and gimbaled nozzle are the two main movable nozzles devices built into current nozzles. Both physically shift the nozzle to deflect the thrust away from the thrust center line of the SRM body. The movable nozzles do not significantly reduce the thrust or the specific impulse of the motor while remaining mass competitive with other mechanical devices. The mass estimation of the movable nozzle TVC actuation system was based on previous nozzle designs and was split into three areas (in kg).

1) A base structural mass

2) System mass scaled to the average motor thrust

3) System mass scaled to the estimated amount of TVC time in use.

For a flex-seal TVC, the mass estimation equation for the actuation system was:

$$\mathbf{M_{Flex\text{-}Seal}} = 11 + (\mathbf{Average\ Thrust})\times 9e\text{-}5 + 0.15\times(\mathbf{Time\ used}) \quad (5.5)$$

For a gimbaled TVC, the mass estimation equation for the actuation system was:

$$\mathbf{M_{Gimbal}} = 13 + (\mathbf{Average\ Thrust})\times 8e\text{-}5 + 0.1\times(\mathbf{Time\ used}) \quad (5.6)$$

The motor nozzle mass was also adjusted by 1.4 times to accommodate the movable nozzle seal or gimbal and the additional mass of the moving nozzle structure.

LITVC is the injection of secondary fluids into the exhaust stream of the SRM nozzle. The injection of a fluid into the nozzle creates a side force based on two components: (1) the force associated with the propellant momentum and reactivity, and (2) the pressure unbalance along the inner nozzle wall resulting from flow detachment and shockwave generation. Injecting reactive fluids yields a higher side force as well as compensating for thrust losses. The expert system will examine a nitrogen tetraoxide injectant with a side-specific impulse of 450 sec. (Sutton, 1985).

For a liquid injected nozzle TVC system, the mass estimation equation (in kg) was based on existing LITVC designs and was broken down into an initial equipment mass, tankage mass, and liquid injectant mass:

$$\mathbf{M_{LITVC}} = 14 + (\mathbf{M_{Tankage}}) + (\mathbf{M_{Injectant}}) \quad (5.7)$$

$$\mathbf{M_{Injectant}} = (\mathbf{Total\ Impulse_{Injectant}}) / (\mathbf{Injectant\ Isp}\times \mathbf{g_0}) \quad (5.8)$$

$$\mathbf{Total\ Impulse_{Injectant}} = 1.05\times(\text{Av.Thrust})\times \text{SIN}(\text{Angle}_{\text{Flow Defl.}})\times(\text{Time of use}) \quad (5.9)$$

With total impulse of the injectant was in units of n-sec. The tankage mass was calculated similar to the SRM motor case. The tank was a composite, 30 cm diameter pressurized cylinder at 0.6895 MPa (100 psi) sized to hold the required injectant volume. The motor nozzle was also adjusted in mass by 1.15 times to accommodate the liquid injector ports for the TVC system and the additional mass of the structure.

Output to HyperCard

After the expert system has finished the propellant grain design, igniter recommendation, and TVC system recommendation, it formats the data into a complete SRM design listing. The design listing is then output to a text file where it is read by the HyperCard user interface. HyperCard then presents the user with the finished design in a printable scrolling field.

CHAPTER 6

SRM-CDOS COMPARISON WITH EXISTING DESIGNS

As a validation of the SRM-CDOS fortran design code, six existing SRM designs were used as comparison case studies. The results of the validation runs and optimization runs are provided in Appendix 4. The following specifications for each separate case motor were entered into the SRM-CDOS program through the HyperCard user interface:

1) Average Thrust (N)
2) Burn Time (sec)
3) Motor Diameter (m)
4) Average Chamber Pressure (MPa)
5) Nozzle Angle (°)
6) Structural Material Type
7) Solid Propellant Type
8) Operational Altitude and Altitude Pressure
9) Real Motor Correction Factors

During the initial runs for each motor case, items 1-7 were given by the manufacturer.

The last two items were determined by separate methods. For item number eight, the operational altitude and pressure were found by selecting an initial condition and observing the resulting nozzle exit diameter. With all the other motor parameters constant, and setting the altitude pressure = nozzle exit pressure, the altitude pressure controlled the exit diameter of the nozzle. In order for the SRM-CDOS to match nozzle size with the case motor, the altitude was basically iterated until the nozzles matched.

For item number nine, the real motor correction factors, ζv, ζd, and sliver factor were found by matching the total impulse of the SRM-CDOS motor to the value of the case motor. By setting the correction factors to values within their acceptable ranges (see Chapter 4) and iterating, the SRM-CDOS total impulse value closely matched the case motor value.

Results of the Case Validations

Once the nine specifications had been entered or found, the SRM-CDOS was run to check the match with the particular case motor. The sample case motors total masses ranged from 895.5 kg for the small Orion 38, to 22,921 kg for the larger Minuteman first stage.

First, a recall the three SRM-CDOS specific motor design assumptions used in the fortran code:

1) neutral thrust profile
2) external nozzle
3) single nozzle design

For an accurate test of the SRM-CDOS, two motors meeting these specifications were used as case studies (Orion 50S-AL and Hercules GEM). Four additional motors, which did not exactly meet the SRM-CDOS assumptions, where also used as case studies to test the flexibility of the SRM-CDOS code.

For all six cases, the SRM-CDOS matched total motor mass, required propellant mass, and motor geometry were well under the targeted 5% error level. Tables 6.1 through 6.6 provide a summary of the six design matching runs.

TABLE 6.1 Orion 50S-AL Case Study Motor

Orion 50S-AL Space Motor			
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**
Vacuum Total Impulse (kN-sec):	35,108.060	35,187.416	-0.06%
Total Loaded Motor (kg):	13,236.500	13,333.190	0.68%
Propellant Mass (kg):	12,157.900	12,234.830	0.63%
Case Assembly Mass (kg):	554.300	563.510	1.66%
Nozzle Assembly Mass (kg):	235.400	241.480	1.79%
TVC System Mass (kg):	0.000	0.000	0.00%
Igniter Assembly (kg):	9.100	10.580	16.32%
Internal Insulation & Liner (kg):	279.600	282.790	-0.43%
Total Inert (kg):	1,078.700	1,098.400	1.24%
Burnout (kg):	1,013.900	974.072	-4.40%
Motor Mass Fraction:	0.920	0.918	-0.22%
Motor Sizing Data			
Total Motor Length (m):	7.919	8.145	2.85%
Case Length (m):	6.497	6.530	0.51%
Nozzle Length (m):	1.650	1.620	-1.82%

TABLE 6.2 Hercules GEM Case Study Motor

Hercules GEM Booster Motor			
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**
Vacuum Total Impulse (kN-sec):	31,536.320	31,643.460	0.34%
Total Loaded Motor (kg):	12,714.600	12,773.734	0.47%
Propellant Mass (kg):	11,741.600	11,739.120	-0.02%
Case Assembly Mass (kg):	394.600	394.190	-0.10%
Nozzle Assembly Mass (kg):	241.500	233.580	-3.28%
TVC System Mass (kg):	0.000	0.000	0.00%
Igniter Assembly (kg):	7.900	5.120	-35.19%
Internal Insulation & Liner (kg):	387.500	401.744	3.68%
Total Inert (kg):	973.100	1,034.620	6.32%
Burnout (kg):	858.500	884.910	3.08%
Motor Mass Fraction:	0.923	0.919	-0.43%
Motor Sizing Data			
Total Motor Length (m):	11.092	11.085	-0.06%
Case Length (m):	10.028	9.975	-0.53%
Nozzle Length (m):	1.107	1.110	0.27%

The Orion 50S-AL and Hercules GEM motors provide good examples with which to test the SRM-CDOS design code. Each has a fairly neutral thrust profile and an external nozzle with no thrust vectoring. Examining Tables 6.1 and 6.2, the SRM-CDOS provides a very accurate design match to the manufacture's listed specifications. The only area which was not matching was the igniter system, which is a very small absolute mass error when compared to the motor as a whole.

The next four tables display motor design comparisons which test the design matching flexibility of the SRM-CDOS code.

TABLE 6.3 Orion 38 Case Study Motor

Orion 38 Space Motor			
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**
Vacuum Total Impulse (kN-sec):	2,183.080	2,187.514	0.20%
Total Loaded Motor (kg):	895.500	908.530	1.46%
Propellant Mass (kg):	770.500	771.030	0.07%
Case Assembly Mass (kg):	39.400	41.590	5.56%
Nozzle Assembly Mass (kg):	48.000	52.440	9.25%
TVC System Mass (kg):	13.700	15.410	12.48%
Igniter Assembly (kg):	1.300	1.656	27.38%
Internal Insulation & Liner (kg):	23.400	26.410	12.86%
Total Inert (kg):	126.000	137.510	9.13%
Burnout (kg):	114.300	119.590	4.63%
Motor Mass Fraction:	0.860	0.849	-1.28%
Motor Sizing Data			
Total Motor Length (m):	1.339	1.354	1.12%
Case Length (m):	0.831	0.837	0.72%
Nozzle Length (m):	0.517	0.517	0.00%

TABLE 6.4 Orion 50 Case Study Motor

Orion 50 Space Motor			
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**
Vacuum Total Impulse (kN-sec):	8,780.350	8,785.814	0.06%
Total Loaded Motor (kg):	3,366.300	3,470.580	3.10%
Propellant Mass (kg):	3,023.500	3,057.480	1.12%
Case Assembly Mass (kg):	137.200	191.600	39.65%
Nozzle Assembly Mass (kg):	104.900	111.381	6.18%
TVC System Mass (kg):	13.700	23.426	70.99%
Igniter Assembly (kg):	5.300	5.622	6.08%
Internal Insulation & Liner (kg):	81.700	81.070	-0.77%
Total Inert (kg):	342.800	413.100	20.51%
Burnout (kg):	314.400	367.740	16.97%
Motor Mass Fraction:	0.900	0.881	-2.11%
Motor Sizing Data			
Total Motor Length (m):	2.657	2.623	-1.28%
Case Length (m):	1.760	1.695	-3.69%
Nozzle Length (m):	0.929	0.927	-0.22%

TABLE 6.5 Orion 50S-GL Case Study Motor

Orion 50S GL Space Motor			
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**
Vacuum Total Impulse (kN-sec):	32,751.070	32,769.852	0.06%
Total Loaded Motor (kg):	13,106.500	13,166.670	0.46%
Propellant Mass (kg):	12,123.900	12,150.880	0.22%
Case Assembly Mass (kg):	479.600	538.050	12.19%
Nozzle Assembly Mass (kg):	166.800	136.480	-18.18%
TVC System Mass (kg):	47.600	50.241	5.55%
Igniter Assembly (kg):	9.100	10.565	16.10%
Internal Insulation & Liner (kg):	279.600	280.465	0.31%
Total Inert (kg):	982.600	1,015.000	3.30%
Burnout (kg):	917.700	913.000	-0.51%
Motor Mass Fraction:	0.930	0.923	-0.75%
Motor Sizing Data			
Total Motor Length (m):	7.093	7.322	3.23%
Case Length (m):	6.497	6.513	0.25%
Nozzle Length (m):	0.797	0.809	1.51%

TABLE 6.6 Minuteman First Stage Case Study Motor

Minuteman First Stage Booster Motor			
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**
Vacuum Total Impulse (kN-sec):	48,200.000	48,022.100	-0.37%
Total Loaded Motor (kg):	22,921.000	22,948.470	0.12%
Propellant Mass (kg):	20,781.000	20,831.030	0.24%
Case Assembly Mass (kg):	1,299.000	1,299.620	0.05%
Nozzle Assembly Mass (kg):	402.000	387.900	-3.51%
TVC System Mass (kg):	70.000	83.037	18.62%
Igniter Assembly (kg):	11.700	6.861	-41.36%
Internal Insulation & Liner (kg):	355.000	377.080	6.22%
Total Inert (kg):	2,139.700	2,117.640	-1.03%
Burnout (kg):	1,933.000	1,952.790	1.02%
Motor Mass Fraction:	0.912	0.907	-0.55%
Motor Sizing Data			
Total Motor Length (m):	7.490	7.638	1.98%
Case Length (m):	6.580	6.677	1.47%
Nozzle Length (m):	0.910	0.951	4.51%

The Orion 38, Orion 50, Orion 50S-GL, and the Minuteman First Stage motors provide good examples with which to test the flexibility of the SRM-CDOS design code. The four different motors provide a wide range of motor masses, 895.5 kg to 22,291 kg, and thrust requirements, 31.89 kN to 865.62 kN, to be examined. Each motor has a fairly neutral thrust profile, but the nozzle design for the first three motors is slightly recessed, and the Minuteman first stage motor has four smaller nozzles instead of a single large one. Each motor also includes thrust vectoring system with movable nozzles.

Examining Tables 6.3 and 6.6, the SRM-CDOS provides an accurate design to the manufacture's listed specifications. The majority of the dominate design areas were matched within the 5% error, including total impulse, total mass, propellant mass, and motor geometry. The SRM-CDOS was able to predict the total motor mass very accurately to within 3.1% for all four of the extra case study motors.

The specific design areas involving case mass, nozzle mass, and insulation had a little broader matching range, 0.05% to 39.65%. Consistent trouble areas were the igniter system mass and the TVC system mass, with matching ranges of 5.5% to 71%. These specific design areas tended to make up less than 12% of the total motor mass. The errors may seem high, but they are dealing with smaller masses, and small mass differences between the design results and the manufacture's specifications will yield a higher percent error. These errors can be accounted for by the simplifications used by the SRM-CDOS code. These specific design areas tend to have greater design detail placed on them by the manufacturer, and that would account for the mass difference noted in tables 6.3 to 6.6. When all of the specific design areas are included into the total motor mass, the relative errors in each tend to balance, resulting in the total mass estimation being very accurate compared to the manufacture's data.

Results of the Case Optimizations

After the SRM-CDOS had matched to each case study motor, the program was configured to do three separate mass optimization runs on each case motor. The first optimization run used the optimization variables of chamber pressure, motor diameter, and nozzle half angle. Each was set in an acceptable range where the optimized motor results would fall. Typical optimization ranges were:

1. Chamber Pressure, 3 MPa to 20 MPa
2. Motor Diameter, 0.5 meters to 5 meters
3. Motor Nozzle Half-Angle, 6 degrees to 25 degrees

The last two optimization runs additionally optimized the structural material selection (all cases, see Table 2.2) and propellant selection (all cases, see Table 2.1).

Figures 6.1 thru 6.6 illustrate the results of the three mass optimization runs as well as the matching run mass. Mass savings are also provided in each figure for each optimization run.

