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ON SPACE PROGRAM PLANNING

Quantifying the effects of spacefaring goals and strategies on the solution space of feasible programs

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ON SPACE PROGRAM PLANNING

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FOR KYLIE

I love you to the moon and back

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ABSTRACT

An organization's space program represents the coalescence of top-level goals and strategies with the specific, lower-level combinations of mission architectures, hardware, and technology. High costs and long lead times necessitate substantial planning and key decisions to be made years, possibly decades, in advance.

Unfortunately, the methods and processes available for planning have historically been unable to provide decision-makers with objective, quantitative information from both the top and lower levels of the space program. Instead, decision-makers are forced to turn to top-level, qualitative assessments, disconnected from any of the fundamental, technical details, where inconsistent comparisons and secondary effects often become the basis of decisions. This leads to misinformed decisions early on that will have the largest impact on the overall cost and schedule of the program. Similarly, the systems designers and specialists at the lower, technical levels are disconnected from any guidance from the over-arching goals of the program. Therefore, these technical efforts tend to be pursued and optimized separately from the space program as a whole.

It is the hypothesis of this research that this disconnect exists but can be parametrically corrected to better support the decision-makers at all levels of a space program. In pursuit of a solution, a prototype decision-support system, *Ariadne*, has been developed to parametrically integrate the top-level effects of goals and strategies with the primary technical details of mission architectures and hardware designs. The capabilities of this prototype system are demonstrated with a case study of Project Apollo, enabling informed decisions concerning launch vehicle requirements, mission architecture selection, and alternative candidate space programs.

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NOMENCLATURE

Abbreviations

AHP	Analytic hierarchy process
AVD	Aerospace Vehicle Design
C.I.	Consistency Index
CD	Conceptual design
CER	Cost estimating relationship
CSM	Command/service module
DARPA	Defense Advanced Research Projects Agency
DDT&E	Design, development, test, and evaluation
EDL	Entry, descent, and landing
EOR	Earth orbit rendezvous
ESA	European Space Agency
EVA	Extra-vehicular activity
GEO	Geostationary equatorial orbit
HyFAC	Hypersonic Research Facilities
IMLEO	Initial mass in low Earth orbit
ISP	Integrated Space Plan
ISRU	*In-situ* resource utilization
JCL	Joint cost and schedule confidence level
LA	Lunar ascent
LEO	Low Earth orbit
LL	Lunar landing
LM	Lunar module
LOI	Lunar orbit insertion

LOR	Lunar orbit rendezvous
MER	Mass estimating relationship
N-S	Nassi-Shneiderman
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
R.I.	Random Index
R&D	Research and Development
SE	Systems Engineering
SEI	Space Exploration Initiative
SMAD	Space Mission Analysis and Design
STAMP	Space Technology Analysis and Mission Planning
STS	Space Transportation System
TAPP	Technology Alignment and Portfolio Prioritization
TEI	Trans-Earth injection
TMI	Trans-Mars injection
TRl	Technology Readiness Level

Greek letters

χ	Manned lunar landing architecture *split ratio*
λ	Payload fraction
ε	Structural coefficient

Subscripts

0	Initial state
f	Final state
man	Man's involvement strategy factor
scale	Scale of the program strategy factor
tech	Technology level strategy factor
time	Pace of the program strategy factor
D	Destiny category of *spacefaring goals*

P	Pragmatism category of *spacefaring goals*
S	Science category of *spacefaring goals*

Variables

ΔV	Change in velocity
C	Total development and production costs
CP	Connection parameter
f_0	*TransCost* systems engineering and integration factor
f_1	*TransCost* technical development standard correlation factor
f_2	*TransCost* technical quality correlation factor
f_6	*TransCost* cost growth factor for schedule deviation
g	Earth's gravitational constant
I_{sp}	Specific impulse
L/D	Lift-to-drag ratio
m	Mass
MR	Mass ratio
N	Number of launches for rendezvous in Earth orbit
NMF	Net mass fraction
O	Program objective
P	Candidate space program performance metric
R_{Br}	Bréguet range
V	Velocity
S	*Implementation strategy* factor
W	Weight of a prioritized *spacefaring goal*

Chapter 1

INTRODUCTION & OBJECTIVES

The catalyst for this research is best stated by T. Keith Glennan, NASA's first administrator:

> As I look back upon these two years of involvement in this exciting activity, I find myself wishing that we could have been operating in support of more clearly understood and nationally accepted goals or purposes... *How can we decide how important it is to spend, on an urgent basis, the very large sums of money required* to put a man into orbit or to explore the atmosphere and surface of Mars or Venus unless we have a pretty firm grasp of what the purpose behind the whole space effort really is? And yet, who knows the answers to this and many similar questions today? *Who is thinking about them and doing something about developing some answers?* [1]

» T. Keith Glennan
NASA's first administrator (1958–1961)
[emphasis added]

It is the goal of this research investigation to contribute towards these two questions from Glennan that will be shown to be as true today as they were in 1961:

» How can an organization decide which objectives to pursue and if they are worth the cost?

» What can be done to develop some answers?

1.1 Research motivation and background

Delivering any payload into space is very difficult; consistently reaching space with increased payloads, even more so. Coordinating the development of the required technologies with the desired missions in space into a coherent program may be the most difficult of all. This difficulty also comes with a high cost and is what led Glennan to pose the first of his questions in the beginning of 1961: *how can we decide which of these pursuits are worth it?*

Space flight has been proposed as a desirable venture for many different reasons: science, prestige, entertainment, survival, manifest destiny, *etc* [2–8]. Unfortunately, with such a wide range of possible motivations, it is typically very difficult for an organization to reach a consensus on what the primary goals should be. For example, NASA and the U.S. space program have long been criticized for failing to set appropriate goals. It seems that there are either the ever-changing near terms goals that are never given enough time to be realized, or the lofty long term goals that are permanently set 20 years in the future [9–12]. In March of this year, 2017, R. Zubrin[1] stated the following:

> The American human spaceflight program is in very bad shape right now. It is operating without a coherent and rational goal, and unless such a goal is embraced and an intelligent plan set forth to achieve it, the drift and waste will only continue until the taxpayers, losing patience, put it out of its misery. [14]

[1] Zubrin is best known for his approach to quickly put men on Mars, *Mars Direct* [13].

It seems that Glennan's first question from 56 years ago is still relevant for the U.S. space program.

The reality is that space programs are, per definition, composed in pursuit of the top-level goals and policies of the organization. If the guidance from these top-level goals is vague, the efforts of the specialists working on the technical realization of the program will be scattered in as many directions as there are specialists. Each specialist pursues greater and greater technical details in the program element they believe is best for the program (mission architecture, hardware, technology, *etc.*), losing the ability to consistently compare between competing alternatives. The paradox is that the top-level decision-makers can only refine the program goals with an understanding of what is technically feasible from the specialists. As the decision-makers look to support their selected goals with technical realities, the inconsistent recommendations from the specialists lead to decisions based on who can shout the loudest or which decision shines the best light on the decision-maker. This problem is visualized in Figure 1.1 and leads to the hypothesis and research objectives provided below.

IT IS THE HYPOTHESIS of this research undertaking that a disconnect between the top-level decision-makers of a space program and the designers and specialists of individual missions, hardware, and technologies can be observed and parametrically remedied to better support decisions being made at all levels of a program.

THE OBJECTIVES OF THIS RESEARCH are provided below:

» The development of a parametric planning methodology to augment top-level decision-makers and enable informed decisions concerning the overall direction of the program;[2]

» Evaluate the effects of a given program objective or mission architecture on the feasibility of synthesized space programs;

» Determine the possible impact of candidate technologies on the entire program as a whole to better inform research investments into enabling technologies.

[2] The scope of this research was originally focused on the synthesis of individual space mission architectures but evolved by necessity to consider the entire space program. The thoughts of the author leading up to this realization are provided in Chapter 2.

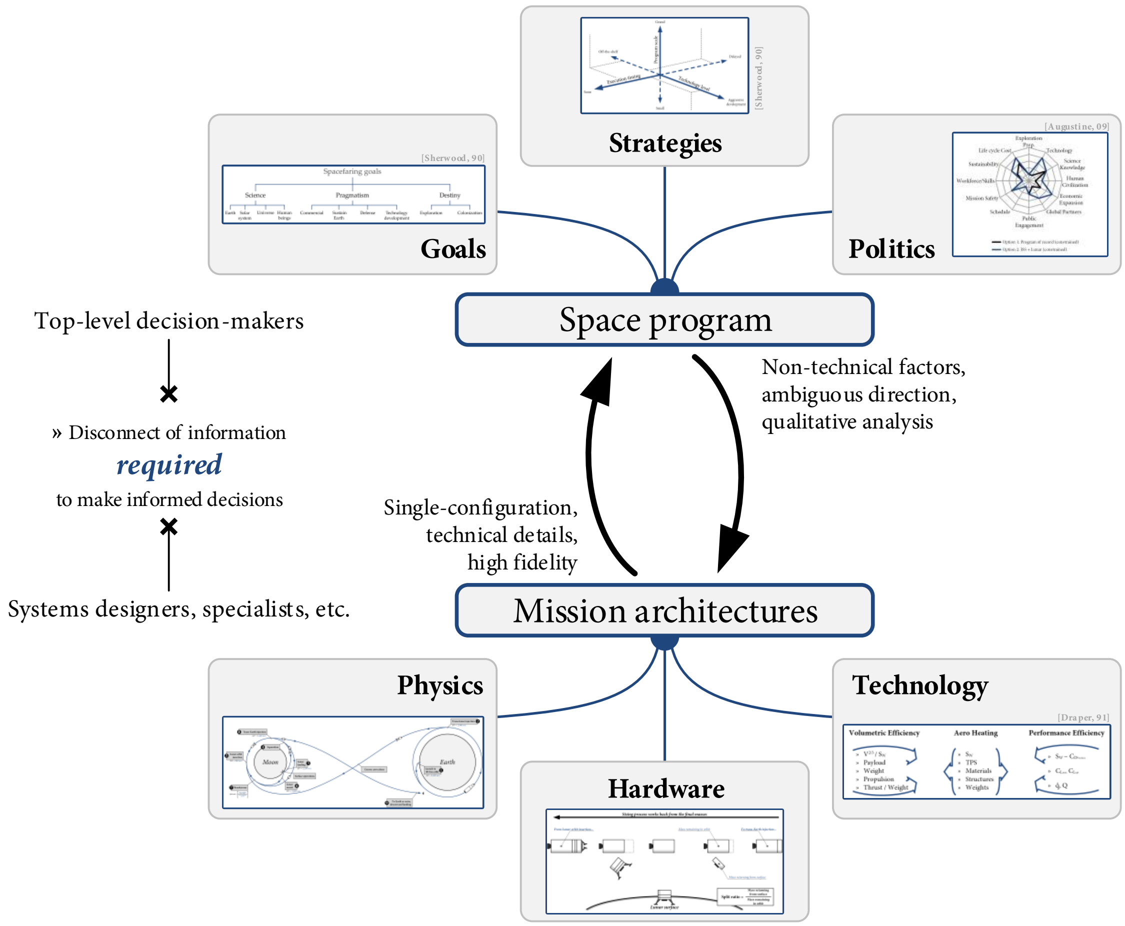

FIGURE 1.1 – Visualization of the hypothetical problem in space program planning: The disconnect of information required vs. available to make informed decisions.

1.2 *Structure of the research investigation*

The rest of this research is laid out in four separate chapters. Each chapter and its purpose in the dissertation are summarized below. The chapters and their contributions can also be visualized in Figure 1.2.

» CHAPTER 2: LITERATURE REVIEW – This chapter contains a comprehensive literature review of the background and topics that led to the problem specified in the hypothesis above and in pursuit of the current set of research objectives. First, a discussion of the author's background in aircraft design is presented, before transitioning to an extensive survey of previously proposed space mission architectures. This survey evolves to also include space program level plans and is followed with a review of the tools and processes available to the strategic planner for space program planning. The chapter closes with a compilation of the desired specifications for the ideal solution concept: a system to augment the decision-maker, parametrically connecting the top-level goals and strategies with the technical details of the required missions, hardware, and technology.

» CHAPTER 3: THE PROPOSED SOLUTION CONCEPT – An ideal system, named *Ariadne*, is proposed to fulfill all of the derived specifications from the previous chapter. The scope of this ideal system is simply too large for a single researcher, so a focus is placed on contributing to portions of the ideal that have not been completed by others. A prototype of the *Ariadne* system is then proposed that concentrates on the parametric connection between the decision makers and the technical specialists. Both the selected existing methods and those newly developed by the author are provided. Finally, the software implementation is introduced with the required end-user specification for the *Ariadne* prototype.

» CHAPTER 4: A CASE STUDY ON PROJECT APOLLO – In an effort to showcase the capabilities of *Ariadne* and its fulfillment of the required specifications and research objectives, several of the critical decisions of Project Apollo are examined. First, the implemented launch vehicle sizing process is validated with the Saturn IB and Saturn V. Then, the three competing mission architectures for manned lunar landing are analyzed and their launch vehicle requirements determined and compared. Finally, alternative program goals, strategies, and objectives are traded and their resultant space programs visualized and compared.

» CHAPTER 5: CONCLUSIONS AND FUTURE WORK – The final chapter summarizes this research as a whole, reflects upon the research objectives, and reiterates the original contributions of this dissertation. An outlook is provided on future opportunities to expand upon the framework established by the *Ariadne* prototype system.

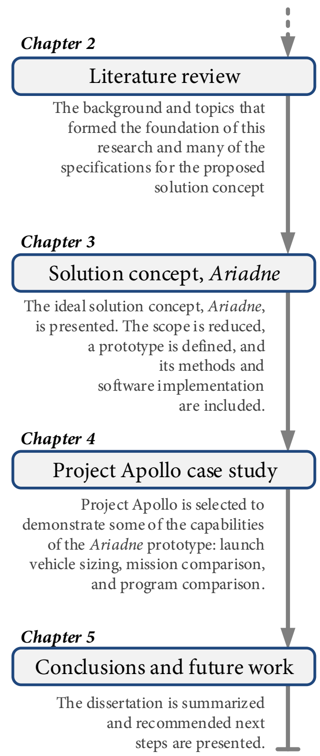

FIGURE 1.2 – Overall structure of the rest of this dissertation.

1.3 *Bibliography*

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Chapter 2

LITERATURE REVIEW

This chapter provides the required background on space program planning, along with aircraft and space mission architecture design, in an effort to identify the specifications for a decision-aid tool for the space program strategic planner as prescribed in the research hypothesis in the previous chapter.

First, an introduction to the aircraft design domain serves to highlight some of the "best-practices" and approaches to design that may not be present in the space flight domain yet. Then, a discussion of space mission architectures and an initiated survey of all previously proposed efforts represents of the early objectives of this research: to apply the best of aircraft design mentalities toward the synthesis of complete space mission architectures. This objective turns out to be less helpful than imagined,[1] thus the scope of this research is expanded to include the parametric planning of the entire space program which was alluded to in Chapter 1.

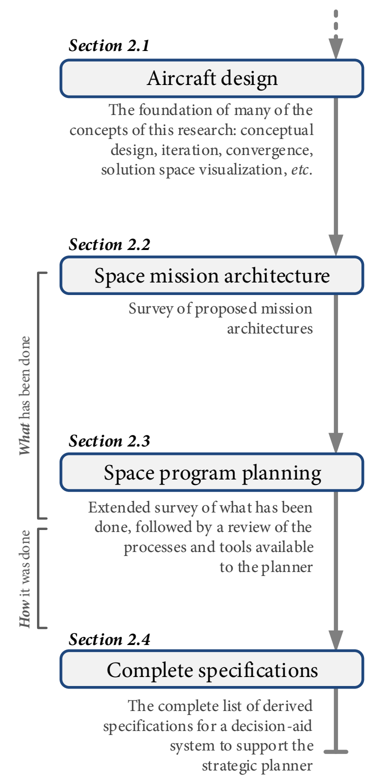

FIGURE 2.1 – Outline of Chapter 2.

[1] See Section 2.2.3.

Figure 2.1 depicts this overall progression of the chapter. Most notable are the two extensive reviews in Sections 2.2 and 2.3. First, a survey of previously proposed architectures and program plans that represent *what* has been planned. Second, a representative review of the tools and methods available for planning; *i.e., how* things are being planned.

Each of the next three sections identify the required specifications for an ideal solution to the problem proposed previously in Chapter 1. These specifications will be compiled and further discussed in the next chapter.

2.1 *Aircraft conceptual design*

This research endeavor began in the Aerospace Vehicle Design (AVD) Laboratory, with a focus on the conceptual design of flight vehicle systems. The prevailing effort in the lab is a pursuit of true synthesis of all the primary disciplines involved in the design of aerospace vehicles.[2] While in this environment, three fundamental design concepts were stressed: the importance of early decisions, multi-disciplinary design iteration, and design convergence. These concepts, along with other important lessons that can be learned from aircraft design, formed the

[2] See the compiled list of past AVD projects provided in Table 2.1.

Project	Source	Customer	Year
Commercial Transport	[1]	NASA LaRC, NIA	2004–2005
Rocketplan XP Space Tourism	–	Rocketplane	2004–2005
SpiritLear SSBJ	[2]	SpiritWing	2005–2006
Reusable Space Access Vehicle	–	NASA LaRC	2006
N+3 Transonic Transport	[3]	NASA	2008–2009
Hypersonic Transport	–	ESA	2009
Truss-Based Wing Aircraft	–	NASA	2009
Hypersonic X-Plane	[4]	NASA LaRC	2010
Manned Satellite Servicing	[5]	NASA, DARPA	2010–2011
Electric Aircraft	–	Lindbergh Foundation	2011–2012
Hypersonic Vehicle Database	[6]	NASA	2011–2012
Tranport Aircraft Mission Research	–	NASA	2013
AVX Assessment	–	Airbus Helicopter	2014
UAV Database	–	Airbus Helicopter	2015

TABLE 2.1 – Compendium of past AVD projects. Updated from E. Haney [7].

foundation for this research in space program planning and are discussed in the rest of this section.

2.1.1 *The importance of early decisions*

The earliest phase of a vehicle design, known as conceptual design (CD), is one of the most critical, and often yet often neglected, phases in the life cycle of a vehicle. Figure 2.2, adapted from B. Chudoba and W. Heinze [8] and W. Fabrycky and B. Blanchard [9], reveals the benefits of early decisions in the development of a new aircraft. Early in the development process, design freedom is at its highest, depicted by the gray line in Figure 2.2. As development progresses, each decision made reduces this design freedom and limits the possible realization of the vehicle. Unfortunately, knowledge of the vehicle and how its elements interact is limited in the beginning (where design freedom is

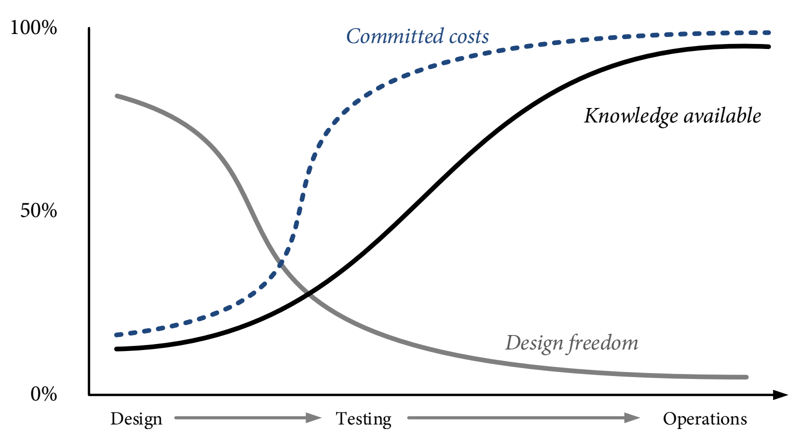

FIGURE 2.2 – A depiction of the importance of early design decisions. Adapted from Chudoba and Heinze [8].

at its highest) and typically only increases as development progresses. So later in the process, when most knowledge is available and thus the decision-maker is most informed about the vehicle, the critical decisions have previously been made, often with inadequate information. Finally, the dotted line in Figure 2.2 represents the committed costs for the development of the aircraft. Early in the conceptual design process, well before any blueprints or detailed schematics are drawn for the aircraft, the development costs have already been determined by early decisions [10]. J. Bernstein includes a similar figure in his dissertation but assumes the rate of knowledge gain is fixed. He proposes improved product development practices to delay the committed costs until later in the development process [11]. However, as shown by Chudoba [12] and G. Coleman [13], a proper,[3] parametric, conceptual design phase can provide an increase in the rate of knowledge gained at the earliest phases of the development, shifting the line representing the knowledge available to the left. This enables decision-makers to make informed decisions when they matter most.

[3] See the next section, Section 2.1.2, for a definition of what constitutes a "proper" design approach.

2.1.2 *The standard to design*

The ladder depicted in Figure 2.3 introduces many of the goals of a proper aircraft design approach. It serves to illustrate the difference between typical single discipline *analysis* (seen at the lowest step), and the upper steps that contribute towards the complete *synthesis* of a multi-disciplinary design.

The second step from the bottom of the ladder, *integrate*, represents the multi-disciplinary nature of design. No single discipline can be allowed to dominate a design, and in order to prevent that, all disciplines must be accounted for and integrated as early as possible in the design process. The exaggerated results of single-discipline-dominant designs are shown in the cartoon in Figure 2.4.

For complex, truly integrated designs, G. Cayley's design paradigm—the sum of individually optimized, decoupled subsystems resulting in an optimized complete design [15]—begins to break down as shown in Equation 2.1:

$$\sum_{i}^{j} \text{Optimum subsystems} \neq \text{Optimum design}\,. \tag{2.1}$$

This integration process can be very difficult and will likely lead to its own set of complications.

The next two steps of the ladder, *iterate* and *converge*, describe the response of the integrated design process to identified constraints. A preliminary set of input values into the multi-disciplinary analysis will produce specifications for a design. These specifications should then

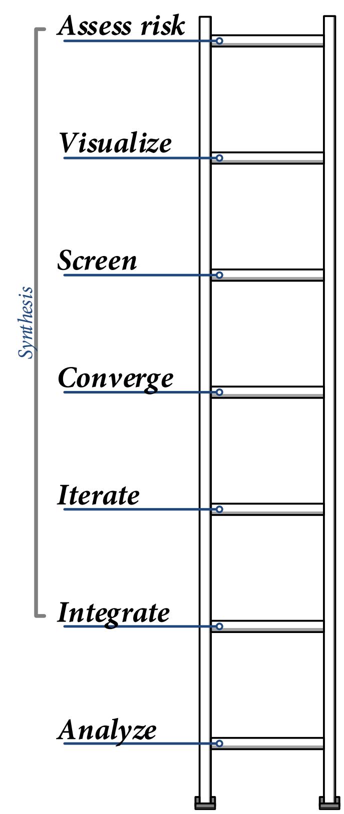

FIGURE 2.3 – Standard ladder to design [14].

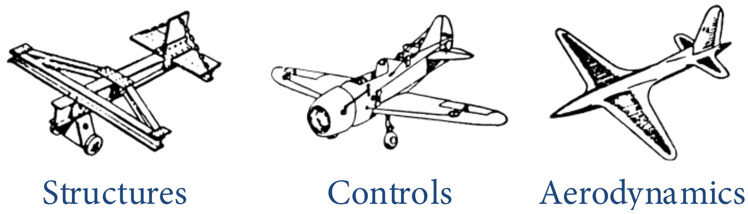

FIGURE 2.4 – Excerpt from C. W. Miller's cartoon, "Dream Airplanes," as seen in Nicolai's text, *Fundamentals of Aircraft and Airship Design: Volume I* [16].

be compared with any constraints on the design; if the design fails to satisfy these constraints, the starting values should be adjusted toward meeting said constraints, and the analysis executed again. This iteration should continue until all identified constraints have been satisfied. Once all constraints are satisfied, the design is described as a converged solution. X. Huang identifies this step, convergence, as the missing step for a majority of aircraft (and space access vehicle) synthesis systems [17].

Continuing to climb the ladder, the next two steps, *screen* and *visualize*, represent the goal of not designing a single vehicle and assuming that it is the ideal solution, but rather to converge many alternative designs from a sweep of inputs. These converged designs can then be compared, subjected to additional constraints, and plotted to visualize the entire spectrum of alternative designs and where they lie in relation to one another.

Only when all the alternative designs have been analyzed and compared can the final step of the ladder finally be reached with confidence: *assessing the risk* of the design. With the entire solution space of converged vehicles visualized, a decision-maker can now make an informed decision about which concept/configuration/design to pursue. For example, a well-informed decision-maker might opt to pursue a sub-optimum performance design due to an observed large decrease in development risk by utilizing well understood technology rather than requiring the successful application of cutting edge technologies to achieve the best possible performing vehicle. With each step of the ladder addressed, this decision-maker can justify and defend any such choices that he makes.

These standard aircraft design principles can readily be applied to hardware in the space domain, and even at the mission and space program level, as will be seen later in this chapter.

LOFTIN DESIGN PROCESS

Mission requirements, design trades, mission profile	
Initial concept research	
Define geometry trade studies: AR, Λ_{LE}, **propulsion system**	
Calculate performance constraints: W/S and T/W	
	Landing field and aborted landing: W/S
	Take-off field length: $T/W = f(W/S)$
	2nd segment climb gradient: T/W
	Climb performance: $T/W = f(W/S)$
	Cruise: $T/W = f(W/S)$
Construct performance matching diagram – based on performance constraints. Select match point: T/W and W/S	
Compute: W_{to}, W_f/W_{to}	
Compute: T, S, **and** fuselage size	
Construct performance map	

FIGURE 2.5 – A N-S representation produced by Coleman [13] of design process developed by Loftin [18].

2.1.3 *Established design processes*

Another benefit of a background in aircraft design is that, due in part to its longer history, there have been more design attempts and proper design methodologies to learn from.

Many of these processes have been decomposed by Chudoba [12] and Coleman [13] and analyzed for their respective strengths and weaknesses, *i.e.*, how well each addressed the individual steps in the ladder shown previously in Figure 2.3. Figure 2.5 depicts one such design process for subsonic aircraft by L. Loftin [18]. Its logic is presented in a Nassi-Shneiderman (N-S) structogram.[4]

[4] For more information on N-S diagrams, see Section B.2 of Appendix B.

Loftin uses a graphical approach to identify the performance of an aircraft based on defined performance constraints: take-off field length, climb performance, *etc.* Clearly identifying these constraints and visualizing them allows a designer to observe any sensitivities of the design and possibly identify any dominant constraint that is limiting the design.

ANOTHER STANDOUT process comes from D. Küchemann and J. Weber [19], who, among other things, developed an approach to compare the performance of different aircraft configurations. Their approach embodies the primary intentions of the conceptual design phase. In their own words, they are "...concerned only with the major trends underlying the main properties of the types of aircraft considered, not with any details [19]." To accomplish this, they used the Bréguet range equation, which, at a constant speed, they define as

$$R_{Br} = V I_{sp} \left(\frac{L}{D} \right) \ln \left[\frac{W}{W - W_F} \right], \tag{2.2}$$

where R_{Br} is the Bréguet range, V is the velocity of the aircraft, I_{sp} is the specific impulse of the aircraft's engine, L/D is the lift-to-drag ratio of the aircraft, W is the aircraft's overall weight, and W_F is the weight of the required fuel. This equation provides multi-disciplinary analysis on its own, integrating the mission requirements (R), aerodynamics (L/D), propulsion (I_{sp}), and structural ($W - W_F$) disciplines into a single equation. Küchemann and Weber calculated the Bréguet ranges for different aircraft configurations and assembled them in the solution space plot shown in Figure 2.6. They acknowledge that much of this space has yet to be explored (when their paper was published in 1968) with any actual design efforts, but it should be very clear that they were able to provide the exact foundation that should be required for any subsequent studies: they have identified the most promising areas for further investigation. This contribution is much more powerful than a perfectly optimized point-design that, unknown to the designer, may lie in a detrimental valley in its respective solution space.

It has been shown by Chudoba [12], Coleman [13], and Huang [17] that even with the added years of experience, aircraft design methodologies still have a lot of room to improve and that the few examples shown previously are, unfortunately, still the exceptions and not the norm. Some start off on the right track: pursuing true multidisci-

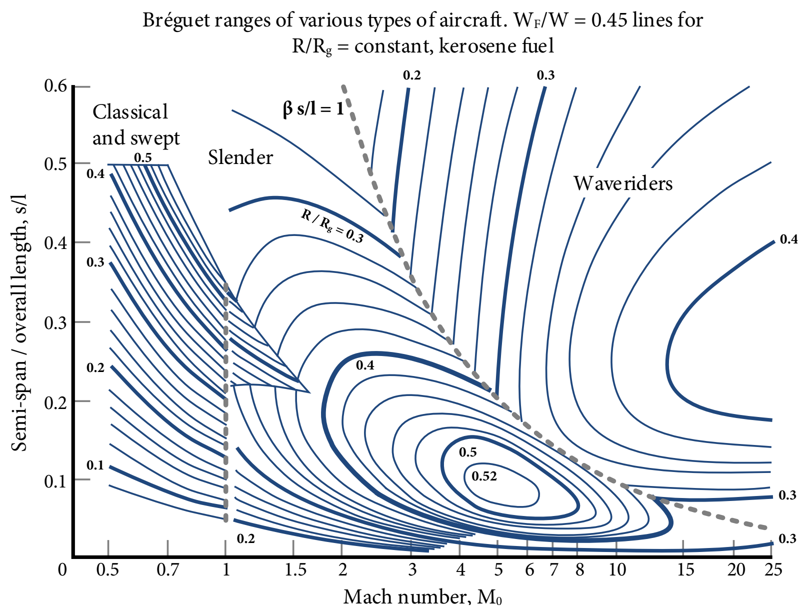

FIGURE 2.6 – Küchemann and Weber's solution space of the Bréguet ranges of three different aircraft configurations. Reproduced from Küchemann and Weber [19].

plinary convergence to the first order, but eventually end up sacrificing their inherent flexibility or multi-disciplinary focus in pursuit of a higher fidelity analysis of a single configuration/aircraft type. Others, in an effort to ensure their perceived competitive edge, package up and hide their analysis, creating a "black-box" mentality that requires absolute trust from any user.

When developing a decision-aid system for the strategic planner in the space domain, it is important to look to previous aircraft design efforts to adopt any applicable "best-practices" (multi-disciplinary, true convergence, *etc.*), and of course to avoid mistakes that have previously been made.

2.1.4 *Applicability to the space domain*

Most of the lessons learned in the previous sections are as applicable to planning and design in the space domain as they are to the aircraft design domain. Figure 2.7 serves to highlight some of the parallels between these two domains.

Aircraft designers are very familiar with designing a vehicle for various segments of the mission's flight profile. In the space domain, this can be seen with the basic sequence of phases in a mission architecture. The hardware required (launch vehicle, spacecraft, *etc.*) can be designed and subjected to performance analysis in each respective phases.

The previously discussed Bréguet range equation,[5] a first order multi-disciplinary method, is mirrored in Tsiolkovsky's rocket equa-

[5] The equation given for the Bréguet range factor in Figure 2.7 is in a slightly different form than before, to emphasize it's multi-disciplinary nature. It is still equivalent to Equation 2.2 given by Küchemann and Weber.

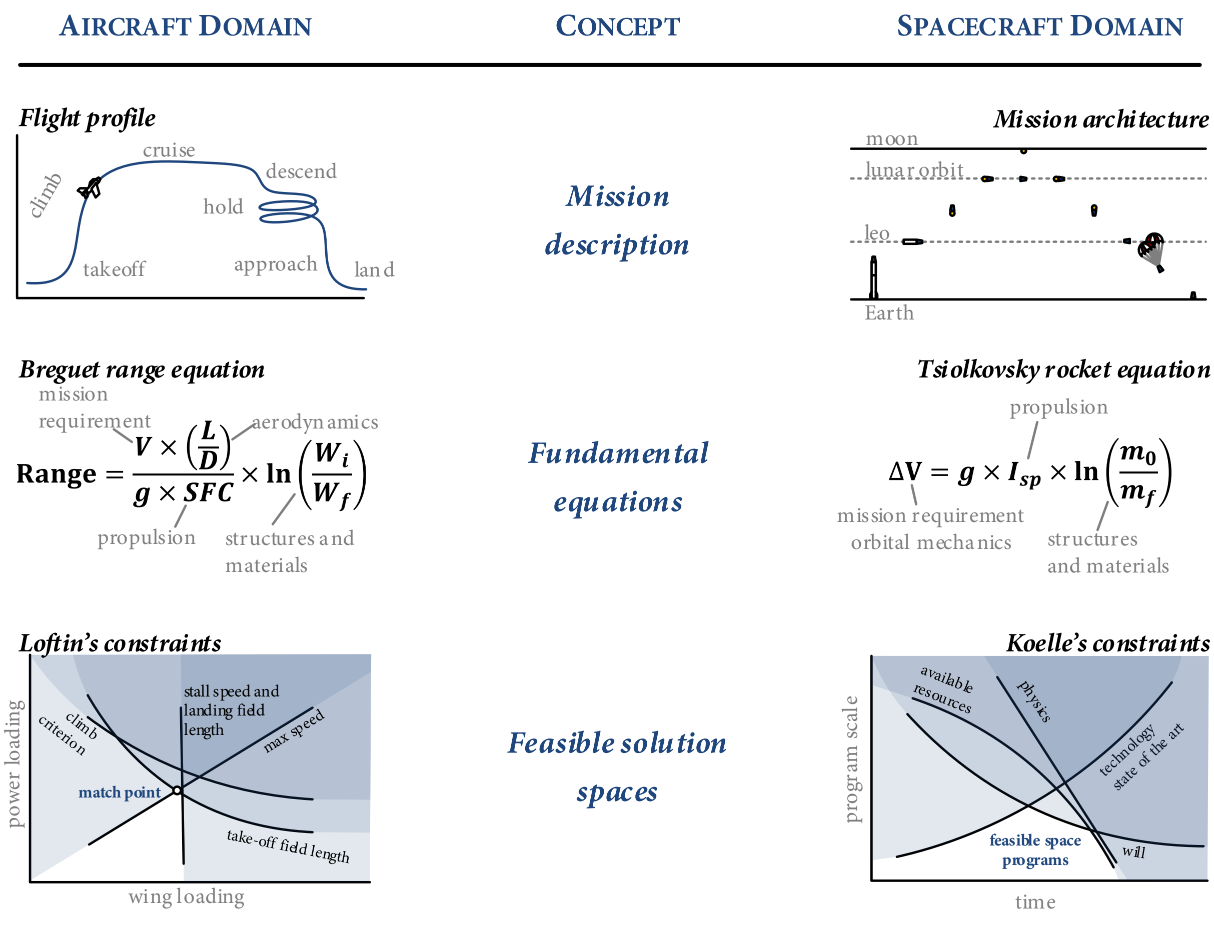

FIGURE 2.7 – Parallels observed between the aircraft domain and the spacecraft domain.

tion, shown in the Figure 2.7 and given below in Equation 2.3. This well-known "rocket equation" provides a comparable means of analyzing a mission phase and sizing the stage necessary to meet a specified velocity requirement. The Tsiolkovsky equation is given as

$$\Delta V = g I_{sp} \ln \left[\frac{m_0}{m_f} \right]. \tag{2.3}$$

In Tsiolkovsky's equation, ΔV is the change in velocity, g is the gravitational constant, I_{sp} is specific impulse of the rocket's engine, m_0 is the initial mass of the rocket, and m_f is the rocket's final mass, after all the propellant has been used [20]. Together this ratio of m_0/m_f is known as the mass ratio (MR) of the rocket.

Finally, Loftin's previously discussed performance constraints on an aircraft's design are echoed by four proposed constraints on the feasibility of an entire space program by H.H. Koelle [21]. His theory is discussed in more detail later in Section 2.3.1 of this chapter, but for now, it is only important to know that his proposed constraints are still only theory at this point. Loftin has successfully quantified and applied his performance constraints to aircraft design and therefore serves as a goal, inspiration, and possibly even a foundation for a future realization of Koelle's work.

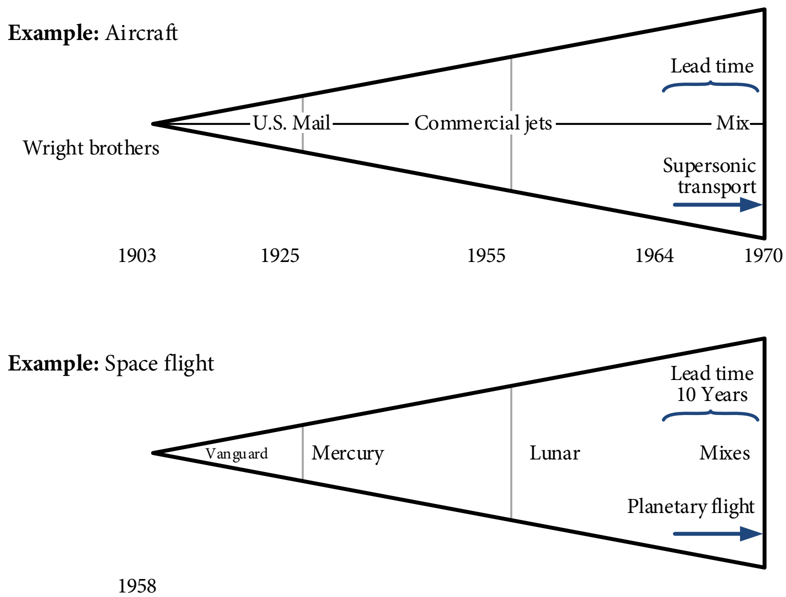

FIGURE 2.8 – Additional comparisons between aircraft and spaceflight. Note the inherent ambiguity in the timeline for the space flight progression. Reproduced from Morgenthaler [21].

G. Morgenthaler also compared the aircraft and space domains, as shown in Figure 2.8. Looking back on Morgenthaler's depiction from the 1960s, it is observed that aircraft have seemed to settle into a working configuration that has led to profitable endeavors by those major players in the commercial aircraft realm. Small improvements will continue to be made, and eventually a large leap may occur, but for now there is no real drive to pursue radically different concepts.

The space domain, on the other hand, has regressed back to its capabilities in the 1960s and is prime for a change from "business-as-usual." This is what SpaceX, Blue Origin and other commercial efforts are attempting with their development of fully-reusable launch vehicles.

W. Hammond concurs with many of the concepts discussed in the previous sections in his two textbooks: *Design Methodologies for Space Transportation Systems* [22] and *Space Transportation: A Systems Approach to Analysis and Design* [23]. In his text on design methodologies, Hammond says that "Whereas poor detailed design engineering can mess up a good concept, the best detailed design engineering will not correct a flawed concept design and selection [22]." This is perfectly reflected in the previously discussed Figure 2.2 that illustrates the rapidly decreasing design freedom as development progresses. In fact, Hammond produces a very similar figure adapted specifically to the development of space transportation systems where the goals of increased knowledge and design freedom in the early design process are highlighted. His depiction is shown reproduced in Figure 2.9. In that same text he also speaks of the decline of the "integrator," the

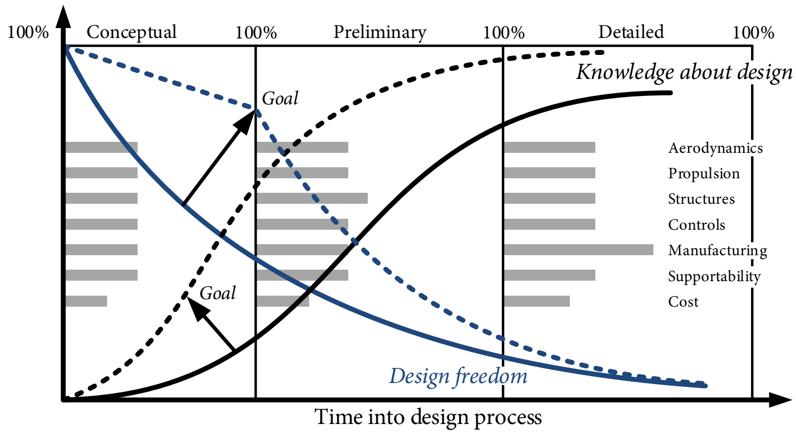

FIGURE 2.9 – Desired goals for the design process: increasing the knowledge available and the design freedom in the early phases of the design. Reproduced from Hammond [23].

multi-disciplinary thinker. Hammond says:

> ...the importance and prestige of analytical specialists soared. Specialists were needed to expand the limits of scientific knowledge and to reach for ever higher performance. The best minds were attracted by the challenges of research and development, which usually meant estrangement from design. As a result, the 'generalist' design engineer's prestige declined. [23]

He is speaking of the prevalent focus on single-disciplinary analysis. These specialists are high performers in their respective fields and are often eventually promoted into positions requiring multi-disciplinary design decisions, for which they are ill-equipped by the very nature of their specialist-mentality.

2.1.5 *Ideal specifications*

This section has identified many of the "best-practice" efforts available in the aircraft design domain. It was made clear that the concepts and thinking behind such approaches can and should be applied in the space domain.[6] From this review, the following ideal specifications for a space program planning system to augment the strategic planner have been derived:

[6] ...if they are not already. Section 2.3.4 will include a discussion of the current state of design/planning processes in the space domain.

» Aid the decision-maker in the earliest phases of design with a parametric, forecasting capability;

» Strive for first order correctness rather than high-fidelity accuracy;

» Multi-disciplinary design synthesis: integration, iteration, convergence, screening, visualizing, and risk assessment.

Element	Definition
Launch	Includes any means a spacecraft uses to get into orbit. The supporting infrastructure of the launcher is also included; launch site, equipment and personnel are all members of this element.[a]
Spacecraft	Consists of the payload, which is the hardware and software that are used to accomplish the objective of a given mission, and the spacecraft bus (the systems that support the payload).
Operation	The people and software that manage the mission on a day-to-day basis. This includes scheduling, data flows, *etc*.
Communications	How all the parts of a given mission communicate with each other.[b]
Ground	The equipment and facilities that communicate with and control any operating spacecraft. Wertz [24] lists this element separately, but it fits readily into the communications element detailed above, so it might be best to combine them in any future research effort.
Orbit	The path of the spacecraft during its mission. Often, this element is very particular for a given mission, but there are several benefits from integrating the orbits of all missions: providing an equal coverage over a body, avoiding collisions, and timing future launches, just to name a few.

[a] Future missions that involve launches off of other heavenly bodies might require special consideration but for now, launches off other heavenly bodies are accounted for in the spacecraft element, since all of that hardware and infrastructure had be transported there through space in the first place.

[b] This capability is easily expanded to other missions and projects, but only if the policies allow the transfer. For example, a military communications infrastructure, no matter how advanced, may not be available for use by commercial launches.

TABLE 2.2 – Definitions of the elements of a mission architecture adapted from Wertz and Larson [24, 25]

2.2 *Space mission architecture design*

One of the early research objectives of this research was the application of the previously defined aircraft design principles to the design and synthesis of space mission architectures.[7] This section explores that original objective, beginning with the proper definitions of the topics involved and ends with the initiation of an extensive survey on previous proposed mission architectures. A metric is introduced in order to quantify the quality of these plans in order to provide a more "structured exposure" to these past efforts. This survey is interrupted by the realization of the need to expand the scope beyond a single mission.

[7] Defined below in Section 2.2.1.

2.2.1 *Traditional definitions*

As noted earlier, a basic space mission architecture can be thought of similarly to the flight profile in the aircraft domain. Due to the inherent complexity and number of contributing elements toward an actual space mission, the complete definition of a mission architecture usually goes much further than that. The definitions in Table 2.2 have been adapted from Wertz and Larson's *Space Mission Analysis and Design* [24] and Wertz's *Space Mission Engineering: the New SMAD* [25] where each of the specified *mission elements* are introduced. These de-

In the definitions given in Table 2.2, there is, and should be, some overlap among the elements. Even so, these generalizations are useful when discussing aspects of any given mission.