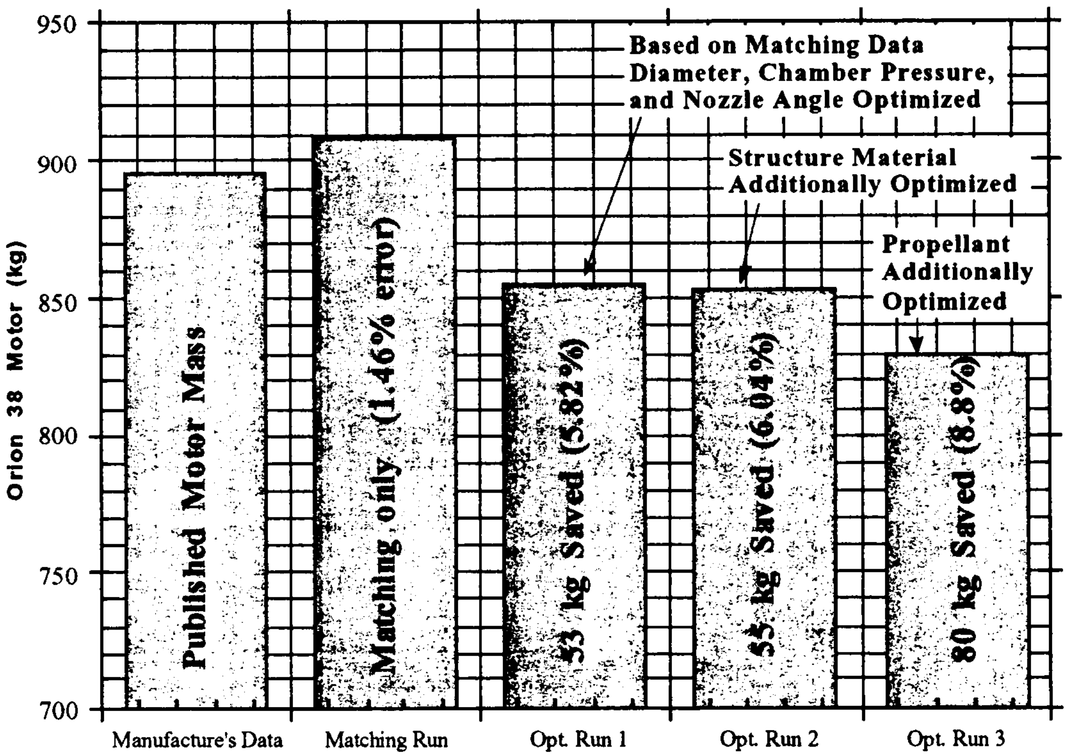

FIGURE 6.1 Orion 38 Optimized Case Study Motor

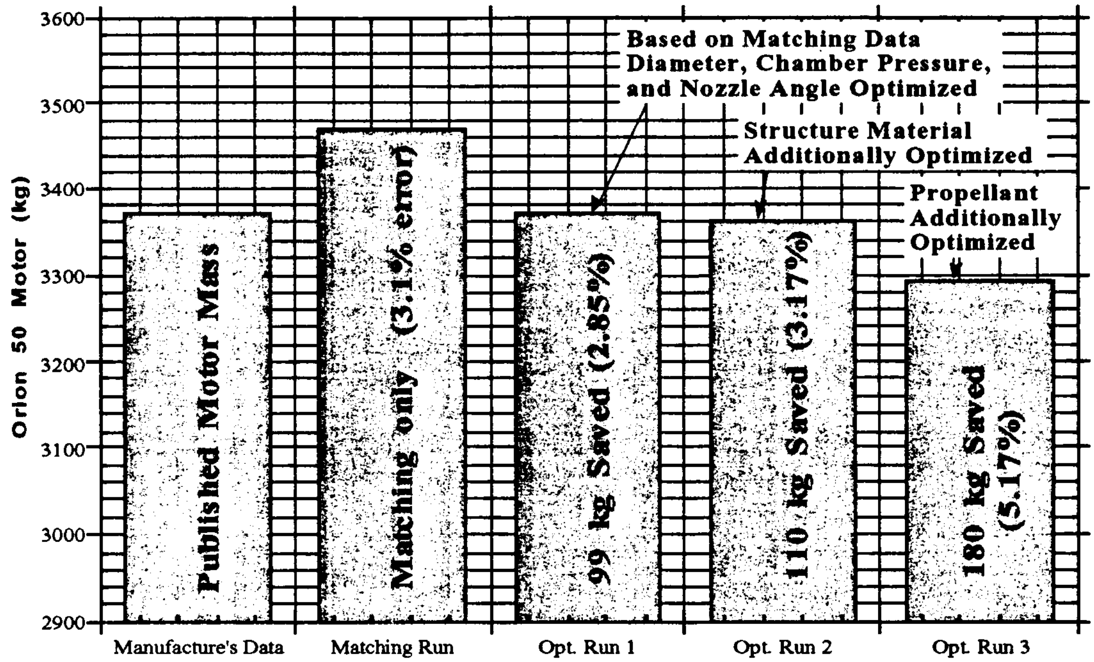

FIGURE 6.2 Orion 50 Optimized Case Study Motor

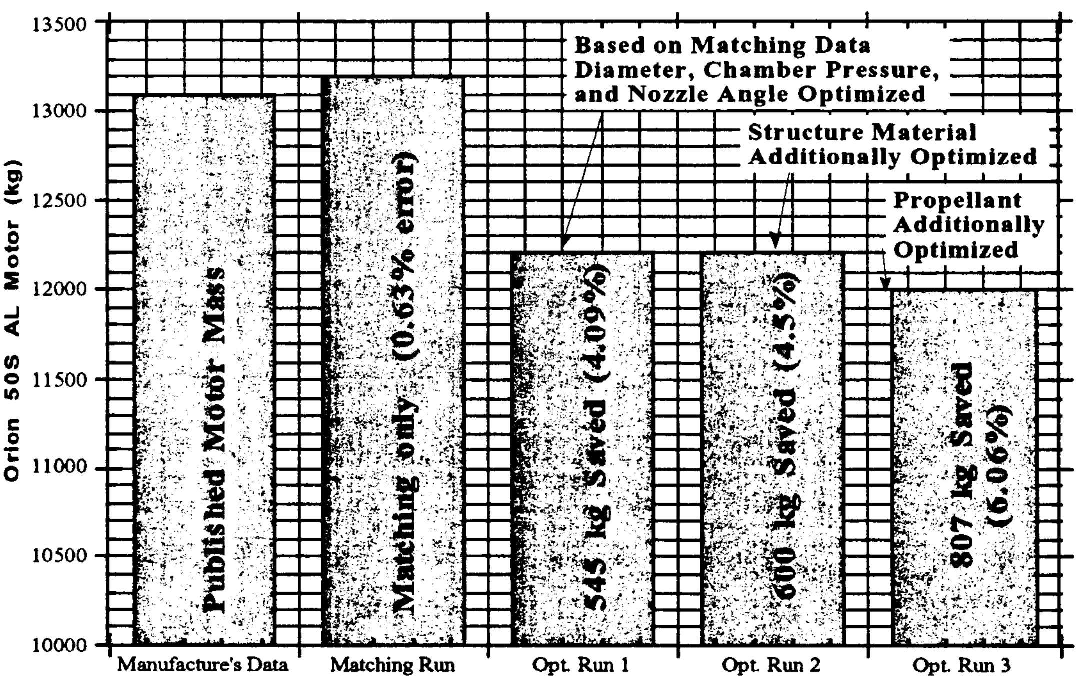

FIGURE 6.3 Orion 50S-AL Optimized Case Study Motor

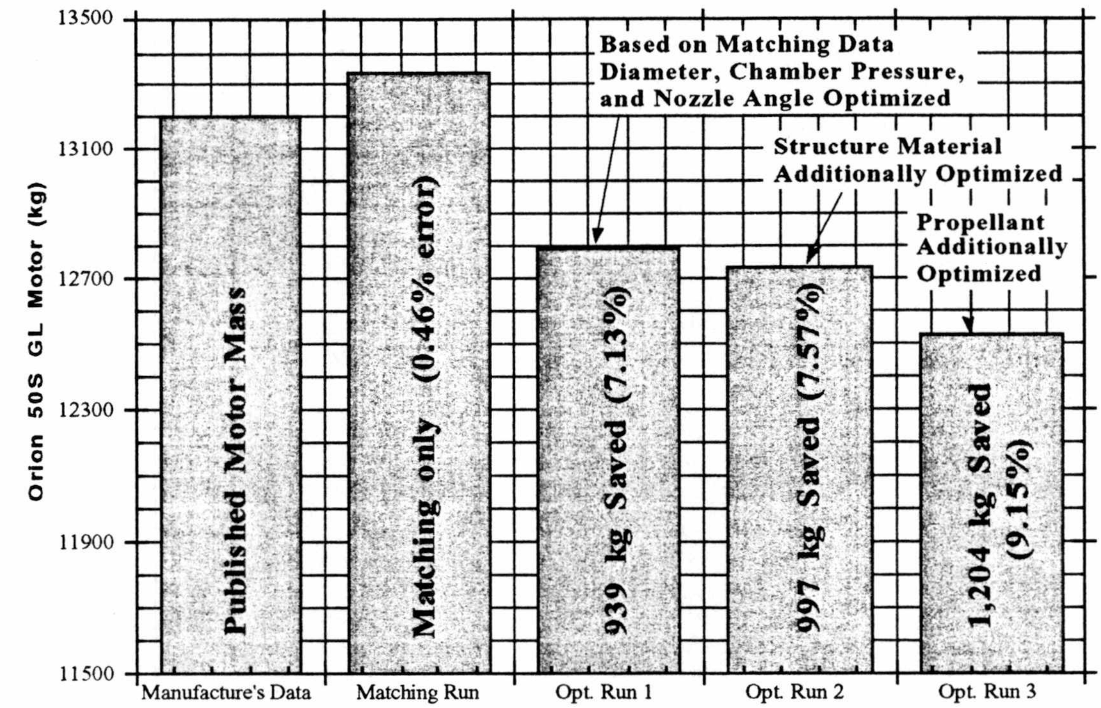

FIGURE 6.4 Orion 50S-GL Optimized Case Study Motor

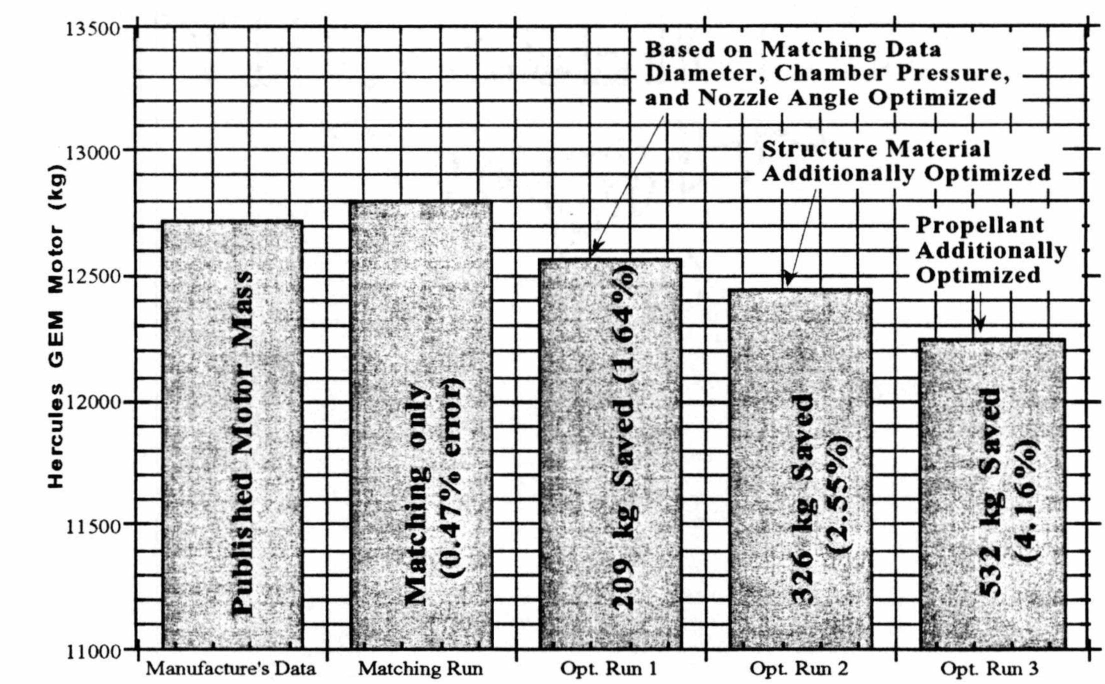

FIGURE 6.5 Hercules GEM Optimized Case Study Motor

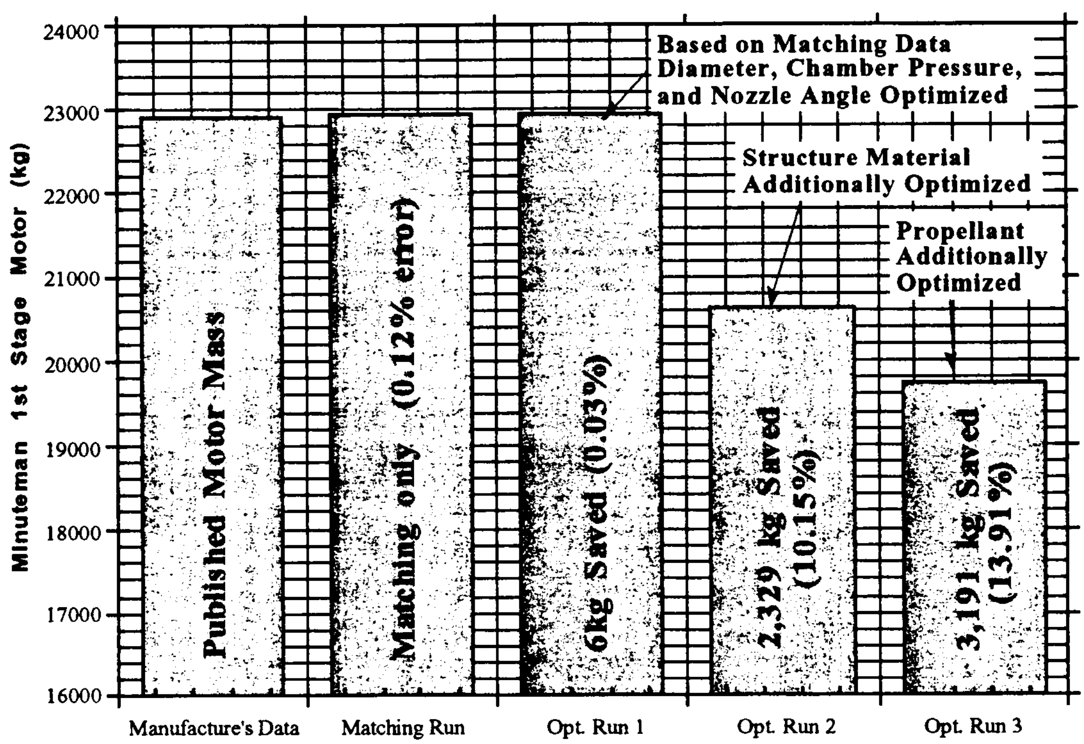

FIGURE 6.6 Minuteman First Stage Optimized Case Study Motor

Examining Figures 6.1 and 6.6, the SRM-CDOS code is able to provide mass savings for each of the six motors entered into the user interface. Up to 13.9% mass savings can bee seen for a fully optimized motor. Note that the mass savings of 13.9% resulted mainly from the Minuteman motor structural material changing from a steel alloy to high strength fiber composites.

CHAPTER 7

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

With the use of computer aided design systems, such as the SRM-CDOS, solid rocket motor design prototypes can be rapidly formulated and mass optimized. SRM-CDOS combines an user interfacing Hypertext control shell, a Fortran optimization code, and an Expert System artificial intelligence database to speed up the design process and maintain an user-friendly computing environment. The utilization of the unique combination these three computer methodologies has been shown to increase user proficiency and productivity by reducing the work load on the design engineer.

The user interface provides a friendly environment for SRM parameter input, and allows easy data modification for varying data runs. The design engineer can quickly obtain rocket motor designs that satisfy the performance mission of the SRM as well as specify the ranges and types of mass optimization runs to be completed.

SRM-CDOS has also shown that a computerized tool can facilitate and define tradeoffs between various aspects of the SRM design and optimization requirements. These tradeoffs are translated into a SRM design which offers substantial mass savings over a non-optimized SRM design. This mass savings can translate into lower costs, increased payload capability, and overall better performance. The SRM-CDOS has succeeded in its main purpose; to create a system which would aide the SRM design engineer in making the best initial design selections and reducing the overall "design cycle time" of a SRM project.

Recommendations

The utilization of the unique combination these three computer methodologies has been demonstrated with very accurate results. As with any project, there is always room for improvement. Future versions of the SRM-CDOS may incorporate the following suggestions to improve the design results.

1) Addition of a user selected thrust profile. This would give the designer the flexibility to tailor the motor's thrust profile to match a specific mission.

2) Increase the number of nozzle case studied to provide better empirical formulas for the estimation of nozzle mass and the estimation of the mass increases required for thrust vector control.

3) Addition of an atmospheric data subroutine to provide the user with an instant reference to an entered altitude.

4) Enable the cost optimization routine in the SRM-CDOS. This would require additional studies into the costs, man hours, and availability of the materials used to build the SRM.

5) Enable the expanded structural analysis routine in the SRM-CDOS. This would allow the option of additional structural analysis criteria, involving bending forces, vibration, and acceleration loads, being added to the motor casing structure design.

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BIBLIOGRAPHY

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APPENDICES

APPENDIX 1

HyperCard SRM-CDOS User Interface Stack

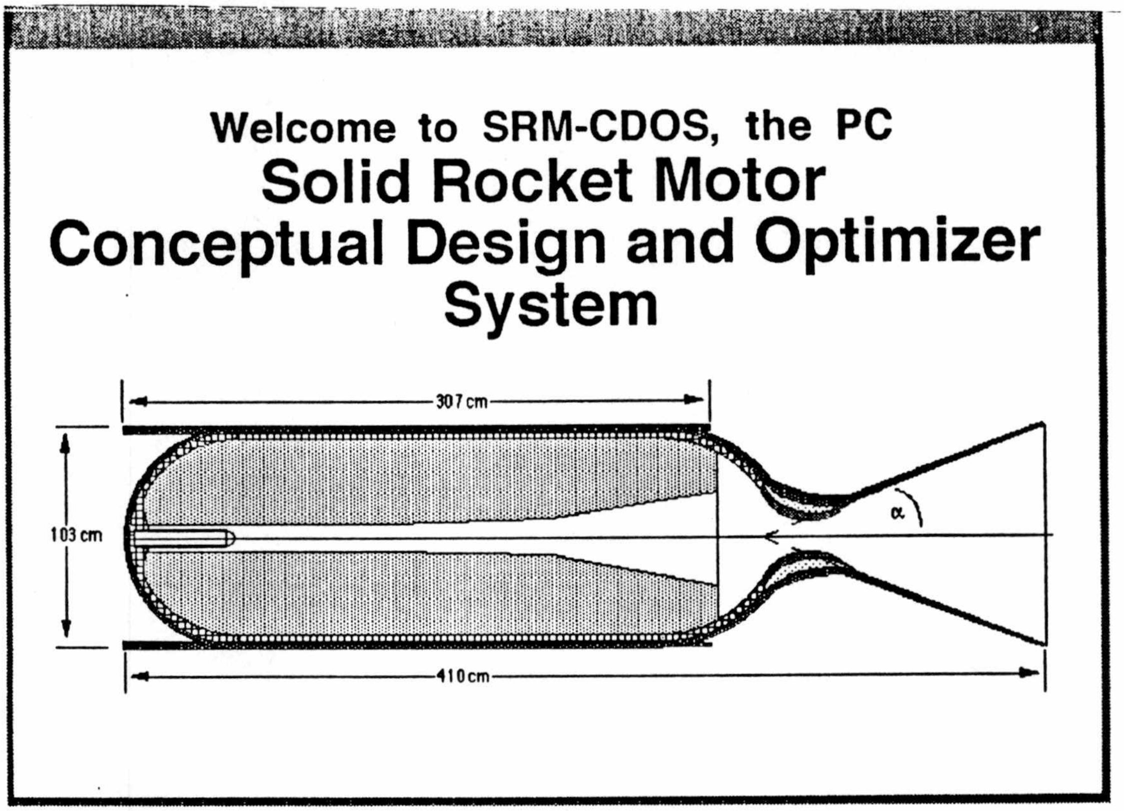

FIGURE A1.1 HyperCard SRM-CDOS User Interface Stack Introduction Card

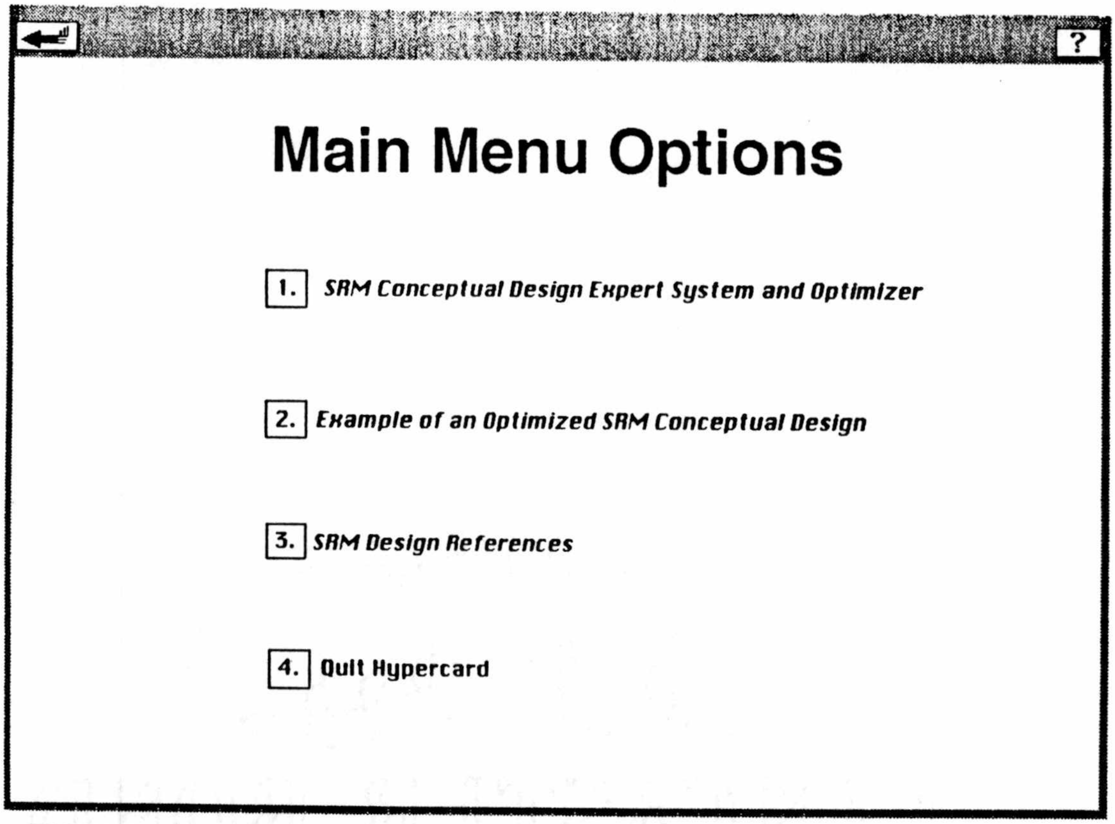

FIGURE A1.2 HyperCard SRM-CDOS User Interface Stack Main Menu Card

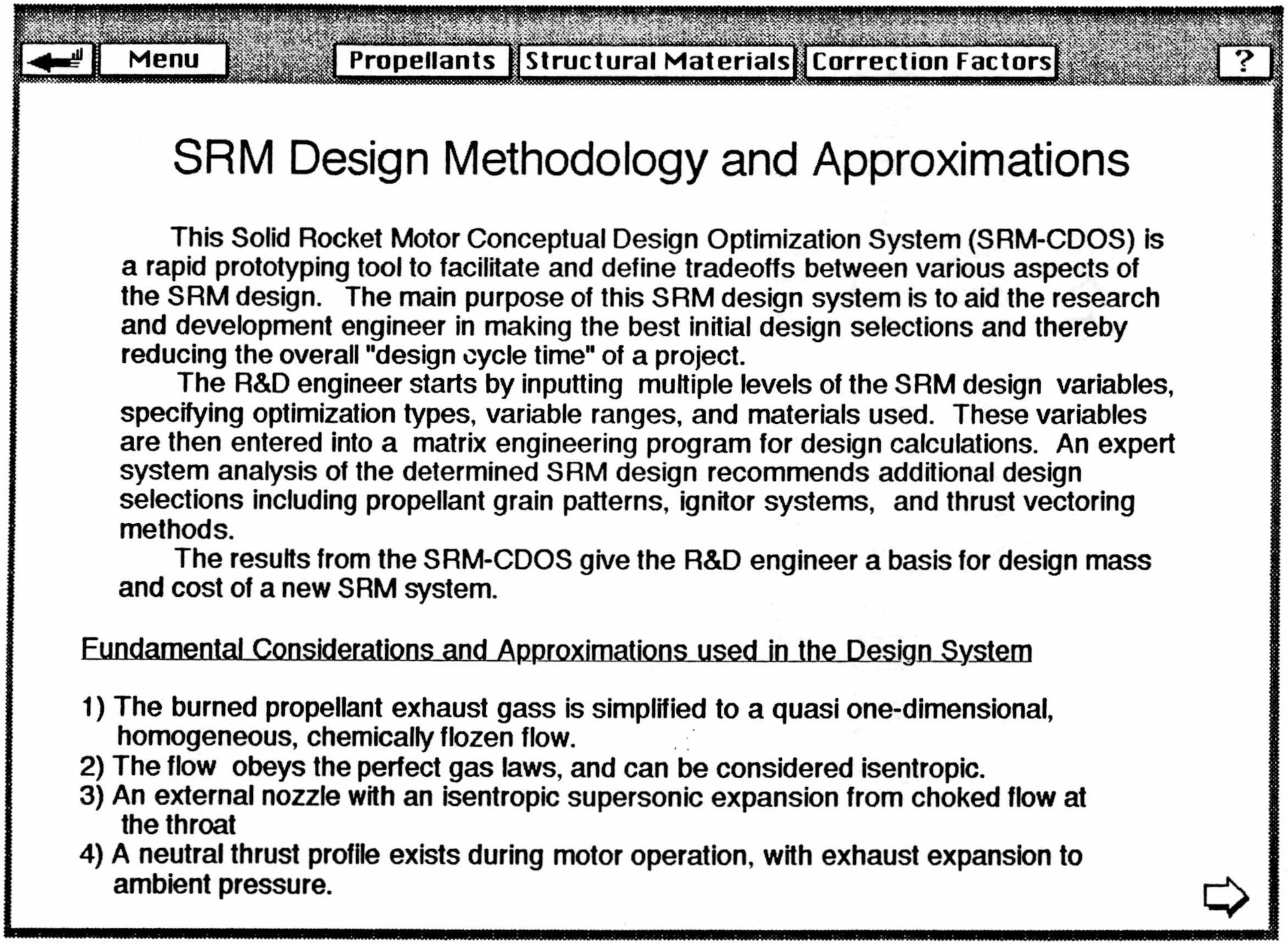

FIGURE A1.3 HyperCard SRM-CDOS User Interface Stack Design Methods Card

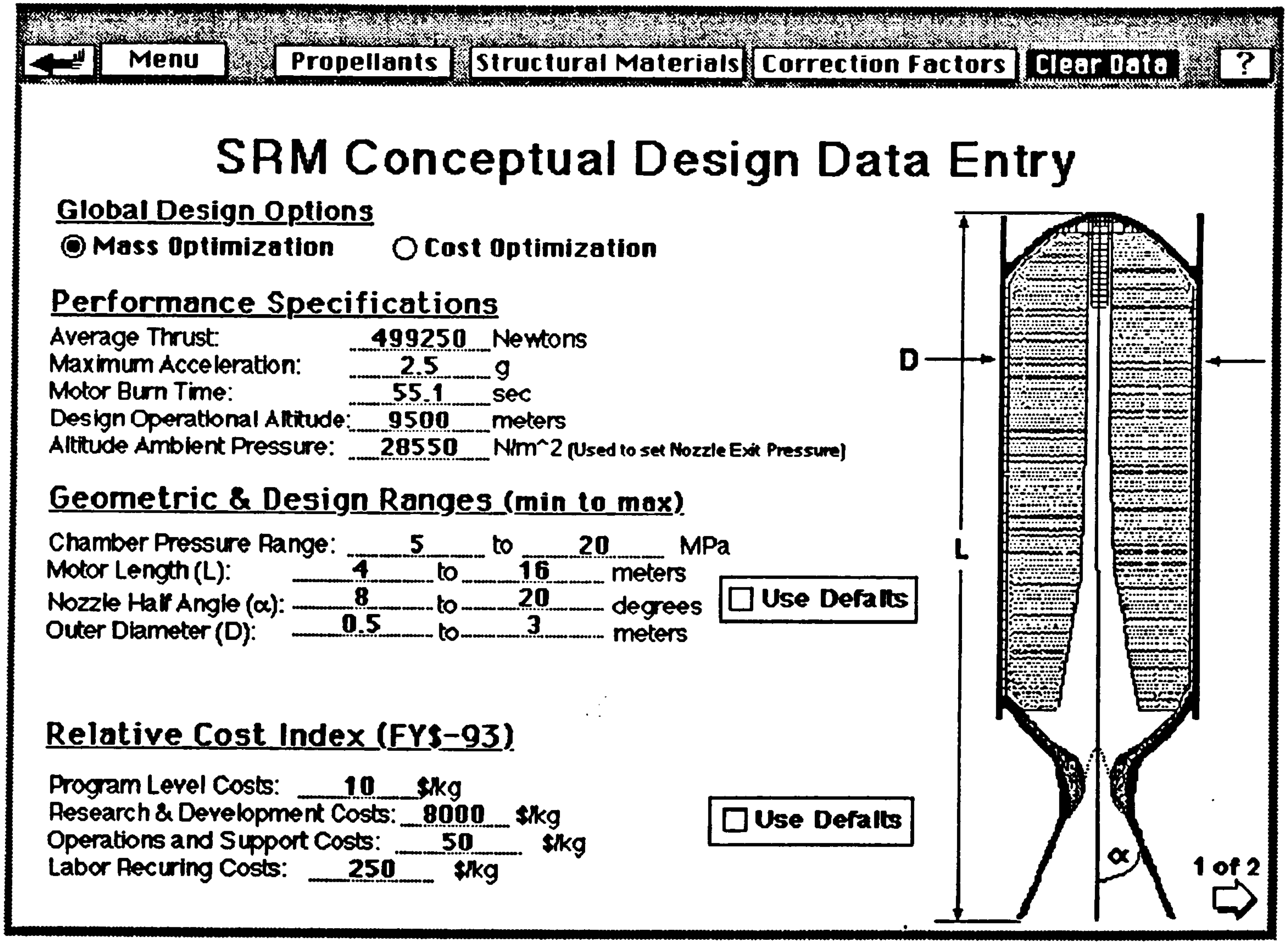

FIGURE A1.4 HyperCard SRM-CDOS User Interface Stack Data Entry Page #1 Card

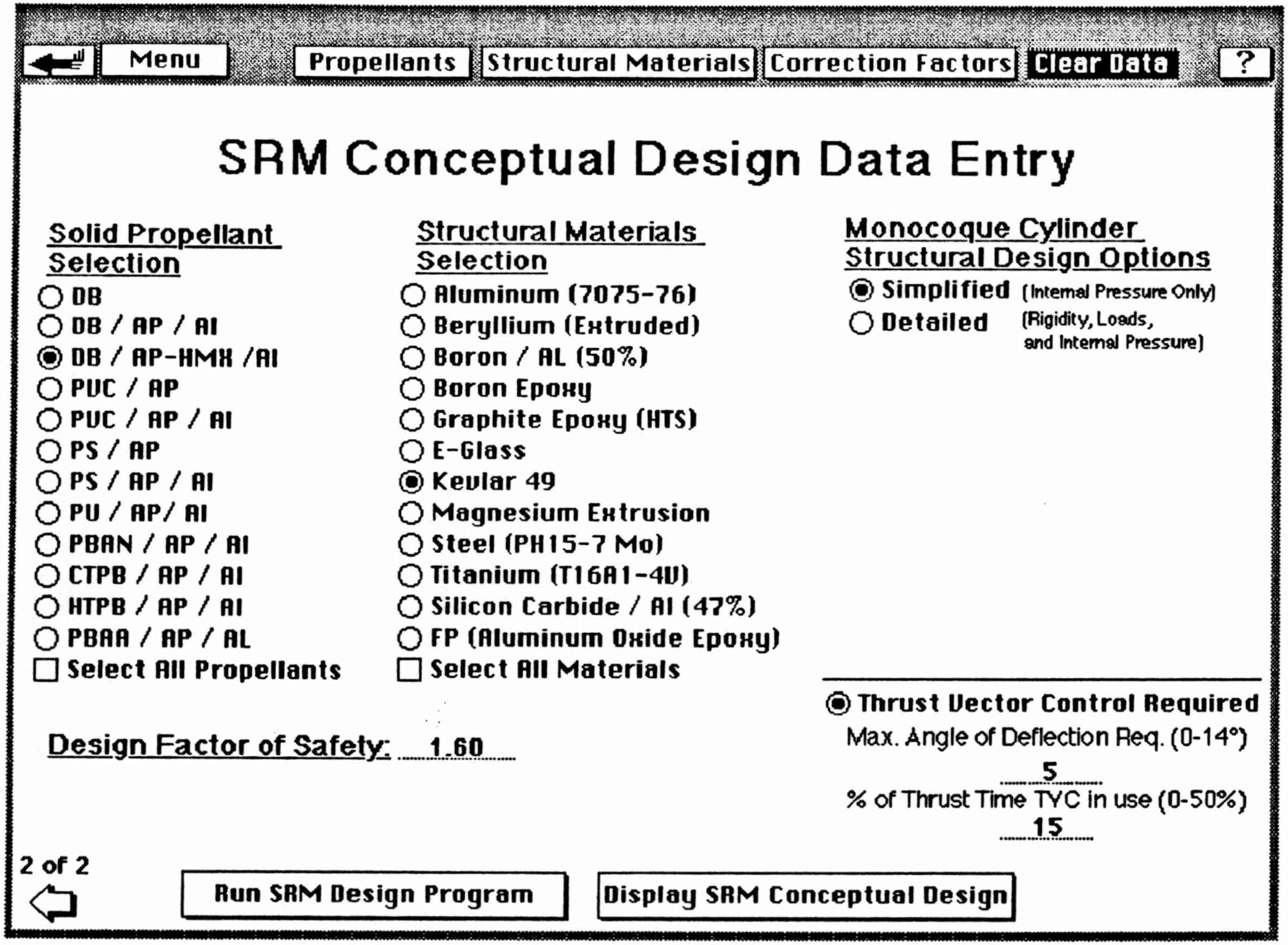

FIGURE A1.5 HyperCard SRM-CDOS User Interface Stack Data Entry Page #2 Card

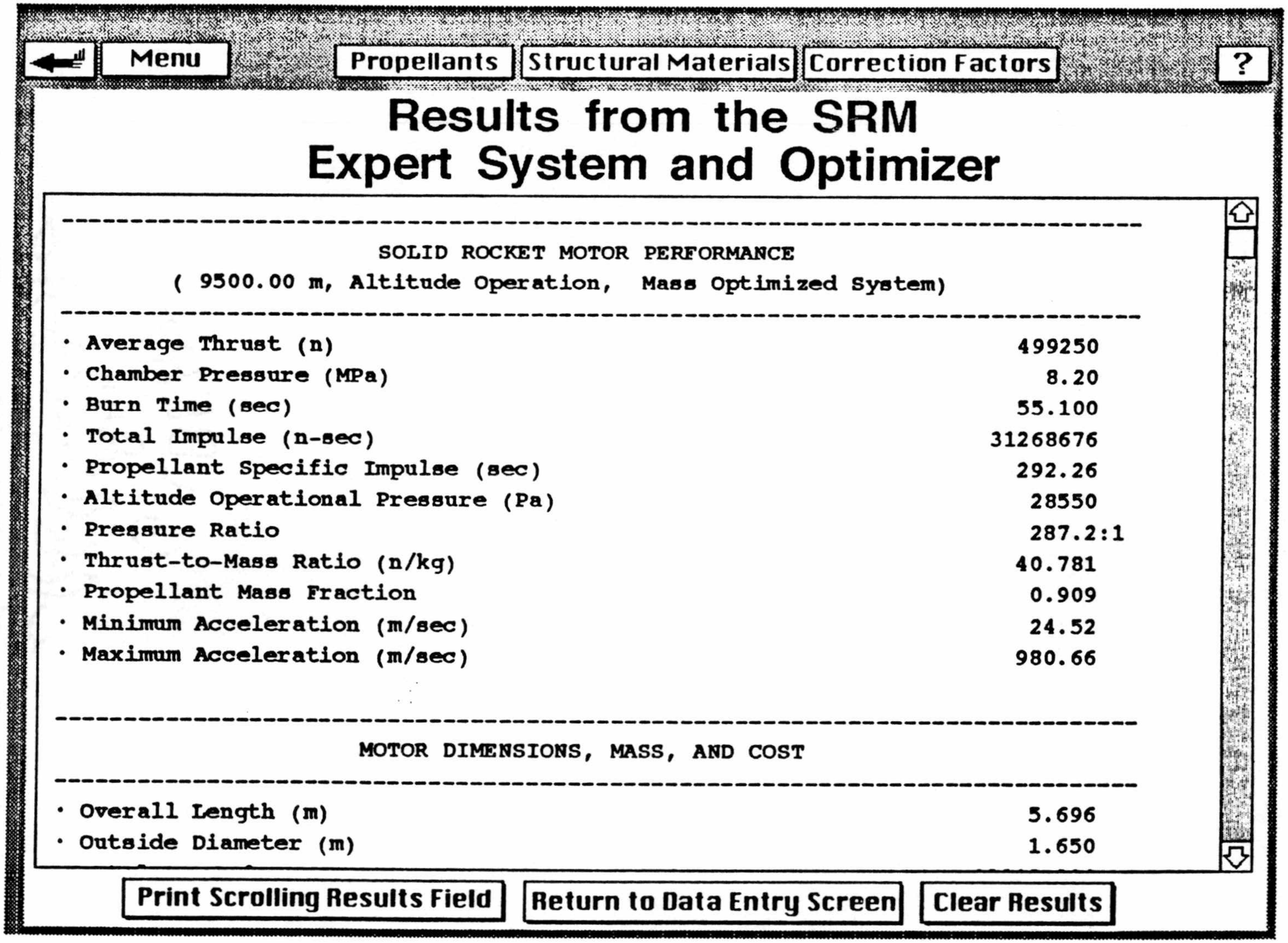

FIGURE A1.6 HyperCard SRM-CDOS User Interface Stack Design Results Card

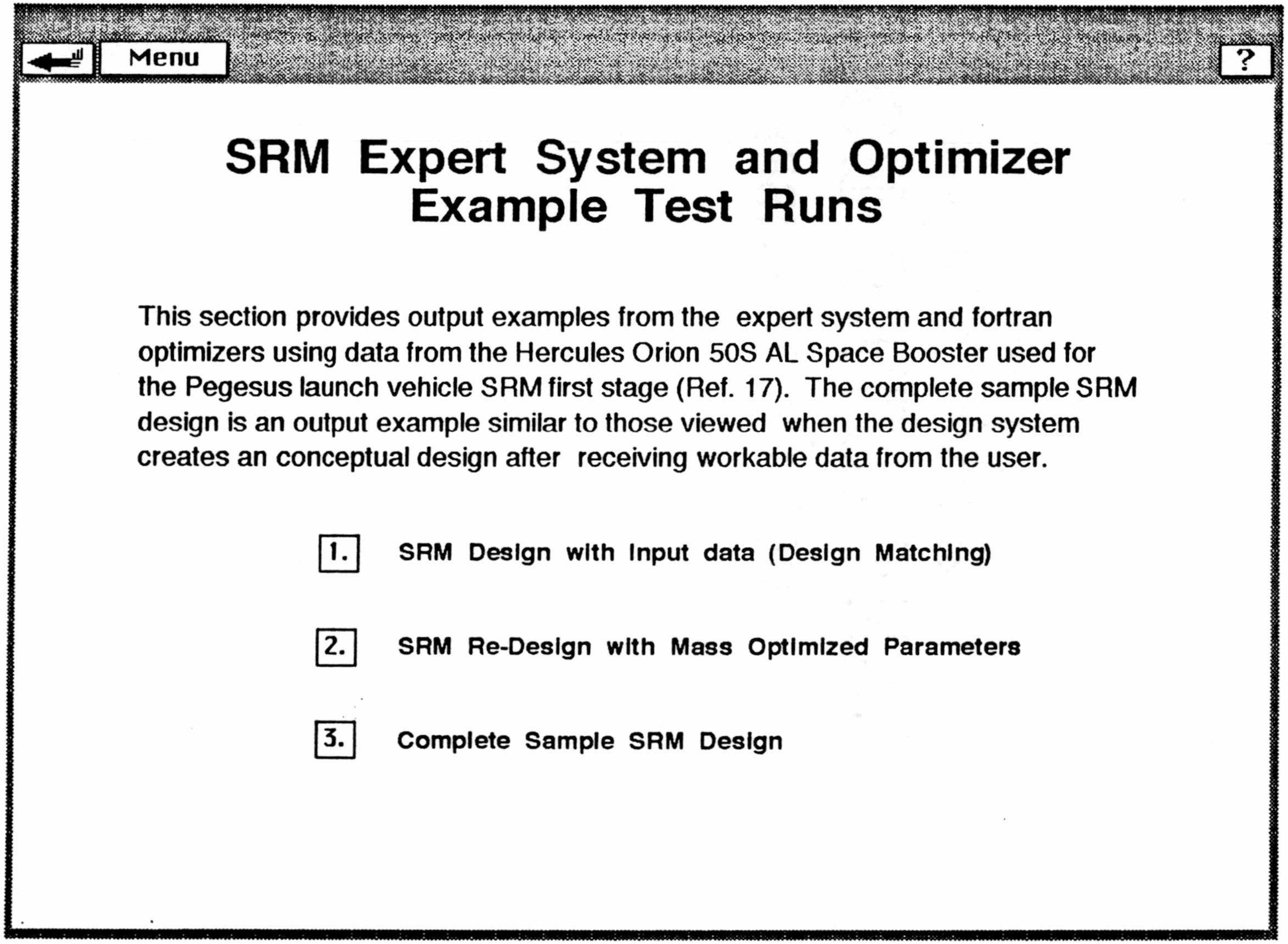

FIGURE A1.7 HyperCard SRM-CDOS User Interface Stack Example Runs Menu Card

Menu | Propellants | Structural Materials | Correction Factors | ?

SRM Design with Input data (Design Matching)

Orion 50S AL Space Launch Booster (Pegasus first stage)

User Entered Manufacture's Data
Burn Time (sec): 72.400
Average Chamber Pressure (MPa): 5.840
Nozzle Angle (°): 22.50
Average Vacume Thrust (kN): 486.000
Structure (type): High Tensile Strength Graphite Epoxy
Propellant (type): HTPB / AP / Al

	Manufacture's Performance Data	SRM-CDOS Data	Data Rel. Error
Vac Total Impulse (kN-sec):	35,108.060	35,187.420	0.23%
Total Loaded Motor (kg):	13,451.000	13,404.885	0.73%
Propellant Mass (kg):	12,157.900	12,332.549	0.63%
Case Assembly Mass (kg):	547.900	553.530	1.66%
Nozzle Assembly Mass (kg):	235.400	240.747	2.58%
TVA System Mass (kg):	0.000	0.000	0.00%
Igniter Assembly (kg):	9.100	10.724	16.26%
Internal Insulation & Liner (kg):	279.600	267.336	1.14%
Total Inert Mass(kg):	1,078.700	1,072.336	1.83%
Burnout Mass (kg):	1,013.900	974.072	-3.93%
Motor Mass Fraction:	0.920	0.918	-0.22%
Motor Sizing Data			
Total Motor Length (m):	7.919	8.145	2.85%
Motor Width (m):	2.550	2.550	0.00%
Case Length (m):	6.497	6.530	0.51%
Nozzle Length (m):	1.650	1.620	-1.82%