Score	Description
0	Element not addressed in architecture
1	Element qualitatively discussed (idea proposed, options mentioned, qualitative trades, *etc.*)
2	Components of the element quantified (launch dates, weights, engine performance, power required, *etc.*)
3	Quantified trades of different components in the element (propulsion options, trajectory options, sensor options, *etc.*)
4	Quantification plus methods for reproducing the values included (launcher sizing, weight estimation methods, *etc.*).[a]
5	Quantified solution space of possible element alternatives.[b]

[a] An outline of the process is also acceptable if the method itself would be too lengthy or proprietary. The main idea is to see evidence that the planner has and can look at the surrounding alternative plans.

[b] Ideally this is fully integrated among all the program elements, but for the sake of this initial survey a solution space for a single element is still awarded this score.

TABLE 2.3 – A metric created for the survey of mission architectures. The goal was a "structured exposure" to the vast amount of material available.

fined elements will be used in the survey introduced in the following section.

2.2.2 *Survey of mission architectures*

In pursuit of the original research objective to develop a synthesis system capable of modeling, comparing and identifying the preferred space mission architecture, a broad survey of proposed mission architectures has been initiated. This includes any sources that identify themselves as an architecture, along with other sources that are judged by the author to represent an architecture as defined by the inclusion of multiple mission elements given in Table 2.2. Hundreds of proposed architectures have been found in various formats: conference proceedings, journal papers, internal presentations, and company technical reports. This survey is not meant to be exhaustive.[8] A sharp focus on many of the individual included studies would likely turn up an additional source or two. It is the contention of the author that this survey does provide a representative set of architectures, extensive enough to observe any characteristic trends that may be present and assess the overall state of previous efforts; *i.e.*, what all alternatives for space activity have been proposed.

[8] It is hard to imagine the effort that would be required to substantiate that claim due to the sheer number of sources produced and the historically fluid definition of what actually constitutes an architecture.

A metric was devised to provide more structure to the survey and move it away from a purely qualitative discussion. The metric was combined with the mission elements defined in Table 2.2 to answer two questions concerning the architecture's coverage and quality:

Importance	Definition	Explanation
1	Equal importance	Two activities contribute equally to the objective
3	Weak importance of one over another	Experience and judgment slightly favor one activity over another
5	Essential or strong importance	Experience and judgment strongly favor one activity over another
7	Very strong or demonstrated importance	An activity is favored very strongly over another; its dominance is demonstrated in practice
9	Absolute importance	The evidence favoring one activity over another is of the highest possible order of affirmation
2, 4, 6, 8	Intermediate values between adjacent scale values	When compromise is needed

TABLE 2.4 – Definition of the pairwise comparison metric. Reproduced from Saaty [26]

1. Coverage » is each mission element addressed?

2. Quality of analysis » how well is each mission element addressed?

These two questions serve to asses an architecture's "ability to convince." The first question is easy enough to answer by simply reviewing the source; if the source mentions an element, then that element is considered addressed. To answer the question of quality, however, requires the scale defined in Table 2.3. Now, for each surveyed source, a score of 1-5 can be awarded for each mission element addressed, depending on the quality of the analysis included.

It quickly becomes apparent that although all the defined elements are required to specify a truly comprehensive mission architecture, they are not all equal early in the decision-making process when it comes to informing the strategic planner or convincing a shareholder. For example, ground equipment and infrastructure are required to some degree for any space mission. However in most cases, it will not be the primary element that determines a mission's feasibility. Having an adequate launch vehicle on the other hand, may often determine the possibility of a mission without the need to consider most of the other elements. Therefore, certain elements needed to be emphasized during the survey, and an architecture's total score should reflect these prioritizations. This manner of prioritization forces the surveyed sources to be viewed from the perspective of the early design phase and the information required to make an informed decision this early in the process. It is understood that not all sources have been necessarily written for this purpose, but, due to the previously discussed impact of early decisions, this should be the ideal lens to view any architecture to assess its "ability to convince" a decision-maker of its feasibility.

THIS REQUIREMENT FOR PRIORITIZATION was the perfect opportunity for the T. Saaty's Analytic Hierarchy Process [26]. The analytic hierarchy process (AHP), is a formal decision framework that excels in including qualitative factors into the decision process. Saaty explains its importance and purpose this way:

> We must stop making simplifying assumptions to suit our quantitative models and deal with complex situations as they are. To be realistic our models must include and measure all important tangible and intangible, quantitatively measurable, and qualitative factors. This is precisely what we do with the analytic hierarchy process. [26]

The AHP begins with the use of pairwise comparisons to assess the relative priority of each element to each of the others. The complete process is given in some detail here, as it shows up again in the solution concept of this research.

These comparisons are quantified from 1–9, with 1 representing no preference between the two options and 9 representing an extreme level of importance of one option over the other. A full description of what each score represents is provided in Table 2.4. For example, when comparing two elements, A and B, if element A has a strong importance over element B, then that comparison receives a score of 5. Inversely, comparing B's importance to A, in turn, receives a score of $1/5$.

Each mission element was compared with the other elements and assigned a priority based on the element's discerned importance to the early design and its contribution towards convincing a decision-maker of an architecture's feasibility. The completed comparisons of all of the mission architecture elements can be found in the decision matrix below:

$$
\begin{array}{r c}
 & \begin{array}{cccccc} \text{Launch} & \text{Spacecraft} & \text{Ground} & \text{Orbit} & \text{Comm.} & \text{Operations} \end{array} \\
\begin{array}{r} \textbf{Launch} \\ \textbf{Spacecraft} \\ \textbf{Ground} \\ \textbf{Orbit} \\ \textbf{Comm.} \\ \textbf{Operations} \end{array} &
\begin{bmatrix}
1 & 5 & 9 & 3 & 9 & 9 \\
1/5 & 1 & 9 & 3 & 5 & 7 \\
1/9 & 1/9 & 1 & 1/9 & 1/3 & 1/3 \\
1/3 & 1/3 & 9 & 1 & 9 & 5 \\
1/9 & 1/5 & 3 & 1/9 & 1 & 1 \\
1/9 & 1/7 & 3 & 1/5 & 1 & 1
\end{bmatrix}
\end{array}
$$

This decision matrix, A, is read by looking at the element in the row and then reading along each column to see how it compares with each other element. Note the values of 1 along the primary diagonal that represent an element being compared with itself.

The AHP then requires the determination of the principle eigenvector, which is given below in Equation 2.4.

$$w = \begin{bmatrix} 0.84 \\ 0.24 \\ 0.02 \\ 0.18 \\ 0.04 \\ 0.04 \end{bmatrix} \tag{2.4}$$

When normalized, this eigenvector leads to the prioritized set of weights for the mission elements given in Equation 2.5. Note that the rows have been rearranged here in order of priority to more easily see how the elements rank among one another.

$$\begin{array}{r c} & \text{Weight} \\ \begin{array}{r} \textbf{Launch} \\ \textbf{Spacecraft} \\ \textbf{Orbit} \\ \textbf{Communications} \\ \textbf{Operations} \\ \textbf{Ground} \end{array} & \begin{bmatrix} 0.476 \\ 0.238 \\ 0.181 \\ 0.041 \\ 0.041 \\ 0.023 \end{bmatrix} \end{array} \tag{2.5}$$

Saaty provides a method to check the consistency of the assigned comparisons that is based on the ratio of two parameters: the Consistency Index and the Random Index. First, the Consistency Index (C.I.) is calculated with the following equation,

$$\text{C.I.} = \frac{\lambda_{max} - n}{n - 1}, \tag{2.6}$$

where n is the number of criteria being compared and λ_{max} is the largest eigenvalue of A'. In this case, n is 6 and λ_{max} was calculated as 6.614 and therefore Equation 2.6 results in a C.I. value of 0.1228.

Next, the Random Index (R.I.) is based on n. In this case with 6 elements being compared, R.I. is given as 1.24. Finally, for the comparisons to be considered consistent, Saaty says that the ratio of of C.I. to R.I. must be less than 0.1. Equation 2.7 shows that the comparisons for the mission elements given above are, in fact, consistent.

$$\frac{\text{C.I.}}{\text{R.I.}} = \frac{0.1228}{1.24} = 0.099 \tag{2.7}$$

With these consistent weights, along with the author-designated score for each element, were used to determine a total score for each architecture's "ability to convince" as seen in Equation 2.8.

$$\text{Ability to convince} = \sum_{i=1}^{6} s_i \cdot w_i\,, \tag{2.8}$$

ID	Date	Architecture	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
1	1953	von Braun - *The Mars Project*	3	3	0	3	0	0	**2.6**	[27]
2	1958	Ehricke - *Instrumented Comets*	1	3	1	4	1	1	**2.21**	[28]
3	1960	U.S. Army - *Lunar Soft Landing Study*	3	2	2	4	2	1	**2.96**	[29]
4	1961	Houbolt - *Lunar Orbit Rendezvous*	3	3	0	3	2	3	**2.86**	[30]
5	1961	Koelle - *Lunar and Martian Mission Requirements*	4	3	0	3	0	1	**3.06**	[31]
6	1961	U.S. Air Force - *LUNEX*	3	2	1	2	1	1	**2.28**	[32]
7	1963	Hammock - *Mars Landing*	1	3	0	2	0	2	**1.57**	[33]
8	1964	Bono - *ICARUS*	2	1	1	2	0	1	**1.64**	[34]
9	1964	Lockheed - *Manned Interplanetary Missions*	3	4	0	3	1	2	**2.92**	[35]
10	1966	Ehricke - *Future Missions*	3	3	0	3	0	2	**2.69**	[36]
11	1966	Bellcomm - *Manned Flybys of Venus and Mars*	2	2	0	3	0	0	**2.03**	[37]
12	1966	Woodcock - *Manned Mars Excursion Vehicle*	1	2	0	4	0	1	**1.96**	[38]
13	1967	Auburn University - *Jupiter Orbiting Vehicle for Exploration*	4	4	3	4	4	2	**3.89**	[39, 40]
14	1968	NASA - *Advanced Mars Orbiter and Surveyor*	2	4	2	4	4	2	**3.04**	[41]
15	1968	Ginzburg - *Interplanetary Spaceflight Missions*	3	3	0	2	3	0	**2.5**	[42]
16	1968	Aerospace Group - *Integrated Manned Interplanetary Spacecraft*	3	4	2	5	2	4	**3.71**	[43–49]
17	1969	Rockwell - *Extended Lunar Orbital Rendezvous (ELOR)*	0	3	1	1	2	2	**1.02**	[50, 51]
18	1969	McDonnell Douglas - *Integral Launch and Reentry Vehicle System*	5	4	1	5	2	2	**4.42**	[52–56]
⋮		⋮							⋮	
300	2015	NASA - *Humans to Mars Orbit*	1	2	0	1	0	2	**1.12**	[57]
301	2015	Purdue University - *Project Aldrin-Purdue*	2	4	4	4	4	2	**3.09**	[58]
302	2015	Moonspike - *Project Moonspike*	2	2	1	4	2	3	**2.61**	[59]

» Full survey can be found in Table A.1 of Appendix A

TABLE 2.5 – Abbreviated list of entries in the mission architecture survey. Each element is scored according to the previously introduced metric, and the final prioritized score calculated with the element weights from AHP.

where s_i is the score awarded to a particular element and w_i is the prioritized weight for each element, i, listed above. This means that the studies will receive a final score between 0 and 5, where 5 represents a more convincing architecture. Table 2.5 contains several examples of this process including the identified architecture name, each of the scored elements, and the final calculated score. This process was completed for several hundred architectures. A portion of the results are given in here in Table 2.5 while the complete list can be found in Table A.1 of Appendix A.

An initial impression from this survey is the sheer number of architectures that have been produced and are available for review. Whatever the reason for the recent lull in space exploration, it is arguably not from a lack of ideas. Though, it certainly could be from the lack of an ability to appropriately compare ideas and identify the best option.

Take the Moon-first vs Mars-first debate: Both of these concepts are considered in many of the recent surveyed architectures. Some of them have become quite famous, even synonymous with the concept

[60]. However, even among those that are able to adequately make a case for their approach, they all lack a consistent means to compare with one another at the top-level. A Mars plan might criticize a Moon plan for its apparent difficulty to utilize *in-situ* resources (at least in a manner similar to the Mars plan). But should that alone really disqualify the Moon plan? Similarly, just because a strength of the Moon plan (*e.g.*, proximity to Earth) is not shared by the Mars plan, it should not unequivocally become the preferred option. This inability to consistently compare, "apples-to-apples," alternate mission architectures is an all-too familiar problem known in the aircraft design domain in their attempts to compare alternative aircraft configurations [12].

It should be noted that this survey only accounts for the technical aspects, per the definitions of the mission elements of a given mission architecture. Other, non-technical factors (political, economic, social, *etc.* [61]), are not included in the survey though it should be clear now that these non-technical factors can sometimes play an equal, possibly even a greater role in assessing an architecture's feasibility. They are often even more difficult to quantify and thus difficult to consistently compare. When this fails, the discussion turns to qualitative arguments on which destination has the most *value*, a term most difficult to define. This concept of a mission's value is one that could be philosophically debated here *ad nauseam*. A number of excellent strides have are being made toward solving this very problem [62–71], though these efforts appear absent from the analysis within the sources surveyed.

At this point, it became apparent to the author that limiting the scope of the research to only a single mission architecture (as opposed to multiple missions, projects, or even programs) could not answer any of the questions posed by Glennan in the introduction of this research. Morgenthaler comments on this issue by stating that:

> ...if we select the optimum booster and spacecraft for each mission (as happens when only single missions are analyzed), we have no guarantee that a space program composed of the totality of these optima is an optimum space program. In fact, if we evolved a space program that did not develop several *standard* spacecraft and boosters, but developed all those that were optimum for the various missions, a disproportionate amount of R and D money would be expended per payload ton delivered to planetary and other destinations. We would not be developing economical space transportation, but providing 'economy at any price.' [21]

This statement by Morgenthaler is another example of the breakdown of Cayley's paradigm, which was discussed earlier.

Additional justifications for this expansion in scope and research objectives are provided in the following section. The completed survey is presented later in Figure 2.14 and includes the technical analysis of space program-level plans, a larger scope that will be defined in Section 2.3.

2.2.3 *Need for an expansion in scope*

In the midst of the survey of mission architectures, it became clear that attempting to synthesize a new mission architecture would present a substantial problem: objectives in space are typically too expensive to expect a unique mission architecture with all unique components synthesized to complete a given objective.[9] For example, a launch vehicle is simply too costly to develop and produce for a one off mission. In fact, one of *Akin's Laws of Spacecraft Design* [72] reads:

> The three keys to keeping a new manned space program affordable and on schedule:
>
> 1. No new launch vehicles
> 2. No new launch vehicles
> 3. Whatever you do, don't develop any new launch vehicles

[9] This was summarized my Morgenthaler previously.

Problems concerning mission design tend to involve a shorter lead time than program-level decisions and, in an effort to remain affordable, become more about the best use of the available components. Any decisions requiring a large or unique element (obviously excluding a mission-specific payload) made at this level are short sighted. Decisions made at this limited scope can cost a organization a lot of money. R. A. Smith described these limited-scope decisions as *piecemeal decisions* that can only be avoided with adequate long-range planning [73]. He attributes a $425 million dollar loss by General Dynamics in the 1950s to piecemeal decision making.

Any hopeful program will have to utilize its developed capabilities across its entire mission portfolio. This observation led to the one of the current research objectives concerning early space program synthesis: integrating all desired future missions and the hardware required into a feasible program plan in order evaluate the effects of selected missions on the whole program.

S. Dole called for a long-range planning capability for NASA in the late 60s and described it as follows:

> The following are salient aspects of long-range planning, which may be defined as the conscious determination of courses of action to achieve prescribed goals:
>
> » The process begins with an examination of long-range objectives and develops from them concrete goals for achievement.
>
> » It establishes policies and strategies.
>
> » It examines the future consequences of present decisions and provides and overall frame of reference for making decisions.

» Above all, it considers a complete spectrum of future alternative strategies and courses of action.

» It does this for extended time periods.

> A number of compelling reasons why NASA should have a centralized long-range planning organization can be cited: long lead times on hardware, narrow and infrequent launch windows for planetary missions, budgetary limitations, potential changes in the future role of NASA and program complexities that suggest the need for assistance to decision makers. [74]

These statements came at the height of Project Apollo. Many had already been thinking about the next steps for the national program, and Dole recommended that, due to the complexities of space missions, the organization should first concern itself with establishing a group to oversee the program as a whole and plan out a coordinated effort for the foreseeable future. This recommendation was made in an effort to avoid inefficient, expensive piecemeal decisions for the future of the space program.

2.2.4 Ideal specifications

The following ideal specifications for a space program decision-aid system have been identified from the initial survey of mission architectures:

» Identification and prioritization of the primary drivers;

» Capable of making consistent comparisons (*apples-to-apples*);

» Consideration of both technical and non-technical factors, but avoid absolute definition of what *value* is.

2.3 Space program planning

The current scope of this research requires a brief discussion of a new term for this domain: planning. Up until now, the focus has been on designing but it will be seen that many of the concepts and lessons-learned can be applied here as well.

The following quote by G. Steiner outlines a preliminary description of planning:

> Planning may be described from four points of view. First, a basic generic view of planning is dealing with the futurity of present decisions. This means that current decisions are made in light of their long-range consequences. It means also that future alternatives open to an organization are examined and decisions made about preferred alternatives. On this basis, guidance is provided for making current operating decisions. There are also many other conceptual views of planning; one

> concept, for example, recognizes planning as reasoning about how you go from here to there.
>
> Planning is also a process. It is a process which establishes objectives; defines strategies, policies and sequences of events to achieve objectives; defines the organization for implementing the planning process; and assures a review and evaluation of performance as feedback in recycling the process.
>
> Planning may be considered from a third point of view–namely, as a philosophy. Planning has been described as projective thought, or 'looking ahead.' Planning in this sense is an attitude, a state of mind, a way of thinking.
>
> Finally, planning may be viewed in terms of structure. Long-range planning, as the term is typically used in the business world, refers to the development of a comprehensive and reasonably uniform program of plans for the entire company or agency, reaching out over a long period of time. It is an integrating framework within which each of the functional plans may be tied together and an overall plan developed for the entire organization. [74]

Steiner is speaking of planning in a business sense, though it should be seen that these points of view can be applied to planning in any domain. As such, the first and second points of view provided by Steiner can readily be tied to both the design mentality introduced at the beginning of this chapter and the space program planning-focused research moving forward. Concerning the first view, Steiner also says that "Planning is not making future decisions. Planning is concerned with making current decisions in light of their futurity [73]." This is right in line with one of the goals of design: increasing the knowledge of the decision-maker as early as possible. Applied to the scope of a space program, planning should be concerned with the technology being researched, hardware being developed, and the missions being designed. Properly accounting for the "futurity" of these components, allows informed decisions to made for the present space program. According to Steiner's second point of view of planning, space program planning should establish the objectives to be accomplished and a means to accomplish them. This "planning as a process" view defines the programmatic, time, and resource requirements to reach desired program objectives. This process provides both transparency and sanity checks along the way for the decision-makers. A given objective might sound good "on paper," but after proper planning details everything required to achieve it, including sacrificing efforts towards other objectives, the decision-makers can now reconsider the program objectives.

A similar concept to planning is that of forecasting. The two terms *planning* and *forecasting* may appear to be interchangeable, but B. Twiss says that:

Term	Definition
Element	single component that contributes towards a space effort (see Table 2.2)
Launch	single effort with a specific vehicle, purpose, and customer
Mission architecture	clearly definable space effort comprising one or more launches
Project	space flight effort consisting of one or more mission architectures, all contributing towards a specific objective
Program	combination of space projects and technology developments established to accomplish broad organizational goals

TABLE 2.6 – Working definitions adapted from Kolle and Voss [62].

> It is essential to make a clear distinction between the forecasting and planning functions, although in practice this may appear academic where both activities are performed by the same person. Forecasting is focused on the outside environment. Its aim is to evaluate what will happen in the future irrespective of the company's own actions. It is inevitably probabilistic.
>
> Planning, in contrast, is deterministic. Combining the forecasts and an evaluation of them with other considerations, it is a statement of what the organization will do and is the basis for managerial action. Its aim is to ensure that there is coordination of all decisions throughout the company. [61]

Thus, according to Twiss's definitions, it can be argued that the forecasts are contained in the information coming from the individual elements of a program plan: *e.g.*, propulsion performance on the next generation launch vehicle, a new technology's effect on the battery capacity of weather satellites, *etc.*

Planning, when looking at the entire space program, must take these forecasts into account as decisions are made for the overall direction of the program. A. Robertson and P. Fatianow speak about this program-level integration and its difficulty, but also to its necessity:

> The enormous scope and complexity of space programs places a great burden on the decision-makers concerned ... In planning to provide hardware for future space programs, it is necessary to be able to determine or estimate what programs will be undertaken and how these programs might be carried out. [75]

For a better understanding of planning specific to space programs, the following discussion adds to the definitions given earlier in this chapter.

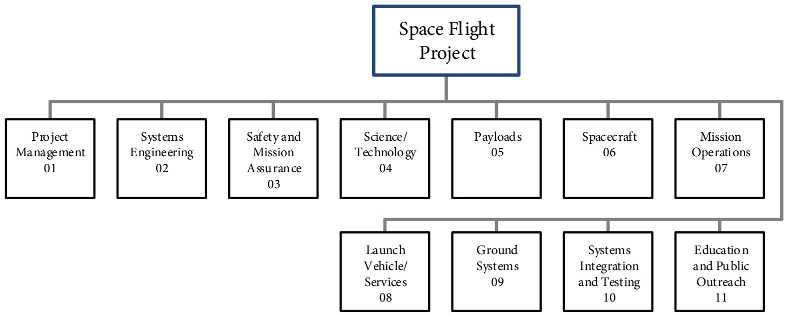

FIGURE 2.10 – NASA's breakdown of the elements involved in planning a given project. Reproduced from NASA [76].

2.3.1 *Additional definitions*

As previously discussed with mission architectures and the mission elements involved, the terminology involved with this topic can be inconsistent and lead to confusion. Thus, for this expanded research objective, the necessary terms and their relation to one another have been defined in Table 2.6. These definitions have been adapted from Koelle and R. Voss [62] to be consistent with the previously defined terms from Larson and Wertz found in Table 2.2. The term *element* refers to Table 2.2's mission architecture elements. A single *launch* consists of all the contributing mission elements.[10] A *mission architecture* is then composed of one or more launches, and a *project* composed of one or more mission architectures. Figure 2.10 depicts NASA's breakdown of the elements involved in a project. A *space program* is the integration of all the elements from each mission architecture and project of the past with current and even future elements yet to be developed. NASA presents the developmental life-cycle of both a program and its relation to projects from a Systems Engineering (SE) standpoint in Figure 2.12. A single mission into space can involve years, even decades, of development and testing before it is ever executed (Figure 2.11).Now imagine the space program responsible for coordinating said mission, in accordance with its respective overarching project and ideally complementing several other projects all working towards the program's ultimate goals.

[10] It is listed separately from the term *mission architecture* here due to complex mission architectures that employ multiple launches to accomplish their objective.

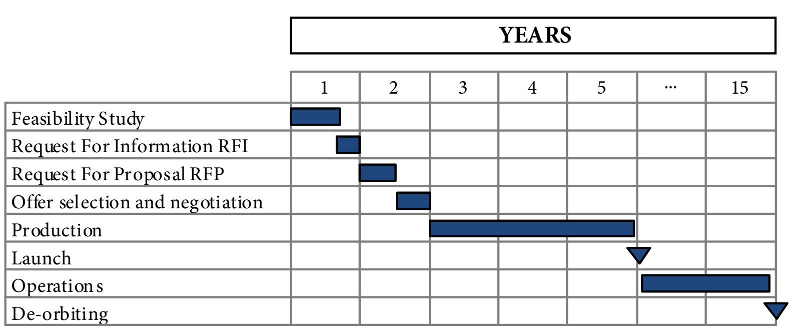

FIGURE 2.11 – Typical decision timeline of a commercial program. Reproduced from Spagnulo and Fleeter [77].

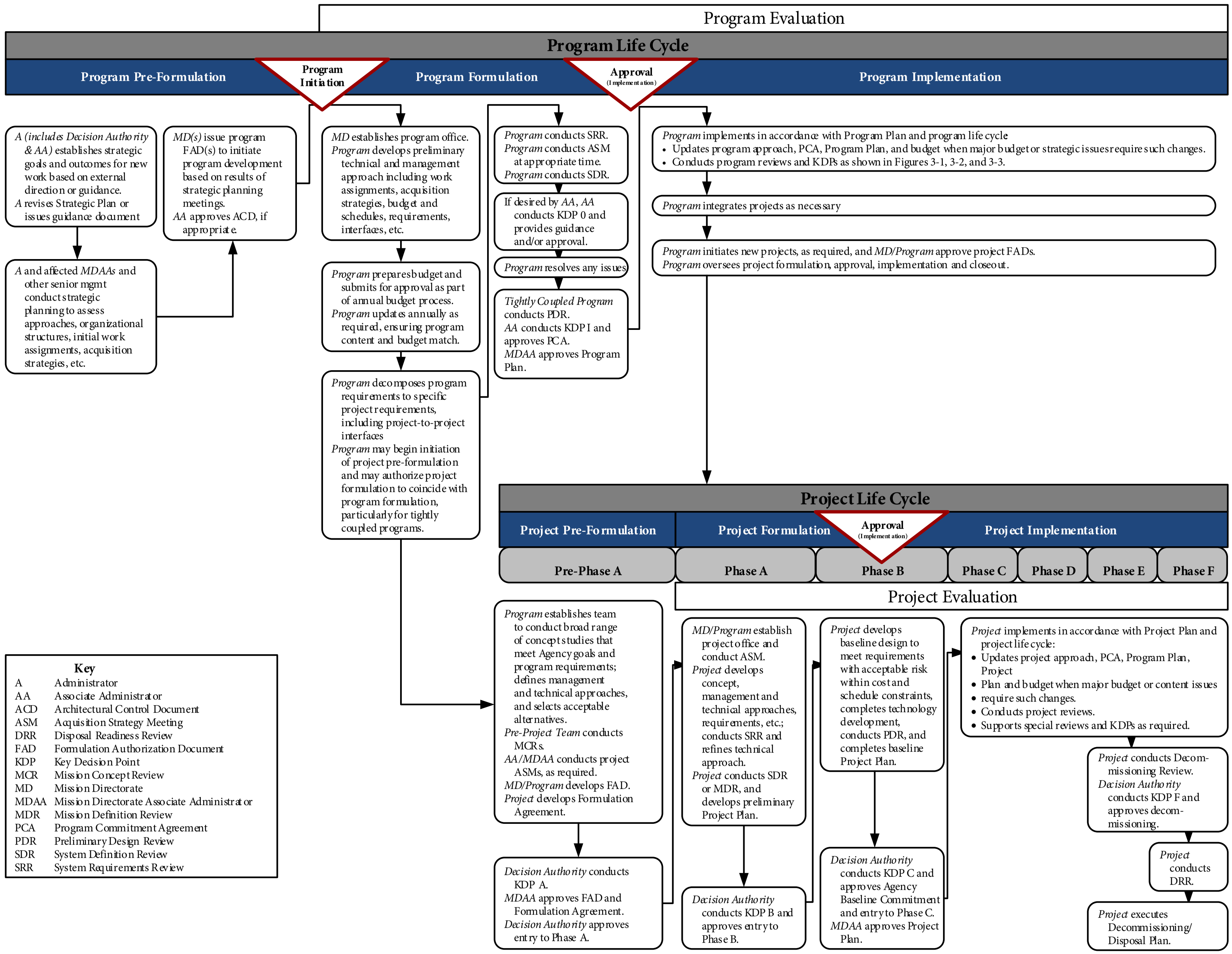

FIGURE 2.12 –
NASA's representation of the interaction between the life-cycles of the program and project. Reproduced from NASA [76].

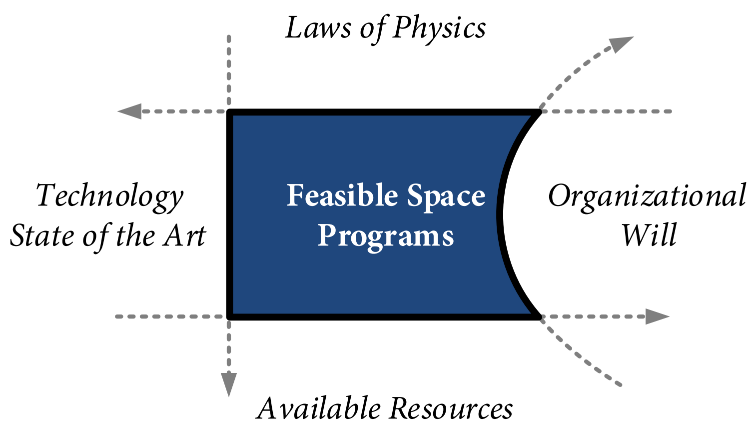

FIGURE 2.13 – The four constraints on the feasible planning space of space programs. Concept originally introduced by H.H. Koelle and figure adapted from Morgenthaler [21].

As an example of this complexity and required lead times, consider what Hammond says about the developmental timeline for the Space Shuttle:

> The design process can take decades, if one includes the tremendous amount of technical papers, trade studies, study contracts, and contractor in-house studies conducted before a Space Transportation System (STS) concept is finalized or a reference design ins baselined for full-scale development. For example, preliminary ground work for the nation's next-generation STS was begun before the first Space Shuttle rolled off the final assembly line in Palmdale California, in 1980. [23]

These decisions about the direction of the space program are too complex with too large an effect on the program to be made under short notice. Recall that these short notice decision are referred to as *piecemeal decision-making* by Smith and dubbed *reaction engineering* by Hammond. In an effort to highlight the necessity of avoiding decisions based on these reactions, he stresses that "It is quite common that nearly 95% of a program's costs are established during the first 5% of the system development process... [23]" This further supports the trends seen previously in Figure 2.2 involving design freedom, knowledge available, and the committed costs.

As previously mentioned, H.H. Koelle, one of the space program planning elite, defined four constraints on the design space of all feasible space programs [21]:

1. The laws of physics;
2. The technological state-of-the-art;
3. The resources available;
4. The organizational[11] will.

These four constraints were depicted by Morgenthaler, another of the elite planners, as shown in Figure 2.13. The first constraint, the *laws*

[11] In the past, even just 20 years ago, there would be no discussion of an organization capable of conducting their own program, hence Koelle originally defined the boundary as "national will." Present companies like SpaceX have their own programs, with separate goals and priorities (albeit still with heavy interaction the national program, for now).

of physics, is very self-explanatory. A planner should not be planning a program that breaks the laws of physics to meet its goals, but rather must plan for the physically possible such as overcoming the force of Earth's gravity and adhering to the fundamentals of orbital mechanics in transit to extra-terrestrial destinations.

The second constraint, the *technological state-of-the-art*, expresses the current industrial capability in areas such as propulsion and structures [78]. As an example, there may be ideal propellants that could exponentially increase the performance of rocket propulsion.[12] But with the current industry capability there may be no way of producing such a rocket. Therefore, a planner must realistically account for the technology available in the target timeline of the space program.

[12] For example, recent efforts in pursuit of a steady state of metallic hydrogen [79] have led to suggestions of using it as a propellant. Relying too heavily on technology breakthroughs can quickly result in the never ending search for *unobtainium* and could hurt the overall program [80].

The third constraint, the *resources available*, readily applies to the program's monetary budget, but also to other aspects like manpower, materials, and time. Planning a large number of missions required to be performed in sequence and in a short amount of time might not even be possible due to the long mission times or specific launch window intervals associated with many extraterrestrial destinations. A planner must account for all of these types of resources.

Finally, the last constraint is the *organizational will*. This constraint has traditionally been the least understood and thus is depicted by a curved inward line in Figure 2.13. This represents the *hazy*, difficult-to-define nature of this constraint in most efforts. Morgenthaler, even in the midst of the Space Race, considered this *organizational will* boundary the haziest of the four [21]. Now, 60 years later, it is still the least understood of the boundaries.

The problem of these four constraints is how intimately they are all intertwined. M. Maier and E. Rechtin inadvertently echo this coupling of Koelle's constraints with a single heuristic in their book *The Art of Systems Architecting* [81]. They state that "Politics, not technology, sets the limits of what technology is allowed to achieve [81]." They argue that politics (*organizational will*) are required in order to achieve funding (*resources*), and that funding in turn in the ultimate constraint on what to what technology and engineering can achieve (*state-of-the-art* and *physics*).

When it comes to planning a space program, the program's feasibility is the principal concern. Each of the boundaries introduced above should be addressed in a space program plan. For now, only the technical aspects (laws of physics and the technology state-of-the-art) of a reviewed program[13] have been accounted for in order to fit within the framework of the previously initiated mission architecture survey introduced earlier. A discussion of the completed survey is included in the following section.

[13] This results in only a partial review of the factors involved but does allow more consistent comparisons with the previously surveyed mission architectures.

2.3.2 *Completed survey of mission architectures and space program plans*

The completed survey of mission architectures and the larger scope program plans is shown in Figure 2.14. The figure contains two plots, both with the architecture/plan date published (time) on the x-axis. The top plot contains a data point for each architecture or plan according to the calculated score for its "ability to convince" by applying the previous metric defined in Table 2.3: a higher score represents a higher quality, more convincing plan. The size of the data point corresponds to the number of supporting sources for that study.[14] The lower subplot is a histogram of the number of studies surveyed based on the first year they were published.

[14] The smallest data points represent studies with a single source and the largest data point represents a study with 12 supporting sources.

The original hope with this survey was observe an prevailing downward trend in performance of studies over time from the "glory days" of the Space Race to the current period of stagnation (at least in manned exploration). Figure 2.14 does not show such a trend. There are certainly many more poor-performing efforts in recent years, but there are still quality efforts observed as well. The proliferation of these inadequate plans could come from a number of reasons, but it likely comes from two key reasons. First, the ease of access to recent articles, studies and plans has become much easier in recent years. It can be seen that there were still inadequate plans proposed in the the 1960s, and there are undoubtedly many more that have been lost or are just much more difficult to track down. The second possibility is that recently, when it appears that the U.S. space program is lacking direction, everybody is offering up ideas to "fix" it. This influx of plans could also explain some of the poor-performance, as a majority of the studies seem to only be concerned with a single element of a proposed mission or program. This could mean that disciplinary experts are adopting the "architecture" terminology and proposing single-discipline solutions based on their area of expertise. Thus, they do not really fulfill the requirements of an architecture defined within this research. The importance of properly defining this growing buzzword is discussed later in this chapter.

A secondary hope of the survey was to visualize a trend in how well the mission architectures compared vs. the space program plans. It would be assumed that, even when only looking at the technical aspects involved, space program plans would have superior coverage of all of the elements involved by the very nature of their large scope and need to compare alternative elements. Unfortunately, there was no visible distinction,[15] and thus no alternative markers were used to denote the two scopes separately in the already crowded Figure 2.14.

[15] This is due either to an under-representation of program-level plans or possibly the previously discussed ambiguously adopted terminology.

So if there is no overall decrease in the ability to convince of techni-

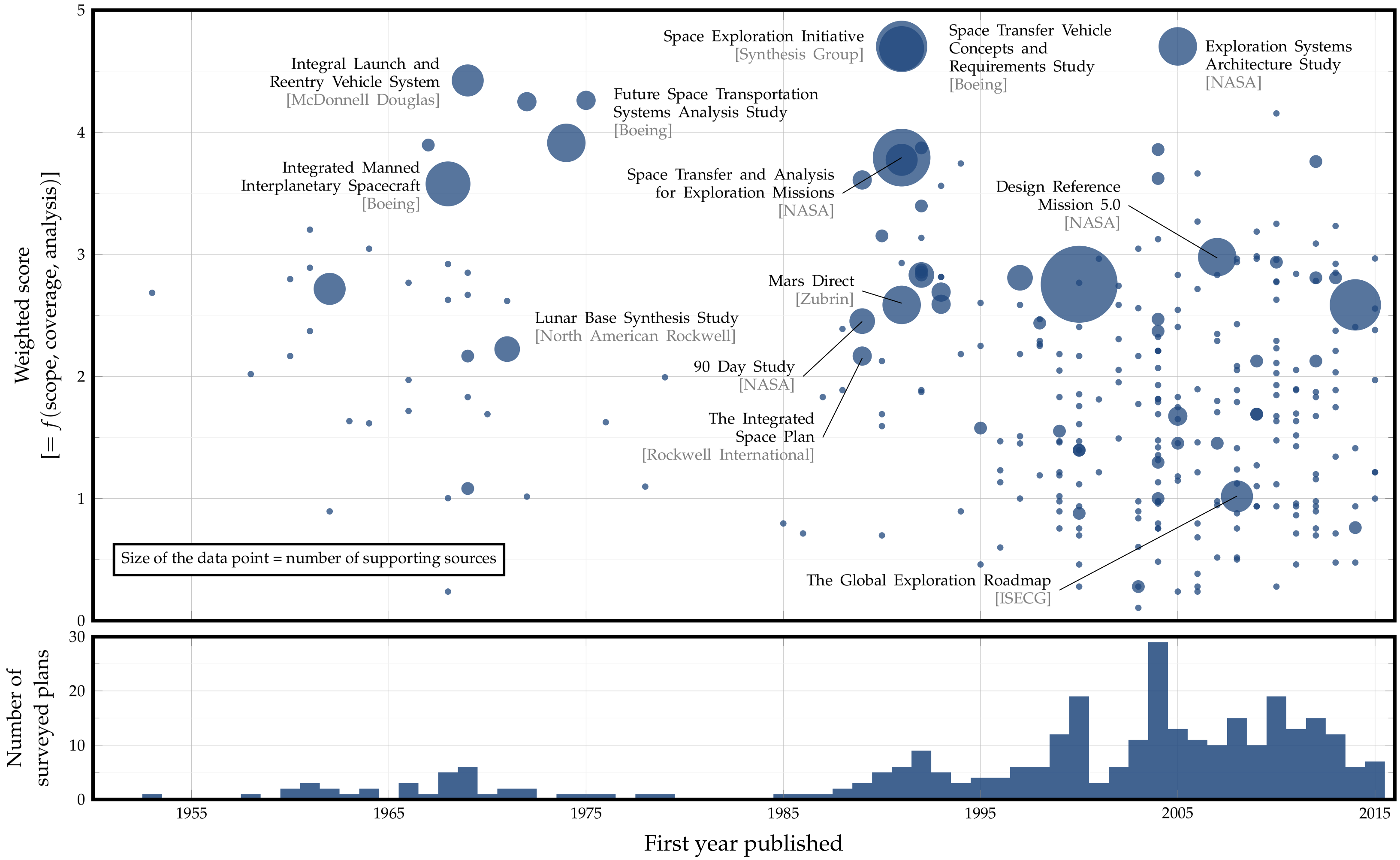

FIGURE 2.14 –
Surveyed mission architectures and program plans addressing the physics and technical boundaries.

cal feasibility of either mission architectures or space programs, which would provide an opportunity for a worthy and original research contribution, what should be the main takeaway of this survey?

All of the initial impressions from only the mission architectures given in Section 2.2.2 still hold true: there is no shortage of ideas, but, due the rarity that any of these plans are pursued, there is obviously an inadequacy of means to compare these ideas and select the "best" one. Program-level plans do not have it figured out any more than designed mission architectures and also should be considered the worse offenders; program plans should, per definition, contain the top-level alternative combination of missions and hardware.

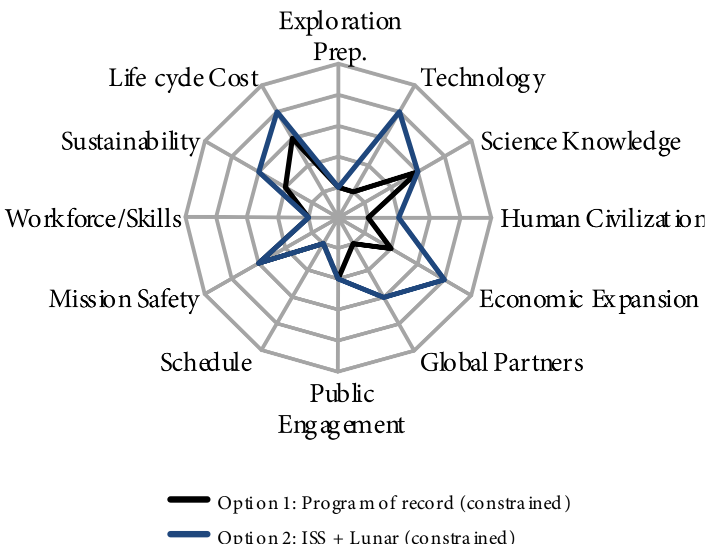

FIGURE 2.15 – A qualitative comparison of two competing programs. Reproduced from Augustine [82].

Some of the better plans did in fact compare alternative programs, but typically only qualitatively when it came to non-technical factors. Figure 2.15 illustrates an example of such a comparison. Unfortunately, even qualitative comparisons between entire programs such as this are rare in the program plans surveyed.

With a completed survey of all of the existing *plans*, it is time to look at all the available *planning* processes; *i.e.*, *how* planning is being done. Stenier describes the difference between these two:

> While the words are similar and interrelated, there is a fundamental difference between planning and plans. Planning is a basic organic function of management. It is a mental process of thinking through what is desired and how it will be achieved. A plan, noted J. O. McKinsey (1932, p. 9) many years ago, is 'the tangible evidence of the thinking on the management.' It results from planning. Plans are commitments to specific courses of action growing out of the mental process of planning. [73]

So far, with the survey of previous mission architectures and program plans in Figure 2.14, much of the "tangible evidence" has been found wanting. A review of space planning processes must be conducted to determine where the fault lies: are planners using appropriate methods to formulate their plans, but failing to communicate in a convincing format? Or are planners left stumbling in the dark, with no proper established parametric process to plan the future of the desired space program? D. Joy and F. Schnebly suggest that non-quantitative approaches[16] provide:

[16] ...like those used by a majority of recent efforts, see Figure 2.14 and 2.15

> ...a less than optimum return on the investment, which results from diffusion of the national space effort in as many directions as there are space scientists and inventors in critical decision making positions. [83]

Essentially, if the case becomes qualitative, there may never be a consensus on a preferred direction, and it is very likely that any selected direction will not be the ideal. The following two sections first discuss a final framework of definitions to clarify this scope and also provide an additional means of screening in the next section: a review of space planning processes is found in the second section.

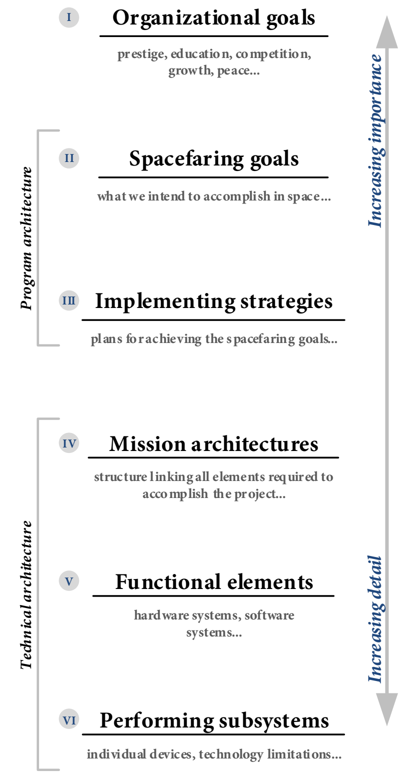

FIGURE 2.16 – The space exploration hierarchy. Adapted from Sherwood [84].