FIGURE A1.8 HyperCard SRM-CDOS User Interface Stack Example Runs #1 Card

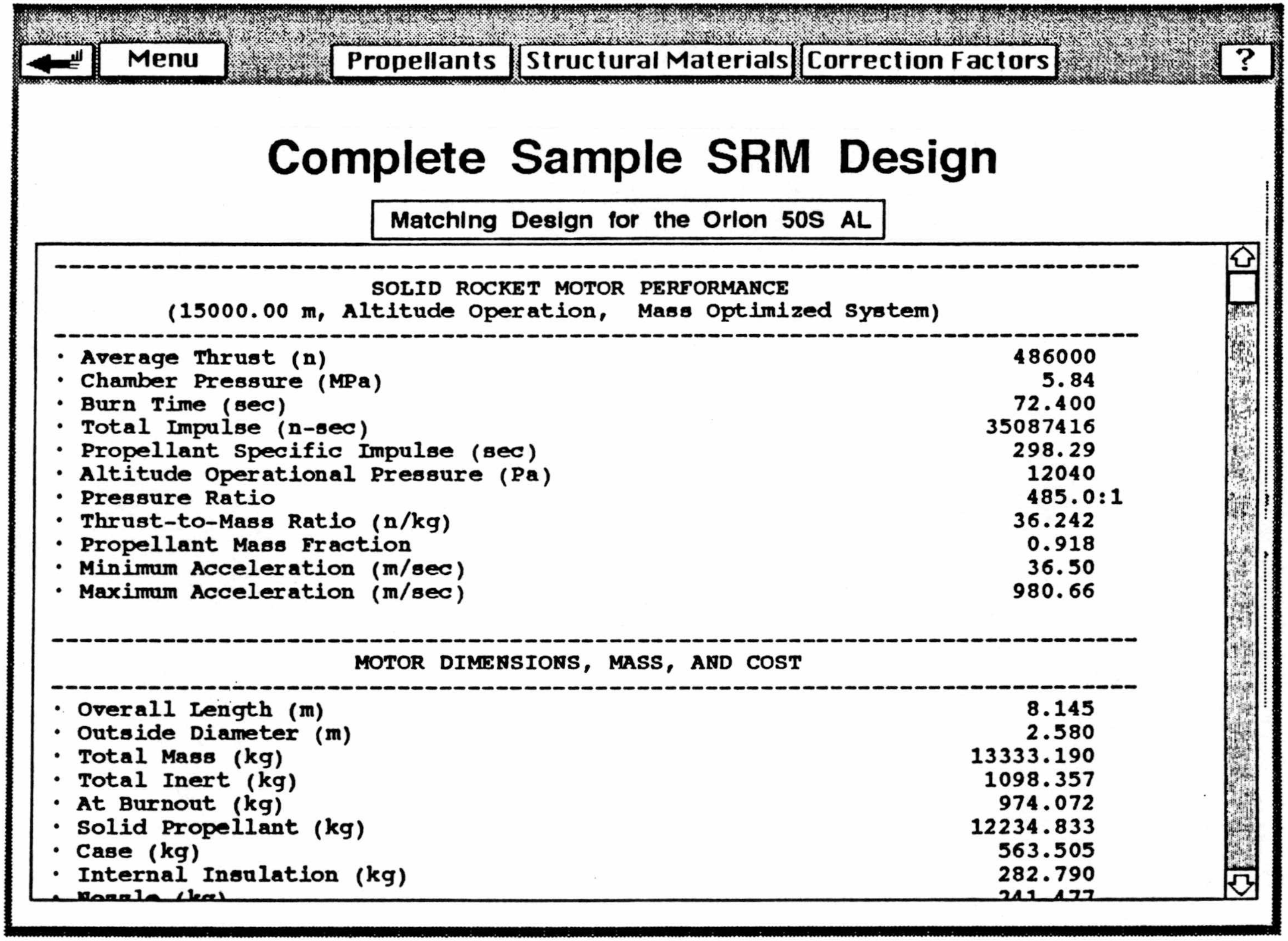

FIGURE A1.9 HyperCard SRM-CDOS User Interface Stack Example Runs #2 Card

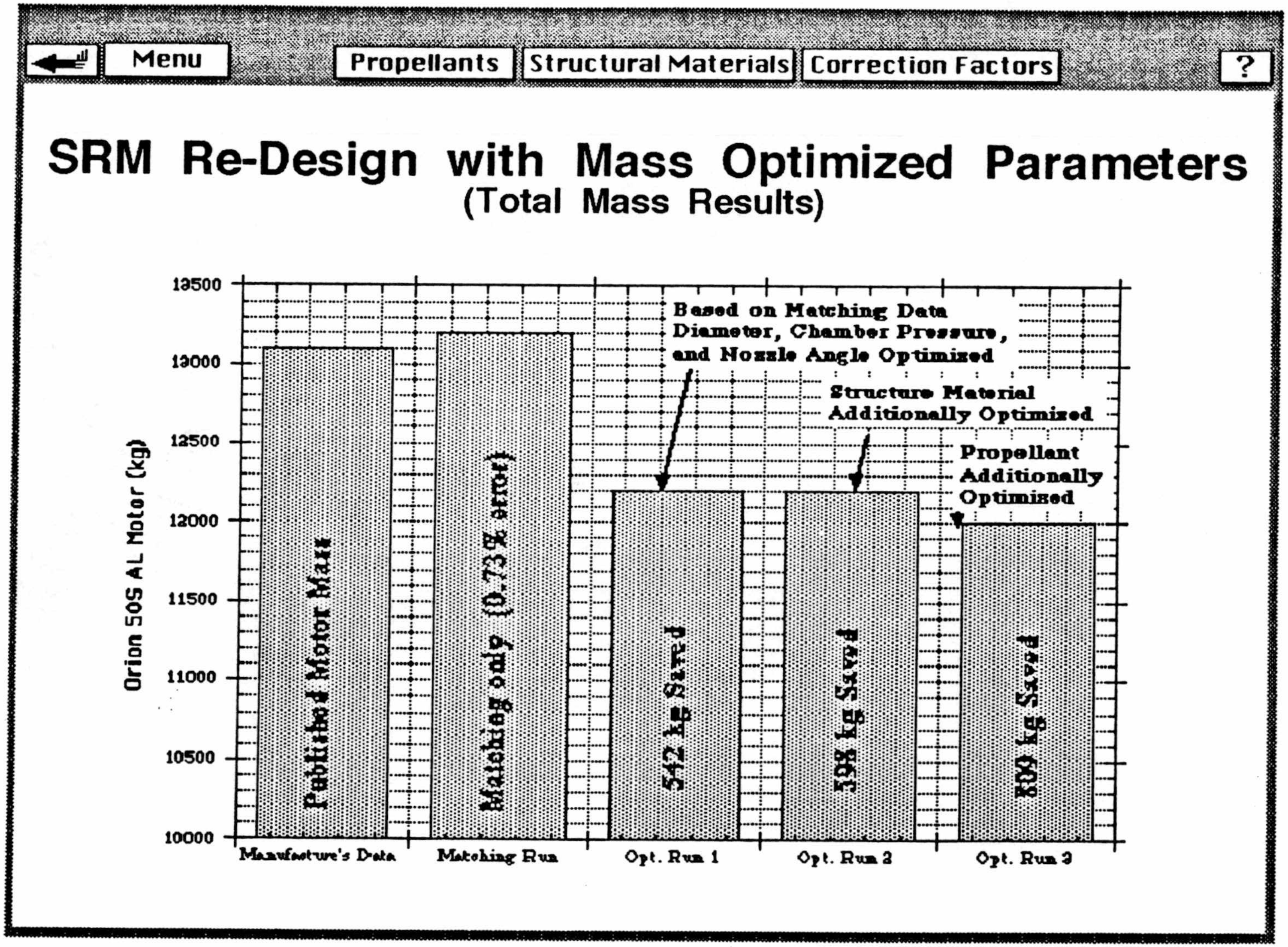

FIGURE A1.10 HyperCard SRM-CDOS User Interface Stack Example Runs #3 Card

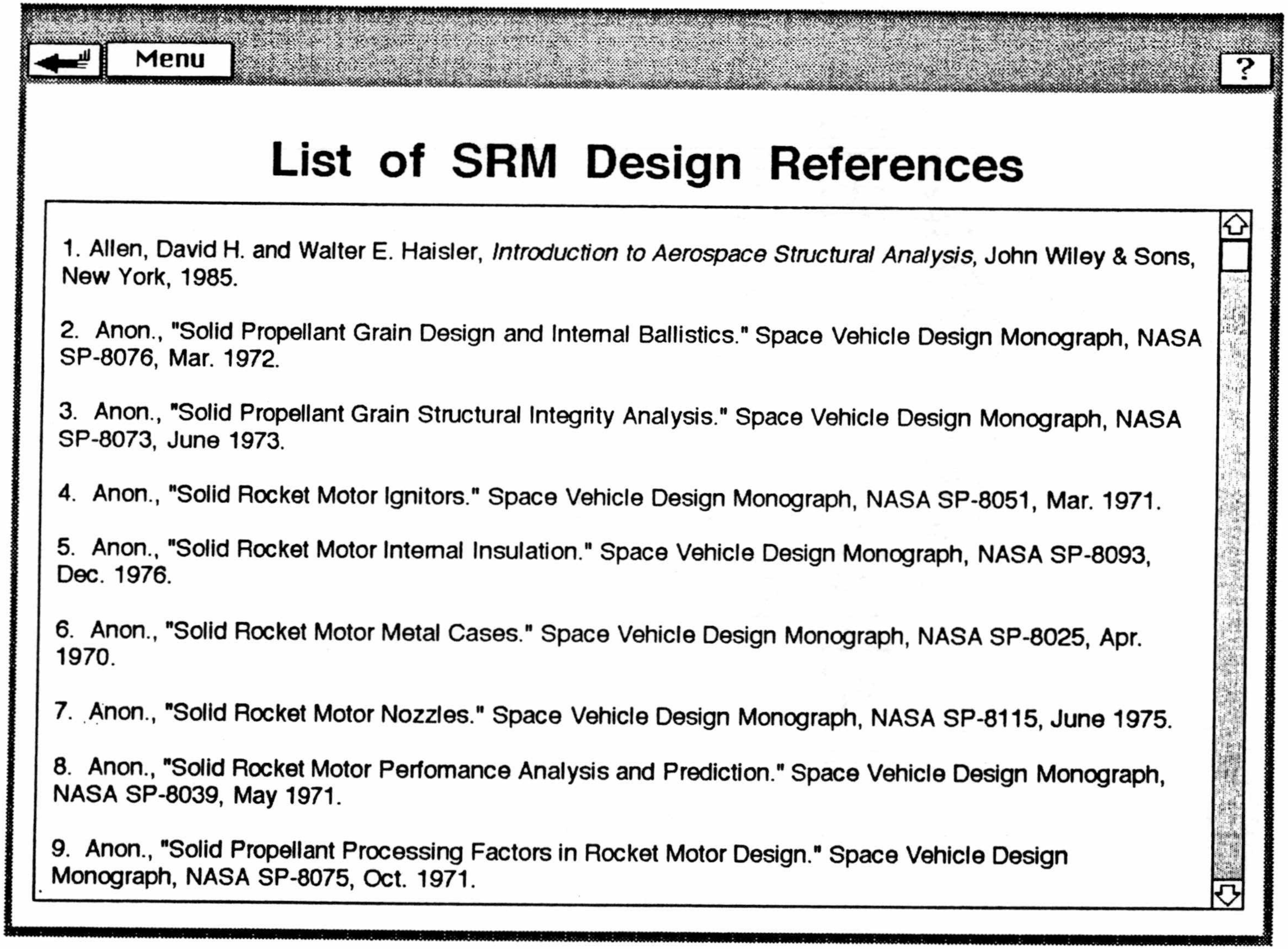

FIGURE A1.11 HyperCard SRM-CDOS User Interface Stack SRM References Card

APPENDIX 2

HyperCard SRM-CDOS Reference Stacks

Propellant Reference Material

Exit Ref.

	Propellant Name	Approx. Burn Constant (a)	Approx. Pressure Exp. (n)	Exhaust Av. Mol. Wt. (M)	Approx. Specific Heat (k)	Approx. Density kg/m^3 (ρ)	Approx. Flame Temp. °C (To)	Propellant Cost ($93) $ / kg
1.)	DB	0.000101	0.3	23.3	1.23	1605	2260	20.67
2.)	DB / AP / Al	0.000036	0.4	28.9	1.18	1800	3600	9.92
3.)	DB / AP-HMX / Al	0.000006	0.49	28.9	1.17	1800	3700	11.57
4.)	PVC / AP	0.000029	0.38	24.5	1.22	1690	2540	2.48
5.)	PVC / AP / Al	0.000046	0.35	25	1.21	1772	3090	2.48
6.)	PS / AP	0.00001	0.43	25.5	1.2	1716	2590	6.2
7.)	PS / AP / Al	0.000044	0.33	25	1.21	1716	2760	6.2
8.)	PU / AP / Al	0.00025	0.21	26.6	1.19	1772	3320	3.31
9.)	PBAN / AP / Al	0.000078	0.33	26	1.2	1772	3200	2.9
10.)	CTPB / AP / Al	0.000021	0.4	25.5	1.2	1772	3200	4.55
11.)	HTPB / AP / Al	0.0000175	0.38	25.5	1.2	1772	3300	8.27
12.)	PBAA / AP / Al	0.000033	0.35	26.6	1.19	1772	3320	9.1

Acronyms and Symbols:

Al, Aluminum
AP, Ammonium Perchlorate
CTPB, Carboxy-Terminated Polybutadiene
DB, Double Base
HMX, Cylcotetramethylene Tetranitramine
HTPB, Hydroxy-Terminated Polybutadiene
PBAA, Polybutadiene-Acrylic Acid Polymer
PBAN, Polybutadiene-Acrylic Acid-Acrylonitrile Terpolymer
PS, Polysulfide
PU, Polyurethane
PVC, Polyvinyl Chloride

Defalt Values

FIGURE A2.1 HyperCard SRM-CDOS Propellant Reference Stack

Structural Reference Material

Exit Ref.

	Structural Material Name	Material Form	Density kg/m^3 (ρ)	Tensile Ult. Str 10^6 N/m^2	Tensile Yld. Str. 10^6 N/m^2	Young's Modulus 10^9 N/m^2 (E)	Manufacture Workability Cost Factor	Materials Cost $ / kg
1.)	Aluminum	7075-T6	2800	523	448	71	1	4.5
2.)	Beryllium	Extruded	1850	620	413	293	5	2200
3.)	Boron / Al	50%	2600	1491	1491	214	4	1600
4.)	Boron Epoxy	.	2010	1337	1337	206	4	320
5.)	Graphite Epoxy	HTS	1490	1340	1340	160	3.5	30
6.)	E-Glass	.	1094	1000	1000	45	3.5	2.5
7.)	Kevlar 49	.	1300	1380	1380	76	3	45
8.)	Magnesium	Extrusion	1770	269	199	44	2	10
9.)	Steel	Ladish D6AC	7600	1550	1344	200	1.5	15
10.)	Titanium	T16A1-4V	4430	1103	999	110	1.9	60
11.)	SiC / Al	37%	2850	1034	1034	172.4	2.6	220
12.)	Al-Oxide Epoxy	50%	3100	1380	1380	200	3.2	300

Casing Liner Type
- ○ **Silica / NBR**
- ◉ **Asbestos / NBR**
- ○ **Mastic**
- ○ **Silica / Phenolic**

Casing Liner Data:
Material Name: EPDM / Aramid
Density kg/m^3: 1269
Safety Factor: 1.9
Cost $/kg: 400

Throat Insert Data:
Material Name: Carbon / Carbon
Density kg/m^3: 1850
Cost $/kg: 650

Defalt Values

FIGURE A2.2 HyperCard SRM-CDOS Structural Reference Stack

Real Rocket Correction Factors

Exit Ref.

Velocity Correction Factor (VCF) Currently Selected: 0.955

VCF is approximately the ratio of the actual specific impulse to the ideal or theoretical specific impulse. Its value ranges between 0.85 and 0.98 with an average near 0.92.

Mass Discharge Correction Factor (MDCF) Currently Selected: 1.01

MDCF is defined as the ratio of mass flow rate in a real rocket vs. an ideal rocket. The value of this factor is usually greater than one (0.98 to 1.15). The actual flow is larger due to:

- The molecular weight and density of the combustion gasses increases slightly while passing though the nozzle.
- Heat transfer to the walls slightly lower the flow temperature, thereby increasing the density of the flow.
- The specific heat and other gas properties change slightly through an actual nozzle.
- Incomplete combustion increases the density of the exhaust gases.
- Nozzle thermal expansion and/or erosion.

Solid Propellant Sliver Factor (SLIVER) Currently selected: 0.058

Propellant sliver is the unburned propellant remaining in the motor case or lost through the nozzle at the time of web burnout. A well designed propellant grain has slivers that are less than 2 percent of the grain mass.

Multiplicative Factors for Motor Development Heritage Currently Selected: 1.1

(Applied to R&D costs only)

- New SRM design with advanced development	> 1.1
- Nominal new design with some prior heritage	1.0
- Major Modification to an existing design	0.7 - 0.9
- Moderate Modifications	0.4 - 0.6
- Basically existing design	0.1 - 0.3