2.3.3 *The space exploration hierarchy*

The broad range of scopes involved in planning has been best classified by B. Sherwood, currently at the Jet Propulsion Laboratory of NASA, in *Programmatic Hierarchies for Space Exploration* [84]. This *space exploration hierarchy* provides a framework for defining and classifying the broad scope of a organization's goals, all the way down to the focused scope of individual subsystems and technologies. The current six levels of the hierarchy are illustrated in Figure 2.16.

This framework is defined here and applied throughout the remainder of this research. The rest of this section expands upon the six levels defined by Sherwood as shown in Figure 2.16 with an emphasis placed on the higher levels that are less familiar to the reader at this point. As previously mentioned, they have all been thoroughly explained by Sherwood so only brief examples, modifications, and supporting sources are provided for each of the six tiers.

» Tier I – Organizational goals

The top-level of the hierarchy, Sherwood defines *national goals* as "...agendas for improving the opportunity and quality of life available to Americans [84]." As was done previously with Koelle's constraints,[17] in an effort to make the hierarchy more generic and applicable today, these *national goals* have been renamed to *organizational goals*, and these goals seek to improve the opportunity and quality of life for the organizations constituents. An organization can represent any nation, international team, company, *etc.* Since this is a space exploration

[17] See Section 2.3.1.

related framework, only those organizational goals that can readily be accomplished through a space program are proposed by Sherwood. Sherwood lists five categories of organizational goals, given in their adapted form below:

» Organizational spirit;

» Education;

» Organizational competitiveness;

» Economic seeding and growth;

» Visibility for peace.

These organizational goals are the purview of top-level decision-makers; those who may not care at all about any of the details of a space program, only the outcomes. An organization might not care to spend the large sums of money required for a lunar exploration mission, but if one could quantify the program's effects on the organization's competitive standing, or the inspired populace that could lead to improved conditions in the future, the decision-makers may certainly begin to entertain the thought.

From here, Sherwood groups the next two tiers, *spacefaring goals* and *implementation strategies*, into what he terms the *program architecture*. The remaining levels, *mission architectures*, *functional elements*, and *performing subsystems*, are grouped into the *technical architecture*. It can seem confusing, having multiple tiers and groups referred to as different types of 'architectures,' but it can help clarify some of the ambiguity of the term when it is used today. Sherwood says that

> Careful definition of the word 'architecture' can help us avoid dabbling in strategy issues when our intention is to synthesis mission architectures. Conversely, it can help us avoid getting wrapped up in hardware and mission profile details when our intention is to synthesize strategies. [84]

» Tier II – Spacefaring goals

The second tier, the first entry in the *program architecture*, is defined as the *spacefaring goals*. According to Sherwood:

> Spacefaring goals are specific, purposeful spacefaring activities which meet objectives of an evolutionary National Space Policy, as those objectives become elaborated by political leaders over time. The spacefaring goals concisely specify exactly what we intend to accomplish in space, clarifying the meaning of that action in historical context, and justifying the undertaking for humankind and America. [84]

Category	Purpose	Example(s)
Science	Understand the Earth	Orbital stations and platforms
	Understand the solar system	Robotic and human probes
	Understand the universe	Orbital and lunar telescopes
	Understand the human species	Permanent human presence
Pragmatism	Develop cis-lunar space commercially	Comsats, prospecting, tourism
	Drive high technology	Extremely challenging tasks
	Sustain Earth	Supply space-energy to Earth
	Build a solar system economy	Recover asteroidal resources, industrialize the Moon
Destiny	Explore	Send people to new places
	Establish viable offworld populations	Settle Mars, establish colonies in orbit

TABLE 2.7 – Three broad categories of spacefaring goals. Adapted from Sherwood [84].

V. van Dyke, author of *Pride and Power - The Rationale of the Space Program* [85], stresses the importance of explicitly defining these goals when he says that "When people do not know what matters to them—what their goals are—failures and frustrations are not surprising [85]." J. Vedda, a proponent of the capability-driven approach to a space exploration program [86, 87], would agree with Sherwood when he calls for goals of capabilities and types of activities as opposed to destination-driven goals. Concerning these destination targets, Sherwood says that:

> If treated as goals themselves, an implicit vagueness of human purpose invites predictable and important questions about why we should want to 'go there,' what we hope to accomplish, and how we hope to gain by having done it. [84]

Sherwood lists some categories of these *spacefaring goals* that can be seen in Table 2.7 along with their purpose and some example activities.

These goals are echoed throughout the history of the U.S. space program. President Eisenhower's 1958 Science Advisory Committee, chaired by Dr. James Killian Jr. described four factors that represented their identified spacefaring goals:

> The first of these factors is the compelling urge of man to explore and to discover, the thrust of curiosity that leads men to try to go where no one has gone before. Most of the surface of the earth has now been explored and men now turn to the exploration of outer space as their next objective.
>
> Second, there is the defense objective for the development of space technology. We wish to be sure that space is not used to endanger our security. If space is to be used for military purposes, we must be prepared to use space to defend ourselves.
>
> Third, there is the factor of national prestige. To be strong and bold in space technology will enhance the prestige of the United States among

> the peoples of the world and create added confidence in our scientific, technological, industrial, and military strength.
>
> Fourth, space technology affords new opportunities for scientific observation and experiment which will add to our knowledge and understanding of the earth, the solar system, and the universe. [85]

The inherent ambiguity in the general terminology of "goals" and "objectives" has led to some overlap when reviewing previous efforts. The committee's identified factors mostly reside in this tier of *spacefaring goals*, though a few can be listed in the tier above with the *organizational goals*.

Similarly, the National Space Act of 1958, the act that created NASA as an organization, lists the following spacefaring goals:

> The aeronautical and space activities of the United States shall be conducted so as to contribute materially to one or more of the following objectives:
>
> 1. The expansion of human knowledge of phenomena in the atmosphere and space;
> 2. The improvement of the usefulness, performance, speed, safety, and efficiency of aeronautical and space vehicles;
> 3. The development and operation of vehicles capable of carrying instruments, equipment, supplies and living organisms through space;
> 4. The establishment of long-range studies of the potential benefits to be gained from, the opportunities for, and the problems involved in the utilization of aeronautical and space activities for peaceful and scientific purposes;
> 5. The preservation of the role of the United States as a leader in aeronautical and space science and technology and in the application thereof to the conduct of peaceful activities within and outside the atmosphere;
> 6. The making available to agencies directly concerned with national defenses of discoveries that have military value or significance, and the furnishing by such agencies, to the civilian agency established to direct and control nonmilitary aeronautical and space activities, of information as to discoveries which have value or significance to that agency;
> 7. Cooperation by the United States with other nations and groups of nations in work done pursuant to this Act and in the peaceful application of the results, thereof; and
> 8. The most effective utilization of the scientific and engineering resources of the United States, with close cooperation among all interested agencies of the United States in order to avoid unnecessary duplication of effort, facilities, and equipment. [88]

These space objectives (read: *spacefaring goals*) have been amended in recent years but are still part of the overarching goals of NASA to this day [89]. These specified *spacefaring goals* can largely be fit into the

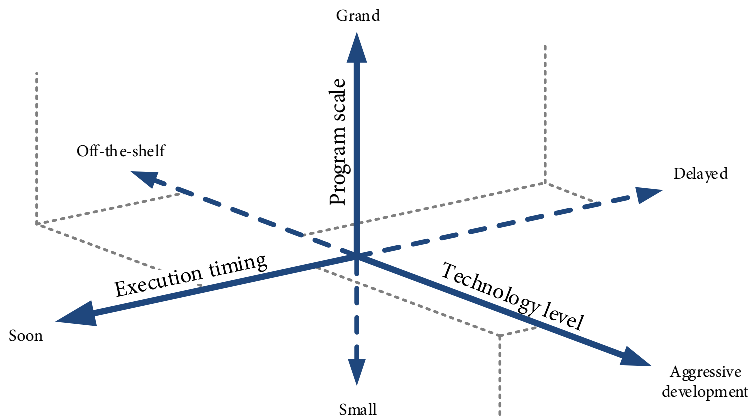

FIGURE 2.17 – Three example independent implementation strategy scales. Reproduced from Sherwood [84].

three primary categories outlined by Sherwood: science, pragmatism, and destiny.

For the sake of the solution concept introduced in the next chapter, these *spacefaring goals* are left as the broad, overarching direction for the program and *space program objectives* are introduced to account for the specific activities that a space program might wish to undertake.

» Tier III – Implementation strategies

Sherwood defines this third tier as *implementation strategies* and it represents the second entry in the *program architecture*. He says:

> Implementing strategies are plans for achieving spacefaring goals. By addressing the relevant opportunities, constraints, values and motives, a complete strategy expresses judgments of how best to navigate through the domain of possible actions. [84]

Twiss agrees and says "...strategy is the path by which the objectives are to be achieved [61]." Steiner, when discussing business strategy, states:

> Strategic planning is the process of determining the major objectives of an organization and the policies and strategies that will govern the acquisition, use, and disposition of resources to achieve those objectives...
>
> Policies are broad guides to action, and strategies are the means to deploy resources. [73]

This aligns perfectly with this space exploration hierarchy. In this case, the organizational and spacefaring goals represent the policies, these 'broad guides to action.' The implementation strategies in turn, are the means to deploy resources, the means to accomplish the the desired goals.

Hammond agrees and suggests that the U.S. program needs to discuss and compare available strategies in pursuit of what he deems to be the ultimate spacefaring goal:

> To maximize the effectiveness of limited resources, the United States needs a national strategy to develop the space frontier...
>
> What is needed now is community discussion of strategic options for space development and a consensus that TI (terrestrial independence) or a similar strategy, is the wisest approach to long-term infrastructure development, if we are to become a space-faring nation and eventually, a solar civilization. [23]

Vedda states that "Principles and goals should be designed to endure, while the strategies and programs supporting them should be allowed to evolve [87]," arguing for fairly constant, overarching space-faring goals with flexible strategies specifying how these capabilities are obtained.

Sherwood identifies three example implementation strategy parameters: program scale, execution timing, and technology level. These three parameters are shown in Figure 2.17. Steiner suggests additional dimensions that could be considered for alternative strategies such as complexity (simple vs complex), coverage (comprehensive vs narrow), and flexibility (readily adaptable or rigid) [73]. Sherwood suggests that any tradable strategy parameters should be mutually independent to first order. This decoupling could lead to comparison sweeps of alternate strategies to better understand their effect on a program.

A final possible implementation strategy to discuss is the level of inter-organizational[18] cooperation. Possibly the most divisive parameter introduced here, many have debated the relative merits and risks of incorporating cooperation. Hammond identifies cost reduction, risk reduction, aggregating resources, and promoting foreign policy objectives and some of the benefits of a strategy with a high level of cooperation. Unfortunately, this strategy could also carry a decrease in flexibility, increased management complexity, and decreased autonomy [23].

[18] Traditionally, this is referred to as *international* cooperation.

At the beginning of the Space Race, refusing to cooperate is what propelled both the U.S. and U.S.S.R. forward. Dr. Sedov was a Soviet delegate at a conference during the Space Race. Another delegate, Dr. Dryden of the U.S., commented to him that it was too bad that the U.S. and U.S.S.R. were competing in space rather than cooperating. "Dr. Sedov is said to have responded that the scientists should be thankful for the competition, for otherwise neither country would have a manned space flight program [85]."

» Tier IV – Mission architecture

For this tier Sherwood uses a term that should be very familiar to the reader and whose elements have previously been defined in Table 2.2. The *mission architecture* is the first tier in the *technical architecture* grouping defined by Sherwood. He defines it as:

> the structure linking all the elements required to accomplish the project: the mission profiles and the operations scenarios, including all hardware and software. [84]

This is perfectly in line with what the previous definitions given by Larson and Wertz in Table 2.2 and Koelle and Voss in Table 2.6.

These final three levels, the *technical architecture*, have been the primary focus of the AVD Laboratory. AVD has historically dealt with Tier V of the hierarchy: the *functional elements* [4, 13, 17]. This tier, from an early-decision making point of view, involves the conceptual design, the sizing, of the hardware involved in a given mission. Most recent efforts have taken the members of the lab into the Tier VI, *performing subsystems*, as they forecast the effect that different technologies might have on hardware (an aircraft) in an effort to advise the decision-makers in their investment decisions [90].

» Tier V – Functional elements

The next tier, also a part of the *technical architecture*, contains the *functional elements*. These are defined by Sherwood as "integrated, procurable end-items: hardware systems, software systems, and operations plans [84]." This is the domain of traditional conceptual design: sizing the hardware required to complete the given mission. Most of the individual elements listed in Table 2.2 belong in this tier.

» Tier VI – Performing subsystems

The final tier of the hierarchy and the last tier included in the *technical architecture* includes the *performing subsystems*. According to Sherwood:

> Performing subsystems are the individual devices and computational codes which will execute specific, well-bounded purposes in SEI.[19] The are the 'widgets' and 'programs' which actually 'do things.' [84]

[19] Sherwood was writing in the context of the beginnings of the Space Exploration Initiative (SEI), thus all of his writing points to accomplishing its goals.

Performing subsystems include the fundamental technologies involved with each *functional element* in the tier above. Properly identifying and understanding these driving technologies and their possible impacts is vital to the success of any long-range planning effort. Figure 2.18

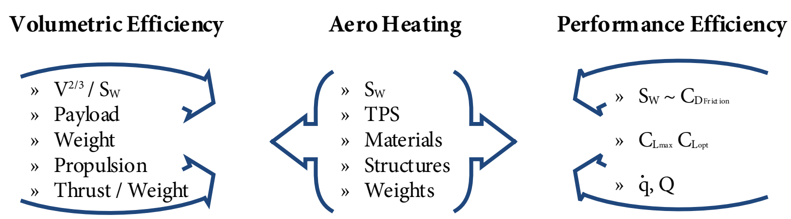

FIGURE 2.18 – Primary variables involved in hypersonic flight and their connections to key technologies. Reproduced from Draper [91].

is an example of some key technology areas and their connections to the primary variables involved in the hypersonic domain, as has been under investigation by the U.S. Air Force. Similar connections can be made on the technologies relevant to space flight.

The ideal, parametric application of Sherwood's hierarchy would result in the depiction shown in Figure 2.19. This concept seems to be something that Sherwood is building towards, though he makes no mention of a means to connect and without this parametric foundation it is difficult to imagine the realization of any practical, integrated system from the framework alone. Imagine, decision-makers at the highest levels identifying and numerically working with the primary driving variables involved with their respective tier. Their tier is parametrically connected with each of the tiers below it, enabling truly consistent comparisons of alternative goals and their effect on the program, as well as quantifying a given technology's ability to help meet stated overarching goals. Figure 2.19 depicts a consistent number of

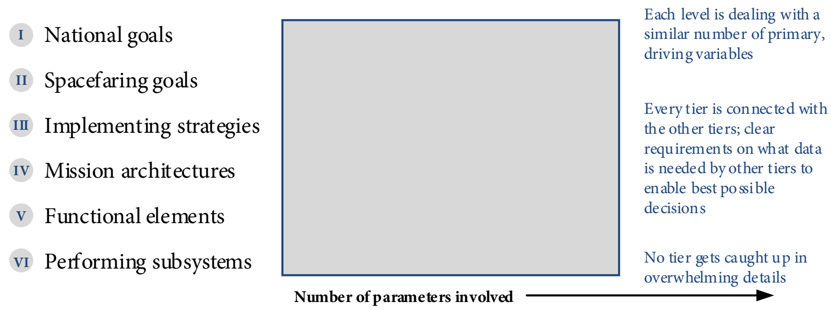

FIGURE 2.19 – The ideal parametric integration of all six tiers of Sherwood's space exploration hierarchy.

parameters at each tier, completely connected with the adjacent tiers. The novelty of such a concept will be made clear in the next section when a review of existing space planning processes reveal how typical process look when viewed with a similar lens.[20]

[20] See Figure 2.23.

2.3.4 *A review of space planning processes*

With the framework of the space exploration hierarchy firmly in place, this section discusses the review of space program planning processes. According to Maier, "If you don't understand the existing systems, you can't be sure that you are building a better one [81]." The objectives of this planning process review are as follows:

» Accelerate the author's understanding of the common approaches used in space program planning and at each tier of the exploration hierarchy;

» Compare the representative processes in an effort to understand their contributions and shortcomings;

» Identify any trends or gaps in coverage of existing space planning processes;

» Identify methods and portions of processes that can be applied to the future solution concept planning system.

This review also contributes towards a working document, a formal Process Library, that can be found in its current state in Appendix B.

The review of planning processes was internally documented in an Excel spreadsheet, serving as a minimal database for the needs of the review. Many attributes were recorded there for each reviewed process: title, author, first year published, a key quote, *etc*. All of the processes reviewed are required to have at least one accessible supporting source as well.

The list found in Table 2.8 represents a thorough literature review of applicable space planning processes. The list given below does not represent every space planning process and emphasis was placed on identifying the unique/milestone processes throughout history. For example, once several similar processes had been included that solely addressed Tiers IV and V of the hierarchy, any additional similar processes were omitted from the review. It is the contention of the author that the planning processes presented here are representative of the entirety of planning processes available and provides an accurate depiction of the "best practice" and typical coverage of processes developed since the 1920s. Noteworthy processes will be selected and subjected to further decomposition and analysis in the remainder of this document.

TABLE 2.8 –
Complete list of reviewed space planning processes.

ID	Year	Primary author(s)	Process name	Source(s)
1	1923	Oberth	The Rocket into Planetary Space	[92]
2	1953	von Braun	The Mars Project	[27, 93]
3	1959	Dergarabedian	Estimating Performance Capabilities of Boost Rockets	[94]
4	1961	Wolverton	Flight Performance Handbook for Orbital Operations	[95]
5	1961	Koelle - *NASA*	Long Range Planning for Space Transportation Systems	[96]
6	1961	Koelle	Mission Velocity Requirements and Vehicle Characteristics	[97]
7	1962	Joy - *Lockheed*	A Comprehensive Analytical Basis for Long-Range Planning Decisions in Future Manned Space and Lunar-Base Programs	[83]
8	1962	Jensen - *Martin Marietta*	Design Guide to Orbital Flight	[98]
9	1962	White	Flight Performance Handbook for Powered Flight Operations	[99]
10	1963	Ehricke - *General Dynamics*	Parametric Mission Analysis	[100]
11	1963	*NASA*	Space Flight Handbook	[101–108]
12	1964	Wood	Aerospace Vehicle Design - Volume II: Spacecraft Design	[109]
13	1964	Purser	Manned Spacecraft: Engineering Design and Operation	[110]
14	1964	Laidlaw	Mass Ratio Design Process	[111]
15	1964	Morgenthaler - *Martin Marietta*	Space Technology Analysis and Mission Planning	[21, 112–119]

Continued on next page

TABLE 2.8 – continued from previous page

ID	Year	Primary author(s)	Process name	Source(s)
16	1964	Ehricke - *General Dynamics*	Space Technology Analysis and Mission Planning	[36, 120–123]
17	1965	*NASA*	A Parametric Study of Mass-ratio and Trajectory Factors in Fast Manned Mars Missions	[124]
18	1965	Koelle - *NASA*	Program Analysis and Evaluation Process	[62]
19	1965	*US Air Force*	The Space Planners Guide	[125, 126]
20	1966	Chamberlain - *NASA*	A Space Mission Success Evaluation Model	[63]
21	1967	Morgenthaler - *Martin Marietta*	On the Selection of Unmanned Probes and Launch Vehicles for Exploration of the Solar System	[127]
22	1968	Novosad - *Martin Marietta*	Integration of Expected Extraterrestrial Resources Into the Design of Space Transportation Systems	[128]
23	1969	*McDonnell Douglas*	Optimized Cost/Performance Design Methodology	[129, 130]
24	1970	Chamberlain - *NASA*	A Methodology to Compare Policies for Exploring the Solar System	[131]
25	1971	Heineman - *NASA*	Fundamental Techniques of Weight Estimating and Forecasting for Advanced Manned Spacecraft and Space Stations	[132]
26	1971	*Lockheed*	Probabilistic Systems Modeling and Cost/Performance Methodologies for Optimization of Vehicle Assignment	[133, 134]
27	1972	*Aerospace Corporation*	Aerospace Vehicle Synthesis Program	[135]
28	1973	*NASA*	Launch Vehicle Estimating Factors for Advance Mission Planning	[136]
29	1974	Glatt	WAATS	[137]
30	1979	*NASA*	Mass Estimating Techniques for Earth-to-orbit Transports with Various Configuration Factors and Technologies Applied	[138]
31	1979	*Naval Research Laboratory*	Optimum Launch Vehicle Sizing	[139]
32	1983	*NASA*	Interplanetary Mission Design Handbook	[140–142]
33	1990	Morgenthaler	Optimal Selection of Space Transportation Fleet to Meet Multi-mission Space Program Needs	[143]
34	1991	Griffin	Space Vehicle Design	[144]
35	1992	Forrest - *Aerospace Corporation*	Launch Vehicle Sizing/Performance Analysis	[145]
36	1992	Larson and Wertz	Space Mission Analysis and Design	[24]
37	1992	Wolf -*DLR*	TRANSYS	[146]

Continued on next page

TABLE 2.8 – continued from previous page

ID	Year	Primary author(s)	Process name	Source(s)
38	1994	Heineman - *NASA*	Design Mass Properties II: Mass Estimating and Forecasting for Aerospace Vehicles Based on Historical Data	[147]
39	1997	Greenberg	STARS: The Space Transportation Architecture Risk System	[148]
40	1997	Blandino	Space System Architecture Code	[149]
41	1997	Koelle	Space Transportation Simulation Model (TRASIM 2.0)	[150]
42	1998	Brown	Spacecraft Mission Design	[151]
43	2000	Larson and Pranke	Human Spaceflight - Mission Analysis and Design	[152]
44	2001	Hammond	Design Methodologies for Space Transportation Systems	[22]
45	2002	Weigel	Bringing Policy into Space Systems Conceptual Design: Qualitative and Quantitative Methods	[153]
46	2002	Kim	Conceptual Space Systems Design Using Meta-heuristic Algorithms	[154]
47	2002	Brown	Elements of Spacecraft Design	[20]
48	2009	Holt - *NASA*	Propellant Mass Fraction Calculation Methodology for Launch Vehicles and Application to Ares Vehicles	[155]
49	2010	Chudoba - *The AVD lab*	AVD$^{\text{sizing}}$	[4, 13, 156]
50	2011	Wertz	The New SMAD	[25]
51	2012	Alber	Aerospace Engineering on the Back of an Envelope	[157]
52	2012	Aguirre	Introduction to Space Systems: Design and Synthesis	[158]
53	2012	Spagnulo	Space Program Management: Methods and Tools	[77]
54	2014	Arney	Modeling Space System Architectures with Graph Theory	[159]
55	2014	Blythe - *NASA*	NASA Space Flight Program and Project Management Handbook	[76]
56	2015	*NASA*	Cost Estimating Handbook - Version 4.0	[160]
57	2015	Sforza	Manned Spacecraft Design Principles	[161]

Using the identified tiers of the space exploration hierarchy, Tiers I–VI, the entire list of processes were screened and classified according to the tiers addressed by each one. The results of this screening can be seen in Figure 2.20. In this figure, the six tiers of Sherwood's hierarchy are listed on the left with individual bars drawn along the *x*-axis representing each reviewed process. The Process ID number located at the base of each bar can be traced to the entries in Table 2.8. The solid fill on the bar, as opposed to the cross-hatched, indicates which processes have been selected for further investigation. These noteworthy

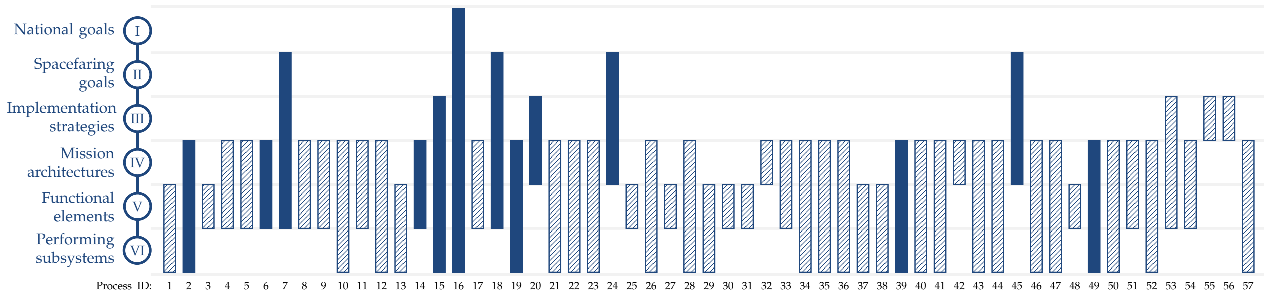

FIGURE 2.20 – A visualization of the program scope addressed by all the reviewed processes.

processes will be discussed later and can also be found in the Process Library in Appendix B.

Of the 57 processes reviewed, it can easily be seen that the top three tiers are very under-represented, even after a directed search on trying to find processes addressing these tiers specifically. If all the omitted processes that dealt only with Tier IV or Tier V were included as well, the discrepancy in representation would be even more extreme. This means that there was rarely any quantification involved in the decisions made at these tiers. In fact, only a single process was found that incorporated the top tier, the *organizational goals* into the planning process.[21]

There is an observable line, a ceiling, that a majority of the processes do not cross: those processes that analyze the *technical architecture* (Tiers IV–VI) do not often venture into the *program architecture* (Tiers II and III). The few that do have all been selected for further investigation to try and better understand their approach.

Several of the more recent efforts require additional explanation. Both of the processes by NASA, the *NASA Space Flight Program and Project Management Handbook* [76] and the *Cost Estimating Handbook* [160], have been written in a way so that they are generic enough to likely be applied to a wide range of tiers. Unfortunately, the methods contained in these sources, in that effort to remain generic, do not provide specific parameters or enough details to be directly applicable. It was the author's opinion that both these efforts best embodied implementation strategies, Tier III, and serve to provide some insight into the inner workings of how NASA works as an organization. They appear to serve more as a guide to an overall mentality, standardizing and integrating all phases of a program's development, and not necessarily as a process with specific inputs and outputs that could be used to plan out and compare future program alternatives. They are not approached in such a way that a N-S representation could readily be constructed from their contents. Additionally, the NASA cost model was included simply because it was one of the latest efforts

[21] This really should not be a surprise. As will be seen, it is rare enough to try and include even just the *spacefaring goals* or *implementation strategies* in the planning process, much less include them along with even more abstract concerns at a the highest level. There are undoubtedly processes out there that focus solely on such a feat, but without a focus on any more details of the space program, they would not be included in such a review as this.

available from NASA, an organization with plenty of space program experience. All other cost-only models were excluded from this review. D.E. Koelle stresses the early integration of cost during planning and development. He says:

> It is important – and this is the distinct different to the past methodology – to start cost analysis at the very beginning of a vehicle design process, and NOT after a detailed design has been established. The usual 'bottom-up' cost estimation with detailed costing of each component and each activity is expensive and time consuming. It also may lead to a cost result which is not acceptable – and the complete process must start again. [162]

Thus, while there are certainly many cost-only processes for the space domain, this review is looking for those that integrate cost into the process as a whole.

» Summary cards of the Process Library

The selected noteworthy processes have been compiled into one-sheet summary cards that contain additional information of the process. The compiled cards represent the beginnings of a reference of space program Process Library.

The process title, published date and type of source are listed first, followed by a quote from the source of its goals and intentions, if available. The levels of the hierarchy addressed are included in a figure in the top-right corner. The primary author and/or organization behind the process are included. A description/summary of the process is provided, along with notable strengths, weaknesses, current status, and the supporting sources. A N-S representation for many of the processes is also included on the summary card, located in the margin on the right. These structograms were constructed verbatim from flow charts within the source process where possible and reverse-engineered if needed.

See Appendix B for the entire collection of summary cards of the completed entries in the Process Library. An example card is included in Figure 2.21, depicting the *Mission Velocity Requirements and Vehicle Characteristics* process from H.H. Koelle and H. Thomae [97].

» Process comparisons

With the selected processes deconstructed, it is now possible to begin drawing comparisons between them. Each process has been reviewed for how well it addresses each of the four constraint's proposed by H. H. Koelle.[22] Figure 2.22 visualizes this comparison.

A simple metric introduced by Coleman [13] in his efforts with aircraft design methodologies has been similarly applied here and is described at the bottom of Figure 2.22. This metric can be seen directly at

[22] These were discussed previously and can be found in Figure 2.13.


Mission Velocity Requirements and Vehicle Characteristics

PUBLISHED DATE: 1961
TYPE: Textbook

> "...the preliminary determination of velocity requirements for individual missions and the characteristic data of the vehicle which is required to carry out the mission of interest" (p. 25–14 [70])

AUTHOR/ORGANIZATION(S):
H. H. Koelle and H. F. Thomae

PROCESS DESCRIPTION:
"The stage specific impulses and the velocity requirement result in an over-all effective mass ratio. Propellant fraction and payload ratio result in an average structural factor. This and the total mass ratio result in the optimum number of stages required, and give a first estimate of the growth factor (take-off weight/payload weight). Combining this with the take-off weight, obtained by dividing the take-off thrust by the take-off acceleration, fives a first estimate of the total weight-carrying capability (weight of instrumentation, guidance and control components, payload containers, and net payload), defined as the dry gross payload.

A preliminary optimization of the propellant loadings of the stages follows, which in turn allows more detailed weight estimates of the subsystems and components. Adding all these weights results in a preliminary vehicle weight breakdown, which then is used for performance check calculations. (p. 25–14)"

STRENGTHS:
Clear identification of the primary drivers in both energy requirements and the launch vehicle's properties. Provides a flow diagram that also demonstrates where the process fits into the larger design process.

WEAKNESSES:
The methods and variables involved use assumed values for other variables and any figures and equations are then based on this assumption. Slightly limits the applicability if the user needed to trade other variables.

STATUS:
The process is still applicable, though no direct follow-up efforts have been observed.

REFERENCE: See [70]

FIGURE 2.16: Koelle and Thomae present a very straightforward approach with many rules-of-thumb and appropriate assumptions given.

FIGURE 2.21 – An example process summary card from the working Process Library on space planning processes.

the bottom of Figure 2.22. Basically, if the constraint is not addressed, the color of the cell will be white. A darker cell represents a more comprehensive coverage of the constraint for the given process.

A couple of observations can be made from Figure 2.22. First, only one of the processes has been seen to fully address all four constraints. The General Dynamics developed *Space Technology Analysis and Mission Planning* (STAMP) process [121–123], a multi-year effort in fulfillment of a NASA contract in the 1960s, scores well on all four of the constraints. Unfortunately, a disconnect between the program and technical architectures limits the potential application of this process. This issue provides an excellent opportunity for an original research contribution. Several portions of the process will no doubt provide guidance and inspiration for the future of this research.

Second, the more common approach taken by a majority of the other processes is to instead focus on only one or two of the constraints. The constraints formed by the *laws of physics* and the *technological state-of-the-art* are often analyzed together like they are in von

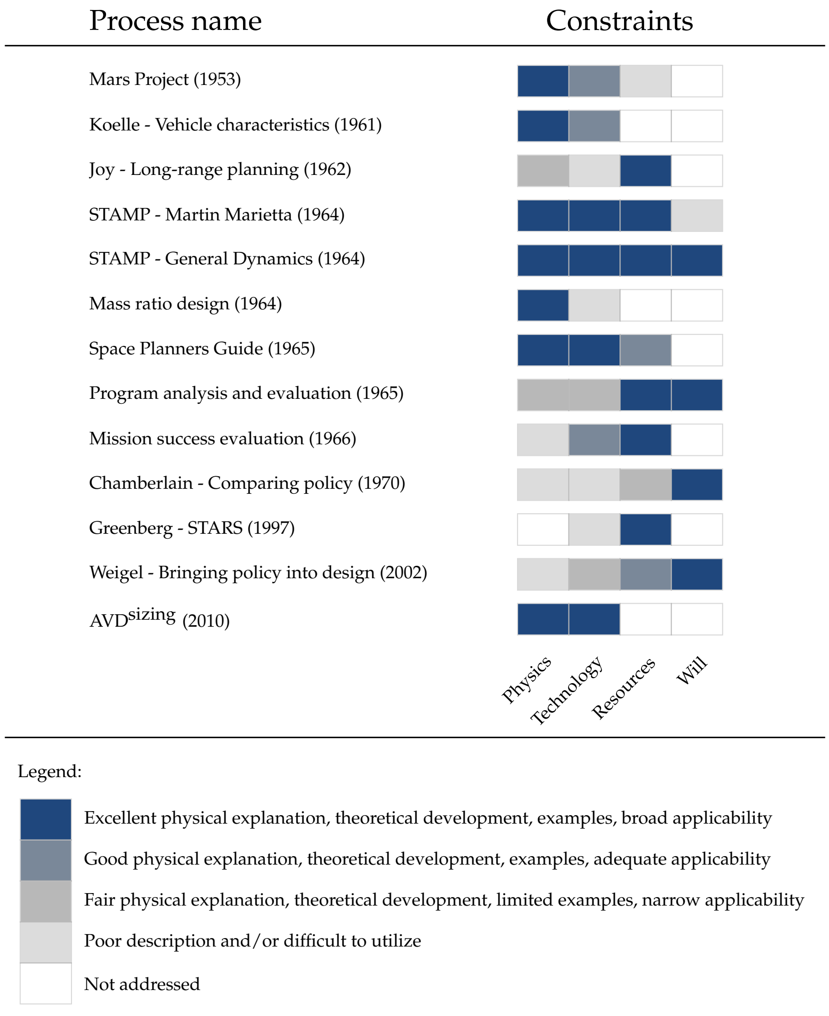

FIGURE 2.22 – A comparison of the identified representative processes and how well they address each of the four *Koelle constraints*.

Braun's Mars Project [27], the Space Planners Guide [125], and the AVD's own current effort, AVD^{sizing} [5]. The *resource* constraint, either cost-estimating or scheduling processes, is usually addressed independently of the others, and is still reasonably well represented considering the intentional omission of purely cost estimating processes. The constraint formed by the *organizational will*, which was previously argued as analogous to the goals and strategies of the program, is the least represented here, but (due in part to the nature of the selection of these particular noteworthy processes) the processes that address that constraint do seem to address it well. Still, even when a quantitative process is used to address this constraint, it is rarely parametrically connected to the other constraints. In the case of several of the other high performers[23] addressing the *organizational will*, it can be seen that

[23] See A. Weigel [153], R. Chamberlain [131], and Koelle and Voss [62].

the other constraints are neglected in favor of this constraint.

Several of the processes apply statistics and probability to evaluate the chances that a given mission or program has to succeed [63, 131, 148]. This is a very valuable contribution, including risk (and more importantly the added cost for any desired risk reduction[24]) into the decision-making process. With this included variability, these processes simulate a large number of programs in the hopes of extracting trends from the accumulated output.

[24] Some have argued that the pursuit of safety at any costs is they primary reason for the decline in manned space activities [163].

Recall the ideal parametric representation of Sherwood's hierarchy was previously depicted in Figure 2.19. The processes in this review tend more towards the parametric representation shown in Figure 2.23

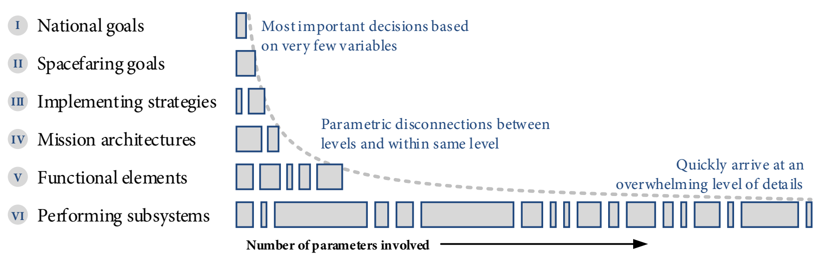

FIGURE 2.23 – The typical parametric breakdown observed in the review of representative space planning processes.

The size of the bar at each tier qualitatively represents the number of parameters involved in the decision process at that tier. The greatest issue with these reviewed processes is depicted in this figure by the vertical separation between the various tiers. This means that the decisions made at the top level, those of the greatest importance and with the largest impact on the rest of the program, are made with inadequate, likely solely qualitative, information, that likely has no connection to the supporting analysis from the tiers below.

At the lower tiers of the hierarchy in Figure 2.23, it can be seen how quickly the number of parameters can grow with hardware and technology details included in the planning process. Such detail typically comes from specialization which leads to a difficulty to consistently compare. This issue is shown in Figure 2.23 by the horizontal breaks in the lower tiers.

Some processes do fare better at balancing the number of parameters involved in the technical architecture,[25] but still fail to identify or incorporate any of the parameters involved in the upper tiers.

[25] The Space Planners Guide [125] properly targets the primary variables only and avoids getting lost in technical details.

Those processes that do attempt to analyze entire space programs often require the entire space program as an input. This can be a significant burden on any user. For example, at the beginning of their analysis, Robertson and Fatianow state that "...it is assumed that the space policy has been formulated and the space objectives have been determined [75]." The space program is then input into their pro-

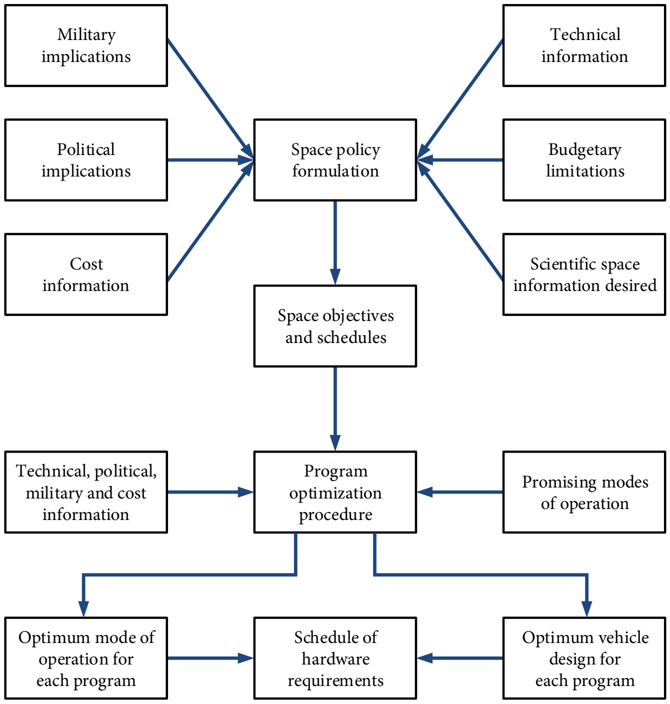

FIGURE 2.24 – Prediction of hardware requirements for space programs. Reproduced from Robertson and Fatianow [75].

cess to be optimized. Figure 2.24 depicts their proposed flow diagram methodology to optimize a space program and determine the required hardware specifications. With no guidance offered on how program objectives should be determined (or what objectives are even available) or how to assemble a program, the user will struggle to produce meaningful results with their process.

» SUMMARY OF RESULTS

This review of planning processes accomplished all of the research goals laid out for it:

» The development of the library has accelerated the author's understanding of the primary drivers involved in the planning of a space program. Common pieces that serve as the "best practice" in addressing particular *Koelle's constraints* or tiers of the space exploration hierarchy have been identified for application in the solution concept proposed in the next chapter.

» An initial comparison of some of the noteworthy processes has been made, revealing an inadequacy in integration within a majority of previous planning efforts. It has also shed light on some unique, legacy processes that could benefit from an update and then be adapted to any modern efforts.

» The initial survey of all the processes revealed a lacking of those pro-

Identified specifications	Reference
Parametric support for the decision-maker in the earliest phases of planning	Section 2.1.5
Capable of integrated, multi-disciplinary correctness	Section 2.1.5
Iteration and convergence, providing a solution space of possible alternatives	Section 2.1.5
Identification and focus on the primary drivers	Section 2.2.4
Capability to consistently compare alternative programs, missions, hardware	Section 2.2.4
Inclusion of both technical and non-technical factors	Section 2.2.4
Integration (both horizontal as well as vertical)	Section 2.3.5
Take variability (reliabilities, scheduling delays, *etc.*) into account	Section 2.3.5

TABLE 2.9 – Complete set of system specifications previously derived in Chapter 2.

cesses that address the upper tiers. Tiers I–III are under-represented and are rarely connected with the lower tiers. Further investigation has identified that even when all the tiers are appropriately addressed, there is no parametric connection between them. This disconnect provides an excellent opportunity for an original research contribution.

» Several key applicable processes have been identified. Portions of the planning that have been thoroughly investigated, such as the hardware sizing, could be used practically in their entirety in a modern decision-support system, leaving the author to develop the understanding and the connections between existing processes.

2.3.5 *Ideal specifications*

This review of space planning processes has identified two additional specifications for a space program planning system to augment the strategic planner:

» Integration (both horizontal as well as vertical);

» Variability: including the probabilities of success, failure, budget fluctuation, *etc.*

2.4 *Compiled list of specifications*

A complete list of the specifications identified in the previous chapter are given in Table 2.9. Recall that these specifications were derived from the 'best practices' from both aircraft design and existing space planning efforts.

2.5 *Chapter summary*

This chapter has detailed the research involved in the development of the specifications for an ideal decision-support methodology to augment the strategic planner of space programs. The literature foundation was in the aircraft design domain, exposure provided from the author's experience working in the AVD Laboratory. A survey of all proposed mission architectures revealed the need to expand the scope beyond just single missions and to think about planning and synthesis at the program level. After the survey of existing plans, the next logical step was to review the tools available to these planners and finalize what capabilities the ideal system should possess, in order to best support the top-level decision-makers.

This chapter contained two original contributions made by this research, summarized below:

» A thorough investigation of over 300 previously proposed mission architectures and space program plans;

» A representative review of available space program planning methodologies.

The next chapter introduces and discusses this proposed solution concept, *Ariadne*.

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Chapter 3

ARIADNE – A PARAMETRIC, VERTICALLY INTEGRATED SPACE PROGRAM PLANNING METHODOLOGY

This chapter contains the methodology for *Ariadne*,[1] a tool to augment the space program strategic planner. First, an ideal solution concept, *Ariadne*, is introduced and briefly discussed along with some of the current 'best practice' approaches that have contributed towards portions of the solution. Due to the large scope of the solution, multiple opportunities are available with this concept for worthy research contributions. These options are compared, a final direction is selected, and a reduced set of specifications are offered.

Then, the prototype solution concept methodology for *Ariadne* is outlined. Each step, along with its inputs and outputs, are defined. Finally, documentation for the supporting methods and the software implementation are provided.

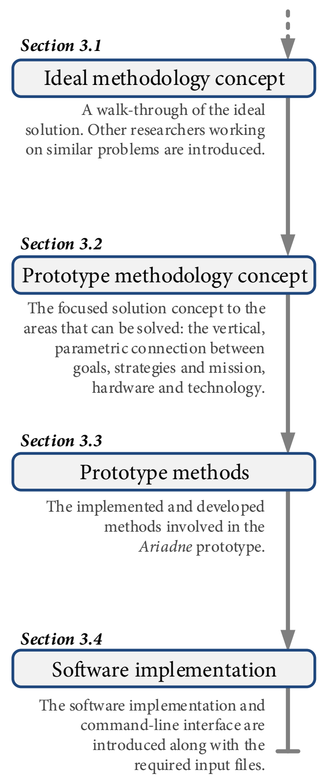


FIGURE 3.1 – Outline of Chapter 3.

3.1 *Ideal methodology concept*

The ideal methodology proposed here begins with the generic design process offered by Torenbeek and has been adapted to fulfill the specifications outlined at the end of the previous chapter.[2] The scope, per definition, is quite large and there are many portions of the solution that have been addressed by other researchers and organizations as was seen in Chapter 2. Most of these are included in the Process Library discussed in Chapter 2 and may be found detailed in Appendix B. These contributors have been identified in order to highlight the missing pieces and thus opportunities for a worthy contribution.

3.1.1 *Walk-through of the ideal solution*

Figure 3.2 is a flow diagram of the proposed ideal *Ariadne* solution concept. It consists of four primary steps: deriving the program objectives, assembling a candidate space program, converging a feasible program, and simulating the space program. These four steps are discussed in turn in the rest of this section. The inputs required of each

[1] The system was named *Ariadne* after a character from the 2010 film, *Inception*. The character plays the role of the *architect*, a planner of the intricate, multi-level, mental heist.

An analogous capability is desired for dealing with the many tiers of the space exploration hierarchy by Sherwood, outlined in the previous chapter.

[2] Another example of applying lessons learned from the aircraft design domain.

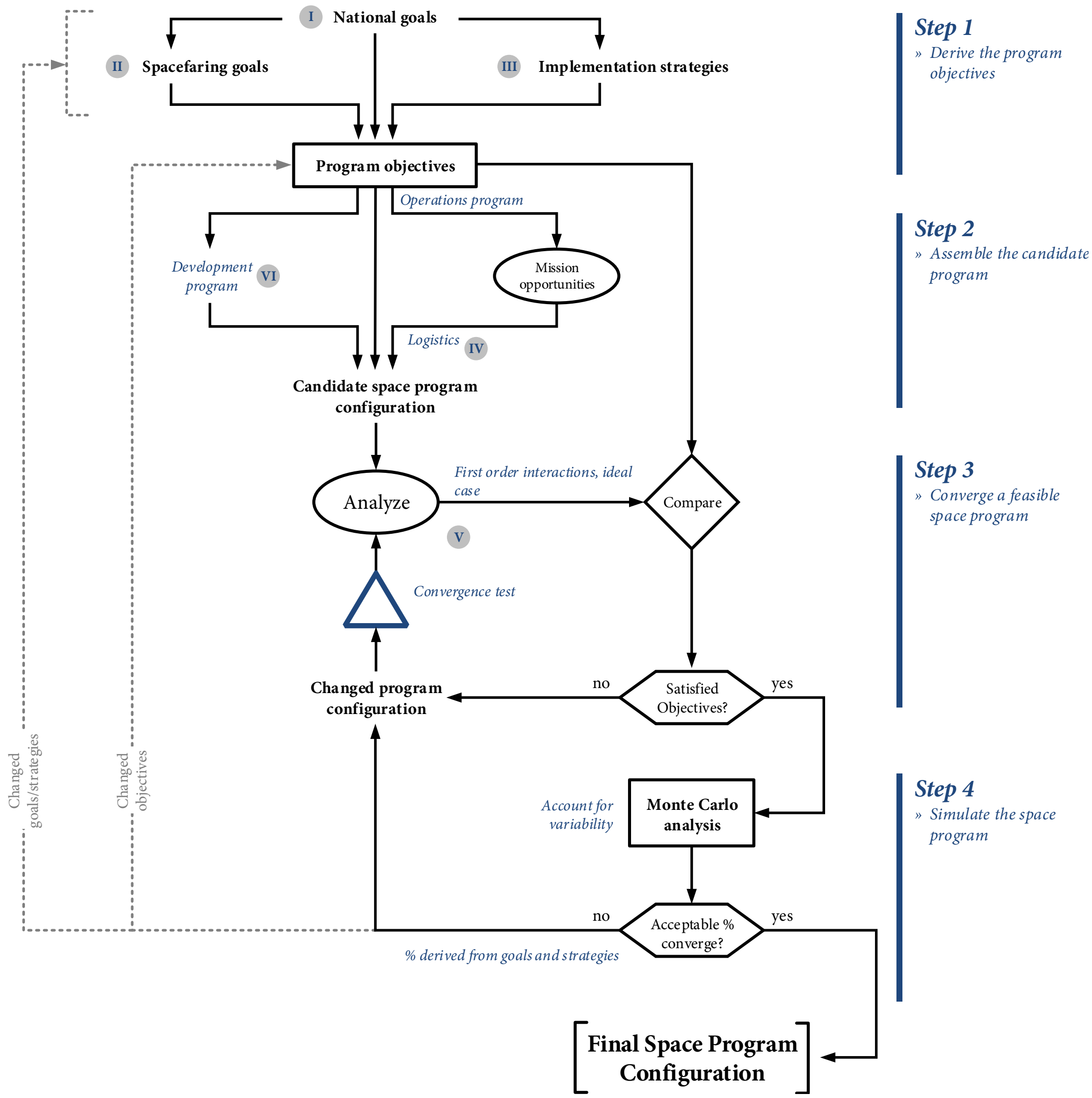


FIGURE 3.2 – A flow diagram of an ideal solution concept. Modeled after the generic design process outlined by Torenbeek [1].

step, a description of the analysis, and the output within the step have been included followed by a brief discussion of existing efforts that are contributing toward the processes involved in the step.

Step 1 » Deriving the program objectives

The first step in the solution concept involves the top three tiers of the previously introduced space exploration hierarchy by Sherwood: national goals, spacefaring goals, and implementation strategies. In order to achieve true integration of the parameters involved at each tier, there must be quantified inputs from these three tiers that are taken into account from the beginning in the form of concrete *space program objectives*.[3]

The objectives are influenced by all three of the top tiers, as seen in

[3] This is a slight departure from Sherwood, who defines *spacefaring goals* as the specific activities. Although they could remain grouped in that tier, the author found the distinction between the broad spacefaring goals and specific space program objectives to be helpful.

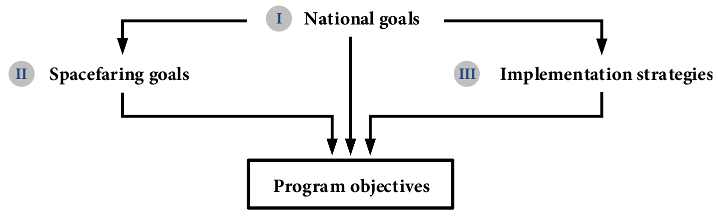


FIGURE 3.3 – The first step in the ideal *Ariadne* solution concept methodology» Derive the program objectives.

Figure 3.3, and provide the specific activities and details that will need to be fulfilled by the missions of the program. For example, while the specified goals and strategies might call for quickly developing LEO commercially, a *space program objective* would define the expected payload requirements and flight rates for a given destination.

The final deliverable from this step is a complete and traceable set of prioritized space program objectives to pursue, a budget constraint, and a schedule constraint.

Many have stressed the importance of these three tiers and their impact on the overall program. Hammond says:

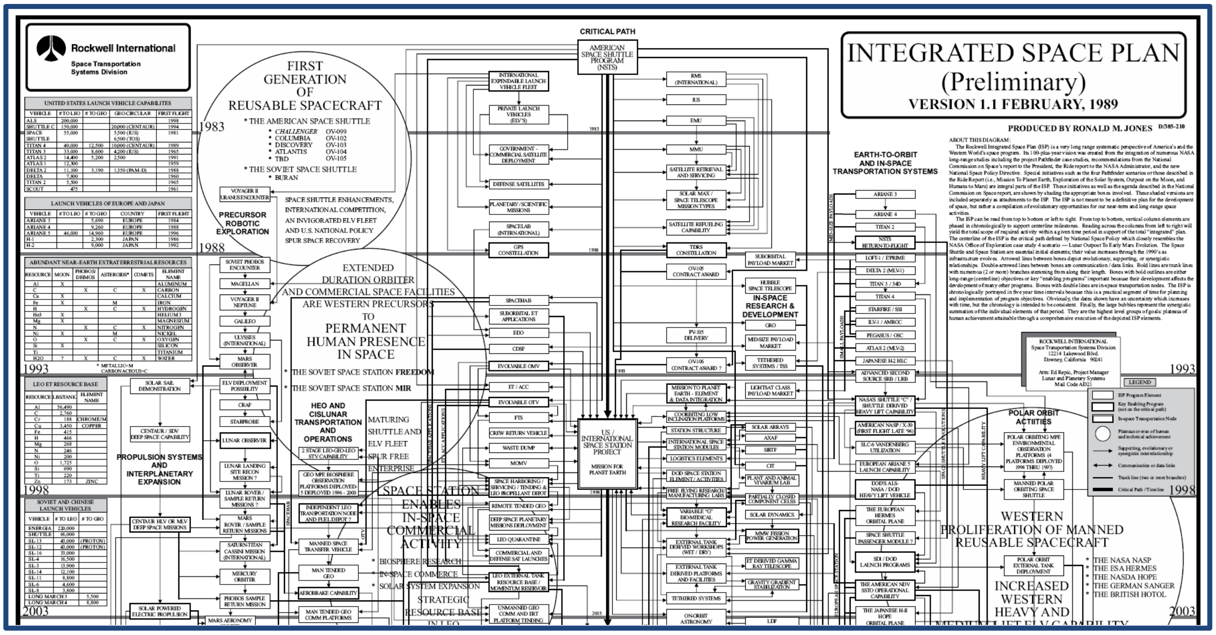
> The future direction of the U.S. Space program needs to be defined and a national space strategy agreed upon and implemented. A healthy U.S. space program depends on setting the right goals. These goals should be specific, must make the costs competitive, and must be long range, while keeping the public interest over many generations. Colonizing the solar system (and, indeed, terraforming Mars) as described in the ISP [Integrated Space Plan] will take many generations and hundreds of years.
>
> Properly set goals must clearly define the ultimate objective. [2]

The Integrated Space Plan (ISP) that he speaks of was the work of Rockwell International in 1989-97. [3, 4] As a road map, the ISP is truly a beautiful visualization, shedding light on a possible progression for mankind in the complex pursuit of becoming a spacefaring civilization. Unfortunately, any processes involved behind the work is hidden from any viewer, and now, years later, when the proposed timelines have shifted and technology has progressed in other, unexpected areas, the Integrated Space Plan is quickly becoming obsolete.

Another group, *Integrated Space Analytics* [5], has recently made efforts to update this roadmap with many of the recent technology developments. They have also updated the delivery of the information and are no longer constrained to just paper. Their website includes links for each of the major milestones to relevant articles and videos. This road map suffers from a similar issue of the Rockwell International ISP: possibly logically sound, but only qualitatively supported.

A. Weigel, in *Bringing Policy into Space Systems Conceptual Design: Qualitative and Quantitative Methods* [6], makes use of influence diagrams like that in Figure 3.5 to illustrate the connection between policy (read: goals and strategies) and several of the driving technical parameters at lower levels. She refers to these as "semi-quantitative methods," and applies them to provide first-order guidance on policy decisions.

Several others, in their efforts to model and analyze complete space programs, have identified the importance of an explicit set of objec-



[Excerpt] – full plan continues through the year 2100…

FIGURE 3.4 – Excerpt from the Integrated Space Plan by Rockwell International [3].

tives when modeling an entire space program, see Koelle and Voss [7], Chamberlain [8], and the two *Space Technology Analysis and Mission Planning (STAMP)* proposals by Martin Marietta [9] and General Dynamics [10]. Common problems are the fixed sources of the objectives and the level of detail to which they are defined. Typically, the selected objectives are based on the latest presidential address [11], law passed [12], or even NASA's original charter [13]. These specified objectives provide more the overarching support of a *spacefaring goal* rather than specific activities expected in a *space program objective* as they are de-

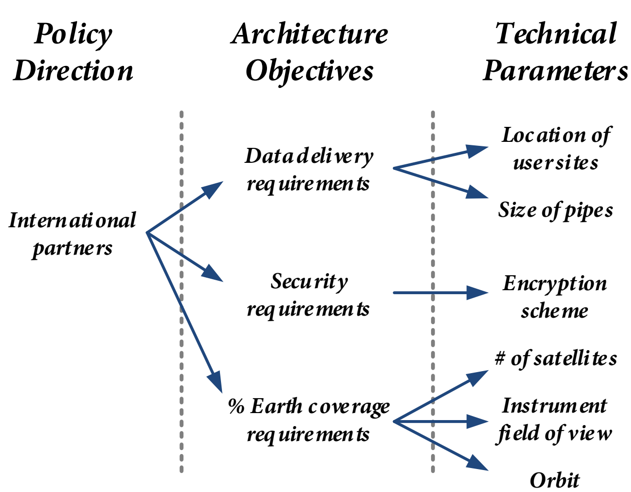


FIGURE 3.5 – Influence diagram illustrating the impact that a given policy direction can have on the lower levels of the hierarchy. Reproduced from Weigel [6].

fined here. Additionally, these processes model the programs that they intend to analyze based on the same static set of goals. How is it possible to inform the decision-maker on goals and strategies (and therefore objectives) to pursue, if a static set of program objectives are used in the analysis?

Two of the reviewed sources did use surveys to prioritize and quantify the desired space program goals and objectives among industry experts [7, 10]. These both happened to be from the 1960s, and their priorities and aspirations unfortunately do not still apply to modern planning efforts. Even if the surveys could simply be re-issued and the priorities updated, the processes themselves had no quantified connection to these goals.

This parametric connection is vitally important during this first step. Dole says:

> ...The point is that a great variety of alternatives should be considered and analyzed, and efforts should be made to explore the complete range of possible alternatives now open to the Agency. The list of alternatives should not be arbitrarily narrowed too early in the long-range planning process. It is hoped that the planning process will uncover new and better policy alternatives as well as elucidate the real nature of the Agency's goals. [14]

This last idea, to uncover new and better policy alternatives is depicted by the dotted line on the left side of Figure 3.2. The process isn't over when the analysis results come in. Rather, alternative sets of goals and strategies need to be traded and compared to better inform those making decisions at these top tiers.

STEP 2 » ASSEMBLING THE CANDIDATE SPACE PROGRAM

With the space program objectives derived, the next step in the ideal process would be to assemble a candidate program to fulfill said objectives. Now ideally this candidate program would consist of two parts: operations and developments.

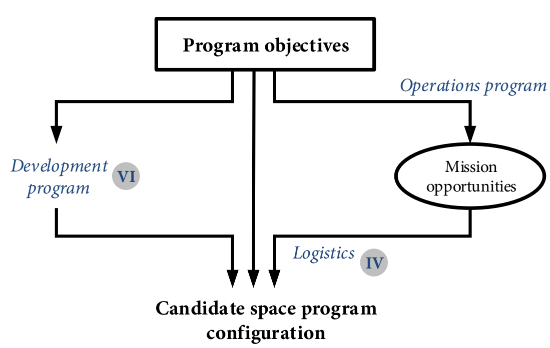


FIGURE 3.6 – The second step in the ideal *Ariadne* solution concept methodology » Assemble the candidate space program.

Operations are the missions, Tier IV in Sherwood's hierarchy, to accomplish specific objectives, or to work out intermediate steps in pur-

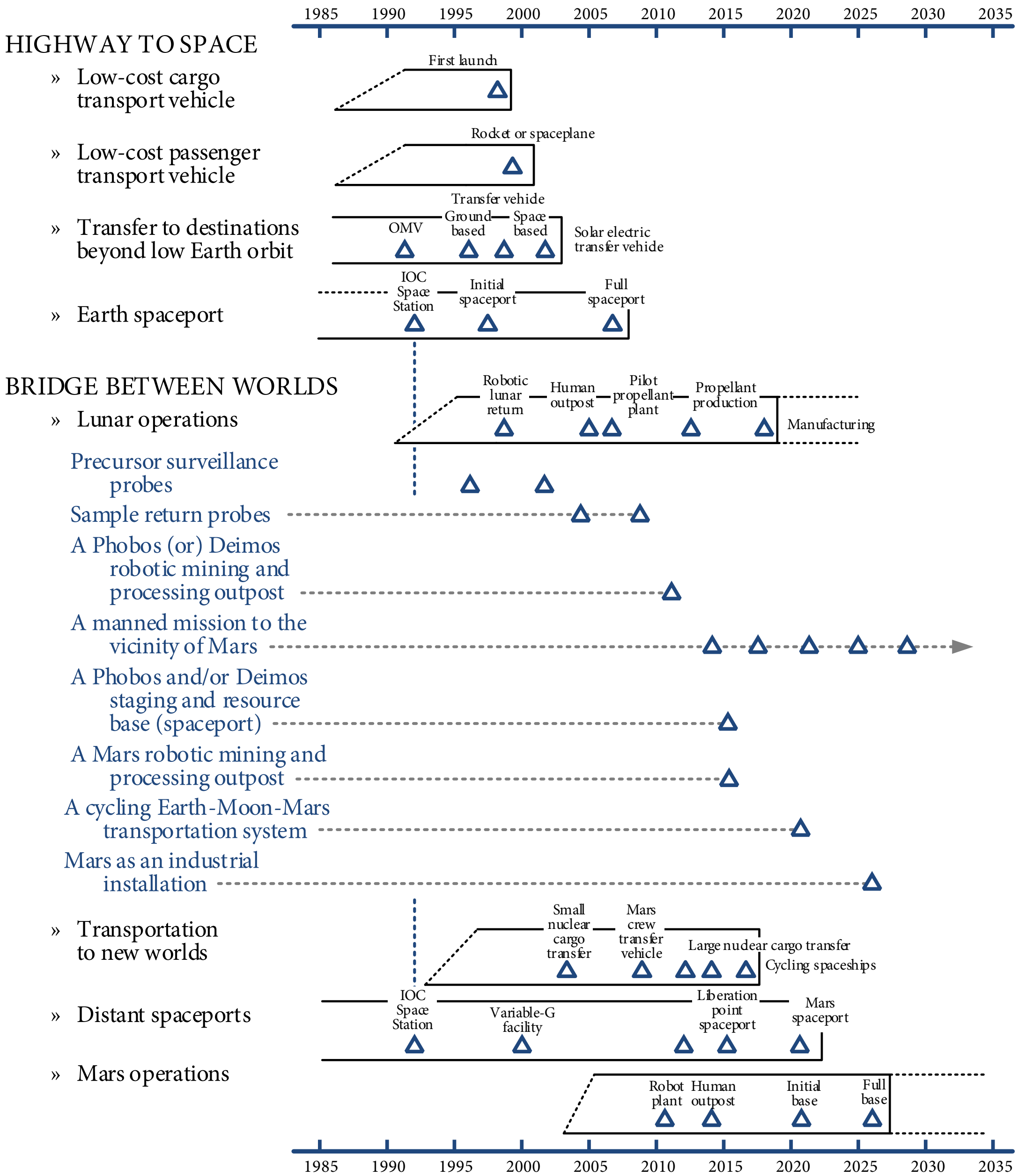


FIGURE 3.7 – Operations plan for 1985-2035 for the establishment of a manned base on Mars. Reproduced from Nolan [15].

suit of a final objective. Figure 3.7 is an example operational overview working towards a fully realized base on Mars.

For missions beyond LEO, opportunities might be limited due to the relative position of the Earth and destination over the length of the program. Possible launch windows and trajectories are well documented [16–20] to practically any desired destination. The mode of completing a given objective, the mission architecture, must be specified at this point, and launch window availability might dictate what architectures are available in the next step. Many researchers are looking at different means of modeling and comparing the wide range of possible mission architectures [21–25].

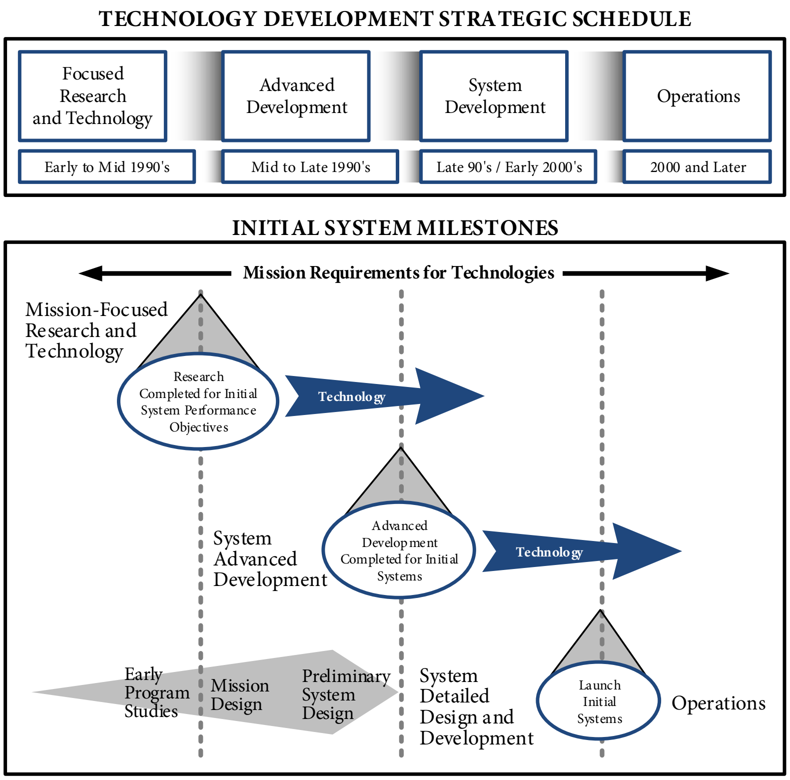


FIGURE 3.8 – Technology development phases. Reproduced from NASA [26].

Running parallel to the program operations in Figure 3.6 is the technology development program. These developments are research tracks targeted at improving the technology for the space program, the *performing subsystems* of Tier VI in Sherwood's hierarchy. Figures 3.8 and 3.9 both represent example overviews of technology development plans. Figure 3.8 depicts the generic phases of technology development and what steps must be reached before it can be operationally applied. Figure 3.9 depicts the expected schedule for the development of a nuclear thermal rocket for in-space propulsion.

The two tracks, operations and developments, run parallel to one

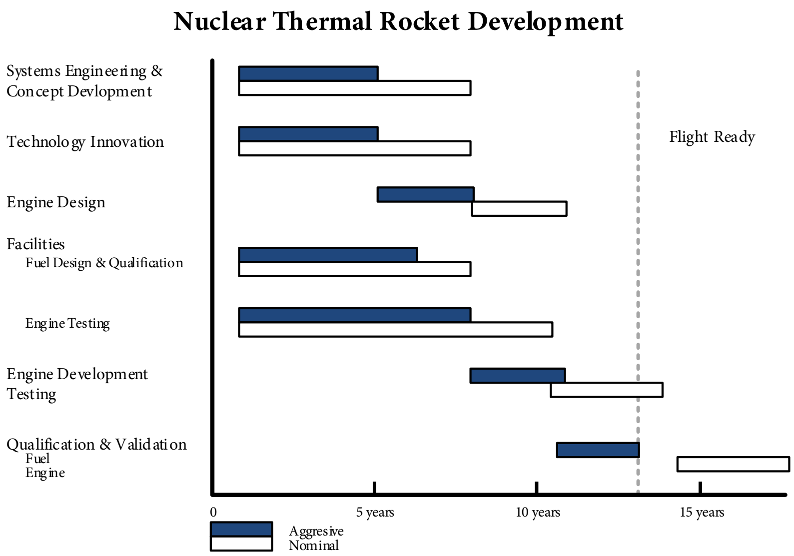


FIGURE 3.9 – Example timeline for the development of a nuclear thermal rocket. Reproduced from the Synthesis Group [27].

another, but they ideally should be tightly integrated. The right technologies can enable previously impossible operations, and purposefully planned operations can identify and prioritize the necessary technologies. The *Technology Alignment and Portfolio Prioritization (TAPP)* [28] method is a current effort by the Advanced Concepts Office of NASA's Marshall Space Flight Center being applied to identify and prioritize relevant technologies towards the desired missions and long range plans of the program.

With the program objectives for both operations and developments scheduled to meet the budget constraint, the candidate program is completely defined and is now ready analyzed and converged into a feasible (under the budget and schedule constraints) space program.

Step 3 » Converging on a feasible space program

A complete candidate space program is input for analysis in this third step. This involves multiple disciplines to be integrated to size the *functional elements*, Tier V, for each mission. Then, these hardware elements (rocket engines, entire launch vehicles, spacecraft, *etc.*) need to be integrated into the rest of the program to maximize the utility of each component. The key element in this step is the blue triangle denoted in Figure 3.10. This triangle represents the convergence check for the program: an objective function that the program can be measured against like minimum cost, maximum payload, *etc.*. During the analysis of this step, all of the variabilities and their effects are ignored: each component is perfectly reliable, scheduling delays are non-existent, and distributed budgets are consistent. Once the candidate program has converged under these ideal conditions, the

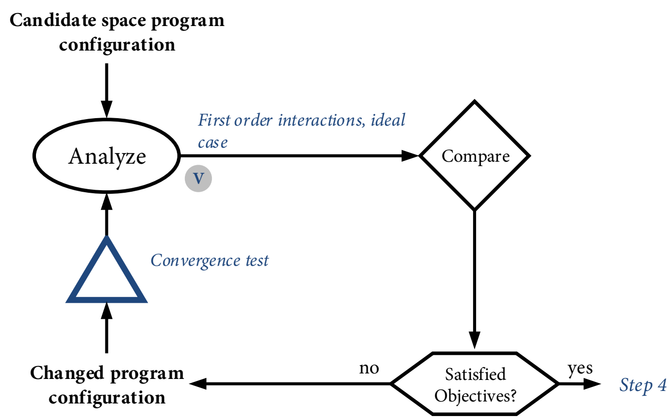


FIGURE 3.10 – The third step in the ideal *Ariadne* solution concept methodology » Converge a feasible space program.

variabilities will be introduced in step 4.

The step was the beginning of many of the better legacy processes from the process review in Chapter 2: General Dynamics [10], Koelle and Voss [7], Joy and Schnebly [29], and Martin Marietta [30]. The problem with these approaches was the lack of a systematic way of assembling the candidate program to begin with. For better or worse, it was left up to the experience of the user to determine the best starting program configuration.

Analysis of the program includes the sizing of any desired in-space elements according to their assigned mission architecture. This is typically done with mass estimating relationships (MER); regression analysis from a database of previous functional elements to determine the approximate mass and dimensions. W. Heineman, Jr. provides an excellent example of this approach in *Design Mass Properties II: Mass Estimating and Forecasting for Aerospace Vehicles Based on Historical Data* [31]. Several others apply a similar approach to determine the mass of desired spacecraft [32–34].

Once the final in-space payload for each mission has been determined, a launch vehicle needs to be designed that is capable of completing each mission. Figure 3.11 depicts an example flow of data between the software used at NASA's Advanced Concepts Office at Marshall Space Flight Center for launch vehicle sizing [35]. The initial sizing is performed with suite of tools: INTROS (INTegrated ROcket Sizing) for the initial weights and dimensions, LVA (Launch Vehicle Analysis) for structures, and POST (Program to Optimize Simulated Trajectories). Others have also contributed to this field of launch vehicle sizing, including the U.S. Air Force [36], the Aerospace Corporation[37], R. Rohrschneider [38], and P. Czysz and J. Vandenkerckhove [39].

As it is analyzed, the cost of the program needs to be estimated. The cost estimation process has been a very well explored field, with many contributions [40–43]. This component of the analysis is crit-

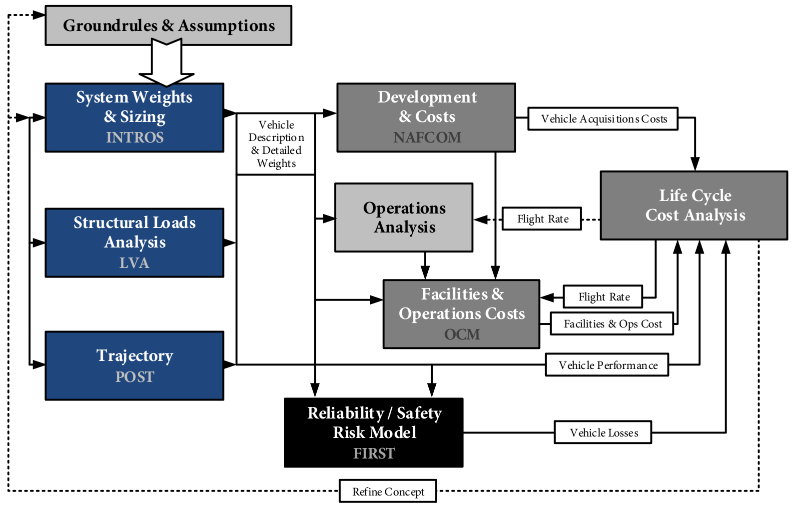


FIGURE 3.11 – Example Earth-to-orbit launch vehicle concept analysis from NASA Marshall's Advanced Concepts Office. Reproduced from NASA [35].

ical for the overall convergence of the program. Previously in this analysis, a launch vehicle was separately sized that was capable of meeting each mission requirement. Developing and producing a single unique launch vehicle for each mission would never be affordable. The program needs to be consolidated, developing only the optimum number of launch vehicles required in order to minimize costs. This consolidation can include a reduction in the number of launchers to be developed, combining missions, reducing required payloads, *etc.*

Once a program successfully satisfies its objectives, schedule, and budget, it is considered converged. Converged programs can now start to be compared with one another. Several metrics for comparison have been pursued [44, 45], the most interesting of which is the concept of a program's value. This concept was considered by Koelle and Voss [7], Chamberlain and Kingsland, Jr. [8] and Joy and Schneebly [29]. Another, more involved means of comparing converged programs is through simulation, which is discussed in final step of this ideal methodology.

Step 4 » Simulating the space program

The final step introduces all of the variabilities into the hardware, missions, technology development schedules, *etc.* A *Monte Carlo* analysis [46] can then be performed to simulate a candidate program thousands of times, this time with the included variabilities. This simulation will shed more light on each program by capturing the consequences of high risk missions, schedule slippage, mission failures, *etc.* Many efforts have been made toward addressing this variabilities by taking the reliability of interacting hardware and mission phases into account [8, 47, 48] and others have addressed the reliabilities of the individual hardware elements [49, 50].

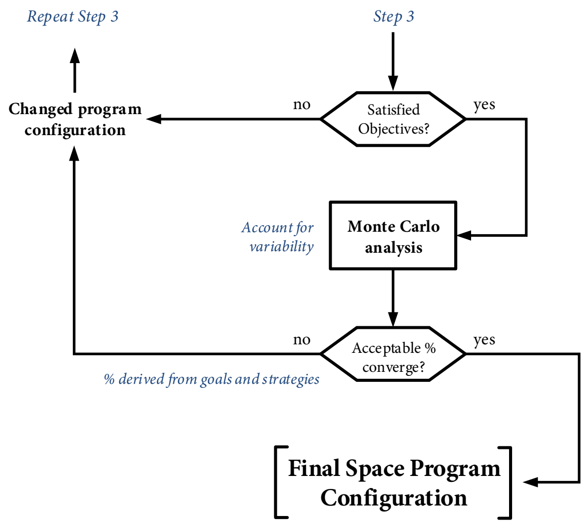


FIGURE 3.12 – The final step in the ideal *Ariadne* solution concept methodology » Simulate the space program.

NASA's *Space Flight Program and Project Management Handbook* [51] offers a applicable metric:

> A joint cost and schedule confidence level (JCL) is a quantitative probability statement about the ability of a program or project to meet its cost and schedule targets. ... Put simply, the JCL is the probability that a project or program's actual cost will be equal to or less than the targeted cost and its schedule will be equal to or less than the targeted schedule date. [51]

The example result of a JCL calculation is shown in Figure 3.13. With such a calculation, a seemingly ideal candidate may be eliminated due to its dependency on a risky mission architecture for a key mission. Accounting for this risk will drive the costs up over budget in most simulations. Another candidate may be removed from consideration due to the high probability of technology development delays for an enabling technology. The only problem with the JCL approach, is that it typically takes place later in the development process when more information is typically available.[4]

[4] Recall Figure 2.2.

Overall » the *Ariadne* ideal methodology

The ideal methodology enables a planner to take policy inputs from the highest tiers, assemble and analyze space program candidates, and consistently compare them with one another. This process could enable better-informed decisions at every level.

A process that was mentioned throughout the above sections is given special attention here as it most closely resembles the specifications sought after in the ideal *Ariadne* methodology: the *Space Technology Analysis and Mission Planning (STAMP)*, a deliverable of a NASA contract by General Dynamics and championed by Kraft Ehricke.

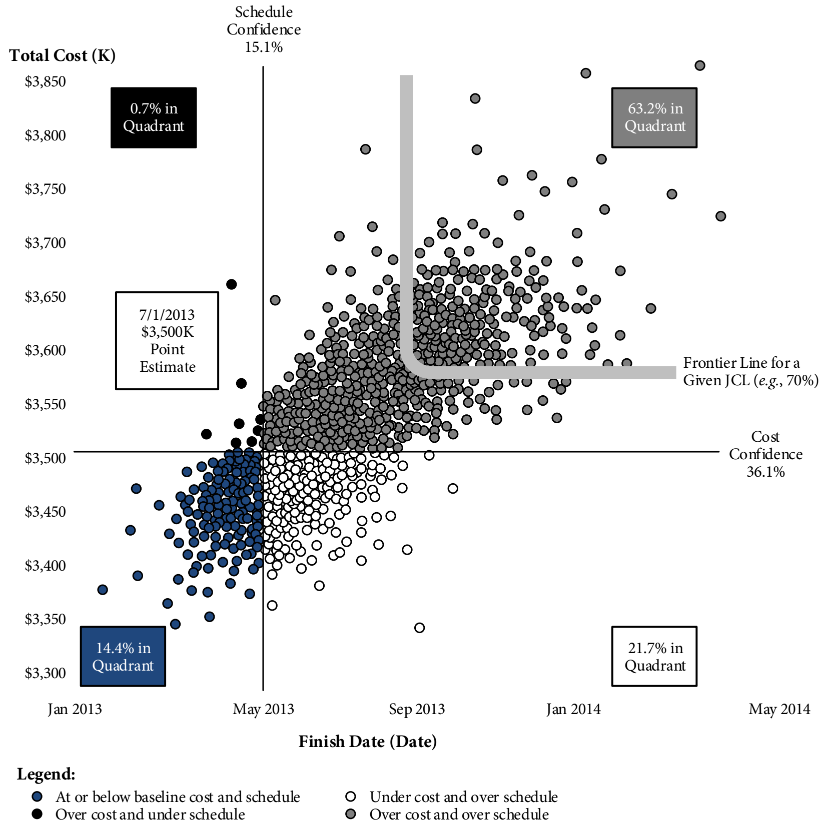


FIGURE 3.13 – An example joint cost and schedule (JCL) calculation result plot. Reproduced from NASA [51].

Their effort fell short in a few areas, most notably a distinct disconnect between the formulation of the objective and the analysis of the program, see Figure 3.14. In between these critical steps, it was up to the planner and his expertise to formulate a space program to fulfill the broad spacefaring goals. This presents two problems.

First, with no defined connection between the goals and the objectives that the planner comes up with, it is left completely to his interpretation on how well the program fulfills the goals. Two different planners would come up with two different plans, and while the STAMP system would be capable of determining which performed better in certain metrics, there would be no way of determining which of the two came closer fulfilling the original goals.

Second, STAMP was hindered computationally at the time. A lot of time and management was involved for a single run of one mission. Often there was not any memory available for larger runs of multiple missions or programs. Ehricke says:

> If storage capability is introduced to permit the hold-over of important results over a number of missions and the subsequent evaluation of these results in a Technological Program Synthesis, it is possible to consider

STAMP [General Dynamics]

- **Define general program objectives**
- **Determine objective weights** – process uses a survey of company employees
- **Calculate *National Space Program Value***
- **Formulate *General Space Programs***
- For each *General Space Program*
  - For each program objective
    - **Determine utility factor** – the program's effectiveness in accomplishing the objective
  - **Determine *Space Program Quality*:** cost-effectiveness, operational effectiveness, ability, and growth
  - **Calculate utility of *General Space Program***
- **Compare *General Space Programs*** – compare against the *National Space Program Value*, as well

- **Define the operational achievements and technology milestones**
- **Define mission objectives**
- For each mission
  - **Calculate performance requirement:** ΔV
  - **Payload analysis**
  - **Propulsion analysis**
  - **Vehicle analysis**
  - **Mission performance analysis and weight calculation**
  - **Vehicle-mission integration**
- **Synthesize projects and sub-programs**
- **Assemble operations program**
- **Assemble development program**
- **Synthesize *National Space Program***

FIGURE 3.14 – The General Dynamics *Space Technology Analysis and Mission Planning* (STAMP).

Priority	Prototype specification
1	Integration, both horizontal (multi-disciplinary) as well as vertical (multi-tier)
2	Inclusion of both technical and non-technical factors
3	Parametric support for the decisionmaker in the earliest phases of planning
4	Iteration and convergence, providing a solution space of possible alternatives
5	Capability to consistently compare alternative programs, missions, and hardware

TABLE 3.1 – Focused and prioritized list of system specifications for the *Ariadne* prototype system.

projects or complete space programs. [52]

Despite these shortcomings, the effort of Ehricke and General Dynamics has yet to even be replicated, much less surpassed.

3.1.2 *Identification of prototype specifications*

The ideal solution methodology is too comprehensive for an individual PhD researcher to implement alone. There are multiple missing pieces that would need to be developed before such an ideal system could be realized. These missing pieces have been identified as possible opportunities for worthy research contributions:

» a parametric connection between the top three tiers and the analysis in step three of the program;

» the synthesis of program operations and technology developments into the ideal space program;

» finally, the modification of JCL-like calculations to be applied at an earlier phase in the development of a program.

Based on the observed available 'best practice' introduced in Chapter 2 and discussed further earlier in this chapter, the primary contribution should be the parametric connection between the goals, strategies, missions and hardware.

The ideal specifications for *Ariadne*, presented previously in Table 2.9, have been focused and prioritized to only highlight those specifications necessary to prove the concept of a vertically integrated decision support system. Table 3.1 provides this reduced list of specifications for the prototype system.

Figure 3.15 represents the goal of the prototype's level of parametric integration. Notice that although two tiers are omitted and the number of parameters between the tiers is not quite balanced, there are no disconnects within any given tier or between tiers. This vertical integration addresses the first desired specification listed in Table 2.9

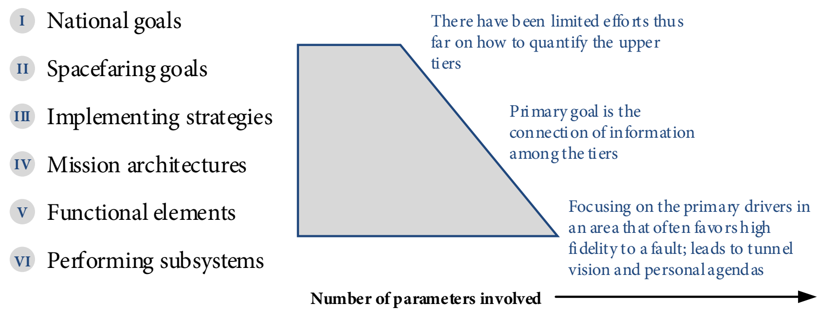


FIGURE 3.15 – The parametric goal for the prototype realization of *Ariadne*.

In the next section, a prototype methodology is introduced to prove the utility of the contribution.

3.2 *Prototype methodology concept*

This section contains a similar introduction to the reduced prototype methodology, a walk-through of its primary steps with the inputs and outputs of each step identified. The applied methods used are then covered (either original or a selected existing method) followed by an overview of the software implementation.

3.2.1 *Prototype flow diagram*

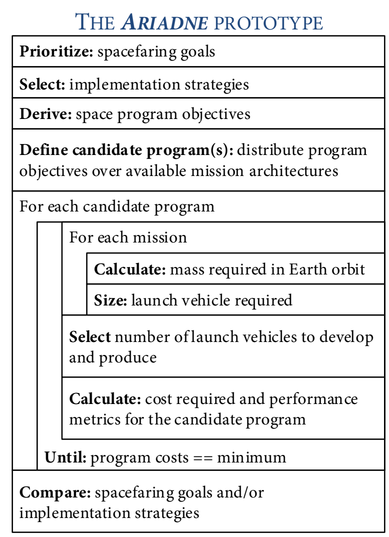


FIGURE 3.16 – Structogram representation of the top-level logic of the *Ariadne* prototype system.

The logic of the *Ariadne* prototype system is shown in Figure 3.16. This structogram provides more detailed steps than the previous ideal methodology and will be discussed in the following subsections.

The prototype flow diagram is also shown in a flow diagram in Figure 3.17, similar to the previously discussed ideal case depicted in Figure 3.2. By comparing the two, it can be seen that the prototype contains the following omissions: the organizational goals from the derivation of program objectives, a technology development track for the program, scheduling and specific mission opportunity constraints, and finally the variability and Monte Carlo simulation of converged programs. All of these omissions allow the focus to be on the vertical integration of goals, strategies, missions and hardware aspects.

3.2.2 *Select and prioritize spacefaring goals*

The AHP was also used for the survey of space mission architectures introduced in Section 2.2.2.

The first step in the *Ariadne* prototype process involves the definition and prioritization of spacefaring goals. For the purposes of this prototype, the selected spacefaring goals have been modified from Sherwood's original listing discussed in Table 2.7 of Chapter 2. By arranging the categories and sub-goals that Sherwood introduced into its

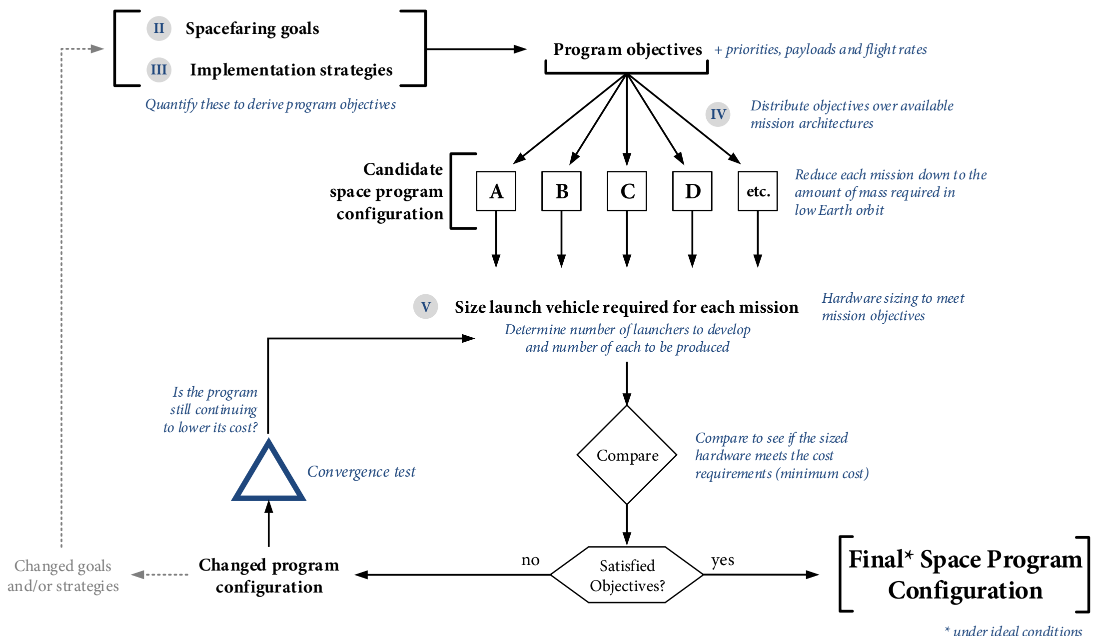


FIGURE 3.17 – Flow diagram representation of the *Ariadne* prototype system.

own hierarchy, the AHP can once again be used for the quantification and prioritization of these goals.

The details of this application are given in the methods description in Section 3.3.1. This first step leads to a consistent set of quantified goals to be used later in the derivation of specific program objectives.