Defalt Values

FIGURE A2.3 HyperCard SRM-CDOS Correction Factors Reference Stack

APPENDIX 3

Expert System If-Then Knowledge Base Code

```
TITLE  SRMCDOS Design Advisory System


STRING Grain
    ANDMounting
    ANDIgniter
    ANDTVC Choice
    ANDLiner Note
    ANDPName
    ANDSName
    ANDIName
    ANDTIName
    AND Optimizer Note
    ANDError


ATTRIBUTE  Mass
    ANDAngle
    ANDTVC Mass


NUMERICTotal Impulse
    ANDBurn Rate
    ANDSpecific Impulse
    ANDPropellant Mass
    ANDWeb Thickness
    ANDWeb Fraction
    ANDPropellant Volume
    ANDBurn Area
    ANDWall Thickness
    ANDLiner Thickness
    ANDEffective Length
    ANDLD Ratio
    ANDFree Volume
    ANDVolumetric Fraction
    ANDThroat Area
    ANDExpansion Ratio
    ANDExit Area
    ANDThroat Diameter
    ANDExit Diameter
    ANDNozzle Length
    ANDTotal Motor Length
    ANDCase Mass
    ANDLiner Mass
    ANDNozzle Mass
    ANDThroat Insert
    ANDTotal Mass
    ANDMin Accel
    ANDMax Accel
    ANDMass Fraction
    ANDProduction Cost
    ANDLiner Flag
    ANDMass Optimized
    ANDCost Optimized
    ANDAverage Thrust
    ANDBurn Time
    ANDAltitude Pressure
```

```
    ANDDesign Altitude
    ANDSafety Factor
    ANDStructural Design
    ANDMax Angle
    ANDTime Used
    ANDPyrogen
    ANDPyrotechnic
    ANDLayerPyro
    ANDIgniter Mass
    ANDTVC Angle
    ANDFuel Mass
    ANDLITVC Mass
    ANDFlex Mass
    ANDGimble Mass
    ANDFinal TVC
    ANDPressure Ratio
    ANDThrustMass Ratio
    ANDMiscell Mass
    ANDTotal Inert
    ANDBurnout
    ANDMotor Diameter
    ANDPDensity
    ANDPExponent
    ANDPTemp
    ANDPHeat Ratio
    ANDPVelocity
    ANDSDensity
    ANDIDensity
    ANDCone Angle
    ANDChamber Pressure
    ANDTVC Option
    ANDRD Cost
    ANDCase Length
    ANDTank Mass
    ANDTank Length
    ANDTank Thickness


INIT   Nitrogen Density := 1442
    ANDNitrogen Tetraoxide := 450
    ANDPi := 3.141592654
    ANDgo := 9.81
    ANDTank Diameter:= 0.3
    ANDTank Pressure:= 649480
    ANDMaterial Strength:= 137000000
    ANDMaterial Density:= 1490


MULTI  Mass
    ANDAngle
    ANDTVC Mass


EXHAUSTIVE    Grain Check


FILE FData.txt
```

```
!-------------------------------
!                    Goals
!-------------------------------

1.  Information Checked
      1.1.  Grain Check
             1.1.1.  Igniter Check
                    1.1.1.1.  TVC Check
                           1.1.1.1.1. Finished
                    1.1.1.2 Finished
             1.1.2.  Finished
      1.2.  Finished
2.  Finished

!------------------------
!              SRM Data Input
!------------------------

RULE   Set Igniter Masses
  THEN Igniter Cal
   AND Pyrogen := 6 + (1.019*(Free Volume)^0.7)*1.1
   AND Pyrotechnic := 4 + (1.019*(Free Volume)^0.7)*1.3
   AND LayerPyro := 1.1 + (1.019*(Free Volume)^0.7)
   AND Fuel Mass := 0
   AND Grain := ""
   AND Igniter :=""
   AND TVC Choice := "**TVC not Required**"
   AND Final TVC := 0
   AND Pressure Ratio := (Chamber Pressure)/(Altitude Pressure)
   AND Chamber Pressure := (Chamber Pressure)/1000000
   AND Case Length := Total Motor Length - Nozzle Length

RULE    Read in the Data from Hypercard
  READ Results
  DATA Total Impulse
  DATA Burn Rate
  DATA Specific Impulse
  DATA Propellant Mass
  DATA Web Thickness
  DATA Web Fraction
  DATA Propellant Volume
  DATA Burn Area
  DATA Wall Thickness
  DATA Liner Thickness
  DATA Effective Length
  DATA LD Ratio
  DATA Free Volume
  DATA Volumetric Fraction
  DATA Throat Area
  DATA Expansion Ratio
  DATA Exit Area
  DATA Throat Diameter
  DATA Exit Diameter
  DATA Nozzle Length
  DATA Total Motor Length
  DATA Case Mass
  DATA Liner Mass
```

```
  DATA Nozzle Mass
  DATA Throat Insert
  DATA Total Mass
  DATA Min Accel
  DATA Max Accel
  DATA Mass Fraction
  DATA Production Cost
  DATA Liner Flag
  DATA Mass Optimized
  DATA Average Thrust
  DATA Burn Time
  DATA Altitude Pressure
  DATA Design Altitude
  DATA Safety Factor
  DATA Structural Design
  DATA TVC Option
  DATA Max Angle
  DATA Time Used
  DATA Motor Diameter
  DATA PName
  DATA PDensity
  DATA PExponent
  DATA PTemp
  DATA PHeat Ratio
  DATA PVelocity
  DATA SName
  DATA SDensity
  DATA IName
  DATA IDensity
  DATA Cone Angle
  DATA TIName
  DATA Chamber Pressure
  DATA RD Cost
  DATA Error
 THEN   Information Entered

RULE   Check Liner1
  IF Liner Flag <>1
  THEN Liner Checked
    ANDLiner Note :=" Liner Type Specified Meets Insulation Requirements"
  ELSE Liner Checked
    ANDLiner Note :=" Possibility of Burn Through Exists with this Liner. An Alternate
may be Needed."

RULE   Optimizer Selection1
  IF Mass Optimized <>1
  THEN Optimizer Checked
    ANDOptimizer Note :=" Cost Optimized System"
  ELSE Optimizer Checked
    ANDOptimizer Note :=" Mass Optimized System"

RULE   Program Run1
  IF Total Impulse <> 0
  THEN Run Checked
  ELSE Run Checked
    ANDFILE Error Display
    AND EXIT
```

```
RULE   Set Igniter Masses
   IF Information Entered
    ANDRun Checked
    AND Igniter Cal
    AND Liner Checked
    ANDOptimizer Checked
   THENInformation Checked

!-------------------------
!              Grain Design
!-------------------------

RULE   End Burner2
  IF  Web Fraction >= 1.0
    ANDVolumetric Fraction >=0.90
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS End Burner CONFIDENCE 85
    ANDGrain := "End Burner"

RULE   Internal-External Burning Tube2
  IF  Web Fraction >=  0.3
    ANDWeb Fraction  < 0.5
    ANDVolumetric Fraction >=0.75
    ANDVolumetric Fraction <0.85
 THEN  Grain Check IS Internal External Burning Tube CONFIDENCE 70
    ANDGrain := "Internal-External Burning Tube"

RULE   Internal Burning Tube4
  IF  Web Fraction >=  0.5
    ANDWeb Fraction  < 0.9
    ANDLD Ratio < 2
    ANDVolumetric Fraction >=0.85
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS Internal Burning Tube CONFIDENCE 95
    ANDGrain := "Internal Burning Tube"


RULE   Segmented Tube4
  IF  Web Fraction >=  0.5
    ANDWeb Fraction  < 0.9
    ANDLD Ratio > 2
    ANDVolumetric Fraction >=0.80
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS Segmented Tube CONFIDENCE 85
    ANDGrain := "Segmented Tube"


RULE   Rod and Tube2
  IF  Web Fraction >=  0.3
    ANDWeb Fraction  < 0.5
    ANDVolumetric Fraction >=0.60
    ANDVolumetric Fraction <0.85
 THEN  Grain Check IS Rod and Tube CONFIDENCE 75
    ANDGrain := "Rod and Tube"
```

```
RULE   Star2
  IF  Web Fraction >=  0.3
    ANDWeb Fraction  < 0.6
    ANDVolumetric Fraction >=0.75
    ANDVolumetric Fraction <0.85
 THEN  Grain Check IS Star CONFIDENCE 95
    ANDGrain := "Star"

RULE   Wagon Wheel2
  IF  Web Fraction >=  0.2
    ANDWeb Fraction  < 0.3
    ANDVolumetric Fraction >=0.65
    ANDVolumetric Fraction <0.70
 THEN  Grain Check IS Wagon Wheel CONFIDENCE 95
    ANDGrain := "Wagon Wheel"

RULE   Dogbone2
  IF  Web Fraction >= 0.2
    ANDWeb Fraction  < 0.3
    ANDVolumetric Fraction >=0.70
    ANDVolumetric Fraction <0.80
 THEN  Grain Check IS Dogbone CONFIDENCE 85
    ANDGrain := "Dogbone"


RULE   Slotted Tube4
  IF  Web Fraction >=  0.5
    ANDWeb Fraction  < 0.9
    ANDLD Ratio > 3
    ANDVolumetric Fraction >=0.85
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS Slotted Tube CONFIDENCE 85
    ANDGrain := "Slotted Tube"


RULE   Conocyl4
  IF  Web Fraction >=  0.5
    ANDWeb Fraction  < 0.9
    ANDLD Ratio > 2
    ANDLD Ratio < 4
    ANDVolumetric Fraction >=0.85
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS Conocyl CONFIDENCE 85
    ANDGrain := "Conocyl"


RULE   Finocyl4
  IF  Web Fraction >=  0.6
    ANDWeb Fraction  < 0.9
    ANDLD Ratio > 1
    ANDLD Ratio < 2
    ANDVolumetric Fraction >=0.85
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS Finocyl CONFIDENCE 85
    ANDGrain := "Finocyl"
```

```
RULE   Spherical2
  IF  Web Fraction >=  0.2
    ANDWeb Fraction  < 0.5
    ANDVolumetric Fraction >=0.90
    ANDVolumetric Fraction <0.95
 THEN  Grain Check IS Spherical  CONFIDENCE 85
    ANDGrain := "Spherical "

RULE   Minimum Grain1
  IF Web Fraction >  0.0
    ANDWeb Fraction  < 0.2
 THEN  Grain Check IS Dendrite  CONFIDENCE 65
    ANDGrain := "Dendrite "

RULE   Minimum Grain2
  IF Web Fraction >=  0.2
    ANDWeb Fraction  < 0.3
 THEN  Grain Check IS Wagon Wheel  CONFIDENCE 65
    ANDGrain := "Wagon Wheel "

RULE   Minimum Grain3
  IF Web Fraction >=  0.3
    ANDWeb Fraction  < 0.6
 THEN  Grain Check IS Star  CONFIDENCE 65
    ANDGrain := "Star "

RULE   Minimum Grain4
  IF Web Fraction >=  0.6
    ANDWeb Fraction  < 1.0
 THEN  Grain Check IS Internal Burning Tube  CONFIDENCE 65
    ANDGrain := "Internal Burning Tube "

RULE   Minimum Grain5
  IF Web Fraction >= 1.0
 THEN  Grain Check IS End Burner  CONFIDENCE 65
    ANDGrain := "End Burner "


!--------------------------
!               Ignitor Design
!--------------------------

RULE   Check Motor Mass1
  IF   Total Mass > 20000
  THEN Mass IS Large

RULE   Check Motor Mass2
  IF   Total Mass < 20000
    ANDTotal Mass > 1000
  THEN Mass IS Medium


RULE   Check Motor Mass3
   IF  Total Mass < 1000
  THEN Mass IS Small
```

```
RULE   Selection Igniter1
   IF   Mass IS Large
  THEN Igniter Check
    AND Igniter := "Pyrogen"
    AND Mounting := "Ground Mounted"
    AND Igniter Mass := Pyrogen

RULE   Selection Igniter2
   IF   Mass IS Medium
    AND Pyrogen < Pyrotechnic
  THEN Igniter Check
    AND Igniter := "Pyrogen"
    AND Igniter Mass := Pyrogen
    AND Mounting := "Forward Mounted, Internal"

RULE   Selection Igniter3
   IF   Mass IS Medium
    AND Pyrotechnic < Pyrogen
  THEN Igniter Check
    AND Igniter := "Pyrotechnic"
    AND Igniter Mass := Pyrotechnic
    AND Mounting := "Forward Mounted, Internal"

RULE   Selection Igniter4
   IF   Mass IS Small
    AND LayerPyro < Pyrogen
    AND Pyrogen < Pyrotechnic
    OR  LayerPyro < Pyrotechnic
    AND Pyrotechnic < Pyrogen
  THEN Igniter Check
    AND Igniter := "Pyrotechnic Sheet"
    AND Igniter Mass := LayerPyro
    AND Mounting := "Grain Mounted, Internal"

RULE   Selection Igniter5
   IF   Mass IS Small
    AND Pyrogen < Pyrotechnic
  THEN Igniter Check
    AND Igniter := "Pyrogen"
    AND Igniter Mass := Pyrogen
    AND Mounting := "Forward Mounted, Internal"

RULE   Selection Igniter6
   IF   Mass IS Small
    AND Pyrotechnic < Pyrogen
  THEN Igniter Check
    AND Igniter := "Pyrotechnic"
    AND Igniter Mass := Pyrotechnic
    AND Mounting := "Forward Mounted, Internal"
```

```
!-------------------------
!              TVC Design
!-------------------------

RULE   TVC Data Prep1
  IF   Max Angle > 0
    AND TVC Option <> 0
  THEN Data Prep
    AND TVC Angle := (Max Angle)/180*Pi
    AND Flex Mass := 11+(Average Thrust)*0.00009 + 0.15*(Burn time)*(Time Used)/100
    AND Gimble Mass := 13+(Average Thrust)*0.00008 + 0.1*(Burn time)*(Time Used)/100
    AND Fuel Impulse := 1.05*(Average Thrust)*SIN(TVC Angle)*(Burn time)*(Time
Used)/100
    AND Fuel Mass := (Fuel Impulse)/(Nitrogen Tetraoxide*go)
    AND Tank Length := 1.05*((Fuel Mass)/(Nitrogen Density)/(Pi/4*(Tank Diameter)^2))
    AND Tank Thickness := 1.5*(Tank Pressure)*(Tank Diameter)/(2*(Material Strength))
    AND Tank Mass := (Tank Thickness)*Pi*(Material Density)*((Tank Diameter)*(Tank
Length)+((Tank Diameter)/2)^2)
    AND LITVC Mass := 14+(Tank Mass)+(Fuel Mass)


RULE   TVC Selection1
  IF  Data Prep
    AND Max Angle <= 5
    AND LITVC Mass < Flex Mass
    AND LITVC Mass < Gimble Mass
  THEN TVC Mass IS LITVC
    AND TVC Check
    AND Final TVC:= LITVC Mass + Fuel Mass
    AND TVC Choice := "Liquid Injection TVC"
    AND Nozzle Mass:= Nozzle Mass*1.15
  ELSE Fuel Mass := 0


RULE   TVC Selection2
  IF  Data Prep
    AND Max Angle <= 5
    AND Flex Mass < LITVC Mass
    AND Flex Mass < Gimble Mass
  THEN TVC Mass IS Flex
    AND TVC Check
    AND Final TVC:= Flex Mass
    AND TVC Choice := "Flex-Seal Nozzle TVC"
    AND Nozzle Mass:= Nozzle Mass*1.4


RULE   TVC Selection3
  IF  Data Prep
    AND Max Angle <= 5
    AND Gimble Mass < Flex Mass
    AND Gimble Mass < LITVC Mass
  THEN TVC Mass IS Gimble
    AND TVC Check
    AND Final TVC:= Gimble Mass
    AND TVC Choice := "Gimbled Nozzle TVC"
    AND Nozzle Mass:= Nozzle Mass*1.4
```

```
RULE   TVC Selection4
  IF   Data Prep
    AND Max Angle < 10
    AND Flex Mass < Gimble Mass
  THEN TVC Mass IS Flex
    AND TVC Check
    AND Final TVC:= Flex Mass
    AND TVC Choice := "Flex-Seal Nozzle TVC"
    AND Nozzle Mass:= Nozzle Mass*1.50

RULE   TVC Selection5
  IF   Data Prep
    AND Max Angle < 10
    AND Gimble Mass < Flex Mass
  THEN TVC Mass IS Gimble
    AND TVC Check
    AND Final TVC:= Gimble Mass
    AND TVC Choice := "Gimbled Nozzle TVC"
    AND Nozzle Mass:= Nozzle Mass*1.50

RULE   TVC Selection6
  IF   Data Prep
    AND Max Angle < 14
  THEN TVC Mass IS Gimble
    AND TVC Check
    AND Final TVC:= Gimble Mass
    AND TVC Choice := "Gimbled Nozzle TVC"
    AND Nozzle Mass:= Nozzle Mass*1.50

!------------------------
!               Send Results to Text File
!------------------------

RULE   Prep1
  IF   Mass IS Large
  THEN Prep done
    AND Miscell Mass := (Total Mass)*0.001
    AND Case Mass := (Case Mass)+(Miscell Mass)
    AND Total Mass := (Case Mass)+(Final TVC)+(Nozzle Mass)+(Liner Mass)+(Propellant
Mass)
    AND Total Inert := (Total Mass)-(Propellant Mass)
    AND Burnout := (Total Mass)-(Propellant Mass)-(Fuel Mass)-0.25*(Liner Mass)-
0.2*(Nozzle Mass)
    AND ThrustMass Ratio := (Average Thrust)/(Total Mass)
    AND Mass Fraction := (Propellant Mass)/(Total Mass)

RULE   Prep2
  IF   Mass IS Medium
    OR Mass IS Small
  THEN Prep done
    AND Miscell Mass := (Total Mass)*0.001
    AND Case Mass := (Case Mass)+(Miscell Mass)
    AND Total Mass := (Case Mass)+(Final TVC)+(Igniter Mass)+(Nozzle Mass)+(Liner
Mass)+(Propellant Mass)
    AND Total Inert := (Total Mass)-(Propellant Mass)
    AND Burnout := (Total Mass)-(Propellant Mass)-0.5*(Igniter Mass)-(Fuel Mass)-
0.25*(Liner Mass)- 0.2*(Nozzle Mass)
```

```
   ANDThrustMass Ratio := (Average Thrust)/(Total Mass)
   ANDMass Fraction := (Propellant Mass)/(Total Mass)
   ANDMax Accel := (Average Thrust)/(Burnout)
   ANDMin Accel := (Average Thrust)/(Total Mass)

RULE   File1
  IF   Prep done
  THEN Finished
   ANDFILE Data Display
   AND EXIT

!-----------------------
!               Display Screens
!-----------------------
DISPLAY Error Display

********************************************************************************
*                                                                              *
*    The SRM-CDOS was unable to design an acceptable Solid Rocket Motor        *
*    with the entered design values. The possible error source is the         *
*    following design area:                                                    *
*                                                                              *
*    [Error(67)]   *
*                                                                              *
********************************************************************************
<END>
!
!DISPLAY Data Display
--------------------------------------------------------------------------------
                        SOLID ROCKET MOTOR PERFORMANCE
        ([Design Altitude] m, Altitude Operation, [Optimizer Note])
--------------------------------------------------------------------------------
· Average Thrust (n)                                  [Average Thrust(15,0)]
· Chamber Pressure (MPa)                              [Chamber Pressure(15,2)]
· Burn Time (sec)                                     [Burn Time(15,3)]
· Total Impulse (n-sec)                               [Total Impulse(15,0)]
· Propellant Specific Impulse (sec)                   [Specific Impulse(15,2)]
· Altitude Operational Pressure (Pa)                  [Altitude Pressure(15,0)]
· Pressure Ratio                                      [Pressure Ratio(15,1)]:1
· Thrust-to-Mass Ratio (n/kg)                         [ThrustMass Ratio(15,3)]
· Propellant Mass Fraction                            [Mass Fraction(15,3)]
· Minimum Acceleration (m/sec^2)                      [Min Accel(15,2)]
· Maximum Acceleration at Motor Burnout (m/sec^2)     [Max Accel(15,2)]

--------------------------------------------------------------------------------
                        MOTOR DIMENSIONS, MASS, AND COST
--------------------------------------------------------------------------------
· Overall Length (m)                                  [Total Motor Length(15,3)]
· Outside Diameter (m)                                [Motor Diameter(15,3)]
· Total Mass (kg)                                     [Total Mass(15,3)]
· Total Inert (kg)                                    [Total Inert(15,3)]
· At Burnout (kg)                                     [Burnout(15,3)]
· Solid Propellant (kg)                               [Propellant Mass(15,3)]
· Case (kg)                                           [Case Mass(15,3)]
· Internal Insulation (kg)                            [Liner Mass(15,3)]
· Nozzle (kg)                                         [Nozzle Mass(15,3)]
· Ignitor (kg)                                        [Igniter Mass(15,3)]
```

```
· Thrust Vector Control System (kg)                 [Final TVC(15,3)]
· SRM Projected Development Cost ($)                [RD Cost(15,2)]
· SRM Projected Production Cost ($)                 [Production Cost(15,2)]
  NOTE:Costs do not include TVC or Igniter systems

-------------------------------------------------------------------------------
                 PROPELLANT SELECTION AND GRAIN PATTERN
-------------------------------------------------------------------------------
· Solid Propellant Type                     [PName]
· Density (kg/m^3)                                  [PDensity(15,0)]
· Burn Rate (m/sec)                                 [Burn Rate(15,5)]
· Burn Rate Exponent                                [PExponent(15,3)]
· Flame Temperature (℃)                             [PTemp(15,0)]
· Specific Heat Ratio                               [PHeat Ratio(15,3)]
· Exhaust Velocity (m/sec)                          [PVelocity(15,0)]
· Grain Type                                [Grain]
· Propellant Volume (m^3)                           [Propellant Volume(15,5)]
· Web Thickness (m)                                 [Web Thickness(15,4)]
· Web Fraction                                      [Web Fraction(15,4)]
· Average Burn Area (m^2)                           [Burn Area(15,5)]
· Effective Length (m)                              [Effective Length(15,4)]
· L/D Ratio                                         [LD Ratio(15,3)]
· Volumetric Fraction                              [Volumetric Fraction(15,3)]

-------------------------------------------------------------------------------
                      MOTOR CASING AND INSULATION
-------------------------------------------------------------------------------
· Case Length (m)                                   [Case Length(15,3)]
· Case Structural Material                  [SName]
· Density (kg/m^3)                                  [SDensity(15,0)]
· Factor of Safety                                  [Safety Factor(15,2)]
· Case Wall Thickness (m)                           [Wall Thickness(15,5)]
· Case Free Volume (m^3)                            [Free Volume(15,5)]
· Insulation Material                       [IName]
· Density (kg/m^3)                                  [IDensity(15,0)]
· Insulation Thickness (m)                          [Liner Thickness(15,4)]
  [Liner Note]

-------------------------------------------------------------------------------
                     MOTOR NOZZLE AND IGNITION SYSTEM
-------------------------------------------------------------------------------
· Nozzle Number and Type                            1, External
· Nozzle Length (m)                                 [Nozzle Length(15,3)]
· Expansion Area Ratio                              [Expansion Ratio(15,1)]:1
· Throat Area (m^2)                                 [Throat Area(15,5)]
· Exit Area (m^2)                                   [Exit Area(15,5)]
· Throat Diameter (m)                               [Throat Diameter(15,5)]
· Exit Diameter (m)                                 [Exit Diameter(15,5)]
· Expansion Cone Half Angle                         [Cone Angle(15,2)]
· Throat Insert Material                   [TIName]
· Shell Structure Material                 [SName]
· Thrust Vector Control System             [TVC Choice]
· Ignitor Type                             [Igniter]
· Ignitor Mounting                         [Mounting]
<END>
!
END
```

APPENDIX 4

SRM-CDOS Design Comparisons and Optimization

Data Runs

The SRM-CDOS design comparisons and optimization data runs, presented in Appendix four, consist of six separate motor case studies. Each consists of an initial data run for the SRM-CDOS to match the characteristics of the case motor. In all six cases, the SRM-CDOS matched the total loaded motor mass within an relative error of 3.5%.

Three additional runs where completed per case motor with increasing amounts of motor parameter and geometry optimization. The first additional run optimized chamber pressure, motor diameter, and nozzle half angle. The second additional run optimized the chamber pressure, motor diameter, and nozzle half angle, as well as the motor's structural material. The final additional run optimized the chamber pressure, motor diameter, nozzle half angle, motor's structural material, and the motor's solid propellant selection.

Due to space limitations, the full optimization runs for all six motors can not be provided in Appendix four. Only a representative motor, the Orion 50S-AL space motor, will be provided in full detail while the other motors will be provided on a single page summary sheet.