3.2.3 *Select implementation strategies*

The next step in the prototype process requires the definition and selection of implementation strategies. Numerous possible strategies were introduced in Section 2.3.3. From these, four implementation strategies have been selected for the prototype:

» The level of technology;

» The pacing of the program;

» The level of man's involvement;

» The relative scale of the program.

A scale is created for each of the four strategies. The user will select a value from each scale to observe and eventually evaluation the effects that each strategy may have on the program solution space.

These four strategies were selected for the prototype due to their logical, primary effects on a program that could be readily modeled and understood. The level of technology involved will affect the recommended performance of the selected launch vehicle candidates, but

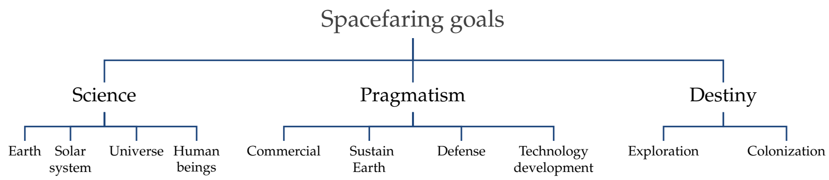


FIGURE 3.18 – Hierarchal arrangement of selected spacefaring goals for the *Ariadne* prototype system.

will also lead to higher development costs. The level of urgency sets the pacing of the program: the flight rate for each operational objective. The level of man's involvement adjusts the prioritization of manned operational objectives accordingly. Finally, the relative scale of the program allows the observation of how sensitive a given program might be to increased payloads, lower flight rates, *etc.* These four strategies are displayed in a radar plot in Figure 3.19 and discussed further in Section 3.3.2.

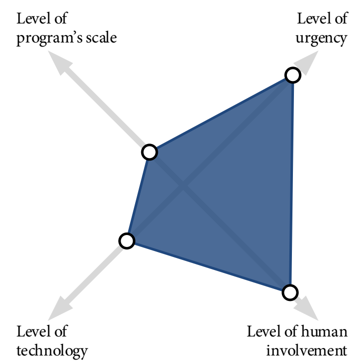


FIGURE 3.19 – A radar plot of the example values for the selected strategy parameter scales.

3.2.4 *Derive space program objectives*

Now, with both the prioritized spacefaring goals and selected implementation strategies defined, the next step in the *Ariadne* prototype process is the derivation of program objectives. This step involves a defined list of program objectives and their connections to both the goals and strategies in order to properly prioritize the objectives and thus inform the planner. The list of selected program objectives has been compiled from multiple sources [53, 54] and is given in Table 3.2.

The list consists of two types of objectives: operations and technology developments (denoted in the first column in Table 3.2).

The operations are further grouped into two categories: operations in Earth orbit, and operations in the vicinity of the moon or other planetary bodies. These objectives, selected from STAMP, Ehricke and General Dynamics [53], have been simplified intentionally for the sake of the prototype. See the 1970 McDonnell Aircraft company study on Hypersonic Research Facilities (HyFAC) [55, 56] for an in depth look at how intricate the determination of program objectives can be implemented.

The returns on investing in a given technology will not be taken into account for this prototype. The technology program objectives are provided here to illustrate how their inclusion would look, as well as inform the planner how much attention is being applied to technology developments (even if their effects are not taken into account). The list of technologies were taken directly from NASA's most recent technology roadmaps [54].

The priority of each objective is determined by its connections to the

	Program Objective	Examples
Earth operations	Unmanned satellite	Satellites for communication, weather, reconnaissance, *etc.*
	Manned satellite	Satellite for reconnaissance, science experiments, *etc.*
	Space station	Manned installation in orbit for science and/or military purposes
	Power plant	Station for generating solar power and providing it to the surface
	Cis-lunar station	Station in cycle between the Earth and moon
	Auxiliary vehicles	Debris disposal craft, tugs, inter-station transportation, *etc.*
	Supplied colony	Permanent manned installation in orbit for colonization purposes
Moon/planet operations	Unmanned satellite	Satellite for survey, experimentation, *etc.*
	Manned satellite	Satellite for reconnaissance, exploration, *etc.*
	Unmanned lander	Rover, drone, *etc.* for scouting and experimentation
	Manned lander	Exploration, experimentation, sample collection
	Orbital station	Manned station in orbit for observation, control of surface drone, *etc.*
	Surface station	Manned station for mining, hub for exploration, *etc.*
	Supplied colony	Permanent manned installation on the surface for colonization purposes
	Self-sufficient colony	Independent colony on the surface of the planet/moon
	Solar probe	Flyby satellite to observe bodies within the solar system
	Interstellar probe	Flyby satellite destined for beyond the solar system
Technology development	Launch propulsion	Solid and liquid rockets, air breathing propulsion, *etc.*
	In-space propulsion	Chemical, non-chemical, and advanced propulsion technologies
	Space power and energy storage	Power generation, storage, management and distribution
	Robotics	Sensing and perception, mobility, autonomy, *etc.*
	Communications and navigation	Optical, radio frequency, *etc.*
	Human health, life support	Environmental control, EVA systems, radiation protection, *etc.*
	Human destination systems	ISRU, habitat systems, human mobility systems, *etc.*
	Science instruments, observatories	Remote sensing instruments and observatories, *etc.*
	Entry, descent, and landing systems	Aeroassist and atmospheric entry, descent and targeting, landing, *etc.*
	Nanotechnology	Engineered materials, energy storage, propulsion, sensors, *etc.*
	Modeling and simulation	Computing, information processing, *etc.*
	Materials, manufacturing	Structures, mechanical systems, *etc.*
	Ground and launch systems	Operational life cycle, environmental protection, reliability, *etc.*
	Thermal management	Cryogenic systems, thermal control and thermal protection systems, *etc.*
	Aeronautics	Global aviation growth, supersonic aircraft, efficiency, *etc.*

TABLE 3.2 – Selected program objectives for the *Ariadne* prototype system. The operational objectives are largely from STAMP [53] and the technology objectives are from NASA [54].

defined spacefaring goals. Primary connections have been identified between the spacefaring goals and the selected program objectives. For example, a program that highly prioritized the science goal of understanding the universe would have a high prioritization on the program objective of unmanned solar probes. This objective is then highlighted for the program planner, with its resultant attributes determined by the selected levels for the implementation strategies.

Each operational objective consists of multiple attributes depending on its type. Single mission objectives, like placing a satellite into orbit or landing a rover on moon, have two attribute scales: payload mass and number of launches per year. Installation objectives, like a space station in LEO or a supplied base on Mars, have three attribute scales:

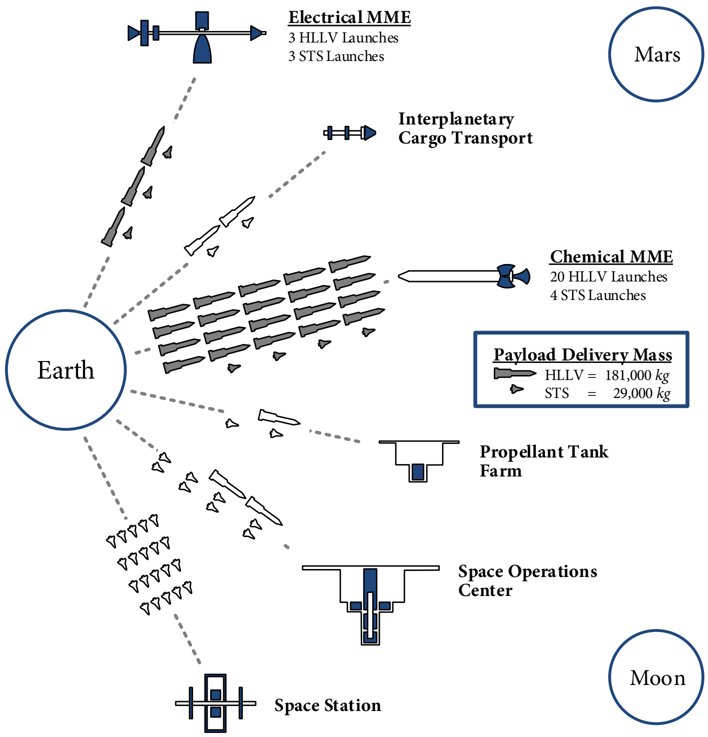


FIGURE 3.20 – Example operational objectives and the required launches for installation/support in the Manned Mars Explorer (MME) study. Reproduced from Nolan [15].

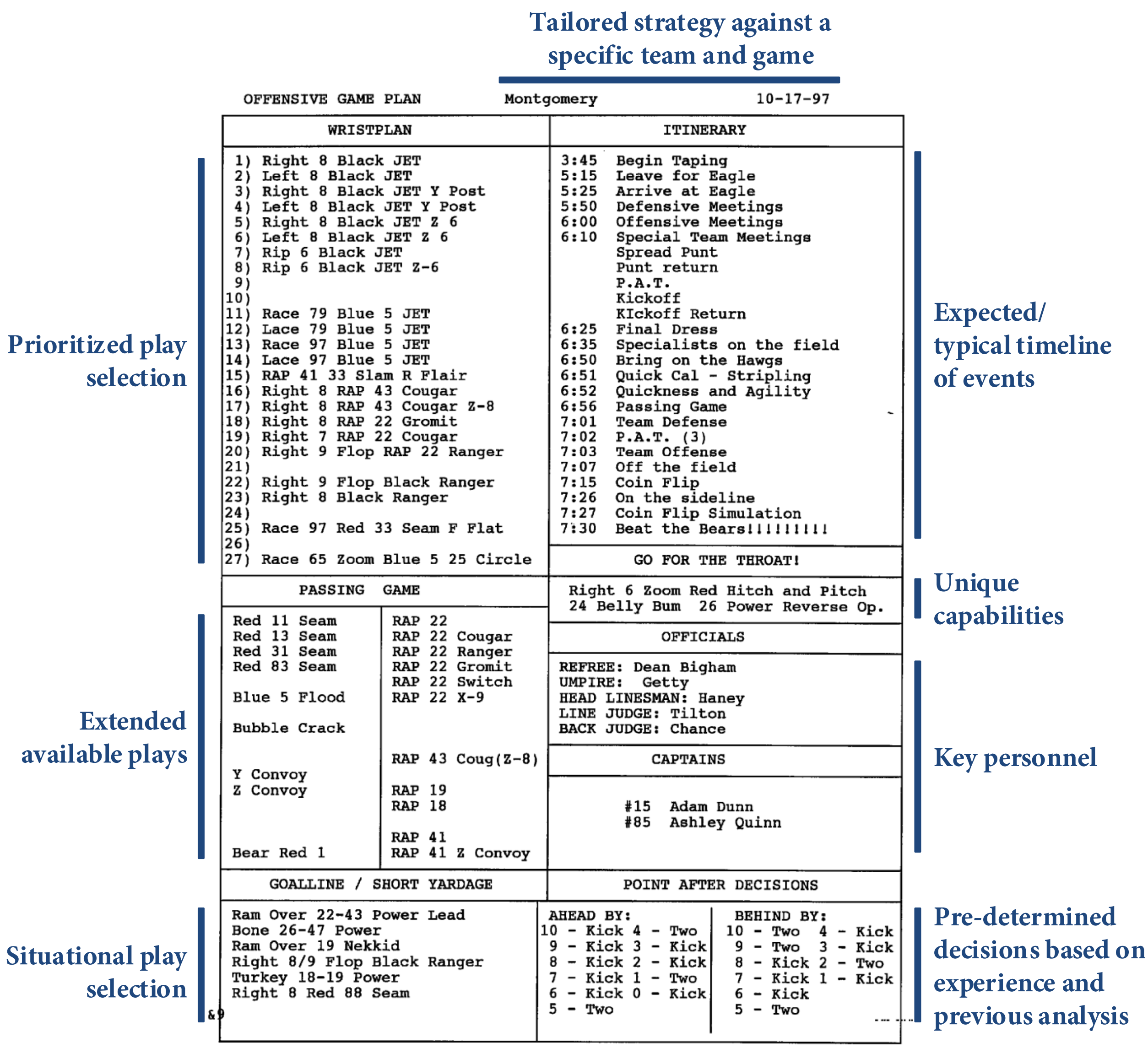
total installation mass, re-supply payload mass, and number of re-supply launches per year. These scales are meant to represent the typical values determined from experience but can also be adjusted to more closely observe a desired range.

For the proof-of-concept prototype, launch rates are taken as the average over the course of the program. This assumption enables program comparison unhindered by detailed schedules.[5] The recommended missions and number of launch for each objective will also be influenced by the selected strategy factors. For example, manned operational objectives are directly influenced by the strategy scale for man's involvement. Likewise, the payload ranges and launch rates are affected by both the scale and pacing strategy factors. The connections between objectives and strategies will be detailed in Section 3.3.3.

[5] If desired, the launch windows for particular destinations dictated by orbital mechanics can be taken into account in the assigned launch rates.

It should be stressed here that the range of a given attribute is more important to define than a focus on any single attribute (*e.g.*, an accurate mission payload). The range of an attribute enables the planner to see what 'typical' values at a particular program scale may look like, and then assemble and analyze the resultant program. The planner can then make an informed decision based on this trend as he assembles the program in the next step.

The final outputs of this step are the prioritized objectives along with their corresponding attributes. This is the information, referred



OFFENSIVE GAME PLAN   Montgomery   10-17-97

WRISTPLAN

1) Right 8 Black JET
2) Left 8 Black JET
3) Right 8 Black JET Y Post
4) Left 8 Black JET Y Post
5) Right 8 Black JET Z 6
6) Left 8 Black JET Z 6
7) Rip 6 Black JET
8) Rip 6 Black JET Z-6
9)
10)
11) Race 79 Blue 5 JET
12) Lace 79 Blue 5 JET
13) Race 97 Blue 5 JET
14) Lace 97 Blue 5 JET
15) RAP 41 33 Slam R Flair
16) Right 8 RAP 43 Cougar
17) Right 8 RAP 43 Cougar Z-8
18) Right 8 RAP 22 Gromit
19) Right 7 RAP 22 Cougar
20) Right 9 Flop RAP 22 Ranger
21)
22) Right 9 Flop Black Ranger
23) Right 8 Black Ranger
24)
25) Race 97 Red 33 Seam F Flat
26)
27) Race 65 Zoom Blue 5 25 Circle

ITINERARY

3:45	Begin Taping
5:15	Leave for Eagle
5:25	Arrive at Eagle
5:50	Defensive Meetings
6:00	Offensive Meetings
6:10	Special Team Meetings
	Spread Punt
	Punt return
	P.A.T.
	Kickoff
	KIckoff Return
6:25	Final Dress
6:35	Specialists on the field
6:50	Bring on the Hawgs
6:51	Quick Cal - Stripling
6:52	Quickness and Agility
6:56	Passing Game
7:01	Team Defense
7:02	P.A.T. (3)
7:03	Team Offense
7:07	Off the field
7:15	Coin Flip
7:26	On the sideline
7:27	Coin Flip Simulation
7:30	Beat the Bears!!!!!!!!!

GO FOR THE THROAT!

Right 6 Zoom Red Hitch and Pitch
24 Belly Bum 26 Power Reverse Op.

PASSING GAME

Red 11 Seam	RAP 22
Red 13 Seam	RAP 22 Cougar
Red 31 Seam	RAP 22 Ranger
Red 83 Seam	RAP 22 Gromit
	RAP 22 Switch
Blue 5 Flood	RAP 22 X-9
Bubble Crack	
	RAP 43 Coug(Z-8)
Y Convoy	
Z Convoy	RAP 19
	RAP 18
	RAP 41
Bear Red 1	RAP 41 Z Convoy

OFFICIALS

REFREE: Dean Bigham
UMPIRE: Getty
HEAD LINESMAN: Haney
LINE JUDGE: Tilton
BACK JUDGE: Chance

CAPTAINS

#15 Adam Dunn
#85 Ashley Quinn

GOALLINE / SHORT YARDAGE

Ram Over 22-43 Power Lead
Bone 26-47 Power
Ram Over 19 Nekkid
Right 8/9 Flop Black Ranger
Turkey 18-19 Power
Right 8 Red 88 Seam

POINT AFTER DECISIONS

AHEAD BY:		BEHIND BY:	
10 - Kick	4 - Two	10 - Two	4 - Kick
9 - Kick	3 - Kick	9 - Two	3 - Kick
8 - Kick	2 - Kick	8 - Kick	2 - Two
7 - Kick	1 - Two	7 - Kick	1 - Kick
6 - Kick	0 - Kick	6 - Kick	
5 - Two		5 - Two	

FIGURE 3.21 – 1997 New Caney Eagle offensive game plan. Decision support for the football coach.

to hereafter as a *dashboard*, provided to the planner as he begins to assemble the candidate space program for analysis.

This feedback can be seen as analogous to a football coach's play card. An example play card can be seen annotated in Figure 3.21. The play card includes the specific strategy for that game, the available capabilities of the team, previous lessons learned,[6] and pre-calculated decisions based on given game scenarios. While the coach is still free to make calls off book, in most cases he will trust the effort and experience that went into the creation of the game plan on the play card.

The coach is forced to use this play card to make his decisions in real time. The space planner is fortunate enough to have a longer lead time, though this means that he is without excuse for being uninformed. He must have the tools at his disposal to make informed decisions.

[6] ...such as *never underthrow the deep ball*.

3.2.5 *Distribute selected objective payloads among available mission architectures*

The next step in the *Ariadne* prototype process is the selection and distribution of mission payloads among available mission architectures. Available mission architectures are those that have been modeled to take the final payload mass as an input, and return the initial mass required in low Earth orbit (IMLEO) for that mission.[7] Installation masses for space stations or ground bases are not assigned at this point. They will be discussed and accounted for in a later step.

[7] This sizing of the in-space elements is discussed in the next step.

Any mission architectures that have been modeled are made available to the planner, though preferred approaches (due to lower mass requirements, *etc.*) can be presented on the dashboard to better inform the planner. It is desired for these preferred approaches to only be selected after an initial comparison has been made, to avoid using a particular mode or approach simply because it is available or familiar.

For the prototype, there has been no destination strategy factor defined. The desired destination of each mission architecture is to be selected by the program planner. So long as the selection is consistently done across any desired analysis sweeps, the programs are still eligible for proper comparison.

LUNAR DIRECT FLIGHT FOR LUNAR LANDING

Input list of maneuvers/phases
Determine all required attributes for each phase: ΔV, t, payload $= f(t)$, I_{sp}, etc.
Input mass at Earth interface (re-entry): $m_{payload}$
Calculate mass required for trans-Earth injection: $m_{TEI} = f(\Delta V, t, I_{sp}, m_{payload})$
Calculate mass required to ascend from lunar surface: $m_{LA} = f(\Delta V, t, I_{sp}, m_{TEI})$
Calculate mass required to land on lunar surface: $m_{LL} = f(\Delta V, t, I_{sp}, m_{LA})$
Calculate mass required for lunar orbit insertion: $m_{LOI} = f(\Delta V, t, I_{sp}, m_{LL})$
Calculate mass required for trans-lunar injection: $m_{TLI} = f(\Delta V, t, I_{sp}, m_{LOI})$
Calculate mass required for Earth launch to orbit: $m_{total} = f(\Delta V, t, I_{sp}, m_{TLI})$

FIGURE 3.22 – N-S representation of a reduced order model for the Direct flight mission architecture.

3.2.6 *Size required in-space elements*

Each modeled mission architecture consists of phases: trans-Earth injection (TEI), lunar orbit insertion (LOI), *etc.* The most basic means of modeling an architecture requires three basic attributes for each phase: the velocity requirement (ΔV), propulsion performance (I_{sp}), and structure ratio (ε). The propulsion performance and structure ratio can be connected and influenced by the selected strategy factor for the level of technology implemented, or they can be modeled separately to trade the effects of altering only one or the other.

This modeling approach uses the rocket equation to determine the propellant required for each phase, beginning with the final payload mass and working backwards. This approach works for most initial estimates since it is the required propellant that primarily dictates the change in mass for each phase.[8] Armed with this information, each assigned final operational objective payload mass can be traced back through each phase of its assigned architecture until it reaches the required mass to be inserted into Earth orbit (IMLEO). An example N-S representation of a mission architecture for a direct lunar landing and return in shown in Figure 4.13. The process begins with the final payload mass, in this case the mass of the return capsule as it re-enters the Earth's atmosphere.[9]

[8] Long duration missions, (*e.g.*, a manned mission to Mars) will require additional considerations due to the increase effect of consumables as a function of time, t.

[9] Future efforts with more trajectory details could begin with only the final payload at touchdown on the Earth's surface. This would require the sizing of the entry, descent, and landing (EDL) elements required based on the payload and re-entry speed.

The detail of the in-space sizing process is detailed in Section 3.3.4. The step ends with the IMLEO determined for each desired mission in the assembled candidate program.

3.2.7 *Size launch vehicle required for each mission*

With the IMLEO determined for each mission in the program, the next step in the *Ariadne* prototype process is to size the required launch vehicles. The number of stages is required as an input characteristic for the launch vehicle for each mission. The propellant for each stage must be specified, which can be traded independently by the planner. The specific impulse (I_{sp}) of each stage is dependent on the propellant selected and can be determined by the technology strategy factor previously selected.

An assumption of this prototype is that the largest sized vehicle will also be used as the launch vehicle for any required installed masses (space stations, ground bases, *etc.*)[10] The required installation mass is divided by the largest payload capability, and the resultant number of launchers are included in the mission distribution.

[10] This may not always be the best assumption *e.g.*, a space program that consists of small satellites to LEO and a large space station. This assumption would lead to possibly hundreds of additional launches of the launch vehicle developed to install the space station, when really it would likely be better to develop a heavy lift capability for just that purpose.

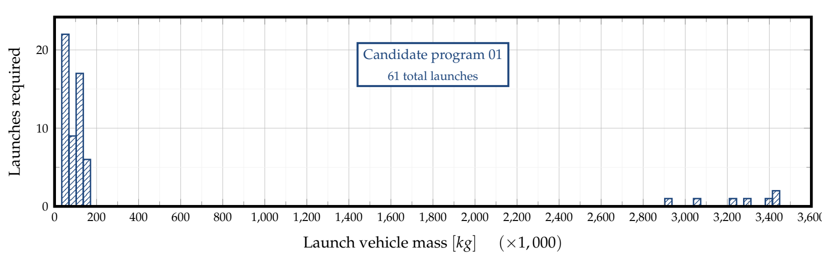


FIGURE 3.23 – An example simple distribution of launch vehicles for a candidate program with 61 payloads required to be delivered to LEO.

The launch vehicle sizing process of the Space Planners Guide was implemented for the prototype and is detailed in Section 3.3.5. The

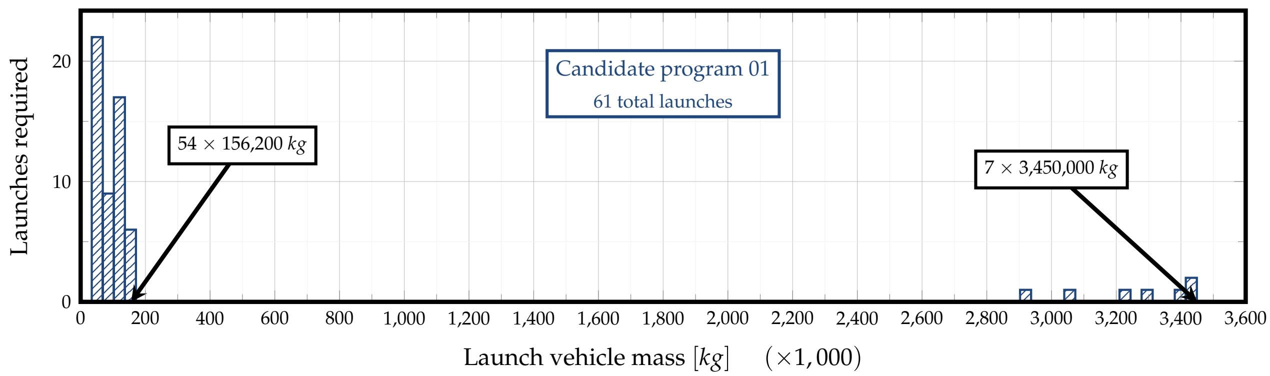


FIGURE 3.24 – Iterating the number of launch vehicles to develop to meet the required number of launches in order to converge on the program with a the minimum cost. In this simple example, two launch vehicles were selected to be developed.

output of this step is the plot found in Figure 3.23 indicating the launch vehicle mass required for each mission in the entire candidate program.

3.2.8 Select number of launch vehicles to be developed and produced to complete program objectives

This step describes the process of converging on the proper number of launch vehicles to achieve the lowest overall program costs. It was discussed previous how impractical it would be to both develop and produce a unique launch vehicle for each mission. Rather, a minimum number of launch vehicles should be developed since development costs are much larger than production costs.

For the example program visualized in the histogram in Figure 3.23, multiple clusters of launch vehicles can be observed. A clustering algorithm, looking for a specified number of clusters, groups the sized vehicles. It begins with a single group, and calculates the required numbers of launch vehicles to develop and produce and then the total launch costs for the program are calculated. With a single grouping, the largest sized launch vehicle is developed and then used for every desired mission. The process then continues, increasing the number of launch vehicles and calculating the total costs until it has converged on the correct number of launch vehicles to develop. Figure 3.24 is a reproduction of the above distribution, this time with the selected launch vehicles to develop and the number of each to produce.

This example candidate was fairly straightforward. But even so, it may be desirable at times to run a sensitivity over a larger set of possible launchers to develop. Figure 3.25 illustrates the convergence and selection process for the example *Candidate program 01*. Note the minimum total cost (denoted by the top line, in blue) is achieved with two developed launch vehicles, but that, even when three or four vehicles vehicles are developed instead, the total costs are not radically different.

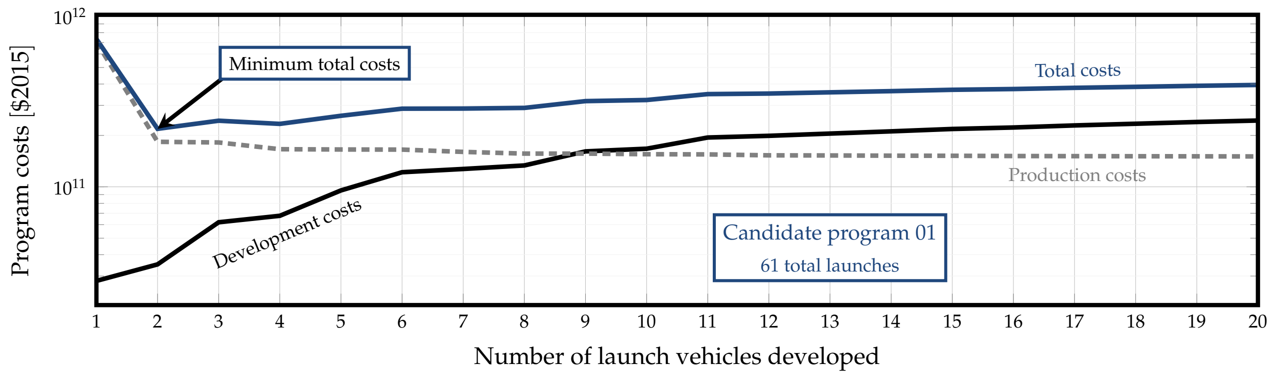


FIGURE 3.25 – Total launch vehicle costs (development and production) as a function of the number of launch vehicles developed.

3.2.9 *Compare converged candidate programs*

In the final step, once each planned program has converged, it is compared against the other candidate space programs to quantify the effects that the initial goals and strategies have on a program. The data output for each converged program includes:

» the selected number of launch vehicles to be developed and produced;

» the total launch vehicle costs for the program;

» the total mass placed in Earth orbit (first order measure of in-space costs and program productivity);

» and the total IMLEO capability (as inevitably some of the launch vehicles will be sized larger than some of their respective payload);

» the percentage of recommended program objectives that were met.

The utility of all of this data and the metrics that can be derived from their combinations will be shown in the case study in Chapter 4. But first, the rest of this chapter contains details of the methods applied and the software implementation of the *Ariadne* prototype.

3.3 *Prototype methods*

The previous section described the overall process of the *Ariadne* prototype system. The inputs and outputs of each step were included but the analysis was saved for this section. This section serves to detail the methods and algorithms behind the previously discussed steps.

3.3.1 *Spacefaring goal prioritization*

The Analytic Hierarchy Process (AHP) was previously used to weigh the elements of an architecture for the survey metric. The AHP is

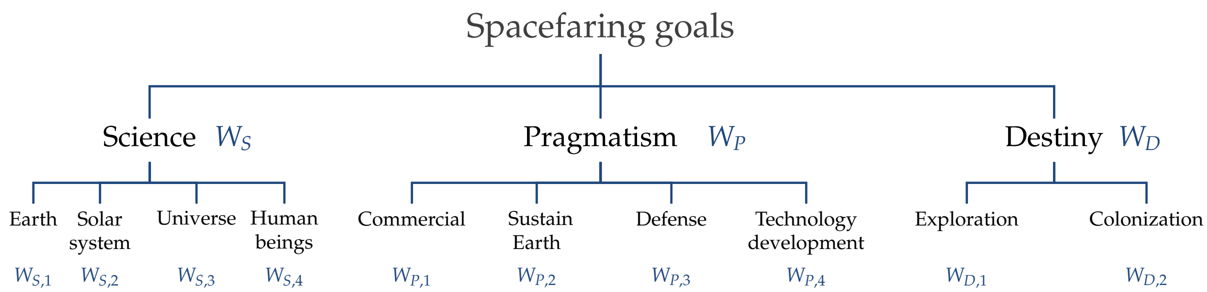


FIGURE 3.26 – Hierarchy of spacefaring goals with assigned variables for the *Ariadne* prototype system.

endorsed by NASA in their *Systems Engineering Handbook* [57]. They say that

> Many different problems can be investigated with the mathematical techniques of this approach. AHP helps capture both subjective and objective evaluation measures, providing a useful mechanism for checking the consistency of the evaluation measures and alternatives suggested by the team, and thus reducing bias in decision-making. [57]

The AHP provides a consistent means of quantifying the goals, but more importantly it provides transparency in the prioritization of the goals. An example set of pairwise comparisons for the top tier (Science, Pragmatism, and Destiny) is shown in the decision matrix below.

$$\begin{array}{r c} & \begin{array}{ccc} \text{Science} & \text{Pragmatism} & \text{Destiny} \end{array} \\ \begin{array}{r} \textbf{Science} \\ \textbf{Pragmatism} \\ \textbf{Destiny} \end{array} & \left[\begin{array}{ccc} 1 & {}^{1}/_{9} & {}^{1}/_{5} \\ 9 & 1 & 3 \\ 5 & {}^{1}/_{3} & 1 \end{array}\right] \end{array} \tag{3.1}$$

The decision matrix is read by looking at the goal listed in the row and then reading along each column to see how it compares with each other goal. With the AHP and linear algebra, these comparisons are transformed into a consistent, prioritized set of weights for the first tier of the spacefaring goal hierarchy, given below.

$$\begin{array}{r c} & \text{Weight} \\ \begin{array}{r} \textbf{Pragmatism, } W_P \\ \textbf{Destiny, } W_D \\ \textbf{Science, } W_S \end{array} & \left[\begin{array}{c} 0.672 \\ 0.265 \\ 0.063 \end{array}\right] \end{array} \tag{3.2}$$

This process is repeated for each sub-tier and an overall weight is assigned to each of the variables depicted in Figure 3.26. Unlike the full realization of the AHP, there is no predefined set of alternative programs being used to compare against the desired goals. Rather, the comparison and prioritization methods are being applied to shape the realization of the resultant space program. These calculated weights for each goal and sub-goal will be used later in the derivation of the space program objectives.

3.3.2 Implementation strategy scales definition

The four selected implementation strategies for the *Ariadne* prototype are: man's involvement, scale, technology, and pacing of the program. Each strategy is represented by a factor on a scale from 1-5. For example, if the selected strategy factor for the pace of the program was 1, the program would be planned for a slower, drawn out program. If the selected factor was 5, the program plan would included high launch rates and quick turn-around times. The factor and scale for each strategy are given below:

$$\text{Program pace: } S_{time} = [1 \text{ (postponed)} - 5 \text{ (near term)}]$$

$$\text{Technology level: } S_{tech} = [1 \text{ (off-the-shelf)} - 5 \text{ (advanced)}]$$

$$\text{Man's involvement: } S_{man} = [1 \text{ (none)} - 5 \text{ (extensive)}]$$

$$\text{Program scale: } S_{scale} = [1 \text{ (small)} - 5 \text{ (grand)}]$$

These factors will be used throughout the rest of the analysis process. They correspond to particular multipliers and attributes of the defined program objectives, as will be seen in the next section.

3.3.3 Derivation of objectives from goals and strategies

The prioritized spacefaring goals and selected implementation strategies are used in the derivation of the space program objectives. The selected objectives for the *Ariadne* prototype were defined in Table 3.2. The first step in the process is to identify any primary connections between the spacefaring sub-goals and the program objectives. This is a binary response for the prototype: the connection ($CP_{i,X,n}$) is assigned the value of 1 if the objective (O_i) primarily contributes toward a given goal ($W_{X,n}$) and 0 if it does not. For example, although a manned space station could be used towards the Science goal of understanding the universe, a manned space station would not be installed primarily for that purpose. A manned station in LEO could, however, be developed to better understand the effects of sustained micro-gravity on man.

There are a couple of important things to note here. First, this is not an exercise in coming up with a creative solution in which a given objective could satisfy a given goal. Only the primary, typical purposes of the selected objectives should be recorded. Second, these connections are not permanent by any means. Per definition, the exploration goal will have to connect with program objectives for destinations that are still being explored. Likewise, the pragmatic goal for commercial development should only connect with any objectives that would primarily contribute towards the current development of industry in space. Commercial opportunities will change along with exploration and technology development, therefore the connections with the corresponding objectives must be allowed to adapt with them. Table 3.3 contains a matrix of the spacefaring goals, the program objectives, and the primary connections denoted by the circles where the each column and row intersect. With all of the identified spacefaring goals and their connections to the selected program objectives determined, the initial priority of the i^{th} objective is given by

$$\begin{aligned} O_{\mathrm{i,\ priority}} &= \left(\sum_{n=1}^{4} W_{\mathrm{S,\ n}} \cdot CP_{\mathrm{i,\ S,\ n}} \right) W_{\mathrm{S}} \\ &+ \left(\sum_{n=1}^{4} W_{\mathrm{P,\ n}} \cdot CP_{\mathrm{i,\ P,\ n}} \right) W_{\mathrm{P}} \\ &+ \left(\sum_{n=1}^{2} W_{\mathrm{D,\ n}} \cdot CP_{\mathrm{i,\ D,\ n}} \right) W_{\mathrm{D}}\,, \end{aligned} \tag{3.3}$$

where W_{S} is the AHP-derived weight of the Science spacefaring goal and $W_{\mathrm{S,\ n}}$ is the weight of the n^{th} sub-goal of the Science goal. Similarly, W_{P} is the weight of the Pragmatism spacefaring goal, $W_{\mathrm{P,\ n}}$ is the weight of the n^{th} sub-goal of the Pragmatism goal, W_{D} is the weight of the Destiny spacefaring goal, and $W_{\mathrm{D,\ n}}$ is the weight of the n^{th} sub-goal of the Destiny goal. The connection parameter is defined as

$$CP_{\mathrm{i,\ X,\ n}} = \begin{cases} 0 & \text{if goal and objective are not connected} \\ 1 & \text{if a primary connection between goal and objective}\,, \end{cases}$$

for each spacefaring goal ($X = S$, P, or D).

The calculated priorities are further influenced by both S_{man} and by S_{tech}. Objectives that involve manned spaceflight will have their priorities multiplied by a value based on the selected value of S_{man}. Table 3.4 describes the strategy scale for man's involvement in a candidate space program. Likewise, the technology development objectives (included only for illustrative purposes in the prototype) are multiplied by a value determined by the selected value of S_{tech}. These values can be seen in Table 3.5 along with the corresponding NASA Technology Readiness Level (TRL).

These program objectives are presented to the planner in order of the priority calculated with Equation 3.3. This informs the planner about the types of objectives that contribute towards his selected goals and strategies.

Additionally, each operational program objective consists of multiple attributes depending on its type. Single mission objectives like launching a satellite in orbit (O_{101}) or landing men on the moon and

			Spacefaring goals											
			Science W_S				Pragmatism W_P				Destiny W_D			
	Variable	Program Objective	Earth $W_{S,1}$	Solar system $W_{S,2}$	Universe $W_{S,3}$	Human beings $W_{S,4}$	Commercial $W_{P,1}$	Sustaining Earth $W_{P,2}$	Defense $W_{P,3}$	Technology $W_{P,4}$	Exploration $W_{D,1}$	Colonization $W_{D,2}$	**Manned objective** S_{man}	**Technology** S_{tech}
Earth operations	O_{101}	Unmanned satellite	○		○		○		○					
	O_{102}	Manned satellite	○			○	○		○				✖	
	O_{103}	Space station	○			○	○		○			○	✖	
	O_{104}	Power plant					○	○						
	O_{105}	Cis-lunar station		○		○	○		○		○	○	✖	
	O_{106}	Auxiliary vehicles					○	○	○					
	O_{107}	Supplied colony					○	○				○	✖	
Moon/planet operations	O_{201}	Unmanned satellite		○	○						○			
	O_{202}	Manned satellite		○			○				○		✖	
	O_{203}	Unmanned lander		○							○			
	O_{204}	Manned lander		○				○			○	○	✖	
	O_{205}	Orbital station		○		○			○		○	○	✖	
	O_{206}	Surface station		○		○		○	○		○	○	✖	
	O_{207}	Supplied colony					○	○	○		○	○	✖	
	O_{208}	Self-sufficient colony						○				○	✖	
	O_{209}	Solar probe		○	○						○			
	O_{210}	Interstellar probe		○	○						○			
Technology development	O_{301}	Launch propulsion							○	○				✖
	O_{302}	In-space propulsion								○				✖
	O_{303}	Space power and energy storage						○		○		○		✖
	O_{304}	Robotics								○	○			✖
	O_{305}	Communications and navigation		○						○				✖
	O_{306}	Human health, life support	○			○				○		○	✖	✖
	O_{307}	Human destination systems				○		○		○		○	✖	✖
	O_{308}	Science instruments, observatories	○	○	○	○				○				✖
	O_{309}	Entry, descent, and landing systems					○		○	○				✖
	O_{310}	Nanotechnology								○				✖
	O_{311}	Modeling and simulation		○	○					○				✖
	O_{312}	Materials, manufacturing						○		○		○		✖
	O_{313}	Ground and launch systems							○	○				✖
	O_{314}	Thermal management								○		○		✖
	O_{315}	Aeronautics	○							○				✖

○ represents a primary connection, CP_i, between the goal and objective
✖ represents an objective to be multiplied by the corresponding strategy factor

TABLE 3.3 – Selected program objectives and their connections with the stated spacefaring goals.

S_{man}	Strategy description	Multiplier
1	Program with no manned efforts	0
2	Program that prefers unmanned activities	0.5
3	Neutral towards man's involvement	1.0
4	Program in favor of manned activities in space	1.5
5	Program practically requires man's presence in space	2.0

TABLE 3.4 – An objective priority multiplier based on the selected strategy factor, S_{man}

S_{tech}	Strategy description	Multiplier
1	Operational technology (TRL 9)	0.9
2	Existing flight tested technology (TRL 8)	1.0
3	Technology in final development stages (TRL 7)	1.1
4	Demonstrated technology (TRL 6)	1.2
5	Advanced technology under development (TRL 4)	1.4

TABLE 3.5 – Program objective priority multiplier based on the selected strategy factor, S_{tech}. The levels are described along with their Technology Readiness Level (TRL) from NASA [58].

returning them (O_{204}) have two attributes: typical mass range (min and max), and typical launch rate range (min and max).

Installation mission objectives like developing a space station in LEO (O_{103}) or a supplied base on Mars (O_{206}) have three attributes: installation mass range, required re-supply launch rate, and re-supply payload mass range.

The objective attributes are directly affected by both S_{scale} and S_{time}. The minimum and maximum values for each value are based on history, experience, or the desired scope under investigation. Realistic and accurate values are always helpful, but more important than specific accurate numbers is the range that is generated. With a proper analysis sweep, this enables the visualization of the entire solution space, not just a single accurate flight rate for a given mission. So long as the attributes are defined consistently (pessimistic/optimistic predictions, historical ranges, *etc.*), then the resultant feedback and eventual programs generated from the data can be appropriately compared.

S_{time}	Program length
1	21 years
2	18 years
3	15 years
4	12 years
5	9 years

TABLE 3.6 – The schedule strategy factor scale, S_{time}, and the corresponding program lengths.

For the minimum and maximum mass-related attributes (payload ranges, re-supply ranges, and installation mass ranges), the selected scale strategy factor is interpolated between the two values to obtain the expected maximum mass value for a given objective. For example, if the unmanned satellite in LEO objective (O_{101}) specified a typical minimum payload mass of 500 kg and a maximum of 5,000 kg, a scale strategy factor of 1 would result in mission payloads of 500 kg to fulfill that objective; an S_{scale} of 3 would recommend typical O_{101} mission payloads around 2,750 kg.

The launch rate attribute should also be consistently defined. When defining the launch rate objective attributes for the *Ariadne* prototype, a typical space program was assumed to be 15 years long. The typical launch rates for each operational objective were assigned based on the 15 year program length. The actual length is dictated by selected schedule strategy factor, S_{time}. The translation from selected strategy to the length of the program is defined in Table 3.6. So a defined re-supply launch rate for a lunar colony (O_{207}) of 3 per year translates to a recommended 45 missions over the course of the program. The launch rates are interpolated with the scale strategy factor similar to the mass ranges. They are then multiplied by the priority of their program objective. For non re-supply missions, this adjusted launch rate is then multiplied by the selected length of the program, S_{time}, to determine how many missions to recommend to the planner. Re-supply mission flight rates depend only on the scale of the installation and should not be condensed or spread out over the adjusted length of the program.

The final deliverable of this step is a single dashboard of all of the calculated information thus far. The priority of each objective is first calculated with Equation 3.3, adjusted accordingly for its connection with manned efforts and technology developments, and then normalized so that the highest priority objective receives a score of 1.00. With this information, the planner is informed on what missions should be prioritized, how many of each mission should be planned, and how large the payloads should be on each mission. The data contained in a given dashboard can be seen in Table 3.7.

A couple of observations can be made from this prototype dashboard. First, the resultant weights for each tier of spacefaring goals are listed at the top of the dashboard. This provides transparency and an initial sanity check of the prioritized program objectives. The goals in this example program stressed the importance of defense and exploration. The selected implementation strategies are also given, reminding the planner of his selected program length, scale, *etc.*

Second, there is an apparent upper tier of objectives with a normalized score between 0.93-1.00. Two of these are technology developments (desired by a defensive space program) and are not considered for the prototype. The other two objectives, a space station in LEO and a manned Earth satellite, are strongly recommended to be included in any candidate program, based on the input set of goals and strategies. From here, the next set of operational objectives are all fairly close around 0.60. A planner could assemble a program that includes each objective or assemble multiple candidates that pursue alternative objectives from here. The dashboard represents a starting point, providing insight into the connections between the goals, strategies, and objectives, but should not restrict the possibilities to be analyzed.