TABLE A4.1 Orion 50S-AL Space Motor Design Summary

Orion 50S AL Space Motor						
User Entered Manufacture's Data						
Average Vacuum Thrust (N):	486,000.000					
Burn Time (sec):	72.400			**SRM-CDOS Optimization Runs**		
Motor Diameter (m):	2.550		Diameter (m):	1.250	1.300	2.400
Average Chamber Pressure (MPa):	5.840		Chamber Pressure (MPa):	18.200	19.500	10.300
Nozzle Angle (°):	20.000		Nozzle Angle (°):	17.000	17.000	16.000
Structural Material Type:	Graphite Epoxy		Structure (type):	Graphite Epoxy	Kevlar 49	Kevlar 49
Solid Propellant Type:	HTPB/AP/AL		Propellant (type):	HTPB/AP/AL	HTPB/AP/AL	DB/AP-HMX/AL
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**	**Data Results**	**Data Results**	**Data Results**
Vacuum Total Impulse (kN-sec):	35,108.060	35,087.416	-0.06%	34,789.468	34,789.468	34,701.540
Total Loaded Motor (kg):	13,236.500	13,326.930	0.68%	12,781.840	12,727.010	12,519.853
Propellant Mass (kg):	12,157.900	12,234.830	0.63%	11,588.980	11,561.330	11,696.200
Case Assembly Mass (kg):	554.300	563.500	1.66%	427.020	422.663	332.290
Nozzle Assembly Mass (kg):	235.400	239.610	1.79%	449.250	439.318	371.787
TVA System Mass (kg):	0.000	0.000	0.00%	0.000	0.000	0.000
Ignitor Assembly (kg):	9.100	10.585	16.32%	4.674	4.744	5.032
Internal Insulation & Liner (kg):	279.600	278.402	-0.43%	311.912	298.955	114.545
Total Inert (kg):	1,078.700	1,092.100	1.24%	1,192.860	1,165.680	823.654
Burnout (kg):	1,013.900	969.280	-4.40%	1,022.696	1,000.710	718.145
Motor Mass Fraction:	0.920	0.918	-0.22%	0.907	0.908	0.934
Motor Sizing Data						
Total Motor Length (m):	7.919	8.145	2.85%	8.402	8.005	4.121
Case Length (m):	6.497	6.530	0.51%	6.579	6.189	2.037
Nozzle Length (m):	1.650	1.620	-1.82%	1.823	1.816	2.084
				Total Motor Mass Savings (kg)		
Design Notes				545.09	599.92	807.08
Liner safety Factor:	1.6	VCF	0.94			
Propellant Constant a:	1.80E-05	MDCF	1.1	**Percent Total Motor Mass Savings**		
Operation Altitude (m):	15000	Sliver Fraction:	0.02	4.09%	4.50%	6.06%
Alt. Pressure (n/m^2):	12040	UTS (kN/m^2):	1340			

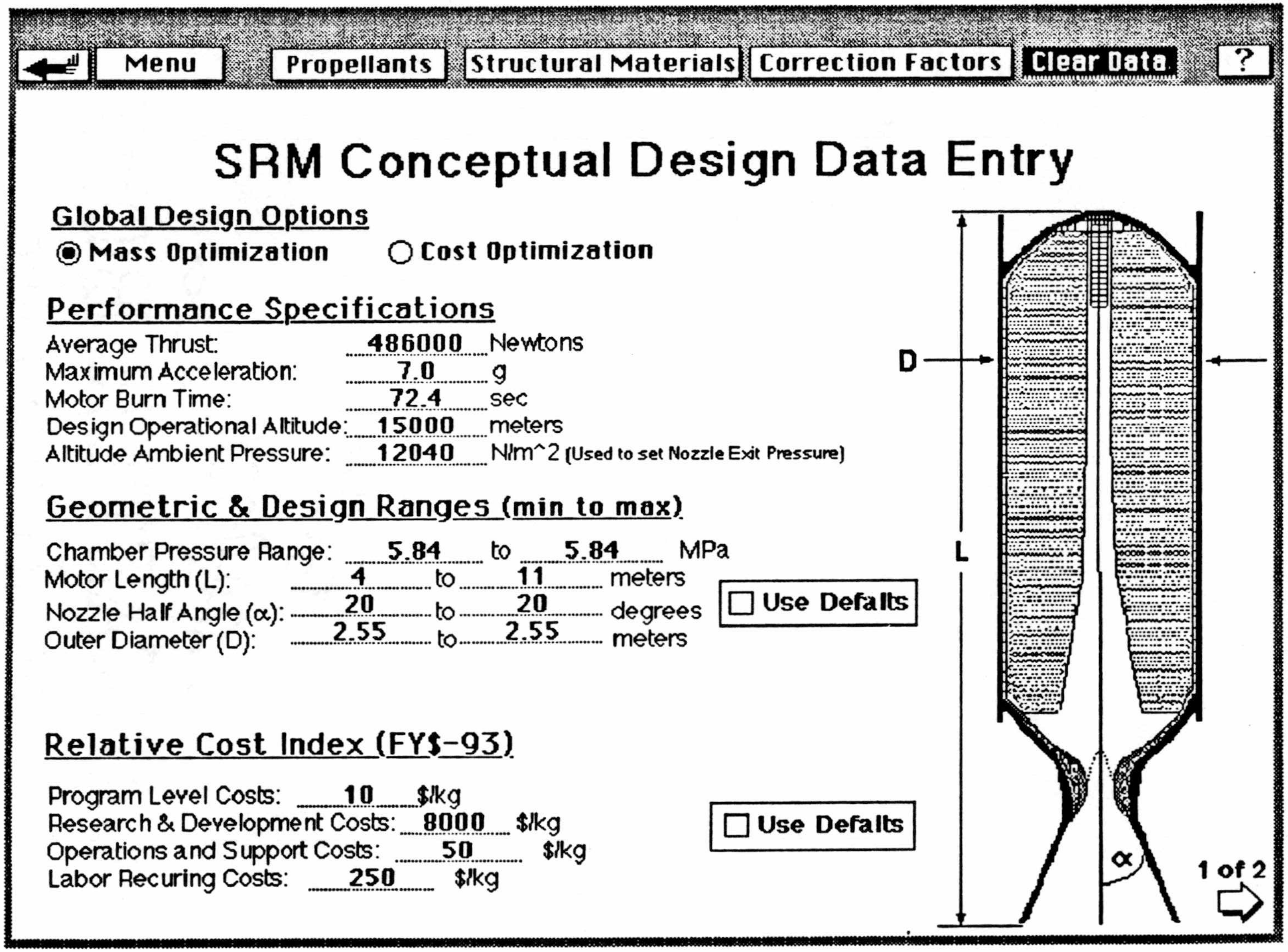

FIGURE A4.1 SRM-CDOS Data Entry Screen #1 for the Orion 50S-AL

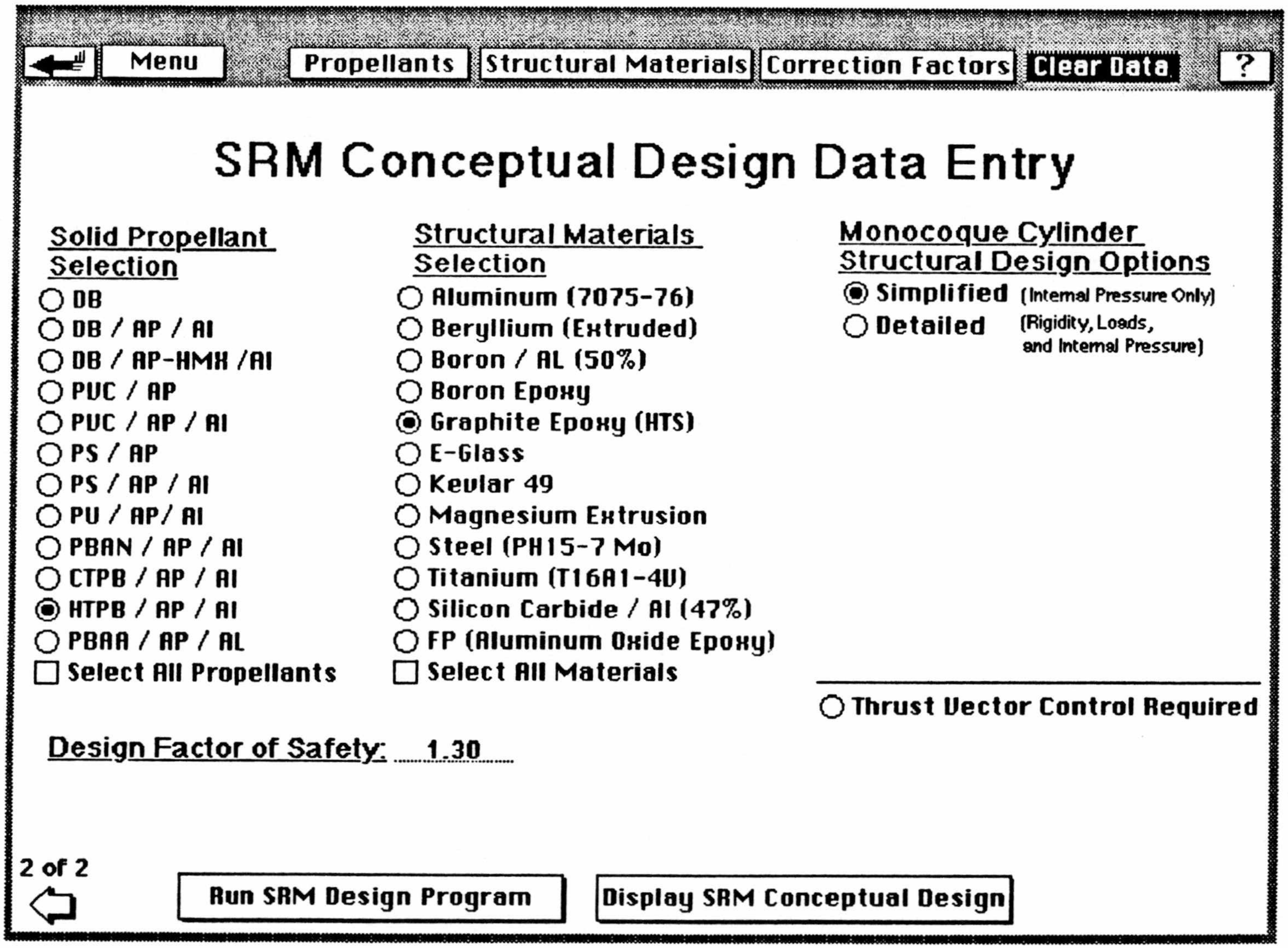

FIGURE A4.2 SRM-CDOS Data Entry Screen #2 for the Orion 50S-AL

TABLE A4.2 Orion 50S-AL Space Motor Design Results

SOLID ROCKET MOTOR PERFORMANCE (15000.00 m, Altitude Operation, Mass Optimized System)	
Average Thrust (n)	486000
Chamber Pressure (MPa)	5.84
Burn Time (sec)	72.400
Total Impulse (n-sec)	35087416
Propellant Specific Impulse (sec)	298.29
Altitude Operational Pressure (Pa)	12040
Pressure Ratio	485.0:1
Thrust-to-Mass Ratio (n/kg)	36.468
Propellant Mass Fraction	0.918
Minimum Acceleration (m/sec^2)	36.47
Maximum Acceleration at Motor Burnout (m/sec^2)	501.40

MOTOR DIMENSIONS, MASS, AND COST	
Overall Length (m)	8.145
Outside Diameter (m)	2.550
Total Mass (kg)	13326.926
Total Inert (kg)	1092.093
At Burnout (kg)	969.278
Solid Propellant (kg)	12234.833
Case (kg)	563.499
Internal Insulation (kg)	278.402
Nozzle (kg)	239.606
Ignitor (kg)	10.585
Thrust Vector Control System (kg)	0.000

TABLE A4.2 (Continued)

PROPELLANT SELECTION AND GRAIN PATTERN		
• Solid Propellant Type	HTPB / AP / Al	
• Density (kg/m^3)		1772
• Burn Rate (m/sec)		0.00491
• Burn Rate Exponent		0.360
• Flame Temperature (°C)		3400
• Specific Heat Ratio		1.200
• Exhaust Velocity (m/sec)		2925
• Grain Type	Wagon Wheel	
• Propellant Volume (m^3)		6.90453
• Web Thickness (m)		0.3556
• Web Fraction		0.2789
• Average Burn Area (m^2)		19.09140
• Effective Length (m)		2.8369
• L/D Ratio		2.561
• Volumetric Fraction		0.409

MOTOR CASING AND INSULATION		
• Case Length (m)		6.530
• Case Structural Material	Graphite Epoxy HTS	
• Density (kg/m^3)		1390
• Factor of Safety		1.30
• Case Wall Thickness (m)		0.00722
• Case Free Volume (m^3)		9.96354
• Insulation Material	EPDM / Aramid	
• Density (kg/m^3)		1269
• Insulation Thickness (m)		0.0042
Liner Type Specified Meets Insulation Requirements		

TABLE A4.2 (Continued)

MOTOR NOZZLE AND IGNITION SYSTEM	
• Nozzle Number and Type	1, External
• Nozzle Length (m)	1.616
• Expansion Area Ratio	34.2:1
• Throat Area (m^2)	0.04619
• Exit Area (m^2)	1.58034
• Throat Diameter (m)	0.24250
• Exit Diameter (m)	1.41850
• Expansion Cone Half Angle	20.00
• Throat Insert Material	Carbon / Carbon
• Shell Structure Material	Graphite Epoxy HTS
• Thrust Vector Control System	**TVC not Required**
• Ignitor Type	Pyrotechnic
• Ignitor Mounting	Forward Mounted, Internal

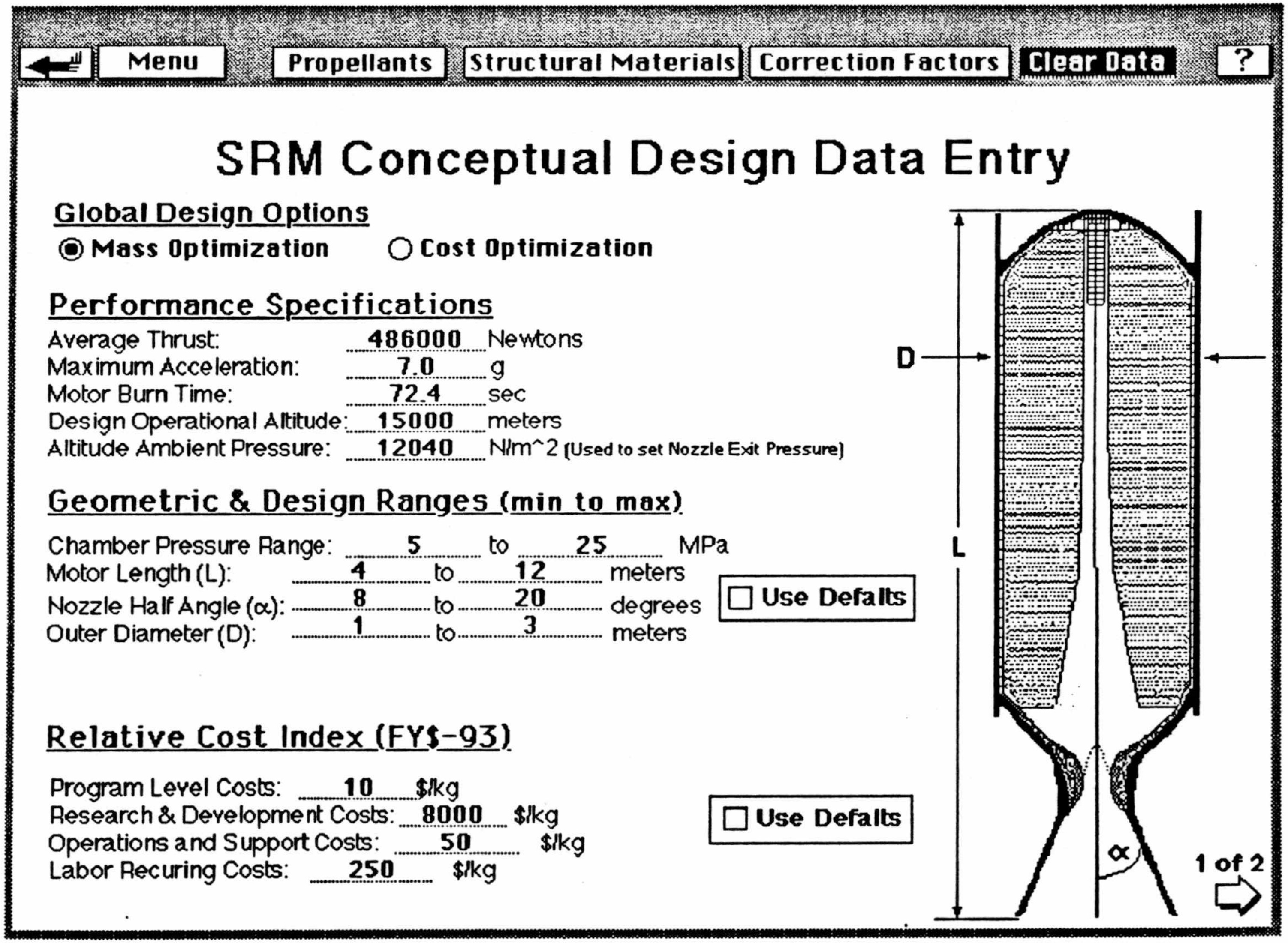

FIGURE A4.3 SRM-CDOS Data Entry Screen #1 for the Orion 50S-AL, First Optimization Run

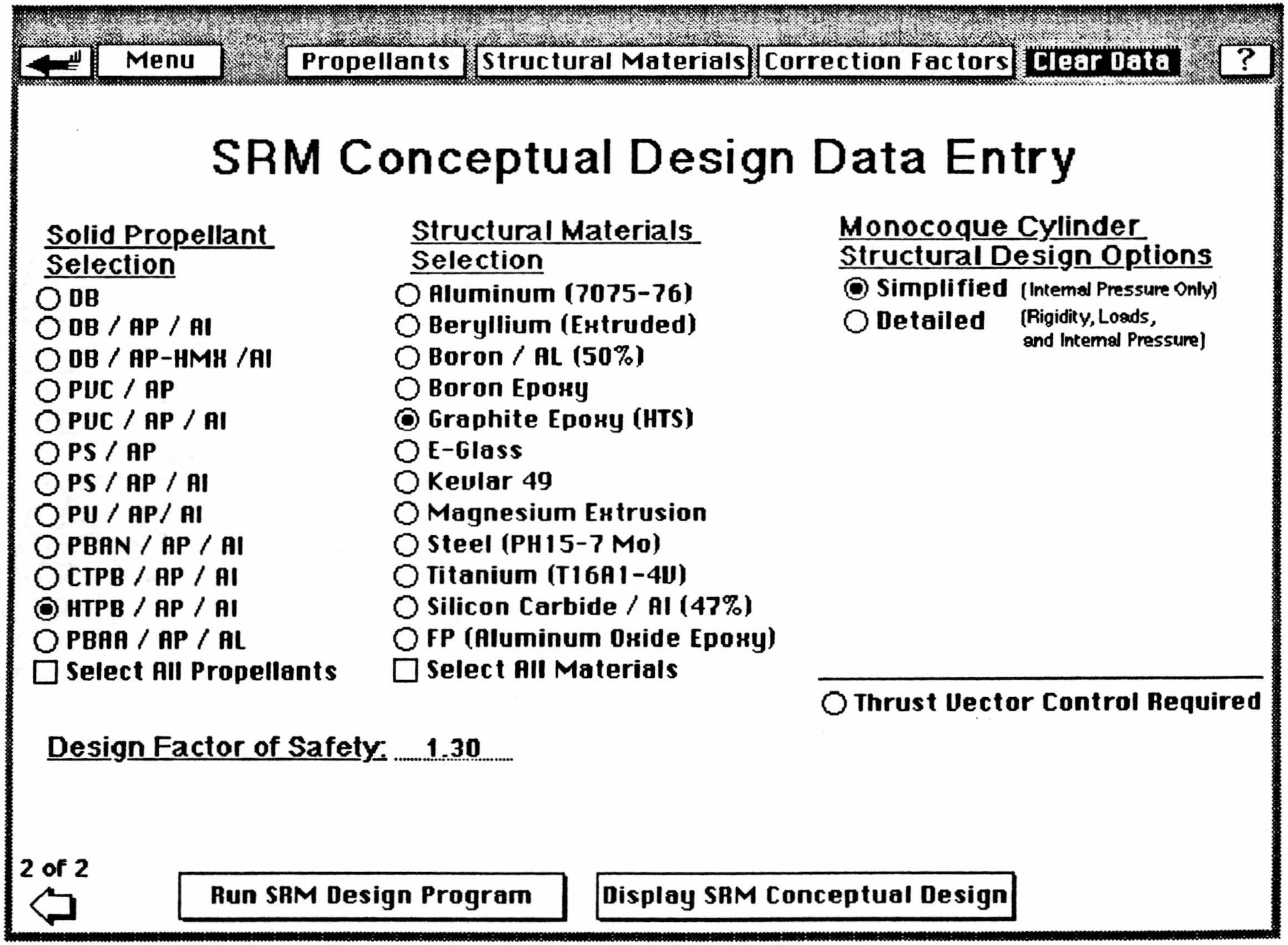

FIGURE A4.4 SRM-CDOS Data Entry Screen #2 for the Orion 50S-AL, First Optimization Run

TABLE A4.3 Orion 50S-AL First Optimization Results

SOLID ROCKET MOTOR PERFORMANCE
(15000.00 m, Altitude Operation, Mass Optimized System)

· Average Thrust (n)	486000
· Chamber Pressure (MPa)	18.20
· Burn Time (sec)	72.400
· Total Impulse (n-sec)	34789468
· Propellant Specific Impulse (sec)	312.24
· Altitude Operational Pressure (Pa)	12040
· Pressure Ratio	1511.6:1
· Thrust-to-Mass Ratio (n/kg)	38.023
· Propellant Mass Fraction	0.907
· Minimum Acceleration (m/sec^2)	38.02
· Maximum Acceleration at Motor Burnout (m/sec^2)	475.21

MOTOR DIMENSIONS, MASS, AND COST

· Overall Length (m)	8.402
· Outside Diameter (m)	1.250
· Total Mass (kg)	12781.836
· Total Inert (kg)	1192.862
· At Burnout (kg)	1022.696
· Solid Propellant (kg)	11588.975
· Case (kg)	427.022
· Internal Insulation (kg)	311.912
· Nozzle (kg)	449.253
· Ignitor (kg)	4.674
· Thrust Vector Control System (kg)	0.000

TABLE A4.3 (Continued)

PROPELLANT SELECTION AND GRAIN PATTERN	
· Solid Propellant Type	HTPB / AP / Al
· Density (kg/m^3)	1772
· Burn Rate (m/sec)	0.00739
· Burn Rate Exponent	0.360
· Flame Temperature (°C)	3400
· Specific Heat Ratio	1.200
· Exhaust Velocity (m/sec)	3062
· Grain Type	Internal Burning Tube
· Propellant Volume (m^3)	6.54005
· Web Thickness (m)	0.5353
· Web Fraction	0.8566
· Average Burn Area (m^2)	12.11355
· Effective Length (m)	5.6821
· L/D Ratio	5.263
· Volumetric Fraction	0.945

MOTOR CASING AND INSULATION	
· Case Length (m)	6.579
· Case Structural Material	Graphite Epoxy HTS
· Density (kg/m^3)	1390
· Factor of Safety	1.30
· Case Wall Thickness (m)	0.01104
· Case Free Volume (m^3)	0.38413
· Insulation Material	EPDM / Aramid
· Density (kg/m^3)	1269
· Insulation Thickness (m)	0.0096

Liner Type Specified Meets Insulation Requirements

TABLE A4.3 (Continued)

MOTOR NOZZLE AND IGNITION SYSTEM	
• Nozzle Number and Type	1, External
• Nozzle Length (m)	1.823
• Expansion Area Ratio	86.5:1
• Throat Area (m^2)	0.01416
• Exit Area (m^2)	1.22528
• Throat Diameter (m)	0.13426
• Exit Diameter (m)	1.24903
• Expansion Cone Half Angle	17.00
• Throat Insert Material	Carbon / Carbon
• Shell Structure Material	Graphite Epoxy HTS
• Thrust Vector Control System	**TVC not Required**
• Ignitor Type	Pyrotechnic
• Ignitor Mounting	Forward Mounted, Internal

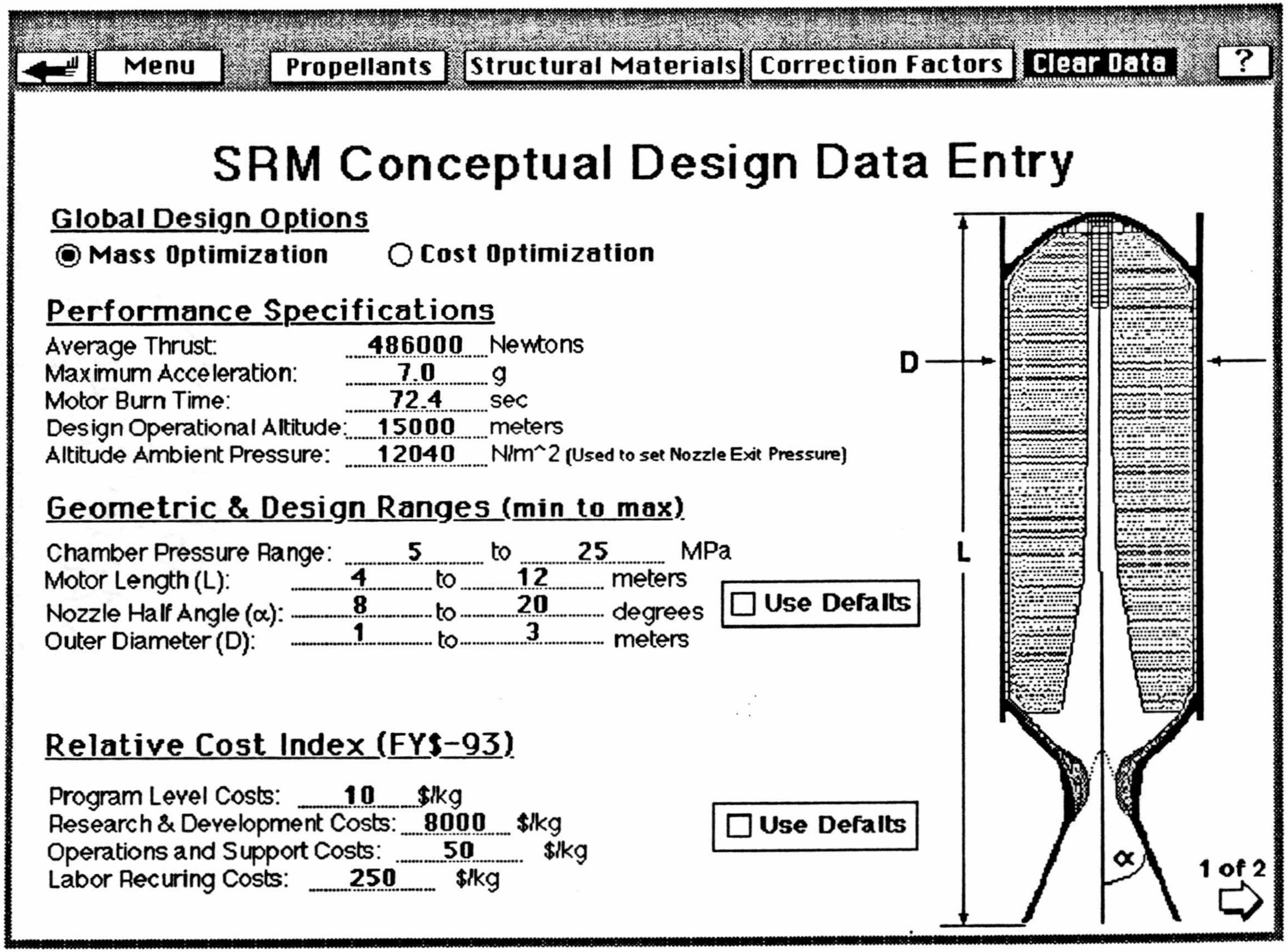

FIGURE A4.5 SRM-CDOS Data Entry Screen #1 for the Orion 50S-AL, Second Optimization Run

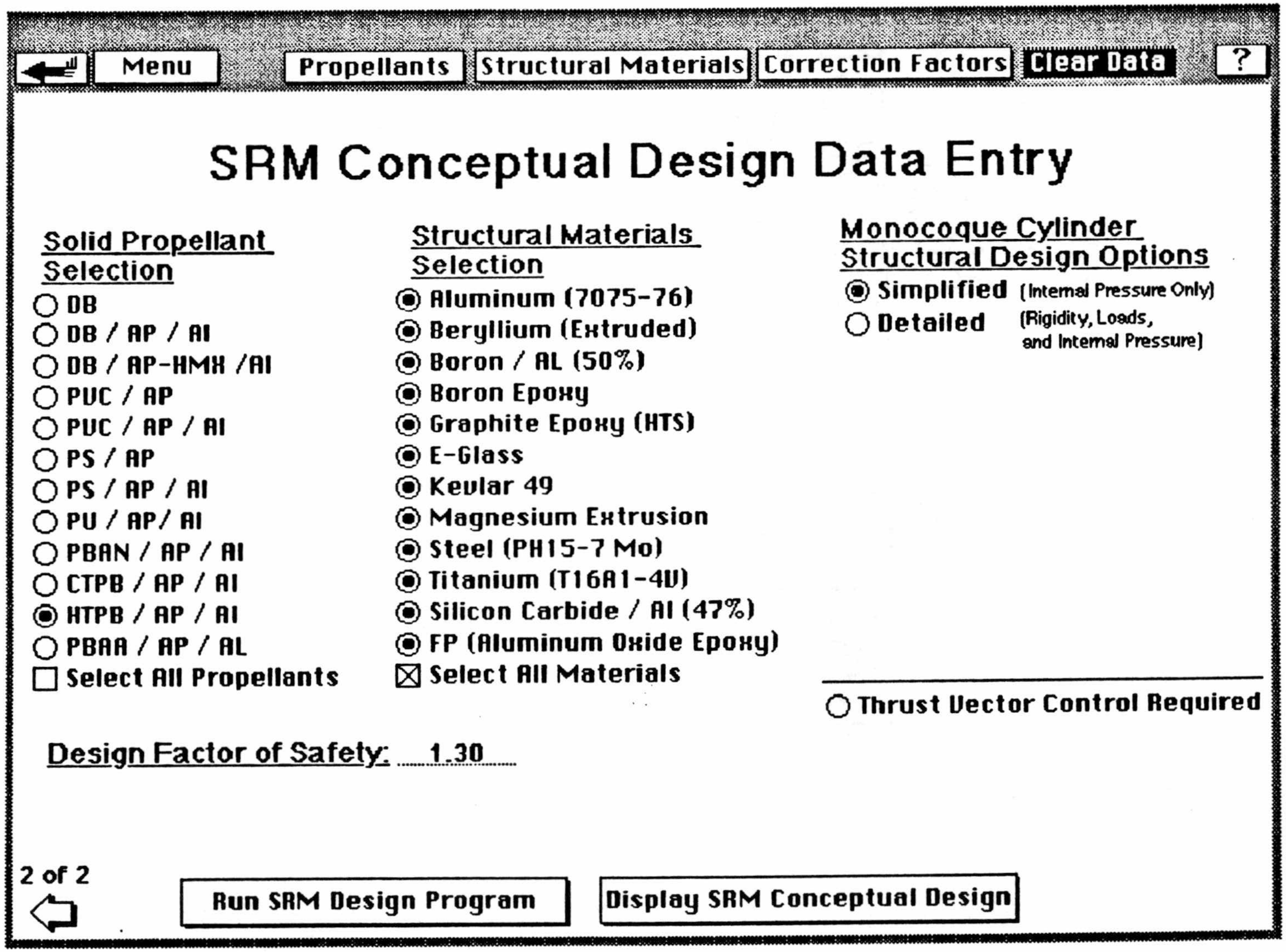

FIGURE A4.6 SRM-CDOS Data Entry Screen #2 for the Orion 50S-AL, Second Optimization Run

TABLE A4.4 Orion 50S-AL Second Optimization Results

SOLID ROCKET MOTOR PERFORMANCE

(15000.00 m, Altitude Operation, Mass Optimized System)

· Average Thrust (n)	486000
· Chamber Pressure (MPa)	19.50
· Burn Time (sec)	72.400
· Total Impulse (n-sec)	34789468
· Propellant Specific Impulse (sec)	312.98
· Altitude Operational Pressure (Pa)	12040
· Pressure Ratio	1619.6:1
· Thrust-to-Mass Ratio (n/kg)	38.187
· Propellant Mass Fraction	0.908
· Minimum Acceleration (m/sec^2)	38.19
· Maximum Acceleration at Motor Burnout (m/sec^2)	485.66

MOTOR DIMENSIONS, MASS, AND COST

· Overall Length (m)	8.005
· Outside Diameter (m)	1.300
· Total Mass (kg)	12727.011
· Total Inert (kg)	1165.681
· At Burnout (kg)	1000.706
· Solid Propellant (kg)	11561.330
· Case (kg)	422.663
· Internal Insulation (kg)	298.955
· Nozzle (kg)	439.318
· Ignitor (kg)	4.744
· Thrust Vector Control System (kg)	0.000

TABLE A4.4 (Continued)

PROPELLANT SELECTION AND GRAIN PATTERN	
• Solid Propellant Type	HTPB / AP / Al
• Density (kg/m^3)	1772
• Burn Rate (m/sec)	0.00758
• Burn Rate Exponent	0.360
• Flame Temperature (°C)	3400
• Specific Heat Ratio	1.200
• Exhaust Velocity (m/sec)	3069
• Grain Type	Internal Burning Tube
• Propellant Volume (m^3)	6.52445
• Web Thickness (m)	0.5488
• Web Fraction	0.8443
• Average Burn Area (m^2)	11.78819
• Effective Length (m)	5.2530
• L/D Ratio	4.761
• Volumetric Fraction	0.936

MOTOR CASING AND INSULATION	
• Case Length (m)	6.189
• Case Structural Material	Kevlar 49 .
• Density (kg/m^3)	1300
• Factor of Safety	1.30
• Case Wall Thickness (m)	0.01194
• Case Free Volume (m^3)	0.44258
• Insulation Material	EPDM / Aramid
• Density (kg/m^3)	1269
• Insulation Thickness (m)	0.0094

Liner Type Specified Meets Insulation Requirements

TABLE A4.4 (Continued)

MOTOR NOZZLE AND IGNITION SYSTEM	
• Nozzle Number and Type	1, External
• Nozzle Length (m)	1.816
• Expansion Area Ratio	91.6:1
• Throat Area (m^2)	0.01318
• Exit Area (m^2)	1.20728
• Throat Diameter (m)	0.12956
• Exit Diameter (m)	1.23982
• Expansion Cone Half Angle	17.00
• Throat Insert Material	Carbon / Carbon
• Shell Structure Material	Kevlar 49 .
• Thrust Vector Control System	**TVC not Required**
• Ignitor Type	Pyrotechnic
• Ignitor Mounting	Forward Mounted, Internal

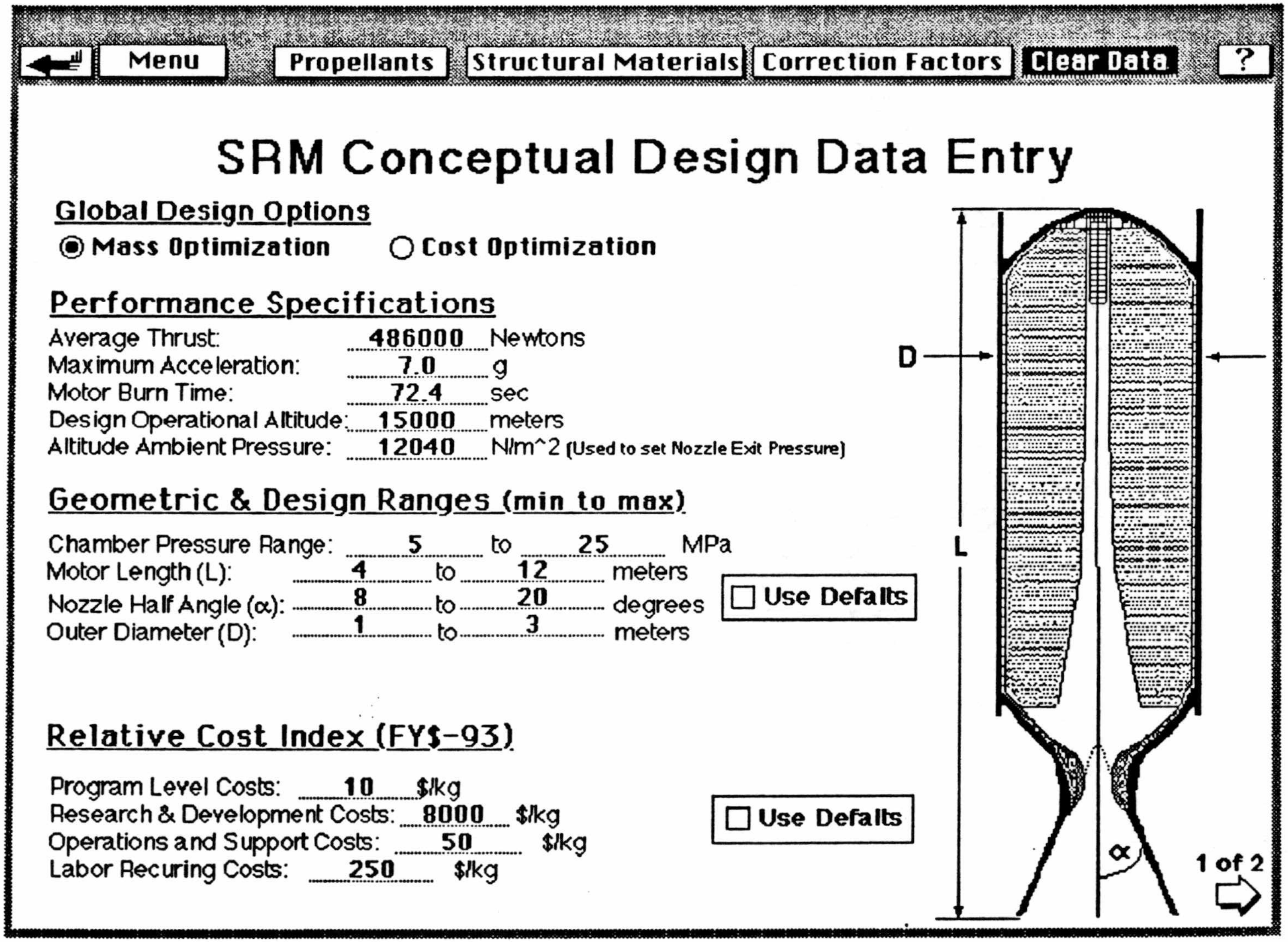

FIGURE A4.7 SRM-CDOS Data Entry Screen #1 for the Orion 50S-AL, Third Optimization Run

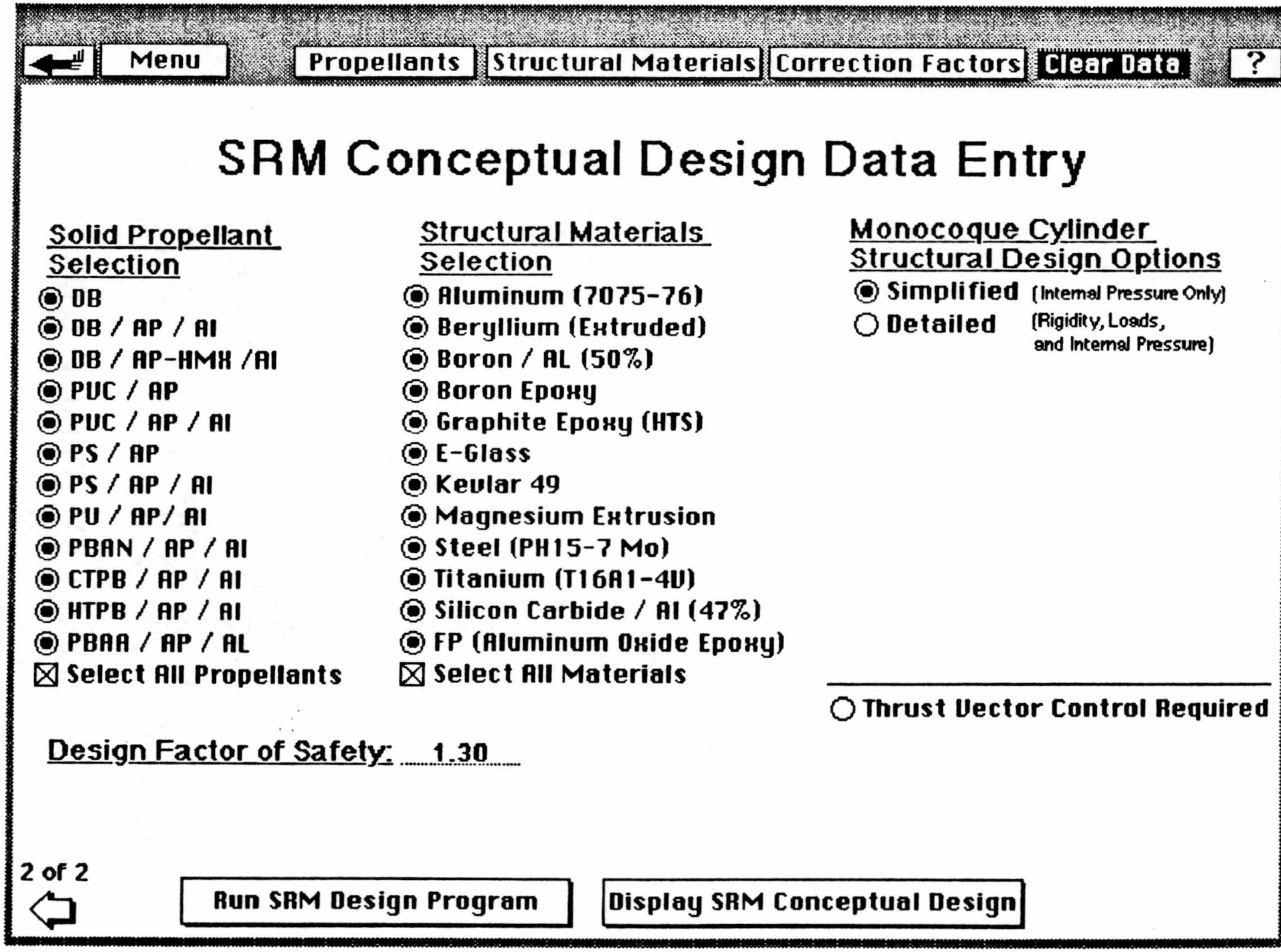

FIGURE A4.8 SRM-CDOS Data Entry Screen #2 for the Orion 50S-AL, Third Optimization Run

TABLE A4.5 Orion 50S-AL Third Optimization Results

SOLID ROCKET MOTOR PERFORMANCE (15000.00 m, Altitude Operation, Mass Optimized System)	
· Average Thrust (n)	486000
· Chamber Pressure (MPa)	10.30
· Burn Time (sec)	72.400
· Total Impulse (n-sec)	34701540
· Propellant Specific Impulse (sec)	308.59
· Altitude Operational Pressure (Pa)	12040
· Pressure Ratio	855.5:1
· Thrust-to-Mass Ratio (n/kg)	38.818
· Propellant Mass Fraction	0.934
· Minimum Acceleration (m/sec^2)	38.82
· Maximum Acceleration at Motor Burnout (m/sec^2)	676.74

MOTOR DIMENSIONS, MASS, AND COST	
· Overall Length (m)	4.121
· Outside Diameter (m)	2.400
· Total Mass (kg)	12519.853
· Total Inert (kg)	823.654
· At Burnout (kg)	718.145
· Solid Propellant (kg)	11696.199
· Case (kg)	332.291
· Internal Insulation (kg)	114.545
· Nozzle (kg)	371.787
· Ignitor (kg)	5.032
· Thrust Vector Control System (kg)	0.000