Weighted spacefaring goals

Science – 0.063	**Pragmatism – 0.672**	**Destiny – 0.265**
Universe – 0.047	Defense – 0.582	Exploration – 0.875
Human – 0.586	Technology – 0.312	Colonization – 0.125
Earth – 0.285	Sustain Earth – 0.068	
Solar system – 0.081	Commercial – 0.038	

Selected implementation strategies

Tech level the program, $S_{tech} = 4$
Man's involvement, $S_{man} = 2$
Scale of the program, $S_{scale} = 3$
Pace of the program, $S_{time} = 3$

Prioritized space program objectives

ID	Objective	Priority	Missions	Max payload [kg]	Installed mass [kg]
O_{103}	Earth space station	**1.00**	18	4,750	77,500
O_{301}	Launch propulsion	**0.95**			
O_{313}	Ground systems	**0.95**			
O_{102}	Earth sat. (manned)	**0.93**	17	4,125	–
O_{307}	Human exploration systems	**0.78**			
O_{306}	Human health	**0.71**			
O_{206}	Planet/moon surface base	**0.70**	5	4,250	755,000
O_{304}	Robotics	**0.70**			
O_{207}	Planet/moon supplied colony	**0.67**	5	40,000	4,000,000
O_{105}	Cis-lunar space station	**0.66**	12	5,000	155,000
O_{204}	Planet/moon lander (manned)	**0.62**	5	8,250	–
O_{106}	Earth orbital aux. vehicle	**0.61**	–	–	3,775
O_{205}	Planet/moon space station	**0.61**	5	4,000	378,750
O_{101}	Earth sat.	**0.58**	72	2,500	–
O_{202}	Planet/moon sat. (manned)	**0.51**	2	9,000	–
O_{303}	Power generation	**0.46**			
O_{312}	Materials	**0.46**			
O_{314}	Thermal	**0.38**			
O_{309}	EDL systems	**0.37**			
O_{315}	Aeronautics	**0.36**			
O_{311}	Modeling/simulation	**0.34**			
O_{305}	Communications	**0.34**			
O_{302}	In-space propulsion	**0.33**			
O_{310}	Nano	**0.33**			
O_{201}	Planet/moon sat.	**0.32**	14	875	–
O_{209}	Solar probe	**0.33**	3	1,137	–
O_{210}	Interstellar probe	**0.33**	1	1,150	–
O_{203}	Planet/moon lander	**0.32**	7	2,500	–
O_{107}	Earth orbital colony	**0.21**	11	8,250	4,000,000
O_{208}	Planet/moon sufficient colony	**0.16**	–	–	8,750,000
O_{308}	Science instruments	**0.11**			
O_{104}	Earth orbital power plant	**0.09**	–	–	15,500

TABLE 3.7 – An example prototype dashboard to inform the program planner.

Mission Phase	Velocity required [km/s]
Earth to LEO (185 km)	9.45
LEO to GEO (35,800 km)	4.95
LEO to lunar flyby	2.94
LEO to lunar orbit	3.88
LEO to lunar landing (soft, no return)	6.43
LEO to lunar landing and return	11.58
LEO to Mars flyby	3.61

TABLE 3.8 – Velocity requirements of modeled phases for the *Ariadne* prototype. Compiled and adapted from [60–64]

3.3.4 *Mission architecture modeling and the sizing of in-space elements*

For the *Ariadne* prototype, the method for modeling mission architectures is kept as basic as possible. Each mission is divided into distinct phases,[11] assembling the architecture in a way that can be easily analyzed with the Tsiolkovsky rocket equation and mass ratio design process. The first step in the architecture modeling process, a specified velocity requirement, ΔV, must be determined for each phase. For this prototype, these velocity requirements are based on conservative estimates for similar maneuvers and not an actual calculated trajectories that require a specific launch date, the positions of the planetary bodies, *etc.* These velocity requirement have been assembled from various sources and are listed in Table 3.8.

[11] Commonly referred to phases include the trans-Mars injection (TMI) and lunar orbit insertion (LOI).

The basic mentality of J. Houbolt[12] was followed on the selection of the velocity requirements. He said:

[12] Houbolt was the champion of the lunar-orbit rendezvous mission architecture used during Project Apollo and will be discussed more in Chapter 4

> It should be noted that a conservative approach has been taken in defining the velocity increments used in this study. In general, non-optimum conditions have been assumed for each phase. While this approach tends to penalize the results somewhat, it is considered to be the logical approach to a parametric study and contributes to the confidence in the results obtained. [59]

In addition to the velocity requirements, two other inputs are required to enable the sizing of in-space elements: propulsion performance (I_{sp}), and the structure factor (ε). Once these are defined, the IMLEO sizing process begins with the final mission payload at its destination (actually, just before its final destination for the purposes of the prototype).

Beginning with the final mission payload, for example a satellite in geo-stationary orbit (GEO), the process applies the rocket equation shown in Equation 3.4, along with the known values for ΔV, g (gravitational constant), and the I_{sp} to solve for the mass ratio, MR, of the phase as seen in Equation 3.5:

$$\Delta V = g I_{sp} \ln (MR) = g I_{sp} \ln \left(\frac{m_0}{m_f} \right), \tag{3.4}$$

$$MR = e^{\left(\frac{\Delta V}{g I_{sp}}\right)}. \tag{3.5}$$

Since $MR = m_0/m_f$, and the final mass at the end of the phase is the payload placed in orbit, simply multiply the mass ratio and final mass together to determine the initial mass at the beginning of the phase which, in this case, is equal to IMLEO. Other architectures might involve more phases but the process is the same: simply start with the final mission payload and solve for the initial mass at the beginning of that phase. This process is repeated for each subsequent phase, using the output initial mass from the last phase as the input final mass for this phase.

Hammond has this to say about the use of similar equations to determine the final required in-space mass:

> The preceding close-form impulsive relations are only a preliminary estimating device, but should yield gross liftoff weights within 10-15% of the actuals, depending upon the experience of the engineer in selecting mission velocity requirement and estimating realistically achievable mass fractions. [65]

Additionally, with an assumed structure factor (ε) input for each phase, it is also possible to determine the individual propellant and structure mass for each phase. The initial mass, m_0, can be defined as

$$m_0 = m_{str} + m_p + m_{payload}\,, \tag{3.6}$$

where m_{str} is the structure mass of the stage, m_p is the propellant mass and $m_{payload}$ is the mass of the phase payload. The final mass, m_f, is defined as

$$m_f = m_0 - m_p = m_{str} + m_{payload}\,, \tag{3.7}$$

The stage structure factor, ε, is an assumed input typically between 0.05 and 0.15 given by

$$\varepsilon = \frac{m_{str}}{m_{str} + m_p}. \tag{3.8}$$

With this equation, it is now possible to solve for the propellant and structure mass required for each phase.

$$m_{str} = \frac{m_p \varepsilon}{(1-\varepsilon)} \tag{3.9}$$

$$m_p = \frac{m_{payload}(MR-1)}{1 - \frac{MR\varepsilon}{(1-\varepsilon)} + \frac{\varepsilon}{(1-\varepsilon)}} \tag{3.10}$$

These parameters in Equations 3.9 and 3.10 provide the details of the mass of a spacecraft (primarily its propellant required) during a phase and the process can be used to determine the total IMLEO required for a given mission.

3.3.5 *Sizing the launch vehicle*

Once each mission has been reduced down to its IMLEO, a launch vehicle must be designed capable of delivering that mass to orbit. As one of the primary drivers on both the cost and schedule of a program, the launch vehicle requirements need to be easily comparable with a minimum of information needed. The launch vehicle sizing process of the Space Planners Guide [36] provides that exact capability.

The Space Planners Guide, also known as simply *the Guide*, is a handbook method developed by the U.S. Air Force in the midst of the Space Race. It states that its purpose is to "...provide a rapid means of both generating and evaluating space mission concepts [36]." The U.S. and the Soviet Union were each attempting to be the first to achieve increasingly difficult milestones in space in an effort to prove their superiority.

Speed and first-order accuracy were critical to the Air Force in devising and ruling out concepts early on before countless resources were invested pursuing false leads. They developed a handbook (the Guide) that allowed the user to step through nomograph-style figures in order to make their estimates. This forgiving approach hides many of the second-order details from the planner, and allows them to see only the effects of the primary drivers. The stated accuracy of the Guide is to be within ± 20 %, which is sufficient for ruling out less feasible options.

The author has conducted multiple studies with the Space Planners Guide [66, 67] and, although some felt it was unable to live up to its lofty goals [68], the Guide does in fact succeed in providing an excellent sizing methodology for expendable launch vehicles which is ideal for the *Ariadne* prototype.

This section will discuss Figure 3.27 which is a N-S structogram representing the entire launch vehicle sizing process used in the Space Planners Guide. The process involves three separate sections: (1) the initial estimates of total and stage weights, (2) the iteration of stage mass ratios in pursuit of velocity capability convergence, and (3) the iteration of stage structure factors until the optimized stage weights can be determined. For the sake of brevity, each nomograph is not reproduced in this section.[13] The reader is encouraged to follow along in chapter V of the Space Planners Guide. The process requires some basic inputs, many of which can be estimated from other methods found in the Guide if they are unknown to the planner: the mass of the payload, the number of stages, and some of the propulsion properties of each stage (specific impulse[14] and propellant density). Typical values for the specific impulses and densities for many of the common propellants are provided by the Guide and can also be found compiled from

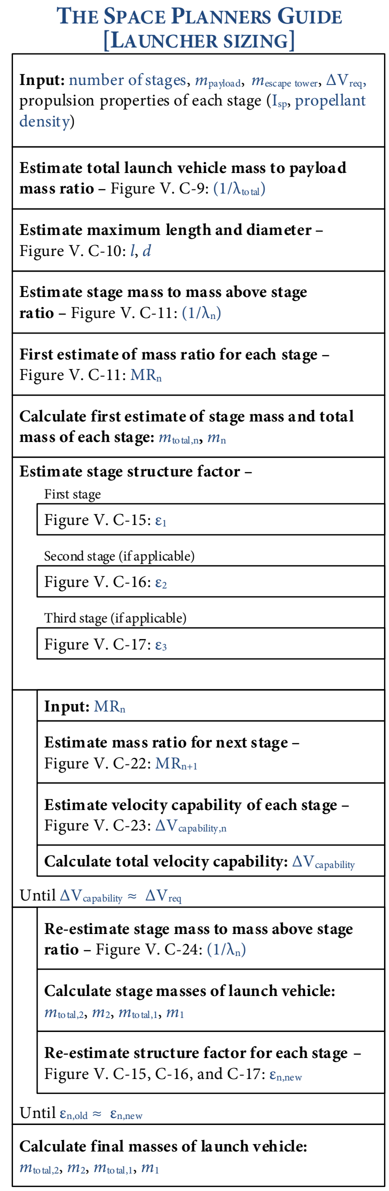


FIGURE 3.27 – N-S representation of the launch vehicle sizing process from the Space Planners Guide [36].

[13] They can, however, be found reproduced in the next chapter for the Saturn IB case study.

[14] use the vacuum rating value

Propllant combinations	Mixture ratio	Density [kg/m^3]	Specific impulse [s]
LH_2-LOX	6.0	360	381
	4.0	281	389
LOX-RP1	2.6	1029	300
	2.4	1023	299
LOX-Methane	3.0	801	308
LOX-UDMH	1.7	977	210
N_2O_4-Hydrazine	1.4	1082	291
N_2O_4-UDMH	2.6	1013	285

Compiled from the Space Planners Guide [36], Sutton [60], and Sforza [69]

TABLE 3.9 – Typical properties of different propellant combinations.

other sources in Table 3.9. The methods of the Guide primarily take the form of nomographs as seen in Figure 3.28. Each nomograph used in the process has been digitized to allow automation and much faster application. This software implementation will be briefly discussed in Section 3.4.

» First section

Following along with Figure 3.27, the launch vehicle sizing structogram, the planner first seeks to obtain a rough estimate of the total launch vehicle mass with nomograph V.C-9 and the average specific impulse of all the stages and the number of stages.[15] The nomograph output is an estimate of the ratio of the launch vehicle total mass to payload mass, which can be used to find the initial guess of the launch vehicle's overall mass. This ratio is the inverse of a commonly used variable know as the payload fraction, λ_{total}, and will be expressed as $1/\lambda_{total}$ for the sake of consistency. This is shown in Equation 3.11:

$$m_{total} = m_{payload} \times \left(\frac{1}{\lambda_{total}} \right) , \tag{3.11}$$

where m_{total} is the total lift off mass of the launch vehicle, $m_{payload}$ is the payload mass and the inverse of the payload ratio, $1/\lambda_{total}$, is the output of nomograph V.C-9.

With an estimate of the total launch mass, the planner can then use nomograph V.C-10 to define the outer limits for the basic dimensions of the launcher.[16] The nomograph provides an estimate for the maximum length and diameter based on the total launch mass. These dimensions are not used elsewhere in the process but could serve as initial constraints on the launch site, manufacturing, support logistics, etc.

Continuing with the process, the inverse of the total payload ratio, along with the number of stages, are used with nomograph V.C-11 to determine the first estimate for the ratio of the stage mass to the mass above the stage.[17] This again, is the inverse of the stage's payload

[15] reproduced in Chapter 4, Figure 4.4

[16] reproduced in Chapter 4, Figure 4.5

[17] reproduced in Chapter 4, Figure 4.6

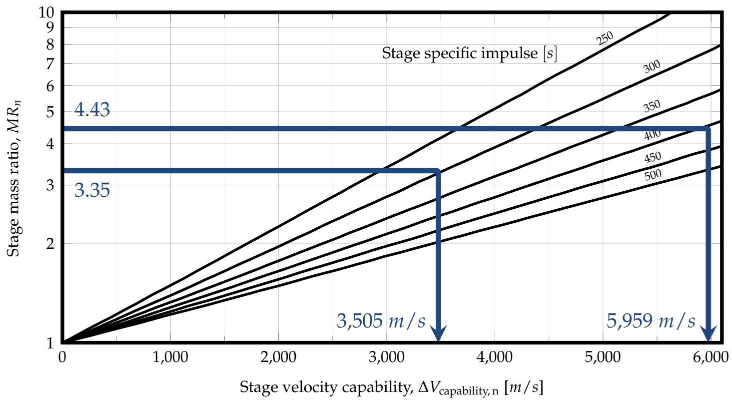


FIGURE 3.28 – Example of a nomograph method from the Guide [36].

ratio, $1/\lambda_n$. For the initial estimate, this value is considered the same for each stage but this will be optimized later in the process.

Similarly, this stage inverse payload ratio, the output of nomograph V.C-11 is used with nomograph V.C-12 to obtain the first estimate for the mass ratio[18] of each stage (again considered equal for now, but will be optimized later).[19]

Using all the output values from nomographs V.C-9 through V.C-12, the planner can now determine the first estimate of the stage masses. Simply begin with the payload mass and multiply it by the ratio obtained from nomograph V.C-11, $1/\lambda_n$. The Guide refers to this mass as the *gross weight* of the stage, which includes the mass of the stage (both propellant and structure) and the mass above the stage though from here on out it will be referred to as the *total mass*. In order to find just the mass of the stage, the planner can subtract the mass above the stage from the stage total mass. This process can be seen for a two stage launch vehicle in the following equations:

$$m_{total,2} = m_{payload} \times \left(\frac{1}{\lambda_2}\right), \tag{3.12}$$

$$m_2 = m_{total,2} - m_{payload}, \tag{3.13}$$

$$m_{total,1} = m_{total,2} \times \left(\frac{1}{\lambda_1}\right), \tag{3.14}$$

$$m_1 = m_{total,1} - m_{total,2}, \tag{3.15}$$

where $m_{total,n}$ is the total mass of the n_{th} stage (including the mass above it), m_n is the stage mass of the n_{th} stage (without the mass above it), and $1/\lambda_n$ is the stage mass to mass above ratio that was found in nomograph V.C-11.[20]

The planner can repeat this process for each stage, starting with the last stage (second or third, depending on the design) until he arrives at the final launch mass, which should be similar[21] to the total launch

[18] where the mass ratio is defined as the ratio of the initial mass over the final mass

[19] reproduced in Chapter 4, Figure 4.7

[20] Remember that $1/\lambda_1 = 1/\lambda_2$ for this initial estimate

[21] Opposing methods from the Guide may often differ slightly when the values are expected to be identical. This is completely normal and aligns perfectly with the philosophy of the Guide.

mass found with nomograph V.C-9.

The final portion of this initial estimate involves the recently determined stage masses and propellant density to estimate the stage's structure factor. The Space Planners Guide provides nomographs for stages one, two, and three for both liquid or solid propellants.[22] The Guide assumes an initial thrust-to-weight ratio for stages one, two, and three of 1.3, 1.0, and 0.75, respectively.

With the estimated structure factors, the structure and propellant masses can be determined for each stage using Equations 3.16 and 3.17:

$$m_{structure} = m_n \times \varepsilon_n \,, \tag{3.16}$$

$$m_{propellant} = m_n \times (1 - \varepsilon_n) \,. \tag{3.17}$$

[22] For the development of the prototype, the author has assumed all liquid stages so only nomographs V.C-15 through V.C-17 have been digitized. See reproduction for the first and second stages in Figures 4.8 and 4.9

» SECOND SECTION

The next portion of the sizing process involves the iteration of the mass ratios of each stage and the determination of the velocity contribution they are capable of providing. The velocity of each stage is then summed and compared to the total velocity required.

Nomograph C.V-22 is used to determine the mass ratio of each stage.[23] The planner enters the nomograph with a guess for the mass ratio of the first stage (from the previous estimation for the first iteration). The structure factor and specific impulse of the stage are used, along with the structure factor and specific impulse of the next stage to determine the stage mass ratio of that stage. This can be repeated for a third stage by entering the nomograph with the second mass ratio to determine the third.

[23] reproduced in Chapter 4, Figure 4.10

The mass ratio and specific impulse for each stage is then used with nomograph V.C-23 to estimate the velocity contribution of the stage.[24] Velocities are summed, then compared to the total velocity required. If the velocity capability is too low, the planner increases their guess for the mass ratio of the first stage and returns to nomograph V.C-22. If the velocity capability is too great, the guess is decreased. Continue until the velocity capability converges with the velocity required.

[24] reproduced in Chapter 4, Figure 4.11

» THIRD SECTION

Once the velocity capabilities have have converged[25] with those required, the stage mass ratios are passed into nomograph V.C-24 along with the structure factor of the stage to determine the new estimate for the stage mass to mass above the stage ratio.[26] These ratios can then be used like they were in the first section of this process (see Equations 3.12 – 3.15) to determine the individual stage masses, this time with the optimized ratios, starting with the last stage. The individual stage masses are used again with nomographs V.C-15 through V.C-17 to arrive at a new value for the stage's structure factor. The planner

[25] The Guide stresses not to pursue extreme accuracy in matching these values, "since the accuracy of the solution is already limited by the approximation of the input data [36]." Since it is digitized and iterations can happen very quickly now, the author's prototype does require a fairly close match, but it is important to remember the amount of approximations already involved.

[26] reproduced in Chapter 4, Figure 4.12

then compares these new structure factors with the original. If the difference is too large, the planner re-enters nomograph V.C-24 with the stage mass ratios and the new structure factors for each stage. The stage masses are calculated again, followed by the stage structure factors. This continues until the new and old structure factor values have converged.

Once they have converged, the planner has successfully sized the required launch vehicle for his given mission. For further discussion and validation of this process, the reader is encouraged to read through Appendix C which contains a comparison of the Guide's sizing process with eight different launch vehicles including past, present and even future hardware.

3.3.6 Launch vehicle cost estimation

With the sizing data for each launch vehicle, it is then desired to be able to estimate the required costs for the development and production of the required number of launch vehicles for each program. In order to make these estimates, D. E. Koelle's *Handbook of Cost Engineering and Design of Space Transportation Systems with TransCost 8.2 Model Description* [70] have been applied. Based on his sales worldwide, Koelle states that "...this seems to be the most widely used tool for cost estimation and cost engineering in the space transportation area [70]." *TransCost* was also used in the beginning of the joint DARPA-NASA effort, *Horizontal Launch Study* [71, 72].[27]

Suppose one of you wants to build a tower. Won't you first sit down and estimate the cost to see if you have enough money to complete it? For if you lay the foundation and are not able to finish it, everyone who sees it will ridicule you...

» Luke 14:28-29 *NIV*

[27] It was eventually replaced with NAFCOM due to its ability to estimate DDT&E and first unit costs at the subsystem level, which was an very important for the study, but not necessary for the *Ariadne* prototype.

The handbook provides several different types of cost estimates: development costs, production costs, ground and operations costs, and complete life cycle costs. For the prototype and the amount of launch vehicle data that is available from the sizing process, only the development and production costs are calculated using cost estimating relationships (CER).

Koelle reports all of his estimated costs in the unit 'work-years' (WYr) which is defined as "the total company annual budget divided by the number of productive full-time people [70]." These values can be converted to actual currency with a conversion factor that Koelle lists per year and per region. The latest conversion factor is for the year 2015, so all reported costs will be in \$2015. The conversion factor is given as 337,100 [$\$/WYr$].

Koelle defines and uses 12 different cost factors, $f_0 - f_{11}$ that are combined with the CERs to estimate the required costs for engines and vehicles. For the prototype, as the objective is to make comparisons at the top level goals and strategies, many of these factors can be ignored. They become critical when comparing alternative launch methods, which would be included in the ideal solution concept, so they are not forgotten.

The factors that have been included in the *Ariadne* prototype are listed below:

» f_0 – project systems engineering and integration factor;

» f_1 – technical development standard correlation factor;

» f_2 – technical quality correlation factor;

» f_6 – cost growth factor for deviation from the optimum schedule.

These factors and the CERs for both engines and stages are discussed below.

» Development costs

To estimate the total development cost for a launch vehicle, the following equation is used:

$$\text{Total development cost} = f_0 \left(\sum \text{Engine development} + \sum \text{Vehicle development} \right) f_6\, f_7\, f_8\,, \tag{3.18}$$

where f_0 is the systems engineering factor based on the number of stages (see below), f_6 is cost growth based on schedule. The factors f_7 and f_8 are assumed to be equal to 1.0 for the purposes of the prototype. The systems engineering factor for development is given as

$$f_0 = 1.04^N\,, \tag{3.19}$$

where N represents the number of stages for the launch vehicle.

S_{tech}	**Description**	f_1
1	Variation of an existing project	0.3 – 0.5
2	Design modifications of existing systems	0.6 – 0.8
3	Standard projects, state-of-the-art	0.9 – 1.1
4	New design with some new technical and/or operational features	1.1 – 1.2
5	First generation system, new concept approach, involving new techniques and new technologies	1.3 – 1.4

TABLE 3.10 – Connection between *Ariadne* technology strategy factor and *TransCost* development standard factor. *TransCost* descriptions reproduced from Koelle [70].

The factor f_1 is development standard factor. Its standard values and their connections to the *Ariadne* technology strategy factor are provided in Table 3.10.

The factor f_2 is defined separately for each system and will be discussed when applicable.

The factor f_6 is based on the schedule of development. Typical development schedules based on the total number of WYr of the launch vehicle are suggested by Koelle. For the purposes of the prototype:

» the assumed development time for launch vehicle under 15,000 WYr is 6 years;

» between 15,000 and 35,000 WYr is 7 years;

» and over 35,000 WYr is 8 years.

These ideal values are compared with an assumed actual value for the development of each launch vehicle (based on the S_{time} strategy factor). The relative difference between the two is used along with a digitized figure by Koelle in order to determine the value of f_6 for each launch vehicle.

The development costs for pump-fed rocket engines is given by

$$\text{Liquid engine development costs} = 277\, m^{0.48}\, f_1\, f_2\, f_3\, f_8\,, \tag{3.20}$$

where m is the mass of the engine, f_2 is based on the number of test fires (and can be assumed equal to 1.0 for now), f_3 and f_8 are not applicable and also assumed to be 1.0. To determine the mass of the required engine, the total thrust of each stage from the sizing output is split by a selected number of engines. Koelle provides a figure based on a database of rocket engines that allows the user to estimate the engine's mass for use in Equation 3.20.

The development costs for expendable vehicle stages is given by

$$\text{Expendable stage} = 98.5\, m^{0.555}\, f_1\, f_2\, f_3\, f_8\, f_{10}\, f_{11}\,, \tag{3.21}$$

where m is the stage mass, f_2 is based on vehicle net mass fraction (NMF) defined below, f_{10} and f_{11} can be assumed equal to 1.0. Koelle defines the NMF as:

$$NMF = \frac{m_{stage,dry} - m_{engines}}{m_{propellant}}\,. \tag{3.22}$$

The equation for f_2 for expendable stages is then given by

$$f_2 = \frac{NMF_{\text{ref}}}{NMF_{\text{eff}}}\,, \tag{3.23}$$

where NMF_{ref} is the average NMF value based on a database of launch vehicles for a given propellant mass, and NMF_{eff} is the calculated NMF value based on the sized launch vehicle stage and engine masses.

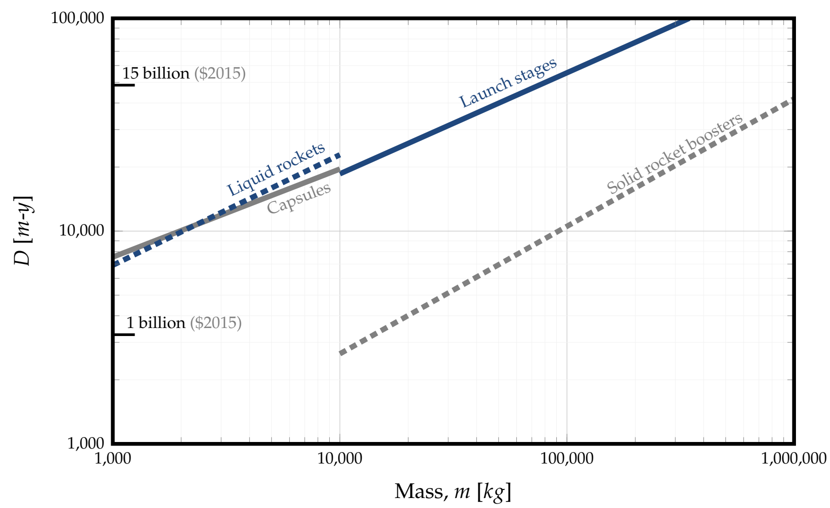


FIGURE 3.29 – Summary figure of the development costs for elements of a space transportation system. Based on Koelle's *TransCost* [70] and reproduced from Sforza [73].

Koelle's handbook contains many other CERs applicable to a wide range of vehicle types, but only expendable launchers with liquid engines have been implemented for the prototype. The plot for each of these CERs for the estimation of development costs has been compiled by Sforza [73] and reproduced by the author in Figure 3.29.

» Production costs

To estimate the total production cost for a launch vehicle, the following equation is used

$$\text{Total production costs} = f_0^N \left(\sum_1^n \text{Stage production} + \sum_1^n \text{Engine production} \right) f_9 \,, \qquad (3.24)$$

where N is the number of stages or system elements, n is the number of identical units per element, and f_9 is not applicable to the prototype and assumed to be 1.0. Again, the prototype only involves expendable stages and liquid rocket engines. The production costs for liquid rocket engines is given by the following two equations:

$$(Cryo)\ \text{Engine production costs} = 3.15\, m^{0.535}\, f_4\, f_8\, f_{10}\, f_{11} \,, \qquad (3.25)$$

$$(Storable)\ \text{Engine production costs} = 1.9\, m^{0.535}\, f_4\, f_8\, f_{10}\, f_{11} \,, \qquad (3.26)$$

where m is the engine mass, and the only other variable that has not been discussed is f_4, the cost reduction factor for series production. This factor is not applied in the prototype although it is a promising candidate for future work.[28]

[28] see Section 5.2 of Chapter 5

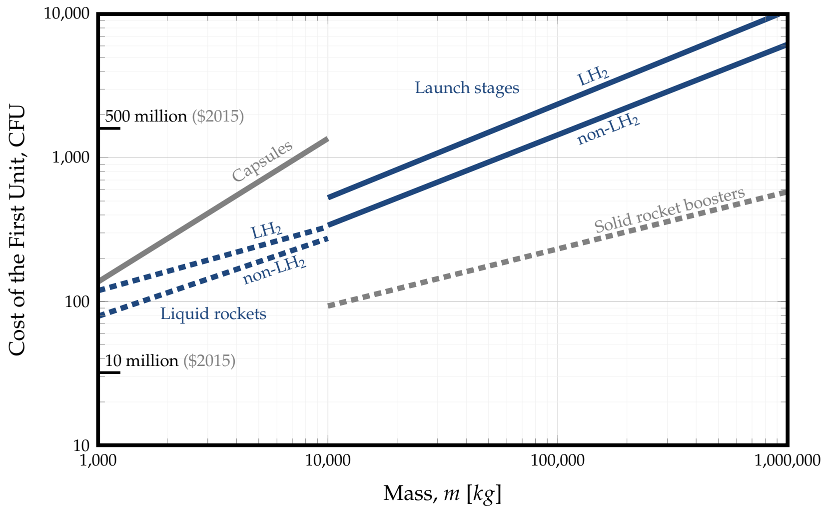


FIGURE 3.30 – Summary figure of the production costs for elements of a space transportation system. Based on Koelle's *TransCost* [70] and reproduced from Sforza [73].

The CERs for expendable stages are given by the following two equations, also divided by cryogenic propellants and storable propellants:

$$(Cryo)\ \text{Stage production costs} = 1.84\, m^{0.59}\, f_4\, f_8\, f_{10}\, f_{11}\,, \tag{3.27}$$

$$(Storable)\ \text{Stage production costs} = 1.265\, m^{0.59}\, f_4\, f_8\, f_{10}\, f_{11}\,, \tag{3.28}$$

The plot for the production costs for these stages and engines, along with other types of systems can be seen in Figure 3.30.

With these CERs in place, the *Ariadne* prototype system is now able to calculate both development and production costs of the fleet of launch vehicles required to place the program IMLEO into orbit. This process is key to the program convergence loop as previously discussed and shown in Figure 3.17.

3.4 *Software implementation*

The *Ariadne* prototype methodology has been developed in the Python 3.6 programming language [74] to demonstrate execution and contribution. This section discusses the overall logic, file structure, required inputs and expected outputs of the software.

3.4.1 *Top-level walk-through of the Ariadne script*

The overall flow of data and the required files for the *Ariadne* prototype can be seen in Figure 3.32. The process is visualized parallel with the

```
ariadne/
├── ahp.py
├── ariadne.py
├── goals_objectives.xlsx
├── II_goals.py
├── III_strategies.py
├── IV_mission_architectures.py
├── program_objectives.py
├── run_default/
│   ├── histogram_01.csv
│   ├── input_candidate_01.py
│   ├── input_goals_01.py
│   ├── input_strategies_01.py
│   ├── output_dashboard_01.txt
│   ├── output_final_01.txt
│   ├── output_LV_01.csv
│   ├── output_LV_01.pdf
│   ├── input_goals_02.py
│   ├── ...
│   └── program_comparison.csv
├── space_planners_guide.py
├── transcost.py
└── V_launcher_sizing.py
```

FIGURE 3.31 – The file directory of the *Ariadne* prototype software implementation

N-S representation previously discussed to emphasize that the overall logic is still the same.

The prototype software is a script that is run from the command line. When starting a new project, several separate runs of the script will be required to set up the initial input files before a full analysis of a program or multiple programs can be run. This inconvenience in execution is justified by an increased level of flexibility and control over the inputs for the user. When a user is more familiar with the program, they should be able to minimize the number of repeated runs.

When in the root of the `ariadne` directory (see Figure 3.31), the primary driver file, `ariadne.py`, can be executed with the following command:

```
~/ariadne/python ariadne.py
```

From here, the script begins to run and asks the user to input a working directory for the program:

```
=====================================================
          -| running: [ariadne.py] v0.3 |-
=====================================================

input name of working directory: run_default
```

The default directory name is `run_default`[29] but the user is free to select any name they would like for organizational purposes. If the directory does not exist (*i.e.*, the start of a new project), the user can request that the directory be created.

[29] the field is auto-populated when run on a Linux system, but unfortunately remains blank when run on Windows.

Once in the new working directory, the script checks for existing candidate programs that are ready to be analyzed and reports to the user:

```
checking ''./run_default'' for input files for
  candidate programs

---> 0 candidate program(s) exist
```

Each candidate requires three input files:

» `input_goals_xx.py`;

» `input_strategies_xx.py`;

» `input_candidate_xx.py`.

The `xx` designation is the ID number of the candidate.[30] The script

[30] Only `input_candidate_XX.py` is technically required by the script to proceed to analysis. Since each candidate input file should only be created from the previous inputs, both goals and strategies are listed as required as well.

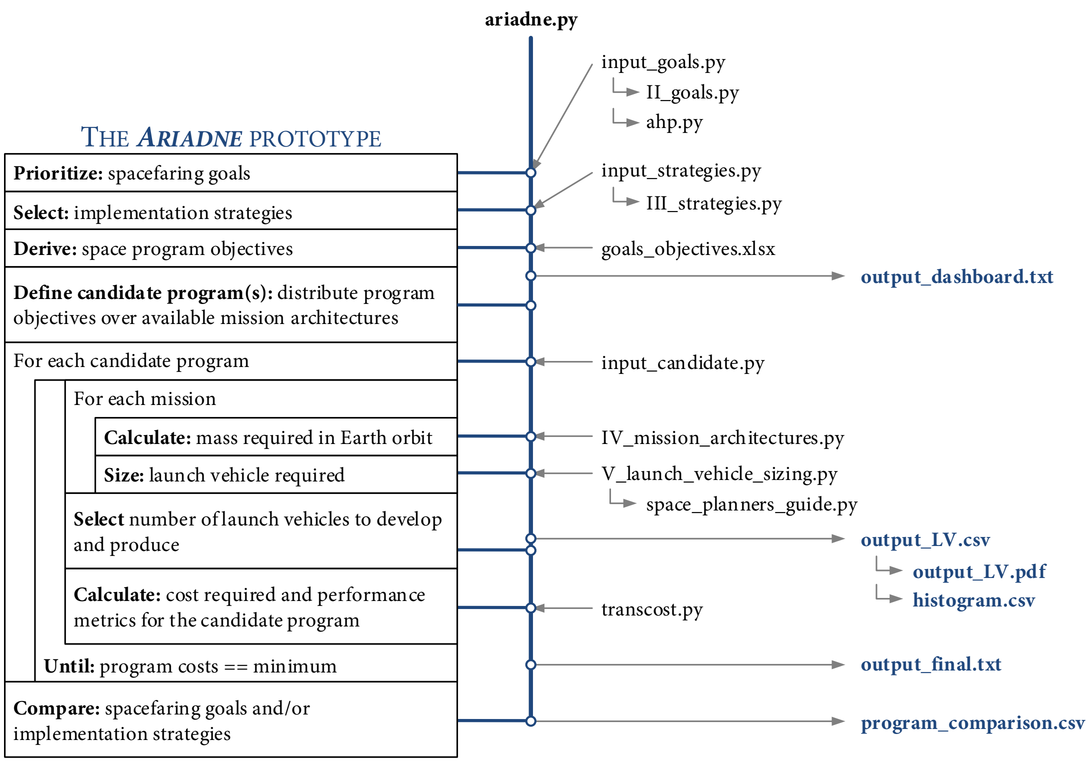


FIGURE 3.32 – Overview of the software implementation of *Ariadne* parallel with the original N-S representation of its overall logic

reports back the number of completed candidate programs currently available in the selected directory.

If the user opts to create either the first or an additional candidate and the file `input_goals_xx.py` is missing, the script will generate a template file and name it the next available ID number. The script then exits, allowing the user to edit the newly created template to properly input the desired spacefaring goals. The input files and their templates are discussed in Section 3.4.2.

Once the user has followed the directions in the template file for `input_goals_xx.py`, the main script is executed again, the proper directory selected, and the user is prompted to see if they would like to continue working on the previous candidate:

```
would you like to continue the one in progress? [y/n]
```

The script will read in the user's goals and compare them using the methods found in `ahp.py`. Any inconsistencies will be reported to the user and the script will exit to allow the user to adjust their input comparisons.

If consistent, the script will continue and check for the selected strategies, `input_strategies_xx.py`. As before, if it is not found, the script can generate a appropriately numbered template and exit to allow the user to modify it with their desired implementation strategies.

When `ariadne.py` is run again with both goal and strategy inputs completed[31] the the script loads an Excel file: `goals_objectives.xlsx`. This spreadsheet contains the primary connections between the spacefaring goals and program objectives as shown previously in Table 3.3. It also contains the remaining information (typical launch rates, payload masses, and installation masses) required to generate a preliminary dashboard to inform the user of recommended program objectives. The dashboard is generated and saved to a text file named `output_dashboard_xx.txt` in the selected directory. The user is then asked if they would like to generate a template for the candidate program:

[31] and the user again opting to continue the analysis in progress

```
no file ''input_candidate_xx.py'' exist.
  create template? [y/n]
```

The user should select to create a template for an appropriate input file for the candidate, `input_candidate_xx.py`. The dashboard priorities and other objective attributes are included at the top of this template in the comments. This information is unique to this candidate and should be used to inform the user's selections of launch rates and payload masses. The user should edit the primary code in this file to assign the desired number of launches and payloads to each available destination. The available modeled mission architectures can be found in `IV_mission_architectures.py`. Once `input_candidate_xx.py` has been appropriately filled out by the user, this candidate program is ready to be analyzed. Running the main script again will produce:

```
---> 1 candidate program exists
would you like to start creating a new
  candidate program? [y/n]
```

The user can select yes and repeat this process to create additional program candidates for analysis as many times as desired. A possible analysis sweep might include using the same consistent set of spacefaring goals (`input_goals_xx.py`) and adjusting the implementation strategies (`input_strategies_xx.py`) for each candidate to see the effects on the overall program. Any desired sweep must be set up manually.

When the desired number of candidate programs have been assembled, a complete analysis of each candidate program will be executed:

```
====================================================
            -| candidate program 01 |-
====================================================

sizing N launch vehicles
[#########-----------------------------------------]
```

A launch vehicle is sized for each mission in the assembled candidate program. The script saves multiple files after this step for additional transparency and troubleshooting:

» `output_LV_xx.csv`;

» `output_LV_xx.pdf`;

» `output_histogram_xx.csv`.

Again, this data is not required during the run and only saved for debugging and plotting purposes. Next, the script proceeds to determine the most cost-effective number of launch vehicles to develop. The largest required launcher is selected to be developed and produced to complete every mission in the program. The module `transcost.py` is used to determine the total launch vehicle costs (both development and production) for the resultant candidate program.

The total launch vehicle cost of the program is stored[32] and the process is repeated again, this time using two launch vehicles. A clustering algorithm is used to group the required launchers into two separate groups, and the maximum of each group is designated to be developed and produced for all of the required launches in its respective group. The final program costs are again calculated and then compared with the costs of the previous program (where only a single launcher was developed). If the costs are lower, the new value is stored and the process repeated again with three launch vehicles. This continues until the total costs begin to increase. At that point, the previously stored program and costs are saved to `output_final_xx.txt` and the script continues to the next candidate program. Listed below is an example of this convergence run for `input_candidate_01.py`:

[32] Technically this cost is first compared against a very large initial cost to produce a negative change in cost. This requirement for the change in cost to be negative is the convergence logic for the iteration.

```
previous run costs with 0 launcher(s) developed:
  1e+16
current run costs with 1 launcher(s) developed:
  1e+11
delta_cost ---> [-]

previous run costs with 1 launcher(s) developed:
  1e+11
current run costs with 2 launcher(s) developed:
  3.6e+10
delta_cost ---> [-]

previous run costs with 2 launcher(s) developed:
  3.6e+10
current run costs with 3 launcher(s) developed:
```

```
  3.7e+10
delta_cost ---> [+]

final total program costs: 36.18 $B

---> selected launch vehicles:
  13 x    44,208 [kg]
   2 x 1,047,258 [kg]

saving program output to ‘‘output_final_01.txt’’
```

This process is repeated for each completed candidate program in the working directory. When every candidate has been run, a master output file is generated with all of the data[33] from each program. All of this data is stored in a single, comma-separated file named `program_comparison.csv` and made available to the user for analysis. Once that final output is saved, the `ariadne.py` script is complete.

[33] goals, selected strategies, number of missions, mass required in LEO, number of launchers selected for development, cost, *etc.*

3.4.2 *Required input files*

The previous section walked through the `ariadne.py` script and introduced three input files that would be required for the analysis of any candidate program to begin. A template for each of the three input files is included here. The comments and representative values in each template serve to inform the user on the file's use and how to properly modify the template to meet the needs of the user.

The first required file is `input_goals_xx.py`. This file handles the input pairwise comparisons between each group of goals previously introduced in the hierarchy. The template for this file is shown below.

Listing 3.1 –
Template file for `input_goals_xx.py`

```
#!/usr/bin/env python3.6
"""
input_goals_01.py

author: Doug Coley

---> template file

this file is automatically generated when needed by the ariadne.py script

it represents a arbitrary, but consistent, set of spacefaring goals for
debugging purposes

the user should follow the guide and make the required pairwise comparisons
between the specified goals

further definitions and documentation can be found in both Chapter 2
```

```
of my dissertation and in the Appendix
"""

def pw_comp():
    """
    handles the initial values for the pairwise comparisons
        ---> see AHP for details

    the following pairwise comparisons should be scored as follows:
    for comparison A_vs_B, imagine a number line between the two
    options

    <A> [-9]---[-7]---[-5]---[-3]---[1]---[3]---[5]---[7]---[9] <B>

    A_vs_B = 3  represents a slight priority for option B over A
    A_vs_B = -7 represents a major priority for option A over B
    """
    # comparisons for the top tier, the spacefaring goals
    Sci_vs_Prag = 9
    Sci_vs_Dest = 5
    Prag_vs_Dest = -3

    spacefaring = [Sci_vs_Prag, Sci_vs_Dest, Prag_vs_Dest]


    # comparisons for the second tier, under science
    Uni_vs_Hum = 7
    Uni_vs_Earth = 7
    Uni_vs_Sol = 3
    Hum_vs_Earth = -3
    Hum_vs_Sol = -9
    Earth_vs_Sol = -5

    science = [Uni_vs_Hum, Uni_vs_Earth, Uni_vs_Sol,
               Hum_vs_Earth, Hum_vs_Sol, Earth_vs_Sol]


    # comparisons for the second tier, under pragmatism
    Def_vs_Tech = -3
    Def_vs_Sust = -9
    Def_vs_Comm = -9
    Tech_vs_Sust = -7
    Tech_vs_Comm = -9
    Sust_vs_Comm = -3

    pragmatism = [Def_vs_Tech, Def_vs_Sust, Def_vs_Comm,
                  Tech_vs_Sust, Tech_vs_Comm, Sust_vs_Comm]


    # comparisons for the second tier, under destiny
    Exp_vs_Col = -7

    destiny = [Exp_vs_Col]

    # return the completed pairwise comparisons in the required format
    # they will be checked for consistency
    #
    # unfortunately, for the prototype, if an inconsistent set of pairwise
    # comparisons are detected, the script simply ends and only tells the user
    # which group, but provides no guidance on which comparisons are the most
    # inconsistent, like a couple of web applications are able to do
```

```
    return([spacefaring, science, pragmatism, destiny])
```

The next required file is `input_strategies_xx.py`. This file is very straightforward and stores the input values, 1–5, for each of the four selected implementation strategies. The template for this file is shown below.

Listing 3.2 –
Template file for `input_strategies_xx.py`

```
#!/usr/bin/env python3.6
"""
input_strategies_01.py

author: Doug Coley

---> template file

this file is automatically generated when needed by the ariadne.py script

allows the user to specify their desired implementation strategies for the
program

each strategy should be assigned a value from 1-5 as follows:

    tech level:
        [off-the-shelf] 1 |---------|---------|---------|---------| 5 [advanced]

    program pace:
            [postponed] 1 |---------|---------|---------|---------| 5 [near term]

    man's involvment:
                 [none] 1 |---------|---------|---------|---------| 5 [extensive]

    program scale:
                [small] 1 |---------|---------|---------|---------| 5 [grand]

"""

def selected_strategies():

    tech = 3
    man = 3
    scale = 3
    time = 3

    return([tech, man, scale, time])
```

The final required file is `input_candidate_xx.py`. This file is the primary driving file for each desired program candidate. In its comment section, it contains the reproduced dashboard information that is derived from the input goals and strategies. When modifying the template and assigning the missions for the candidate space program, a user should consult with this dashboard first. The template for this file is shown below.