TABLE A4.5 (Continued)

PROPELLANT SELECTION AND GRAIN PATTERN		
· Solid Propellant Type	DB / AP-HMX / Al	
· Density (kg/m^3)		1800
· Burn Rate (m/sec)		0.01638
· Burn Rate Exponent		0.490
· Flame Temperature (°C)		3700
· Specific Heat Ratio		1.170
· Exhaust Velocity (m/sec)		3026
· Grain Type	Internal Burning Tube	
· Propellant Volume (m^3)		6.49789
· Web Thickness (m)		1.1863
· Web Fraction		0.9886
· Average Burn Area (m^2)		5.44525
· Effective Length (m)		1.4614
· L/D Ratio		0.849
· Volumetric Fraction		0.902

MOTOR CASING AND INSULATION		
· Case Length (m)		2.037
· Case Structural Material	Kevlar 49 .	
· Density (kg/m^3)		1300
· Factor of Safety		1.30
· Case Wall Thickness (m)		0.01164
· Case Free Volume (m^3)		0.70537
· Insulation Material	EPDM / Aramid	
· Density (kg/m^3)		1269
· Insulation Thickness (m)		0.0045
Liner Type Specified Meets Insulation Requirements		

TABLE A4.5 (Continued)

MOTOR NOZZLE AND IGNITION SYSTEM	
· Nozzle Number and Type	1, External
· Nozzle Length (m)	2.084
· Expansion Area Ratio	59.2:1
· Throat Area (m^2)	0.02503
· Exit Area (m^2)	1.48163
· Throat Diameter (m)	0.17852
· Exit Diameter (m)	1.37349
· Expansion Cone Half Angle	16.00
· Throat Insert Material	Carbon / Carbon
· Shell Structure Material	Kevlar 49 .
· Thrust Vector Control System	**TVC not Required**
· Ignitor Type	Pyrotechnic
· Ignitor Mounting	Forward Mounted, Internal

TABLE A4.6 Orion 38 Space Motor Design Summary

	Orion 38 Space Motor					
User Entered Manufacture's Data						
Average Vacuum Thrust (N):	31,890.000					
Burn Time (sec):	68.400			**SRM-CDOS Optimization Runs**		
Motor Diameter (m):	1.930		Diameter (m):	1.300	1.300	1.350
Average Chamber Pressure (MPa):	3.780		Chamber Pressure (MPa):	3.800	3.800	3.600
Nozzle Angle (°):	24.000		Nozzle Angle (°):	14.000	14.000	14.500
Structural Material Type:	Graphite Epoxy		Structure (type):	Graphite Epoxy	Kevlar 49	Kevlar 49
Solid Propellant Type:	HTPB / AP / Al		Propellant (type):	HTPB / AP / Al	HTPB / AP / Al	DB/AP-HMX/AL
		Matching				
	Manufacture's	**SRM-CDOS**				
Motor Performance Data	**Data**	**Data**	**Rel. Error**	**Data Results**	**Data Results**	**Data Results**
Vacuum Total Impulse (kN-sec):	2,183.080	2,187.514	0.20%	2,124.510	2,124.510	2,126.826
Total Loaded Motor (kg):	895.500	908.530	1.46%	855.670	853.675	828.595
Propellant Mass (kg):	770.500	771.030	0.07%	748.660	748.660	722.235
Case Assembly Mass (kg):	39.400	41.590	5.56%	14.230	12.998	13.066
Nozzle Assembly Mass (kg):	48.000	52.440	9.25%	63.200	62.446	63.390
TVA System Mass (kg):	13.700	15.410	12.48%	15.410	15.410	15.410
Ignitor Assembly (kg):	1.300	1.656	27.38%	1.320	1.320	1.339
Internal Insulation & Liner (kg):	23.400	26.410	12.86%	12.850	12.846	13.459
Total Inert (kg):	126.000	137.510	9.13%	107.012	105.019	106.360
Burnout (kg):	114.300	119.590	4.63%	90.499	88.659	89.724
Motor Mass Fraction:	0.860	0.849	-1.28%	0.875	0.877	0.872
Motor Sizing Data						
Total Motor Length (m):	1.339	1.354	1.12%	1.554	1.554	1.531
Case Length (m):	0.831	0.837	0.72%	0.631	0.631	0.604
Nozzle Length (m):	0.517	0.517	0.00%	0.923	0.923	0.927
				Total Motor Mass Savings (kg)		
Design Notes				52.86	54.86	79.93
Liner safety Factor:	2	VCF	0.965			
Propellant Constant a:	2.10E-05	MDCF	1.08	**Percent Total Motor Mass Savings**		
Operation Altitude (m):	20500	Sliver Fraction:	0.02	5.82%	6.04%	8.80%
Alt. Pressure (n/m^2):	4980	UTS (kN/m^2):	1340			

TABLE A4.7 Orion 50 Space Motor Design Summary

	Orion 50 Space Motor					
User Entered Manufacture's Data						
Average Vacuum Thrust (N):	119,740.000					
Burn Time (sec):	73.300			**SRM-CDOS Optimization Runs**		
Motor Diameter (m):	2.550		Diameter (m):	1.850	1.850	2.100
Average Chamber Pressure (MPa):	3.780		Chamber Pressure (MPa):	10.200	10.600	7.600
Nozzle Angle (°):	20.000		Nozzle Angle (°):	16.500	16.000	15.500
Structural Material Type:	Graphite Epoxy		Structure (type):	Graphite Epoxy	Kevlar 49	Kevlar 49
Solid Propellant Type:	HTPB / AP / Al		Propellant (type):	HTPB / AP / Al	HTPB / AP / Al	DB/AP-HMX/AL
		Matching				
	Manufacture's	**SRM-CDOS**				
Motor Performance Data	**Data**	**Data**	**Rel. Error**	**Data Results**	**Data Results**	**Data Results**
Vacuum Total Impulse (kN-sec):	8,780.350	8,785.814	0.06%	8,700.240	8,689.191	8,678.710
Total Loaded Motor (kg):	3,366.300	3,470.580	3.10%	3,371.510	3,360.660	3,291.138
Propellant Mass (kg):	3,023.500	3,057.480	1.12%	2,960.910	2,953.013	2,923.819
Case Assembly Mass (kg):	137.200	191.600	39.65%	118.825	114.298	103.021
Nozzle Assembly Mass (kg):	104.900	111.381	6.18%	184.210	186.342	170.809
TVA System Mass (kg):	13.700	23.426	70.99%	23.426	23.426	23.426
Igniter Assembly (kg):	5.300	5.622	6.08%	4.607	4.603	4.780
Internal Insulation & Liner (kg):	81.700	81.070	-0.77%	79.532	78.980	65.283
Total Inert (kg):	342.800	413.100	20.51%	410.600	407.646	367.320
Burnout (kg):	314.400	367.740	16.97%	351.571	348.332	314.447
Motor Mass Fraction:	0.900	0.881	-2.11%	0.878	0.879	0.888
Motor Sizing Data						
Total Motor Length (m):	2.657	2.623	-1.28%	2.210	2.238	2.231
Case Length (m):	1.760	1.695	-3.69%	1.101	1.095	0.980
Nozzle Length (m):	0.929	0.927	-0.22%	1.109	1.144	1.251
				Total Motor Mass Savings (kg)		
Design Notes				99.07	109.92	179.44
Liner safety Factor:	5.5	VCF	0.945			
Propellant Constant a:	1.80E-05	MDCF	1.09	**Percent Total Motor Mass Savings**		
Operation Altitude (m):	18000	Sliver Fraction:	0.02	2.85%	3.17%	5.17%
Alt. Pressure (n/m^2):	7410	UTS (kN/m^2):	1370			

TABLE A4.8 Orion 50S-GL Space Motor Design Summary

Orion 50S GL Space Motor						
User Entered Manufacture's Data						
Average Vacuum Thrust (N):	451,920.000					
Burn Time (sec):	72.500					
Motor Diameter (m):	2.550			**SRM-CDOS Optimization Runs**		
Average Chamber Pressure (MPa):	5.810		Diameter (m):	1.300	1.350	2.500
Nozzle Angle (°):	24.000		Chamber Pressure (MPa):	22.200	23.700	11.200
Structural Material Type:	Graphite Epoxy		Nozzle Angle (°):	13.000	13.000	12.000
Solid Propellant Type:	HTPB/AP/AL		Structure (type):	Graphite Epoxy	Kevlar 49	Kevlar 49
			Propellant (type):	HTPB/AP/AL	HTPB/AP/AL	DB/AP-HMX/AL
		Matching				
	Manufacture's	**SRM-CDOS**				
Motor Performance Data	**Data**	**Data**	**Rel. Error**	**Data Results**	**Data Results**	**Data Results**
Vacuum Total Impulse (kN-sec):	32,751.070	32,769.852	0.06%	31,760.310	31,760.310	31,699.658
Total Loaded Motor (kg):	13,106.500	13,166.670	0.46%	12,228.120	12,169.600	11,962.231
Propellant Mass (kg):	12,123.900	12,150.880	0.22%	11,013.010	10,983.220	11,134.436
Case Assembly Mass (kg):	479.600	538.050	12.19%	482.230	475.998	356.971
Nozzle Assembly Mass (kg):	166.800	136.480	-18.18%	393.120	383.560	312.479
TVA System Mass (kg):	47.600	50.241	5.55%	50.241	50.241	50.241
Igniter Assembly (kg):	9.100	10.565	16.10%	4.660	4.727	5.125
Internal Insulation & Liner (kg):	279.600	280.465	0.31%	284.850	271.860	102.980
Total Inert (kg):	982.600	1,015.000	3.30%	1,215.010	1,186.381	827.795
Burnout (kg):	917.700	913.000	-0.51%	1,062.940	1,039.340	736.992
Motor Mass Fraction:	0.930	0.923	-0.75%	0.901	0.903	0.931
Motor Sizing Data						
Total Motor Length (m):	7.093	7.322	3.23%	7.405	7.072	3.625
Case Length (m):	6.497	6.513	0.25%	5.891	5.563	1.881
Nozzle Length (m):	0.797	0.809	1.51%	1.513	1.509	1.744
				Total Motor Mass Savings (kg)		
Design Notes				938.55	997.07	1,204.44
Liner safety Factor:	1.6	VCF	0.94			
Propellant Constant a:	1.80E-05	MDCF	1.1	**Percent Total Motor Mass Savings**		
Operation Altitude (m):	9000	Sliver Fraction:	0.02	7.13%	7.57%	9.15%
Alt. Pressure (n/m^2):	30740	UTS (kN/m^2):	1340			

TABLE A4.9 Hercules GEM Booster Motor Design Summary

Hercules GEM Booster Motor						
User Entered Manufacture's Data						
Average Vacuum Thrust (N):	499,250.000					
Burn Time (sec):	55.100			**SRM-CDOS Optimization Runs**		
Motor Diameter (m):	1.031		Diameter (m):	1.000	1.000	1.650
Average Chamber Pressure (MPa):	5.630		Chamber Pressure (MPa):	9.900	10.900	8.200
Nozzle Angle (°):	20.000		Nozzle Angle (°):	14.500	14.500	15.000
Structural Material Type:	Graphite Epoxy		Structure (type):	Graphite Epoxy	Kevlar 49	Kevlar 49
Solid Propellant Type:	HTPB / AP / Al		Propellant (type):	HTPB / AP / Al	HTPB / AP / Al	DB/AP-HMX/AL
Motor Performance Data	**Manufacture's Data**	**Matching SRM-CDOS Data**	**Rel. Error**	**Data Results**	**Data Results**	**Data Results**
Vacuum Total Impulse (kN-sec):	31,536.320	31,643.460	0.34%	31,233.378	31,233.378	31,268.676
Total Loaded Motor (kg):	12,714.600	12,773.734	0.47%	12,564.370	12,447.860	12,242.214
Propellant Mass (kg):	11,741.600	11,739.120	-0.02%	11,235.430	11,181.670	11,128.299
Case Assembly Mass (kg):	394.600	394.190	-0.10%	609.070	564.400	515.743
Nozzle Assembly Mass (kg):	241.500	233.580	-3.28%	348.519	333.982	359.103
TVA System Mass (kg):	0.000	0.000	0.00%	0.000	0.000	0.000
Igniter Assembly (kg):	7.900	5.120	-35.19%	4.422	4.369	4.987
Internal Insulation & Liner (kg):	387.500	401.744	3.68%	366.930	363.443	234.082
Total Inert (kg):	973.100	1,034.620	6.32%	1,328.940	1,266.190	1,113.915
Burnout (kg):	858.500	884.910	3.08%	1,165.290	1,106.350	981.081
Motor Mass Fraction:	0.923	0.919	-0.43%	0.894	0.898	0.909
Motor Sizing Data						
Total Motor Length (m):	11.092	11.085	-0.06%	10.929	10.852	5.696
Case Length (m):	10.028	9.975	-0.53%	9.419	9.344	4.181
Nozzle Length (m):	1.107	1.110	0.27%	1.511	1.507	1.515
				Total Motor Mass Savings (kg)		
Design Notes				209.36	325.87	531.52
Liner safety Factor:	2.1	VCF	0.895			
Propellant Constant a:	1.75E-05	MDCF	1	**Percent Total Motor Mass Savings**		
Operation Altitude (m):	9500	Sliver Fraction:	0.02	1.64%	2.55%	4.16%
Alt. Pressure (n/m^2):	28550	UTS (kN/m^2):	1370			

TABLE A4.10 Minuteman I First Stage Booster Motor Design Summary

Minuteman First Stage Booster Motor						
User Entered Manufacture's Data						
Average Vacuum Thrust (N):	865,600.000					
Burn Time (sec):	52.600			**SRM-CDOS Optimization Runs**		
Motor Diameter (m):	1.660		Diameter (m):	1,720.000	1.500	2.500
Average Chamber Pressure (MPa):	5.380		Chamber Pressure (MPa):	6.000	26.300	11.200
Nozzle Angle (°):	15.000		Nozzle Angle (°):	11.500	10.000	12.000
Structural Material Type:	Ladish Steel D6AC		Structure (type):	Ladish Steel D6AC	Kevlar 49	Kevlar 49
Solid Propellant Type:	PBAN / AP / Al		Propellant (type):	PBAN/AP/AL	PBAN/AP/AL	DB/AP-HMX/AL
		Matching				
	Manufacture's	**SRM-CDOS**				
Motor Performance Data	**Data**	**Data**	**Rel. Error**	**Data Results**	**Data Results**	**Data Results**
Vacuum Total Impulse (kN-sec):	48,200.000	48,022.100	-0.37%	47,682.560	47,602.492	47,641.592
Total Loaded Motor (kg):	22,921.000	22,948.470	0.12%	22,942.580	20,619.340	19,757.132
Propellant Mass (kg):	20,781.000	20,831.030	0.24%	20,488.310	18,977.264	18,011.248
Case Assembly Mass (kg):	1,299.000	1,299.620	0.05%	1,455.940	909.623	995.059
Nozzle Assembly Mass (kg):	402.000	387.900	-3.51%	549.520	418.776	441.352
TVA System Mass (kg):	70.000	83.037	18.62%	83.037	83.037	83.037
Igniter Assembly (kg):	11.700	6.861	-41.36%	6.927	6.899	5.014
Internal Insulation & Liner (kg):	355.000	377.080	6.22%	365.770	231.140	221.422
Total Inert (kg):	2,139.700	2,117.640	-1.03%	2,454.290	1,642.080	1,745.884
Burnout (kg):	1,933.000	1,952.790	1.02%	2,252.930	1,500.640	1,599.751
Motor Mass Fraction:	0.912	0.907	-0.55%	0.893	0.920	0.912
Motor Sizing Data						
Total Motor Length (m):	7.490	7.638	1.98%	7.543	4.399	4.343
Case Length (m):	6.580	6.677	1.47%	6.262	2.917	2.879
Nozzle Length (m):	0.910	0.951	4.51%	1.281	1.483	1.463
				Total Motor Mass Savings (kg)		
Design Notes				5.89	2,329.13	3,191.34
Liner safety Factor:	1.6	VCF	0.955			
Propellant Constant a:	8.10E-05	MDCF	1.01	**Percent Total Motor Mass Savings**		
Operation Altitude (m):	0	Sliver Fraction:	0.056	0.03%	10.15%	13.91%
Alt. Pressure (n/m^2):	101000	UTS (kN/m^2):	1550			

APPENDIX 5

Trademarks

Trademarks

The following, legally named, computer programs and computer hardware were mentioned during the course of this thesis.

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APPENDIX 6

Equation Nomenclature, Acronyms, and Abbreviations

Equation Nomenclature

a = burn rate empirical constant

A_b = average propellant burn area

A_t = throat area

A_e = exit area

b = web thickness

b_f = web fraction

c = exhaust velocity

C_f = thrust coefficient

c^* = characteristic exhaust velocity

D = motor diameter

D_e = exit diameter

D_i = inside grain diameter

D_o = outside grain diameter

D_t = throat diameter

e = elliptical eccentricity

E = Young's Modulus

f = factor of safety

F = motor thrust

$\bar{F}$ = average motor thrust

g_o = gravity constant

I_{sp} = specific impulse

I_t = total impulse

k = specific heat ratio

L/D = motor length / motor diameter ratio

L_e = effective motor length

L_n = nozzle length

M = average molecular weight

M_c = motor case mass

M_{ic} = igniter mass

M_{init} = inital mass of system

M_l = motor liner mass

M_n = nozzle mass

M_p = propellant mass

M_{psl} = mass propellant sliver loss

M_t = total motor mass

$\dot{m}$ = mass flow

$\tilde{m}_{prop}$ = propellant local chamber mass flux

n = pressure exponent

p_a = ambient pressure

p_e = exit pressure

p_o = chamber pressure

r = propellant burn rate

R_o = universal gas constant

t_b = motor burn time

T_a = ambient temperature

T_e = nozzle exit temperature

T_f = propellant flame temperature

V_e = nozzle exit velocity

V_{cf} = case free volume

V_{fr} = volumetric fraction

V_p = motor propellant volume

V_o = chamber flow velocity

wf = web fraction

α = nozzle cone divergence half angle

ε = expansion ratio

λ = nozzle correction factor

ξ = energy conversion factor

ζ_d = mass discharge correction factor

ζ_f = thrust correction factor

ζ_v = velocity correction factor

ρ_c = case material density

ρ_l = casing liner density

ρ_p = propellant density

$\sigma_\varnothing$ = hoop stress

σ_l = longitudinal stress

τ_{mc} = motor case thickness

τ_l = insulative case liner thickness

Acronyms and Abbreviations

AIAA = American Institute of Aeronautics and Astronautics

Al_2O_3 = Aluminum Oxide

Al = Aluminum

AP = Ammonium Perchlorate

C = Carbon

Cl =Chlorine

CTPB = Carboxy-Terminated Polybutadiene

DB = Double Base

H_2 =Hydrogen

H_2O = Water

HCl = Hydrochloric Acid

HMX = Cylcotetramethylene Tetranitramine

HTPB = Hydroxy-Terminated Polybutadiene

N_2 =Nitrogen

O_2 = Oxygen

PBAA = Polybutadiene-Acrylic Acid Polymer

PBAN = Polybutadiene-Acrylic Acid-Acrylonitrile Terpolymer

PS = Polysulfide

PU = Polyurethane

PVC = Polyvinyl Chloride

SiC = Silicon Carbide

SRM = Solid Rocket Motor

SRM-CDOS = Solid Rocket Motor Conceptual Design Optimization System

VITA

VITA

James Brill Clegern was born March 16, 1969 in Enid, Oklahoma, in the United States of America. He traveled with his family over much of the United States and Europe as a military dependent, attending elementary school in Texas and Germany, junior high in Ramstein, Germany, and Olney, Maryland, and graduating from Sherwood High School in Silver Springs, Maryland, 1987. He received his Bachelor of Science Degree in Aerospace Engineering from the University of Maryland, College Park (UMCP), in 1991, with a Distinguished Graduate Commission into the United States Air Force as a Second Lieutenant. While on educational delay, he started at the University of Tennessee Space Institute (UTSI), Tullahoma, Tennessee, in January 1992, where he pursued a Masters of Science Degree in Aerospace Engineering.

He was awarded the rank of Eagle Scout in June 1985. While at UMCP, he was chapter Treasurer (1990-91) and President (1991) of the aerospace engineering honorary society Sigma Gamma Tau (ΣΓΤ), and Student Post President (1991) of the Society of American Military Engineers (SAME). While at UTSI, he was the American Institute of Aeronautics and Astronautics (AIAA) chapter Treasurer (1992), and chapter Vice President (1992-93), and the Student Government Association (SGA) Vice President of Records and Finance (1992-1993). He has been a member of SAME since 1987 and a member of AIAA since 1988.

Personal interests include Macintosh computer programing and operation, USAF, camping, community service, cycling, model rocketry, volleyball, scouting, and scuba diving.