Listing 3.3 –
Template file for `input_candidate_xx.py`

```
#!/usr/bin/env python3.6
"""
input_candidate_01.py

author: Doug Coley

---> template file

this file is automatically generated when needed by the ariadne.py script

the derived dashboard based on both input_goals_01.py and
input_strategies_01.py is included here:

================================================================================
                     -|  II. Weighted spacefaring goals  |-
================================================================================

            |   1.0   0.11    0.2  |            Science      -  |0.0629|
            |   9.0    1.0    3.0  |            Pragmatism   -  |0.6716|
            |   5.0   0.33    1.0  |            Destiny      -  |0.2654|

            |   1.0   0.14   0.14   0.33  |     Universe     -  |0.0467|
            |   7.0    1.0    3.0    9.0  |     Human        -  |0.5875|
            |   7.0   0.33    1.0    5.0  |     Earth        -  |0.2851|
            |   3.0   0.11    0.2    1.0  |     Solar        -  |0.0807|

            |   1.0    3.0    9.0    9.0  |     Defense      -  |0.5822|
            |  0.33    1.0    7.0    9.0  |     Technology   -  |0.3117|
            |  0.11   0.14    1.0    3.0  |     Sustain      -  |0.0684|
            |  0.11   0.11   0.33    1.0  |     Commercial   -  |0.0376|

            |   1.0    7.0  |                   Exploration  -  |0.8750|
            |  0.14    1.0  |                   Colonization -  |0.1250|

================================================================================
                     -|  III. Implementation strategies  |-
================================================================================

            Level of man's involvement: 3 (x1.5 priority multiplier)
             Level of technology used: 3 (x1.2 priority multiplier)
                         Scale of the program: 3
                   Pace of the program: 3 (12 year program)

================================================================================
                     -|  Prioritized list of objectives  |-
================================================================================

   ID    Name                                 Rank     Flights   Payload    Installed
--------------------------------------------------------------------------------
  [103]  Earth space station                  0.76       18.0      4,750       77,500
  [301]  Launch propulsion                    0.72
  [313]  Ground systems                       0.72
  [102]  Earth sat. (manned)                  0.71       17.0      4,125
  [307]  Human exploration systems            0.59
  [306]  Human health                         0.54
  [206]  Planet/moon surface base             0.53        5.0      4,250      755,000
  [304]  Robotics                             0.53
  [207]  Planet/moon supplied colony          0.51        5.0     40,000    4,000,000
```

```
  [105] Cis-lunar space station          0.50       12.0      5,000     155,000
  [204] Planet/moon lander (manned)      0.47        5.0      8,250
  [106] Earth orbital aux. vehicle       0.46                               3,775
  [205] Planet/moon space station        0.46        5.0      4,000     378,750
  [101] Earth sat.                       0.44       75.0      2,500
  [202] Planet/moon sat. (manned)        0.39        2.0      9,000
  [303] Power generation                 0.35
  [312] Materials                        0.35
  [314] Thermal                          0.29
  [309] EDL systems                      0.28
  [315] Aeronautics                      0.27
  [311] Modeling/simulation              0.26
  [305] Communications                   0.26
  [302] In-space propulsion              0.25
  [310] Nano                             0.25
  [201] Planet/moon sat.                 0.24       14.0        875
  [209] Solar probe                      0.24        3.0      1,137
  [210] Interstellar probe               0.24        1.0      1,150
  [203] Planet/moon lander               0.24        7.0      2,500
  [107] Earth orbital colony             0.16       11.0      8,250   4,000,000
  [208] Planet/moon sufficient colony    0.12                         8,750,000
  [308] Science instruments              0.08
  [104] Earth orbital power plant        0.07                            15,500

================================================================================


the user should also consult IV_mission_architectures.py to see what mission
architectures have been modeled and are available

"""
import numpy as np

def program_missions(program):
    """
    here each mission is assigned to the program.
    currently, for the prototype, each architecture is hard coded to its
    description, e.g., there is only a single architecture (LOR) connected
    to the lunar_land_return attribute -- to run a trade of competing
    architectures, the user will have to modify the code at the beginning of
    ariadne.py

    each destination/architecture needs to be assigned a NumPy array
    of payloads

    the random function has been called to introduce some variation in the
    specified payloads, though they can also be all manually input as well

    for the function below:
        program.leo = np.random.normal(0.7, 0.1, 2)*1000
     - 'leo' is the destination that is tied to an architecture in
       IV_mission_architectures.py
     - 'np.random.normal(0.7, 0.1, ' is the randomization function to split
       vary the payloads a little bit like they typically would be; if desired,
       look up the function in the NumPy documentation for more information on
       how to vary the distribution
     - '2)' is the number of launches desired; should be influenced (but not
       required to be) by the dashboard in the intro comments above
     - '1000' is the desired payload; again should be inflected by the derived
       dashboard found in the comment above
    """
```

```
    np.random.seed(0)

    # just making sure that the selected strategies that will be needed later
    # are preserved
    program.strat_time = 3
    program.strat_tech = 3
    program.strat_man = 3
    program.strat_scale = 3

    # current destinations with modeled architectures
    program.leo = np.random.normal(0.7, 0.1, 2)*1000
    program.leo = np.append(program.leo, np.random.normal(0.7, 0.1, 4)*1500)
    program.leo = np.append(program.leo, np.random.normal(0.7, 0.1, 7)*1000)
    program.lunar_land_return = np.random.normal(0.7, 0.1, 1)*3000

    # program.geo = np.random.normal(0.7, 0.1, 5)*2500
    # program.lunar_flyby = np.random.normal(0.7, 0.1, 5)*2500
    # program.lunar_orbit = np.random.normal(0.7, 0.1, 5)*2500
    # program.lunar_land = np.random.normal(0.7, 0.1, 5)*2500
    # program.mars_flyby = np.random.normal(0.7, 0.1, 5)*2500

def program_installations(program):
    """
    in addition to missions that are flown per year, several of the program
    objectives also require an installation mass (e.g., a space station in
    LEO)

    for the prototype, only installation masses in LEO are accounted for
    (meaning the mass input below is the mass that is going to LEO, no
    architectures are used to determine IMLEO based on whatever final
    destination payload)

    each installation mass will be divided up by the capability of the largest
    launch vehicle developed, and the total number of launches added to the
    number of that vehicle to be produced
    """

    program.installations = [10000]
```

3.4.3 *Python requirements*

Several additional Python packages were applied in this software implementation. Most are included in many of the pre-bundled, scientific distributions of Python or they can be installed individually with Python's built in package manager, `pip`. The following packages are required to run `ariadne.py`:

» NumPy

» Pandas

» matplotlib

» readline (or pyreadline on Windows)

» scikit-learn

3.5 *Chapter summary*

This chapter introduced an ideal solution concept, *Ariadne*, to the complete list of specification identified in Chapter 2. After a walk-through of its components and the identification of other relevant efforts to parts of the problem, the grandiose list of specifications was reduced down to the list given in Table 3.1. A prototype methodology for a focused *Ariadne*, the required methods, and software implementation were then discussed. It is the contention of the author that the prototype methodology and implementation developed in this chapter are capable of fulfilling the entirety of the prototype specifications. The validation of this claim can be found in the following chapter which contains a three part case study on Project Apollo.

The following is a summary of the original contributions made by this research in this chapter:

» A parametrically connected, space program synthesis prototype, *Ariadne*. The prototype accounts for the parametric influences from spacefaring goals, implementation strategies, mission architectures, and launch vehicle development.

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Chapter 4

PROJECT APOLLO – A CASE STUDY IN SPACE PROGRAM PLANNING

In order to demonstrate the contributions of the *Ariadne* prototype system, the U.S. space program as it led up to Project Apollo has been selected as the most fitting case study. Twiss has the following to say about the selection of an appropriate study:

> The choice of the first studies should also be considered carefully. Experience suggests that the following guidelines are likely to provide a sound basis for their selection.
>
> 1. They should be limited in scope and conducted with data that are readily available using relatively simple techniques.
> 2. Their results should be clear and unambiguous.
> 3. They should relate to a current technological problem of significance.
> 4. They should give support to a programme advocated by an influential manager in the company. [1]

Project Apollo fulfills each of his recommended guidelines. First, Apollo properly limits the scope for analysis down to a single, albeit major, project for which there is an incredible amount of validation data from both historical and technical sources [1]. The *Ariadne* prototype is also developed on the foundation of simple methods integrated into a synthesis methodology. Second, the result of the study, demonstrating the value of integrated decision support, will be clearly seen. Third, the case study directly relates to a current problem as nations and organizations are still struggling to steer their space programs the “best” way forward. Finally, Twiss’ last point concerns establishing credibility when first introducing the solution to an audience unfamiliar with the approach. Project Apollo provides many opportunities to step through the critical decisions of the space program to first validate the process and then demonstrate the capabilities of the *Ariadne* solution at work.

Furthermore, the manned requirement of Apollo provides the impetus for clearly defining the goals and strategies involved in the years leading up to the lunar landing. General Dynamics, in their STAMP report, said:

> Of the two principal categories of operational space flight, namely manned and non-manned programs, the former is by far the most expensive. It is,

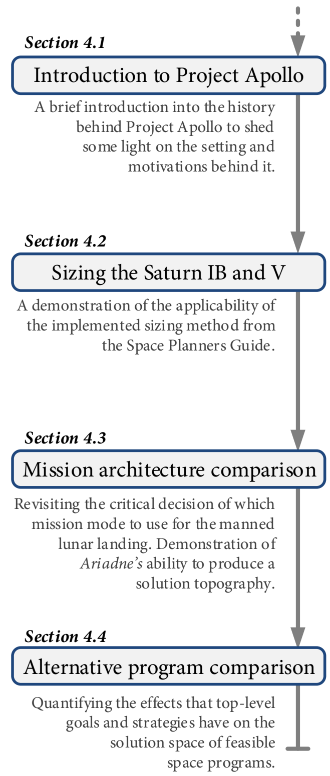

FIGURE 4.1 – Outline of Chapter 4.

[1] See Table 4.1.

Ignition masses [*kg*]	Apollo 11	Apollo 12	Apollo 13	Apollo 14	Apollo 15	Apollo 16	Apollo 17	Average
S-IC stage								
propellant	2,145,796	2,147,525	2,148,265	2,150,629	2,142,237	2,155,070	2,152,888	2,148,916
structure	132,890	133,056	133,039	132,678	131,765	133,017	132,789	132,748
Total	**2,278,687**	**2,280,582**	**2,281,304**	**2,283,307**	**2,274,001**	**2,288,086**	**2,285,677**	**2,281,663**
S-IC/S-II interstage								
dry	5,206	5,220	5,195	5,171	4,120	4,577	4,525	4,859
S-II stage								
propellant	443,235	446,059	452,082	452,102	452,212	456,635	455,654	451,140
structure	36,730	36,961	35,861	35,912	36,283	36,902	36,903	36,507
Total	**479,964**	**483,020**	**487,944**	**488,014**	**488,495**	**493,536**	**492,556**	**487,647**
S-II/S-IVB interstage								
dry	3,663	3,638	3,665	3,656	3,642	3,654	3,637	3,651
S-IVB stage								
propellant	107,095	106,254	106,843	106,149	108,618	108,453	108,584	107,428
structure	12,024	12,219	12,143	12,118	12,181	12,130	12,110	12,132
Total	**119,119**	**118,472**	**118,985**	**118,268**	**120,798**	**120,583**	**120,694**	**119,560**
Instrument unit	1,939	1,940	2,042	2,043	2,035	2,042	2,028	2,010
CSM adapter	1,792	1,796	1,790	1,797	1,798	1,797	1,797	1,795
Lunar Module	15,095	15,223	15,192	15,279	16,437	16,437	16,448	15,730
Command and Service Module	28,806	28,830	28,937	29,233	30,357	30,368	30,364	29,556
Escape System	4,042	4,066	4,078	4,095	4,131	4,158	4,130	4,100
Total spacecraft	**49,735**	**49,915**	**49,998**	**50,404**	**52,723**	**52,759**	**52,739**	**51,182**
Total vehicle	**2,938,312**	**2,942,788**	**2,949,134**	**2,950,865**	**2,945,815**	**2,965,239**	**2,961,858**	**2,950,573**

TABLE 4.1 – Ground ignition weights of some of the Apollo missions. Adapted from Orloff [3].

> therefore, especially important to clearly define the objectives of manned programs. [2]

Clearly defining objectives and how they are affected by early decisions is one of the primary deliverables of the *Ariadne* system.

THE REMAINDER OF THIS CHAPTER first provides a very brief introduction to the history of Apollo and the U.S. space program. After that, the case study is divided into three separate parts:

» validation of the launch vehicle sizing process;

» validation of the mission architecture modeling process;

» comparison of program alternatives leading up to Project Apollo.

With the three parts of the case study completed, the results are discussed and compared against the original specifications for the prototype previously provided in Table 3.1. An outline of the progression of this chapter is also provided in Figure 4.1.

4.1 *Introduction to Project Apollo*

Project Apollo is arguably the most famous space effort in the history of space exploration due to its overall scale and what it managed to accomplish in such a short amount of time. The early U.S. space program succeeded in placing two men on the moon and returning them

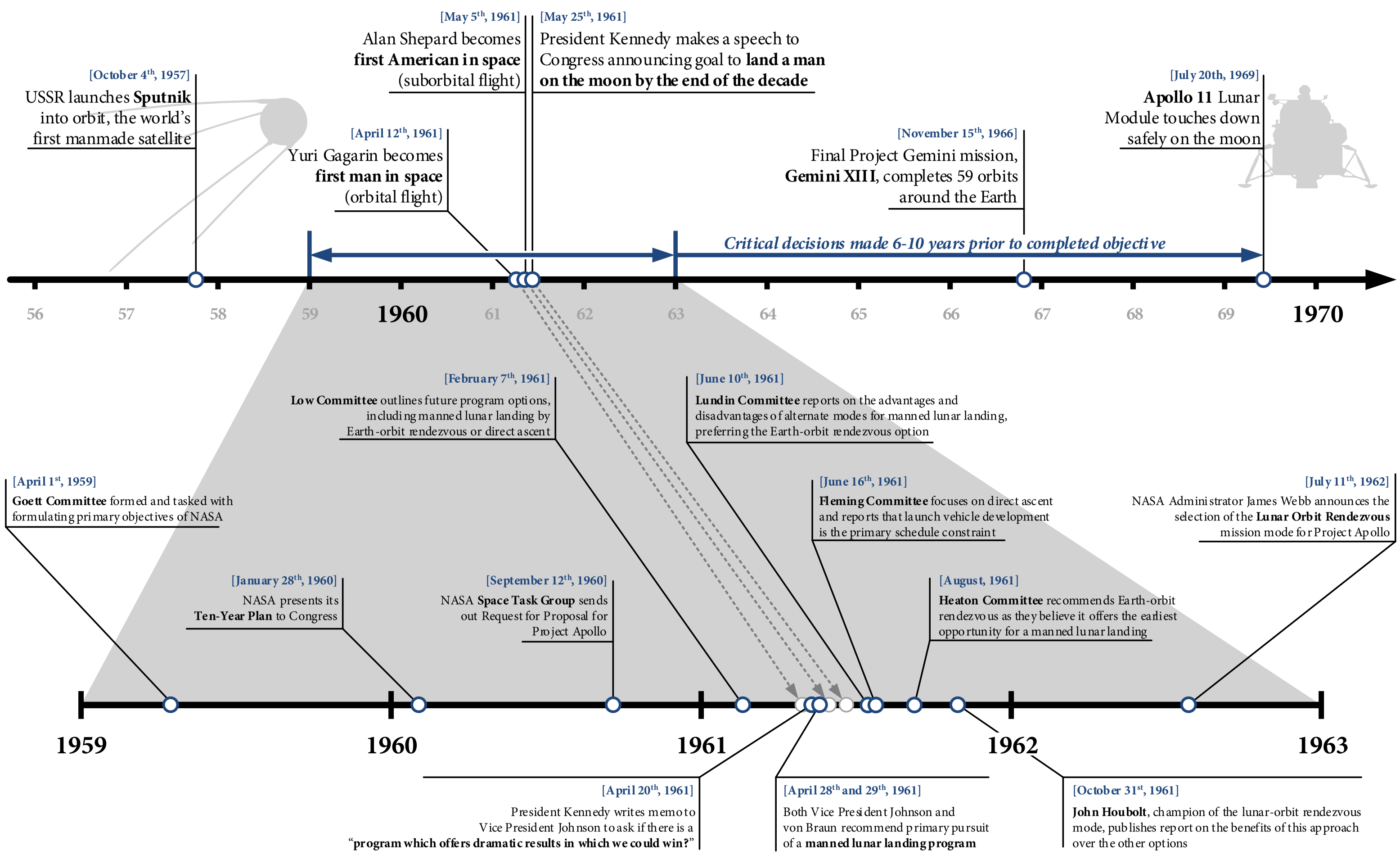

FIGURE 4.2 –
A timeline depicting some of the key milestones in the US space program in the 1960s.

safely to Earth only eight years after the first man had ventured into space.

Most people are familiar with the speech President Kennedy made to Congress in 1961:

> ...I believe that this nation should commit itself to achieving the goal, before this decade is out, of landing a man on the moon and returning him safely to the earth. No single space project in this period will be more impressive to mankind, or more important for the long-range exploration of space; and none will be so difficult or expensive to accomplish. We propose to accelerate the development of the appropriate lunar space craft. We propose to develop alternate liquid and solid fuel boosters, much larger than any now being developed, until certain which is superior. We propose additional funds for other engine development and for unmanned explorations—explorations which are particularly important for one purpose which this nation will never overlook: the survival of the man who first makes this daring flight. But in a very real sense, it will not be one man going to the moon—if we make this judgment affirmatively, it will be an entire nation. For all of us must work to put him there. [4]

What most people do not remember is just how little the U.S. had accomplished in space at the time of that speech. Less than three weeks before his speech, Alan Shepard became the first American in space on his sub-orbital flight in a Project Mercury spacecraft. Yet, Kennedy still set the nation on a course to land two men on the moon and return them safely by the end of the decade. Figure 4.2 is a timeline of the events leading up to the lunar landing. Along the top of the figure are the more well known milestones: Sputnik in 1957, Gagarin in 1961, *etc.*, all the way to the Apollo 11 lunar landing in July of 1969. The lower portion of the figure zooms in on a timeline of the years 1959 through 1963. This is when the decisions were made that put man on the moon in 1969; when multiple committees met and tried to determine what path the U.S. space program should take.

With his speech, Kennedy seized an opportunity to overtake the Soviet space efforts with a manned lunar landing. After he committed the nation in May of 1961, it still took over a year and the efforts of at least three different committees before consensus was reached on how the objective would even be accomplished.

These early decision are the ones that need to be investigated with the *Ariadne* system. How large a launch vehicle is required to complete the desired objectives? What mission architecture should be used to accomplish this manned lunar landing? What might alternative programs look like prior to Kennedy's definitive declaration? The rest of the chapter serves to illustrate how the *Ariadne* system, the proposed solution concept of this research, can be used to provide insight into such questions.

4.2 *Sizing the required launch vehicles*

This first section of the case study validates the application of the Space Planners Guide sizing method introduced in Section 3.3.5. It begins with an introduction to the family of launch vehicles that were developed for Project Apollo: specifically the Saturn IB and Saturn V. Then, a walk-through of the sizing process to estimate the launch vehicle required for a mission similar to that of the Saturn IB is presented. The historical values and the estimated launch vehicle are compared and discussed. The results of the sizing process for a launch vehicle similar to the Saturn V are then presented and discussed.

THE OBJECTIVES for this section of the case study are to:

» Validate the capability of the *Ariadne* prototype system, with its digitized version of the Space Planners Guide launch vehicle sizing process, to properly size the launch vehicle required for desired missions;

» Familiarize the reader with some of the details of the process at work every time a program is analyzed with the *Ariadne* system.

4.2.1 *Introduction to the Saturn family of launch vehicles*

The Saturn launch vehicle development program was officially established on December 31, 1959. Within a month it was designated as one of the highest priorities of the nation. The Saturn family of launch vehicles were to be developed as a building block concept: elements (stages and engines) first developed for an initial launch vehicle could be directly applied to subsequent stages as the program increased their capability and developed larger launch vehicles [6]. There were at least seven planned Saturn configurations from the C-1 all the way up a possible C-8 [7–9]. As goals became more clear and payload estimates more realistic, only three configurations emerged from the Saturn program: the Saturn I, Saturn IB, and the Saturn V.[2] Much has been written on both the history and technical details on the family of Saturn launch vehicles [5, 6, 12–15], most notably R. Bilstein's *Stages to Saturn: A Technological History of the Apollo/Saturn Launch Vehicle* [16].

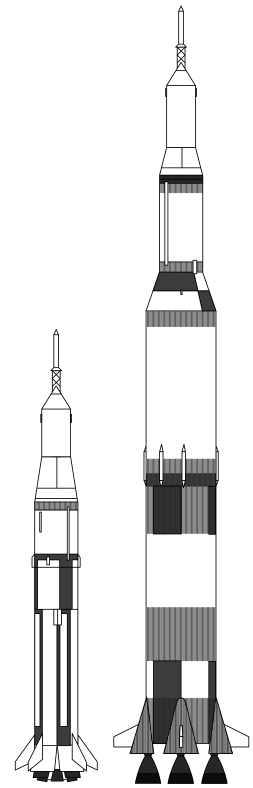

FIGURE 4.3 – The Saturn IB and Saturn V. Adapted from NASA [5].

[2] Though, subsequent derivatives of the Saturn V were also under investigation; some even prior to the first launch of the Saturn V. [10, 11]

4.2.2 *Sizing demonstration with the Saturn IB*

The digitized Space Planners Guide sizing process from the *Ariadne* prototype system is applied here to estimate the launch vehicle required to perform a mission similar to the Saturn IB[3] to illustrate the utility of the Guide and to help validate the results of the process. The required inputs, assumptions, and final deliverables are all provided.

[3] Why not walk-through the process with the Saturn V? The Guide's sizing process optimizes the stages of a launch vehicle for a mission to a 185 *km* orbit. The actual Saturn V used two full stages (S–IC and S–II) and only a partial burn of the third stage (S–IVB) to reach orbit. After some diagnostics in LEO, the third stage was then re-ignited for the trans-lunar injection burn. Thus the Saturn IB more closely resembles the mission modeled by the Guide though with some tweaking of the assumptions, a planner can, in fact, reach a respectable estimate for the Saturn V launch mass. This process will be illustrated later in this section.

Inputs required	
Destination	LEO[a]
Payload mass	18,600 [kg]
Escape tower mass	4,000 [kg]
Number of stages	2
Propellant of each stage[b]	
first	RP-1/LOX
second	LH_2/LOX
I_{sp} of each stage[c]	
first	296 [s]
second	419 [s]
Thrust-to-weight ratio	
first	1.3
second	1.0

[a] 185 [km] orbit
[b] used in determining the bulk density of the propellant
[c] using the vacuum rating

TABLE 4.2 – Listing of the required inputs for sizing process and the values selected for the Saturn IB. Compiled from multiple sources [3, 5, 12].

To begin the sizing process, a number of inputs must be supplied. Table 4.2 contains all of the inputs that are required for the Saturn IB. The mission of the Saturn IB was to low Earth orbit to test out many of the components and systems that would be required for the future lunar landing. Including an assumption of the velocity losses due to drag and gravity, the total velocity required for its mission is about 9,450 m/s. For the input characteristics of the propulsion systems, the selected specific impulse values are the vacuum rating reported for the Rocketdyne H-1 and J-2. The initial thrust-to-weight ratios are suggested assumptions by the Guide and also supported by H. H. Koelle and Thomae [17].

As discussed in Section 3.3.5, the process begins with the method provided by nomograph V.C-9 which relates the average specific impulse of the launch vehicle and the number of stages to the total inverted payload ratio of the launch vehicle, $1/\lambda_{total}$. Using the nomograph shown in Figure 4.4, the method begins with the average specific impulse of both stages along the x-axis, drawing a line up until the number of stages is reached. Then, a straight line is drawn to the y-axis to determine the total launch vehicle mass to the payload ratio, $1/\lambda_{total}$. This is shown for the Saturn IB in Figure 4.4 with the input and output denoted by the bold arrow. With an average specific impulse of 357.5 s for the two stages, the resulting $1/\lambda_{total}$ ratio is 32.7. This ratio is multiplied by the input payload mass of 18,600 kg (provided in Table 4.2) and results in an initial estimate of the total launch mass of 608,220 kg.[4]

With the initial estimate of the total launch vehicle mass, the siz-

[4] Later, when comparing the estimated values with the historical values, it will be seen that this initial estimate is very close for the Saturn IB. This should not come as a surprise, as the Guide's nomographs were probably generated with some of the data from the Saturn launchers and the Saturn IB flew the exact mission under consideration by the Guide. This use of this nomograph alone does not hold up, however, for all launch vehicles, especially not more recent efforts.

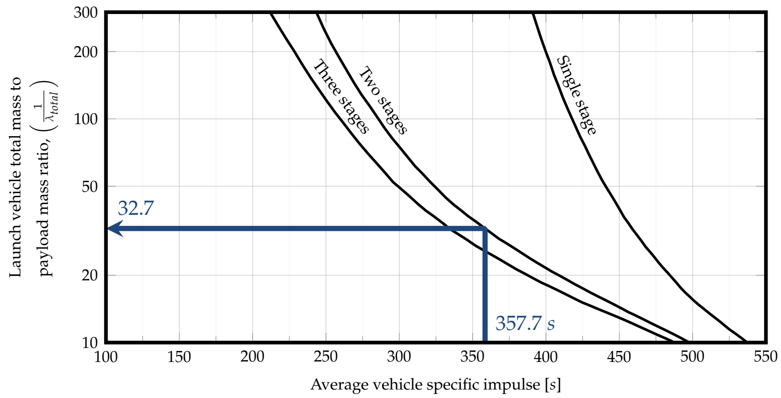

FIGURE 4.4 – Staging and specific impulse effects on payload performance. Originally nomograph V.C-9 from the Guide [18].

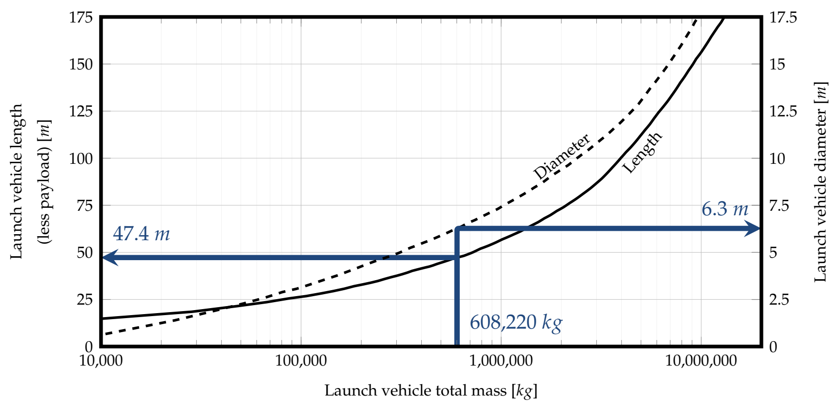

FIGURE 4.5 – Launch vehicle maximum dimensions. Originally nomograph V.C-10 from the Guide [18].

ing process then uses nomograph V.C-10 to provide the maximum expected values for both the diameter and length of the launch vehicle. This method can be seen in Figure 4.5 for the initial estimate of the Saturn IB. Using the total launch mass of 608,220 *kg*, the maximum length is about 47.4 *m* and the maximum diameter of the launch vehicle is about 6.3 *m*.

The next step in the process uses the evaluated value for $1/\lambda_{total}$ and the number of stages with nomograph V.C-11 from the Guide. The ratio is used as an input on the y-axis and followed to the right until it reaches the selected number of stages. A line is then drawn straight down to yield the first estimate for the stage mass to mass above the stage ratio, $1/\lambda_n$. This is shown in Figure 4.6 for the estimated Saturn IB. With an input ratio of 32.7, the output $1/\lambda_n$ is 5.6. This value will be used for both stages to determine the initial estimates of the individual stage masses. The stage ratios will be optimized in the second half of the sizing process.

The method provided in nomograph V.C-12 converts the $1/\lambda_n$ found

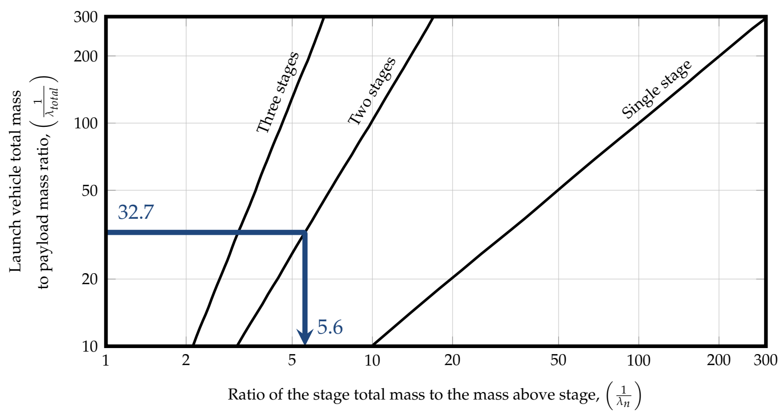

FIGURE 4.6 – Stage to payload mass ratios. Originally nomograph V.C-11 from the Guide [18].

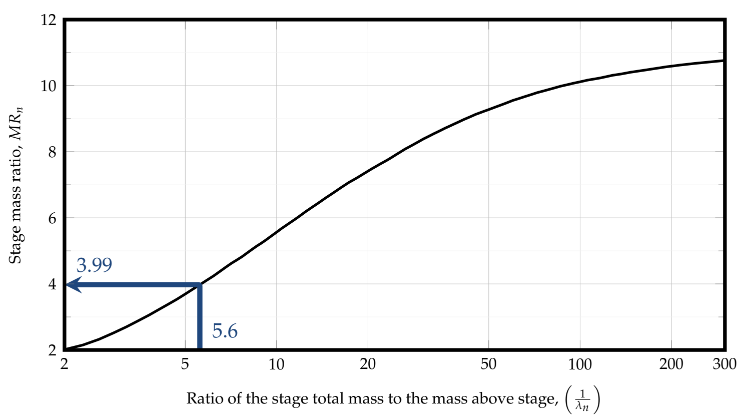

FIGURE 4.7 – Stage to payload mass ratio conversion to stage mass ratio. Originally nomograph V.C-12 from the Guide [18].

in the last figure into an estimate for the mass ratio of each stage. Figure 4.7 illustrates this process using the estimated values for the Saturn IB. The stage mass to mass above the stage ratio of 5.6 is used, resulting in an initial estimate of the mass ratio, MR, of 3.99 for both stages. Again, this assumption of equal ratios for each stage will be corrected in the second half of the process.

These stage ratios, along with the previously discussed Equations 3.12 – 3.15, provide the initial estimate of the contributing masses of each stage.

The next two figures, Figure 4.8 and 4.9, are used along with the newly estimate stage masses to determine the structure factor for each stage. For the first stage, with an assumed initial thrust-to-weight ratio of 1.3 and the propellant bulk density of RP-1/LOX equal to 1,009 kg/m^3, the resultant structure factor is 0.067 as can be seen in Figure 4.8. For the second stage, with an assumed initial thrust-to-weight ratio of 1.0 and the propellant bulk density of LH_2/LOX equal to 325

First estimate - Saturn IB	Variable	Value [kg]
Payload mass	$m_{payload}$	18,600
Second stage		
Stage mass	m_2	85,913
Propellant mass	$m_{p,2}$	77,150
Structure mass	$m_{s,2}$	8,763
Total mass	$m_{total,2}$	104,513
First stage		
Stage mass	m_1	482,747
Propellant mass	$m_{p,1}$	450,403
Structure mass	$m_{s,1}$	32,344
Total mass	$m_{total,1}$	587,261

TABLE 4.3 – The estimated values for the Saturn IB launch vehicle from the first portion of the Space Planners Guide sizing process.

kg/m^3, the resultant structure factor is around 0.10 as can be seen in Figure 4.9.

With the stage masses and structure factors estimated, the specific masses of the propellant and structure of each stage can be found with Equations 3.16 and 3.17. The compiled results of this initial estimate are shown in Table 4.3.

The next part of the sizing process attempts to optimize the mass of each stage to reach the target orbit, $\Delta V_{required}$, with the minimum mass launch vehicle required. The optimum sizing nomograph, V.C-22, with the final iterated value for the Saturn IB is shown in Figure 4.10. This nomograph is paired with V.C-23 for this portion of the process, found in Figure 4.11 and discussed below.

The iterative optimization process begins with an input of a mass ratio for the first stage. Initially the planner should start with the estimated value found in the first part of the process, in this case 4.0. The path to take is illustrated by the bold arrow in Figure 4.10. Proceed upward from the initial mass ratio value located on the x-axis until the structure factor of the stage is reached. Then move to the right until

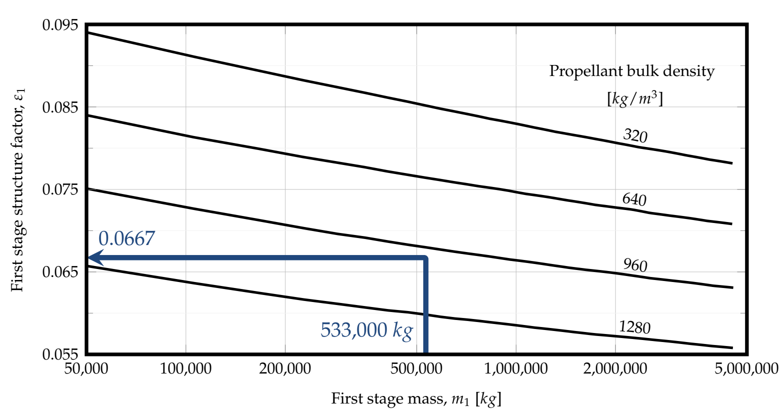

FIGURE 4.8 – Liquid first stage structure factor. Originally nomograph V.C-15 from the Guide [18].

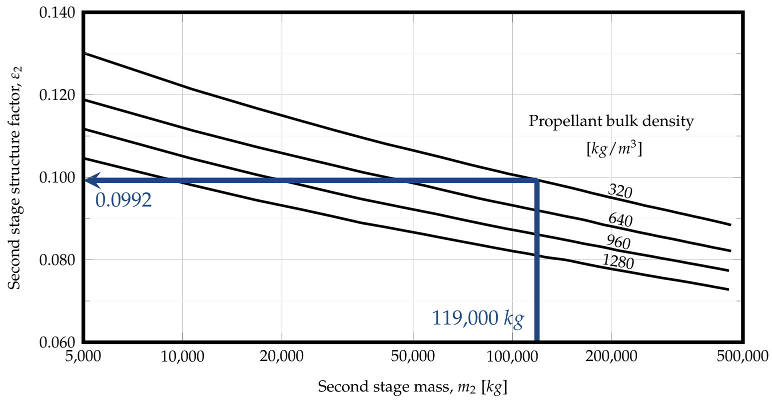

FIGURE 4.9 – Liquid second stage structure factor. Originally nomograph V.C-16 from the Guide [18].

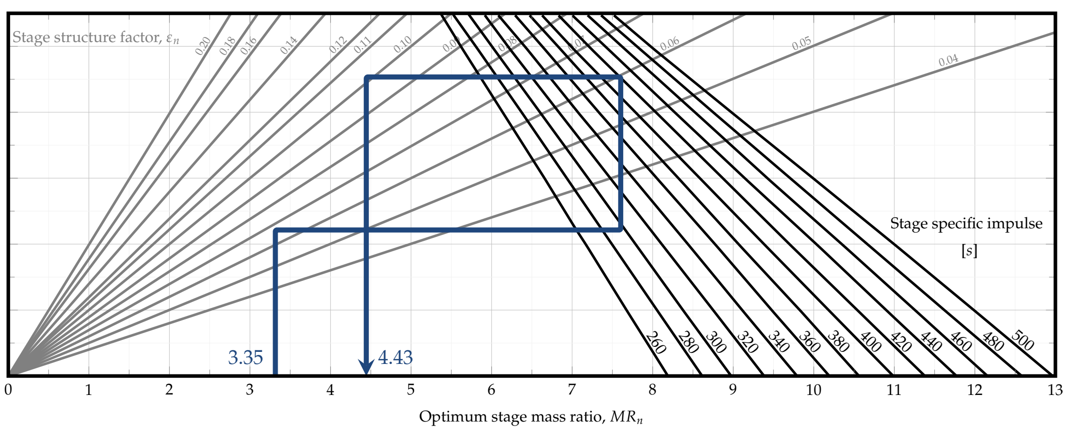

FIGURE 4.10 – Optimum launch vehicle sizing nomograph. Originally nomograph V.C-22 from the Guide [18].

the stage's specific impulse is found. Now proceed upward until the specific impulse of the second stage is found. Move to the left until the second stage structure factor is encountered, then proceed straight down for the new estimate of the mass ratio for the second stage.

Now, with the mass ratio, MR_n of each stage, the velocity capability of stage n is determined with nomograph V.C-23. The initial mass ratio is adjusted until the velocity capability and the velocity required are sufficiently close.[5] Figure 4.10 illustrates the final iteration for the estimate of the Saturn IB. The initial input of the mass ratio of the first stage is 3.35. This results in a mass ratio of 4.43 for the second stage. With a MR_1 of 3.35 and specific impulse of 296 s, the velocity capability of the first stage is around 3,505 m/s. With a MR_2 of 4.43 and specific impulse of 419 s, the velocity capability of the second stage is around 5,959 m/s. Adding these two velocities results in a total velocity capability of 9,464 m/s, sufficiently close to the velocity requirement.

The final portion of the sizing process uses nomograph V.C-24, reproduced here with the Saturn IB values in Figure 4.12. Here the newly

[5] The Guide never recommends a target percent difference, though it does warn about chasing a sense of false accuracy by targeting a small difference for convergence. In this case study, the author iterated until the velocity capability converged within 0.5% of the velocity required. This is probably what the Guide recommends against, although the author has the benefit of digitized nomographs so subsequent iterations were not time consuming and the author remains perfectly aware of the many approximations that have been used to get to this point.

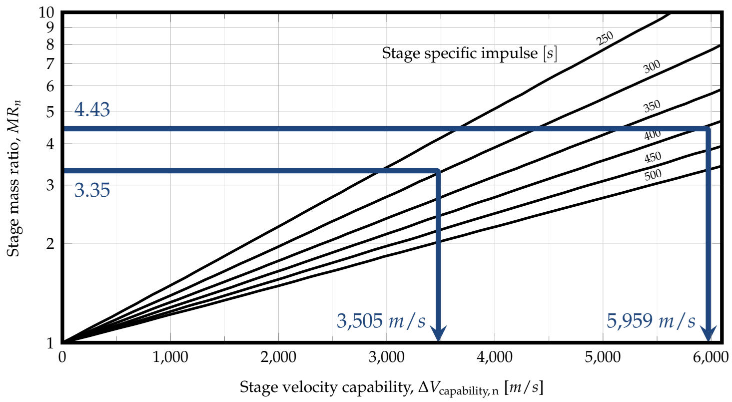

FIGURE 4.11 – Stage velocity versus mass ratio and specific impulse. Originally nomograph V.C-23 from the Guide [18].

optimized stage mass ratios of 3.35 and 4.43 for the first and second stages, respectively, are input along the x-axis and followed upward until they meet the current stage structure factor, then straight left to the y-axis to determine a new estimate for the stage mass to mass above the stage ratio, $1/\lambda_n$. These values are then used to estimate the stage masses with Equations 3.12 - 3.15.

The resultant stage masses are then passed into nomographs V.C-15 and V.C-16, Figure 4.8 and 4.9, to obtain the new estimate for the structure factor of each stage. Once these new and old values are sufficiently close, the entire launch vehicle mass breakdown is calculated again, this time representing the final estimate for the launch vehicle. The final iteration for this study of the Saturn IB is shown in Figure 4.12 and the resultant stage masses are provided in Table 4.4.

The Space Planners Guide was used to make two estimates for the masses of the Saturn IB: an initial estimate that assumes equal stages mass ratios, and a final estimate that optimizes the launch vehicle for the minimum launcher mass required to place the desired payload in a

Final estimate - Saturn IB	**Variable**	**Value [kg]**
Payload mass	$m_{payload}$	18,600
Escape tower mass	m_{tower}	4,000
Second stage		
Stage mass	m_2	117,998
Propellant mass	$m_{p,2}$	106,317
Structure mass	$m_{s,2}$	11,682
Total mass	$m_{total,2}$	136,598
First stage		
Stage mass	m_1	539,083
Propellant mass	$m_{p,1}$	502,965
Structure mass	$m_{s,1}$	36,119
Total mass	$m_{total,1}$	675,682

TABLE 4.4 – The final estimated values for the Saturn IB launch vehicle.

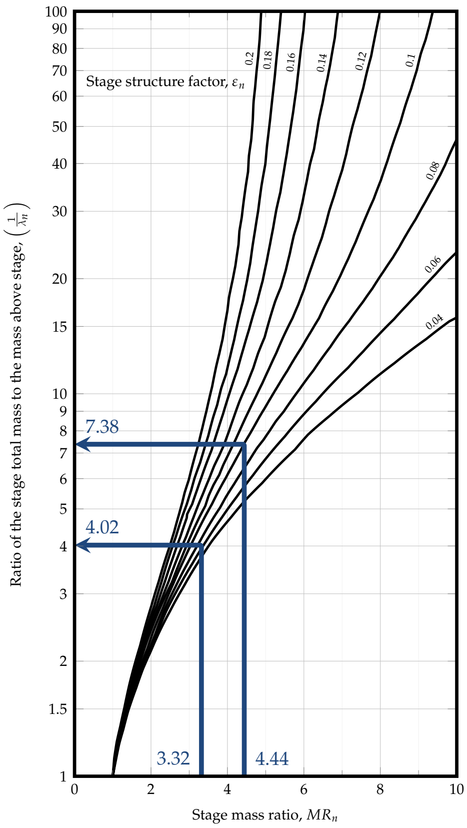

FIGURE 4.12 – Stage mass ratio vs stage mass ratio. Originally nomograph V.C-24 from the Guide [18].

185 *km* orbit. Table 4.5 shows the historical masses of the actual Saturn IB along with both estimates and the percent error of the estimate.

As can be seen, both estimates do an excellent job of estimating the mass of the total launch vehicle with very little information. For the Saturn IB, the initial estimate is actually closer than the optimized estimate. This does not suggest, however, that a planner should only use the first estimation method for any desired studies efforts.[6] The final estimate is still within the target accuracy of the Guide, and fairs much better on the individual masses of the second stage. As mentioned before, the initial estimate also comes from nomographs built upon data from launch vehicles that existed at the time of the Guide's development for a similar mission, likely including the Saturn IB which explains its accuracy. The next section looks at the versatility of the sizing process by comparing the results of two different Saturn V sizing studies.

[6] Considering the Saturn V and several of the studies in Appendix C (Table C.1 and C.2), a majority of final estimates are much closer to the actual values of the launch vehicle than the initial estimate.

4.2.3 *Results of the sizing Saturn V*

The primary reason for using the Saturn IB for the walk-through is that its mission more closely mirrored the "default" capabilities of the

Saturn IB	Actual	First estimate	% Error	Final estimate	% Error
Velocity capability	9,450 m/s	–	–	9,465	0.01
Payload mass	18,600 kg	18,600	–	18,600	–
Escape tower mass	4,000 kg	–	–	4,000	–
Second stage					
Stage mass	116,500 kg	85,913	-27	117,998	2
Propellant mass	105,300 kg	77,150	-27	106,317	2
Structure mass	10,600 kg	8,763	-18	11,682	11
Total mass	135,100 kg	104,513	-23	136,598	2
First stage					
Stage mass	452,300 kg	482,747	5	539,083	18
Propellant mass	414,000 kg	450,403	7	502,965	20
Structure mass	40,500 kg	32,344	-22	36,119	-13
Total mass	**587,900** kg	**587,261**	**-1**	**675,682**	**15**

TABLE 4.5 – Comparison of sizing results with the actual Saturn IB [19, 20].

Space Planners Guide sizing process: namely a multi-stage launch vehicle optimized to reach LEO. The real Saturn V did in fact launch to a similar orbit, but it did so using two full stages and a partial burn of its third stage. The vehicle remained in Earth orbit for a final check of its systems before re-igniting the third stage on a trans-lunar injection burn. Thus, a pure estimate of a three-stage vehicle with similar propulsion performance, optimized for obtaining an 185 *km* orbit would have very little in common with the actual Saturn V.

In light of this, two different studies are presented: a two-stage variant of the Saturn V that requires a lower total velocity,[7] and a three-stage variant that requires a higher, trans-lunar injection velocity. The largest disadvantage of the Space Planners Guide is that the data and methods that went into the creation of the nomographs are not made available. Thus, when the target velocity is altered substantially, the user cannot be certain which nomographs are sensitive to the change. The results of this study of these two variants of the Saturn V are shown in Table 4.6. For each study, the approximate actual values for the Saturn V are listed in a column with the percent error for the initial and final estimates shown in a single column in the format (initial estimate % error / final estimate % error).

[7] This velocity is an assumption based on the actual reported speed at the cutoff of the second stage of the Saturn V, with an added estimate for drag and gravity losses.

A couple of notes about the results shown in Table 4.6 and the analysis behind them:

» The estimated stage masses of both the two-stage and three-stage versions are larger than the range of data provided in the nomographs of the Guide (see Figure 4.8 and 4.9) so the maximum reported value in each case is used for the rest of the analysis. Many of these required approximations appear to be reasonable as the nomograph trend-line levels out, though ideally an alternative method[8] should be found for such large stages.

[8] For values just outside the limits of the nomographs, extrapolation could serve the expansion process well.

Launch vehicle	Saturn V (2-stage) Actual	% Error*	Saturn V (3-stage) Actual	% Error*
Number of stages	2	– / –	3	– / –
Velocity capability, m/s	8,750	– / 0.01	12,800	– / 0.12
Payload mass, kg	135,000	– / –	45,800	– / –
Escape tower mass, kg	4,000	– / –	4,000	– / –
Third stage				
Stage mass, kg	–	–	116,000	-32 / 96
Propellant mass, kg	–	–	104,300	-31 / 98
Structure mass, kg	–	–	11,700	-40 / 73
Total mass, kg	–	–	161,800	-23 / 69
Second stage				
Stage mass, kg	481,400	54 / 106	481,400	-56 / 148
Propellant mass, kg	439,600	54 / 106	439,600	-56 / 148
Structure mass, kg	41,800	56 / 109	41,800	-52 / 151
Total mass, kg	616,400	47 / 88	616,400	-45 / 138
First stage				
Stage mass, kg	2,233,800	81 / 0	2,233,800	-74 / -39
Propellant mass, kg	2,066,100	84 / 1	2,066,100	-74 / -38
Structure mass, kg	167,700	50 / -16	167,700	-77 / -48
Total mass, kg	**2,896,200**	**71 / 17**	**2,896,200**	**-68 / -2**

All listed *actual* values are approximate and have been compiled from [3, 6, 20–22]

* For the sake of brevity, the percent error for the initial and final estimates are given in the format **(initial % / final %)**

TABLE 4.6 – Comparison of the Saturn V with the sizing result estimates for two Saturn-like studies.

» The estimates for the individual stages masses are quite far off from the actual values of the hardware. This is likely due to the simplified optimization process of the Guide that seeks to only minimize the total mass of the launch vehicle. The actual designers of the Saturn V had to concern themselves with additional criteria beyond just minimizing mass: maximum aerodynamic pressure, maximum g-loading, abort conditions, *etc.*

» The initial estimates for the total mass of each launch vehicle are too high for the two-stage variant and too low for the three-stage variant. However, the final estimates for the total mass of the launch vehicle for both studies is within the expected accuracy of the Guide's methods, emphasizing the necessity of following through with the process beyond that initial estimate.

4.2.4 *Conclusions*

This first section of the Project Apollo case study has demonstrated the validity of the digitized sizing process implemented in the *Ariadne* prototype system. The Guide's sizing process is capable of accurately sizing the total lift-off mass for both the Saturn IB and the Saturn V launch vehicles. Appendix C contains additional sizing studies compared with additional launch vehicles (past, present, and future) to validate the process beyond this Project Apollo case study.

The current implementation of *TransCost* for the prototype contains too many omissions to estimate costs that can be directly compared to the actual costs of the vehicles in the Saturn family.[9] It will be applied in the next two sections of this Project Apollo case study when competing mission architectures or programs are considered that will require different classes of vehicles to be developed and different numbers of each to be produced. In those cases, a relative comparison of total program costs through *TransCost* will prove to be valuable.

[9] The full suite of *TransCost* could undoubtedly handle such a comparison but the intricacies of such an implementation are left for future work.

The objectives of this section of the case study have been completed: the sizing process has been demonstrated on the Saturn IB. The capabilities and limitations of the process has been illustrated with the inclusion of the Saturn V and its sizing results. The sizing process is considered valid to the level of fidelity required by the *Ariadne* prototype system and is able to perform the needed calculations for the rest of the Project Apollo case study.

4.3 *Comparison of mission architectures*

This section of the case study will serve to validate the mission modeling approach of the *Ariadne* prototype. First, an introduction is given to the historical significance of the mission architecture decision of Project Apollo. Mission models are presented for the three competing architectures along with two primary connecting parameters that allow for a quantified trade between all three approaches. The phases of each mission architecture are stepped through to calculate the required IMLEO for each architecture. The required launch vehicles are then sized and multiple solution space comparisons between the alternative architectures are presented.

Lunar Direct Flight for Lunar Landing

Input list of maneuvers/phases
Determine all required attributes for each phase: ΔV, t, payload $= f(t)$, I_{sp}, etc.
Input mass at Earth interface (re-entry): $m_{payload}$
Calculate mass required for trans-Earth injection: $m_{TEI} = f(\Delta V, t, I_{sp}, m_{payload})$
Calculate mass required to ascend from lunar surface: $m_{LA} = f(\Delta V, t, I_{sp}, m_{TEI})$
Calculate mass required to land on lunar surface: $m_{LL} = f(\Delta V, t, I_{sp}, m_{LA})$
Calculate mass required for lunar orbit insertion: $m_{LOI} = f(\Delta V, t, I_{sp}, m_{LL})$
Calculate mass required for trans-lunar injection: $m_{TLI} = f(\Delta V, t, I_{sp}, m_{LOI})$
Calculate mass required for Earth launch to orbit: $m_{total} = f(\Delta V, t, I_{sp}, m_{TLI})$

FIGURE 4.13 – Structogram representation of a reduced order model for the Direct flight mission architecture.

THE OBJECTIVES for this section of the Project Apollo case study on mission architecture modeling are to:

» Demonstrate the reduction of alternative architectures down to the minimum number of primary parameters through logical assumptions;

» Introduce the *split ratio* (χ), one of the primary parameters involved when comparing manned lunar landing architectures;

» Develop a generic parametric architecture to visualize the solution space of the mission architectures under consideration during the early years of Project Apollo.

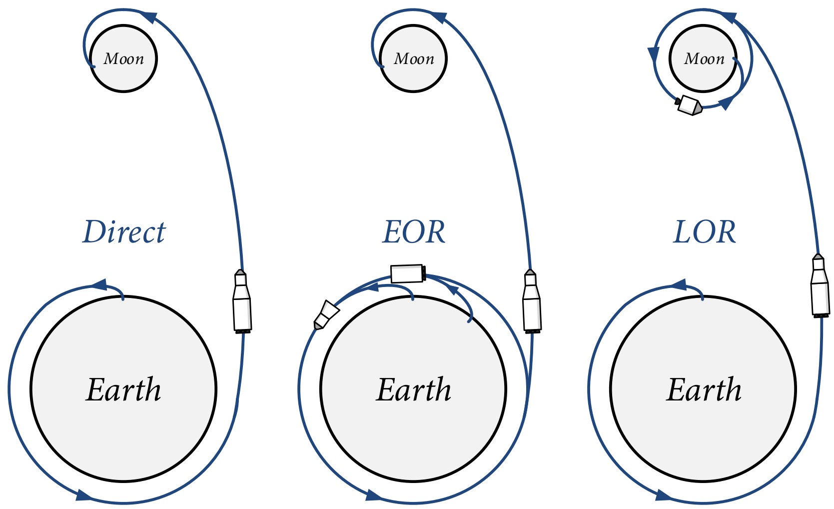

FIGURE 4.14 – Illustration of the three primary candidate mission architectures for a manned lunar landing. Adapted from NASA [23].

4.3.1 *Background of the mission architecture decision*

One of the main reasons that Project Apollo was selected as a case study was the critical decision on which approach to take to actually land on the moon. Three primary alternative mission architectures were proposed: Direct flight (Figure 4.13), Earth orbit rendezvous (Figure 4.15), and lunar orbit rendezvous (Figure 4.16).

The Direct flight architecture for a manned lunar landing offers the most straight-forward approach. A launch vehicle takes off from Earth straight into a trans-lunar injection. The spacecraft then maneuvers into a lunar orbit, before descending to the surface. The astronauts could then disembark and conduct their planned mission (explore, experiment, *etc.*) before launching again into lunar orbit. From there, one final burn would place them on a trajectory towards Earth where they would enter, descend and land (EDL) back on the surface. This mission architecture is shown on the left of Figure 4.14.

Earth orbit rendezvous (EOR) is very similar to the Direct flight architecture except that it divides the required payload over multiple launches to rendezvous and combine in Earth orbit before continuing on to the moon for a landing and return. This mission architectures enables the use of smaller launch vehicles, but at the cost of requiring multiple coordinated launches and subsequent rendezvous for the mission to be successful. The EOR architecture is depicted in the middle of Figure 4.14.

The final mission architecture considered was the lunar orbit rendezvous (LOR). This approach requires a spacecraft to be placed into lunar orbit. Once there, this spacecraft would separate into two smaller spacecraft: one spacecraft containing the men to land on the moon

EARTH ORBIT RENDEZVOUS FOR LUNAR LANDING

Input list of maneuvers/phases
Determine all required attributes for each phase: ΔV, t, payload $= f(t)$, I_{sp}, etc.
Input mass at Earth interface (re-entry): $m_{payload}$
Calculate mass required for trans-Earth injection: $m_{TEI} = f(\Delta V, t, I_{sp}, m_{payload})$
Calculate mass required to ascend from lunar surface: $m_{LA} = f(\Delta V, t, I_{sp}, m_{TEI})$
Calculate mass required to land on lunar surface: $m_{LL} = f(\Delta V, t, I_{sp}, m_{LA})$
Calculate mass required for lunar orbit insertion: $m_{LOI} = f(\Delta V, t, I_{sp}, m_{LL})$
Calculate mass required for trans-lunar injection: $m_{TLI} = f(\Delta V, t, I_{sp}, m_{LOI})$
Divide mass in Earth orbit by the planned number of launchers: $m_{LEO} = m_{TLI} / N$
Calculate mass required for Earth launch of split payload to orbit: $m_{total} = f(\Delta V, t, I_{sp}, m_{LEO})$

FIGURE 4.15 – Structogram representation of a reduced order model for the Earth orbit rendezvous mission architecture.

and the other to remain in lunar orbit. With the mission on the surface complete, the astronauts would launch back into lunar orbit and rendezvous with the craft that remained in orbit. Then, together, the total spacecraft would return to Earth. This architecture is shown as the rightmost mission profile in Figure 4.14.

W. Amster spoke to a number of considerations when comparing these competing mission modes. He said:

> The direct mission is simplest in operation and requires the largest launch vehicle. The Earth orbit rendezvous missions can use smaller launch vehicles at the expense of increased complexity in orbital operations. This type of operation places a particularly severe burden on launch facilities and procedures since several vehicles must be readied for launching almost simultaneously...
>
> The lunar orbit rendezvous uses only one of the smaller vehicles but requires a rather intricate rendezvous in lunar orbit. [24]

At the time of this decision (early 1960s) no spacecraft had ever performed a rendezvous [25]. Thus, the planners in favor of the Direct architecture preferred its architectural simplicity over the unknown difficulty of rendezvous in Earth orbit. A rendezvous in lunar orbit, 380,000 *km* away, was enough for many to ignore it from consideration. This was the hill that LOR had to climb.

A number of accounts cover this mission architecture decision and the eventual selection of the lunar orbit rendezvous approach [7, 8, 26–29]. The following section applies the mission modeling approach introduced in Section 3.3.4 to each of these three competing mission architectures.

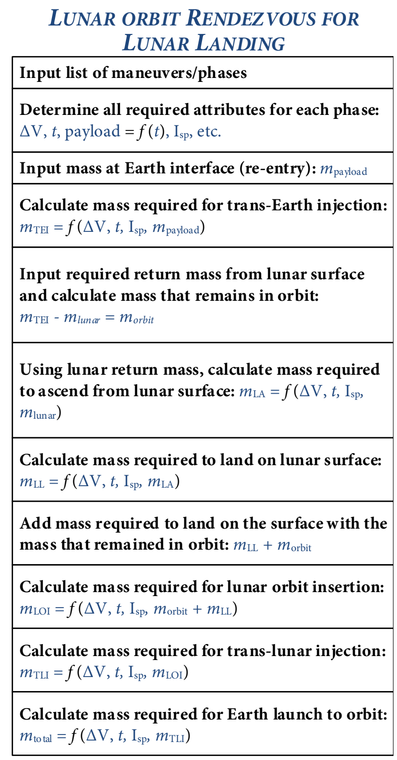

FIGURE 4.16 – Structogram representation of a reduced order model for the lunar orbit rendezvous mission architecture.

4.3.2 *Developing a parametric model*

The mission architecture modeling process divides a manned lunar landing mission into the following phases:

» Launch to Earth orbit;

» Trans-lunar injection;

» Lunar orbit insertion;

» Descent and landing on the lunar surface;

» Ascent from lunar surface;

» Trans-Earth injection;

» Re-entry, descent, and landing back on Earth.

A couple of key assumptions are made in order to focus this analysis only on the primary differences between the three architectures. First,

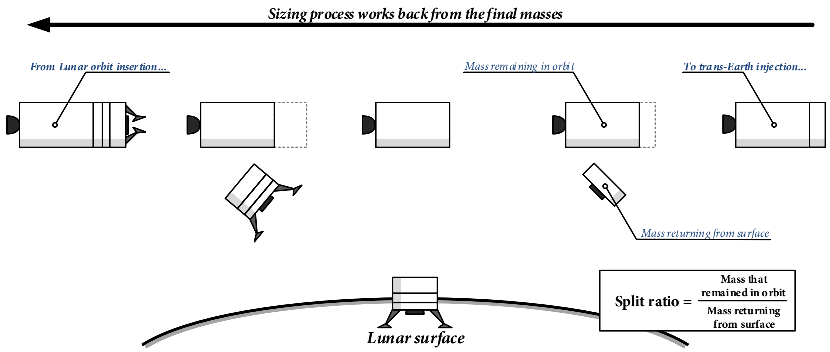

FIGURE 4.17 – Depiction of the *split ratio* (χ) parameter for modeling the competing manned lunar landing architectures.

the final Earth re-entry, descent and landing phase is assumed to be accounted for in the final mission payload mass. This means that for Apollo, the final payload mass used in the analysis is the complete capsule with three astronauts prior to Earth re-entry.[10] Second, only the required propulsion systems are sized for each phase. Time, a notable primary driver in power generation and life support systems, is explicitly ignored for this comparison. It will be seen that this is acceptable for the short duration moon trip but would not hold for the longer travel times to Mars or elsewhere. The use of a generous structure factor accounts for a majority of this mass without getting lost in the details.

[10] The ideal *Ariadne* system would only require the number of astronauts; the required capsule, heat shield, recovery devices, *etc.* would be parametrically sized to account for the required passengers.

With these assumptions in place, the only differences between the three competing architectures comes down to two parameters: the number of launch vehicles used (n), and the *split ratio* (χ). This split ratio is depicted in Figure 4.17. Both parameters are discussed below. Beyond these two parameters, any improvement made to optimize one of the architectures could likely be repeated for the others.[11]

[11] *e.g.*, during the Apollo missions, the mass of the lunar ascent vehicle was ejected before the rest of the Command/Service Module (CSM) made the TEI burn. This reduces the amount of propellant required for that maneuver, ultimately lowering the overall IMLEO. Such an action is desirable but ultimately unnecessary for inclusion when the goal is the initial comparison between competing architectures.

As was discussed previously in Section 3.3.4, when sizing the in-space element for a given mission architecture, the process begins with the final payload mass and works backwards through each phase, calculating the propellant required to complete all of the required maneuvers. For manned lunar landing architectures, a critical point occurs in lunar orbit, right after the astronauts have ascended from the surface. For the Direct and EOR architectures, the mass returning from the surface is in fact the entire spacecraft (the entire crew, supplies for the journey back to Earth, heat shield and parachutes for EDL at Earth, *etc.*), but for LOR a portion of the total spacecraft mass remains in orbit while the other lands and returns. The *split ratio* (χ) of a lunar landing architecture, is therefore defined as

$$\chi = \frac{\text{Mass that remained in orbit}}{\text{Mass returning from surface}}. \tag{4.1}$$

Essentially, the split ratio is the percentage of mass that remained in orbit while a lunar module (LM) landed on the surface and returned to orbit. For the Direct and EOR architectures, no mass remained in orbit so $\chi = 0$. The LOR architecture then includes any architecture where the split ratio, $\chi > 0$.

The other parameter, the number of launches to LEO, involves dividing the final calculated IMLEO into n payloads and then sizing a launch vehicle capable of launching this reduced payload. Of course, while EOR gains the benefit of requiring a smaller launch vehicle, multiple launches are required.

A manned lunar landing architecture based on the split ratio and number of launches (χ and N) is depicted in Figure 4.18. This parametric representation of a generic lunar landing and return architecture will allow the generation and comparison of consistent solution spaces. Figure 4.19 represents an example of one such solution space.[12] This Space Planners Guide-inspired nomograph compares the Direct mission architecture with LOR for a range of selected split ratios. The x-axis represents the final mission mass, *i.e.,* the re-entry capsule for the selected mission. The solid black lines represent alternative mission architectures based on their split ratio, from 0 (Direct flight) to 0.90. By drawing a line up from the mission payload to the desired mission architecture, and then left to the y-axis, a planner can obtain an estimate of the required launch vehicle. This figure will be discussed further in the next section and reproduced with the appropriate validation data.

Manned Lunar Landing

Input list of maneuvers/phases
Determine all required attributes for each phase: ΔV, payload, I_{sp}, etc.
Input mass at Earth interface (re-entry): $m_{payload}$
Calculate mass required for trans-Earth injection: $m_{TEI} = f(\Delta V, I_{sp}, m_{payload})$
Input split ratio to find return mass from lunar surface and mass that remains in orbit: $m_{TEI} \times \chi = m_{orbit}$ $m_{TEI} \times (1 - \chi) = m_{lunar}$
Using lunar return mass, calculate mass required to ascend from lunar surface: $m_{LA} = f(\Delta V, I_{sp}, m_{lunar})$
Calculate mass required to land on lunar surface: $m_{LL} = f(\Delta V, I_{sp}, m_{LA})$
Add mass required to land on the surface with the mass that remained in orbit: $m_{LL} + m_{orbit}$
Calculate mass required for lunar orbit insertion: $m_{LOI} = f(\Delta V, I_{sp}, m_{orbit} + m_{LL})$
Calculate mass required for trans-lunar injection: $m_{TLI} = f(\Delta V, I_{sp}, m_{LOI})$
Divide mass in Earth orbit by the planned number of launchers: $m_{LEO} = m_{TLI} / N$
Calculate mass required for Earth launch of split payload to orbit: $m_{total} = f(\Delta V, I_{sp}, m_{LEO})$

FIGURE 4.18 – Structogram representation of a reduced order, parametric model for a manned lunar landing mission architecture

[12] This solution space was generated with the mission model described here and the sizing process of the Space Planners Guide. The input launch vehicle characteristics were similar to those used for the Saturn V sizing in the previous section.

4.3.3 *Walk-through comparison of the Direct, EOR, and LOR mission architectures*

This section steps through the generic manned lunar landing architecture depicted in Figure 4.18. Another depiction of this architecture is provided in Figure 4.20. The dashed paths represent the possible alternative maneuvers outside of the Direct flight architecture: multiple launches for EOR and lunar orbit activities for the LOR architecture.

A walk-through will show that a first order comparison of these competing mission architectures can be made with only an assumed re-entry payload mass, split ratio, and desired number of launches.

Remember, to size IMLEO, the final mass is assumed and the process works backwards through each of the mission phases. Thus, following along with Figure 4.20 in reverse, the process begins with an assumed mass at Earth re-entry[13] and seeks to determine the mass prior to trans-Earth injection.[14]

[13] Point 9 in Figure 4.20.

[14] Point 8 in Figure 4.20.

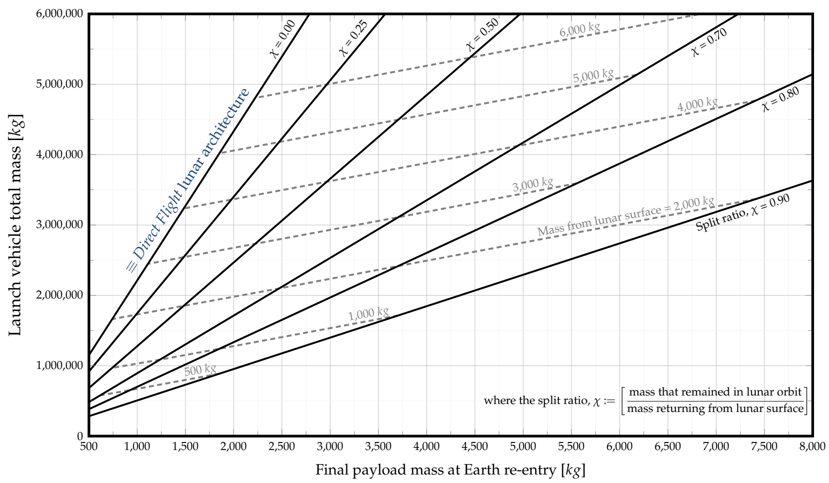

FIGURE 4.19 – Comparison nomograph of the Direct flight architecture with the LOR architecture with a range of selected split ratios.

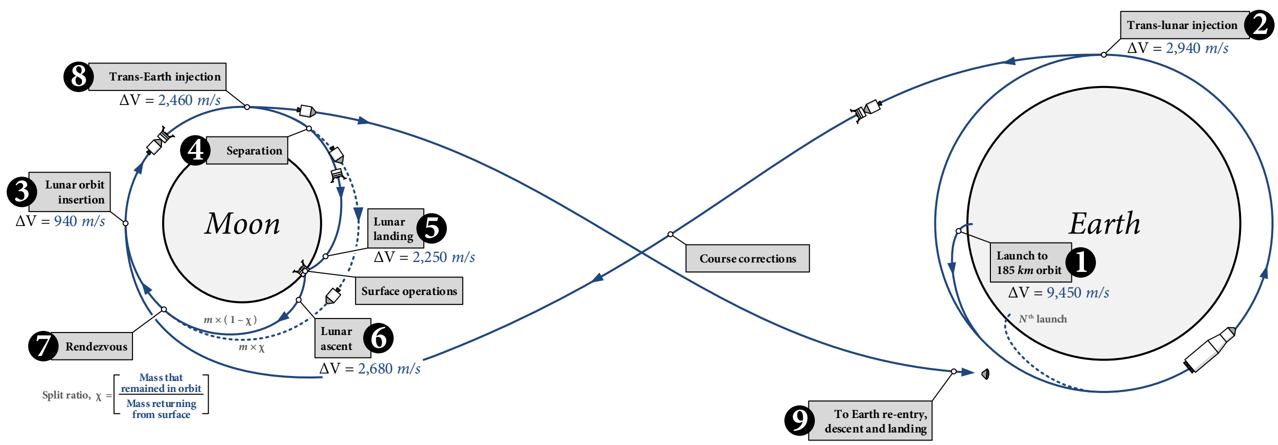

FIGURE 4.20 – Mission profile for a generic manned lunar landing architecture. Adapted from an LOR profile by Micklewait [30].

» Trans-Earth injection (TEI)

For the case study, the assumed re-entry mass of the Direct flight[15] mission architecture is 2,500 *kg*, approximately the mass of the Gemini capsule (2 astronauts) uprated with a heavier heat shield to handle the increased speeds upon re-entry.

Based on the actual Apollo missions, the modeled LOR mission architecture uses three astronauts, with one remaining in lunar orbit while the other two land and return. The assumed re-entry mass is 6,000 *kg*, approximately the re-entry mass of the actual Apollo command module.

The TEI phase is the final kick of the spacecraft out of Lunar orbit on a return trajectory to re-enter Earth's atmosphere. The velocity requirements, assumed propulsion performance, along with the resultant mass breakdown of the propulsion system required for the phase

[15] Through these phases, the Direct flight and EOR mission architectures are identical until the IMLEO is divided up for multiple launches. Thus, for the sake of simplicity, only the Direct flight architecture will be mentioned in the text. The summary table for each phase will include the EOR data as a reminder.

Phase » Trans-Earth injection (TEI)	**Direct/EOR**	**LOR**
Phase final mass, *kg*	2,500	6,000
Velocity required, *m/s*	2,460	2,460
Specific impulse, *s*	300	300
Mass ratio	2.307	2.307
Structure factor	0.1	0.1
Structure mass, *kg*	425	1,019
Burnout mass, *kg*	2,925	7,019
Propellant mass, *kg*	3,822	9,173
Phase initial mass, *kg*	**6,747**	**16,192**

TABLE 4.7 – Comparison of Apollo mission architectures » Trans-Earth injection phase

Phase » Lunar ascent	**Direct/EOR**	**LOR**
Phase final mass, *kg*	6,747	2,591
Velocity required, *m/s*	2,680	2,680
Specific impulse, *s*	300	300
Mass ratio	2.486	2.486
Structure factor	0.1	0.1
Structure mass, *kg*	1,334	512
Burnout mass, *kg*	8,081	3,103
Propellant mass, *kg*	12,008	4,611
Phase initial mass, *kg*	**20,088**	**7,714**

TABLE 4.8 – Comparison of Apollo mission architectures » Lunar ascent phase

are shown in Table 4.7. For comparison, the mass of the average Apollo mission prior to the TEI burn was around 16,400 *kg* [3, 31].

» Ascent and rendezvous

Referring again to Figure 4.20, working backwards from the TEI phase, the next step is where three mission architectures differ. For Direct flight, the entire spacecraft must be propelled from the lunar surface into orbit.[16]

LOR takes advantage of a rendezvous: mass of a spacecraft from the lunar surface meetings returning to a spacecraft that remained in orbit around the moon.[17] This is modeled with the previously discussed split ratio, χ. For the case study, an χ of 0.84 was used. This means that prior to the TEI burn, 84% of the spacecraft's mass had remained in orbit while the other 16% had just returned from the lunar surface. This selected ratio leads to a comparable return mass with the actual Apollo lunar lander ascent stage, though the full range of possible χ values will be shown at the end of this process in Figure 4.21. For this LOR example, the mass that remained orbit is about 13,600 *kg* and the mass that returned from the surface was 2,591 *kg*.

Table 4.8 shows the mass properties of the spacecraft for the lunar ascent phase. Already, the benefits of the rendezvous architecture are

[16] Point 6 in Figure 4.20

[17] Point 7 in Figure 4.20

Phase » Lunar landing	Direct/EOR	LOR
Phase final mass, kg	20,088	7,714
Velocity required, m/s	2,550	2,550
Specific impulse, s	300	300
Mass ratio	2.378	2.378
Structure factor	0.1	0.1
Structure mass, kg	3,633	1,395
Burnout mass, kg	23,722	9,109
Propellant mass, kg	32,700	12,557
Phase initial mass, kg	**56,422**	**21,667**

TABLE 4.9 – Comparison of Apollo mission architectures » Lunar landing phase

becoming evident. The Direct flight has to carry everything needed for the trip back and re-entry down to the surface, leading to a much larger lander.

» Lunar landing

Continuing backwards through Figure 4.20, the next phase consists of the lunar landing.[18] All the architectures behave similarly through this phase though it can be seen how the larger masses in the Direct flight architecture lead to larger propellant requirements, and thus larger masses for the next phase. The mass breakdown of the spacecraft during the phase can be seen in Table 4.9.

[18] Point 5 in Figure 4.20

» Separation

Prior to the lunar landing phase, both mission architectures are in lunar orbit. This is were LOR architecture first splits the spacecraft in two. Since the sizing process is happening in reverse, this phase is where the mass sized for the lunar landing (21,667 kg as given in Table 4.9) and the mass that remained in orbit (13,600 kg) are added back together as the final mass to be used in the analysis of the lunar orbit insertion phase (LOI). This phase does not apply to the Direct flight architecture.

» Lunar orbit insertion

The breakdown of the masses involved in the lunar orbit insertion phase[19] can be seen in Table 4.10. For comparison, the average mass of the spacecraft on the Apollo missions at this point was about 45,150 kg, so the minimal mission model is with 11% of the actual Apollo Command/Service Module (CSM) [3, 31].

[19] Point 3 in Figure 4.20

» Trans-lunar injection

The final phase in these lunar landing architectures is the trans-Lunar injection burn.[20] All of the previous phases have assumed a

[20] Point 2 in Figure 4.20

Phase » Lunar orbit insertion (LOI)	**Direct/EOR**	**LOR**
Phase final mass, *kg*	56,422	35,268
Velocity required, *m/s*	940	940
Specific impulse, *s*	300	300
Mass ratio	1.376	1.376
Structure factor	0.1	0.1
Structure mass, *kg*	2,462	1,539
Burnout mass, *kg*	58,884	36,806
Propellant mass, *kg*	22,158	13,850
Phase initial mass, *kg*	**81,042**	**50,657**

TABLE 4.10 – Comparison of Apollo mission architectures » Lunar orbit insertion phase

Phase » Trans-lunar injection (TLI)	**Direct/EOR**	**LOR**
Phase final mass, *kg*	81,042	50,657
Velocity required, *m/s*	2,940	2,940
Specific impulse, *s*	421	421
Mass ratio	2.041	2.041
Structure factor	0.1	0.1
Structure mass, *kg*	10,603	6,627
Burnout mass, *kg*	91,645	57,284
Propellant mass, *kg*	95,424	59,647
Phase initial mass, *kg*	**187,069**	**116,931**

TABLE 4.11 – Comparison of Apollo mission architectures » Trans-lunar injection phase

propulsion performance (I_{sp}) of 300 s, a fairly typical specific impulse for storable propellants that are used for in-space maneuvers. During the actual Apollo missions, however, this phase was propelled by the Saturn IV stage, which used LOX–LH_2 as a propellant and had an I_{sp} of 421 s. Thus, this is the I_{sp} value that was used for the TLI phase in this case study.

Prior to the burn, the mass in Earth orbit has been calculated in Table 4.11. This value represents the IMLEO of the architecture and provides an initial means of comparing competing approaches. The actual Apollo hardware, prior to TLI, was around 135,000 kg, so now the LOR-sized IMLEO is about 12 % less than the actual.[21]

The mass required by the Direct flight architecture is substantially larger, about 46 %, even with the smaller crew and smaller return capsule. This larger mass, of course, will require a larger launch vehicle to place it into orbit. This is where the benefits of the EOR architecture become apparent. The IMLEO of the Direct architecture is split up over multiple launches (N) which rendezvous in LEO before initiating the TLI burn. Early plans called for as many as a 14 launches of smaller launch vehicles to rendezvous in LEO to be capable of completing the mission [29]. Eventually it was determined that the mission could be completed with two launches of the Saturn V launch vehicle that was

[21] This is still remarkably close considering how little information is required. The discrepancy is likely from the included structure on the Saturn IV stage after its 1^{st} burn to get into orbit.

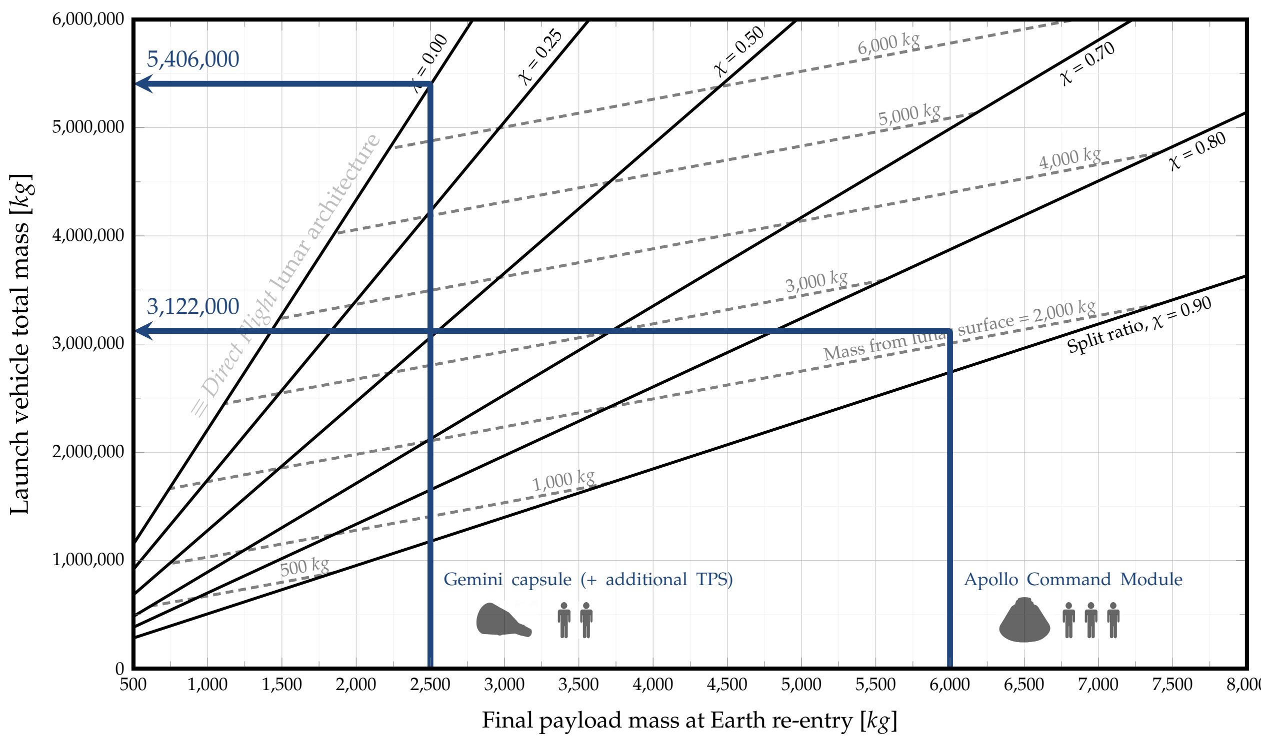

FIGURE 4.21 – Validation data on the comparison nomograph of the competing architectures for manned lunar landing.

under development at the time. In order to properly compare LOR vs. EOR, costs must be taken into consideration, which will be addressed in the following section along with a summary and discussion of the rest of the results of this walk-through.

4.3.4 *Results and discussion*

The previous walk-through comparison of the Direct, EOR, and LOR mission architectures identified the fundamental strength of the LOR approach, especially when compared to the Direct flight architecture. These two architectures can be compared for the entire sweep of possible split ratios as seen in nomograph-form in Figure 4.21. The two assumed payloads are depicted as inputs along the x-axis. From here, each is traced upward until the appropriate split ratio (denoted by the solid black lines) is reached. Then, a line is drawn straight across to the y-axis to determine the total mass of the required launch vehicle. Since this is a study of Project Apollo, the launch vehicles sized for this figure have two-stages with propellant selection and propulsion performance similar the the Saturn V.

The two resulting launch vehicles are as expected and predicted from studies during those early years working on Apollo [24, 32, 33]. The LOR architecture could be completed with a launch vehicle similar to the Saturn V while the Direct flight architecture would likely require a new class of launcher, from the NOVA launch vehicle family. More important than these two specific cases, however, is the ability to observe other possible solutions in the topography around them. For example, the gray dashed lines represents lines of constant mass returning from the lunar surface. Assume that a extensive study has been conducted and determined that the absolute minimum lunar ascent module that an organization is capable of developing for two astronauts is 2,000 *kg*. This is close to the Apollo lunar ascent stage. However, for this hypothetical situation, instead of using a third astronaut in lunar orbit like Apollo, the mass that remains in orbit is further reduced by remaining unmanned while the two astronauts descend to the surface. Trace the dashed gray line representing a constant return mass of 2,000 *kg* down to the left and it can be seen that as it nears the re-entry mass of the strengthen Gemini capsule, the launch vehicle required is now about two-thirds the size of the Saturn V. Of course such an approach would involve a whole new set of risks, but at least with the visualization of the solution space, the strategic planner is aware of the possibility and can make a better informed decision on what needs to be done next.

As previously mentioned, to properly compare EOR with both Direct and LOR architectures, the cost of the required launch vehicles required for the mission must be taken into account. Fortunately, the *Ariadne* system is capable of estimating the launch vehicle costs (both development and production). Figure 4.22 represents the solution space between all three competing architectures and other combinations of χ and N. The assumed final mission payload mass in Figure 4.22 is the uprated Gemini capsule, 2,500 *kg*. A sweep of N EOR launches, from 1–3 and a sweep of χ from 0 to 0.9 created the 15 alternative mission architectures shown in the figure. The final payload is applied to each architecture and the in-space sizing process applied to determine the IMLEO/N, located along the x-axis. The required launch vehicle(s) for a given mission are then sized and the total costs estimated, located on the y-axis. To enable easier comparisons, the total cost have been normalized to the lowest cost architecture. In Figure 4.22, this reference architecture is the one located in the bottom left corner. This mission architecture involves three launches to LEO before heading to the moon where it uses a split ratio of 0.9, which is an even higher split than was used on the Apollo missions. Obviously, such an architecture would involve additional hardware complexities and risks that would have to be evaluated to determine if the cost savings are worth it or not. Conversely, the Direct flight architecture is shown in the upper right corner of Figure 4.22. This architecture is the simplest approach but also the most expensive (around 2.5 times the cost of the reference

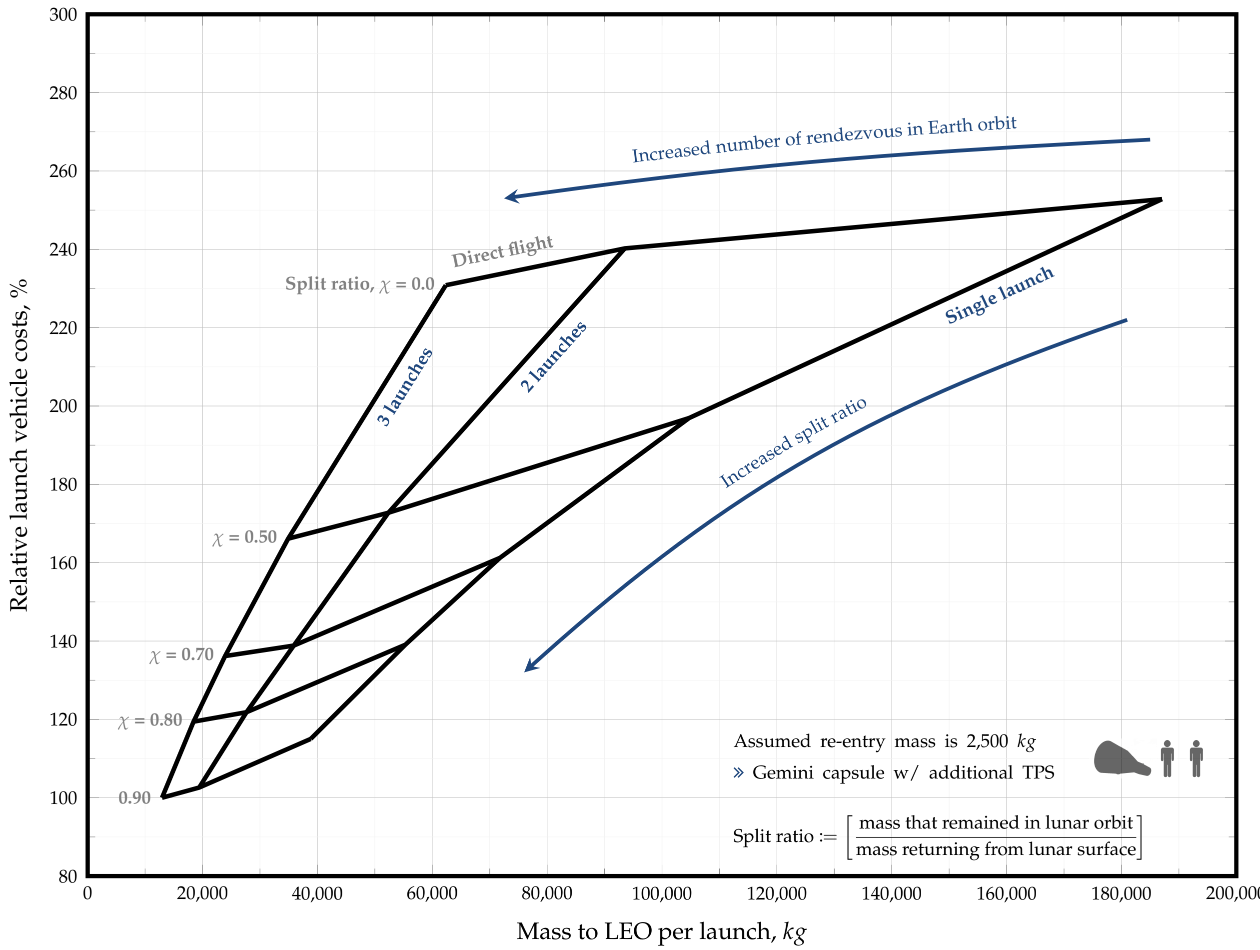

FIGURE 4.22 – A parametric trade of χ and N for all three mission architectures competing for Project Apollo.

mission architecture) and the sheer size of the launch vehicle required would give rise to its own unique challenges.

Figure 4.23 includes all three of the competing architectures with the assumed payloads used previously in the walk-through of the phases. The solution space formed by the solid lines represents the uprated Gemini capsule payload and the dashed lines represent the 6,000 *kg* Apollo capsule. Each architecture has been labeled and the required launch vehicle costs can now be consistently compared.

The mission modeling section of the case study has fulfilled the previously stated objectives. A generic, reduced order lunar mission model was created that primarily differed in only two parameters: the number of launch vehicles desired and the newly defined split ratio. A manned lunar landing and return mission was analyzed using the new parametric model for the Direct, EOR, and LOR mission architectures to determine the IMLEO for each. Finally, the entire solution space between these approaches and others was visualized and validated in Figures 4.21 – 4.23. The impact of the mission architecture on the rest

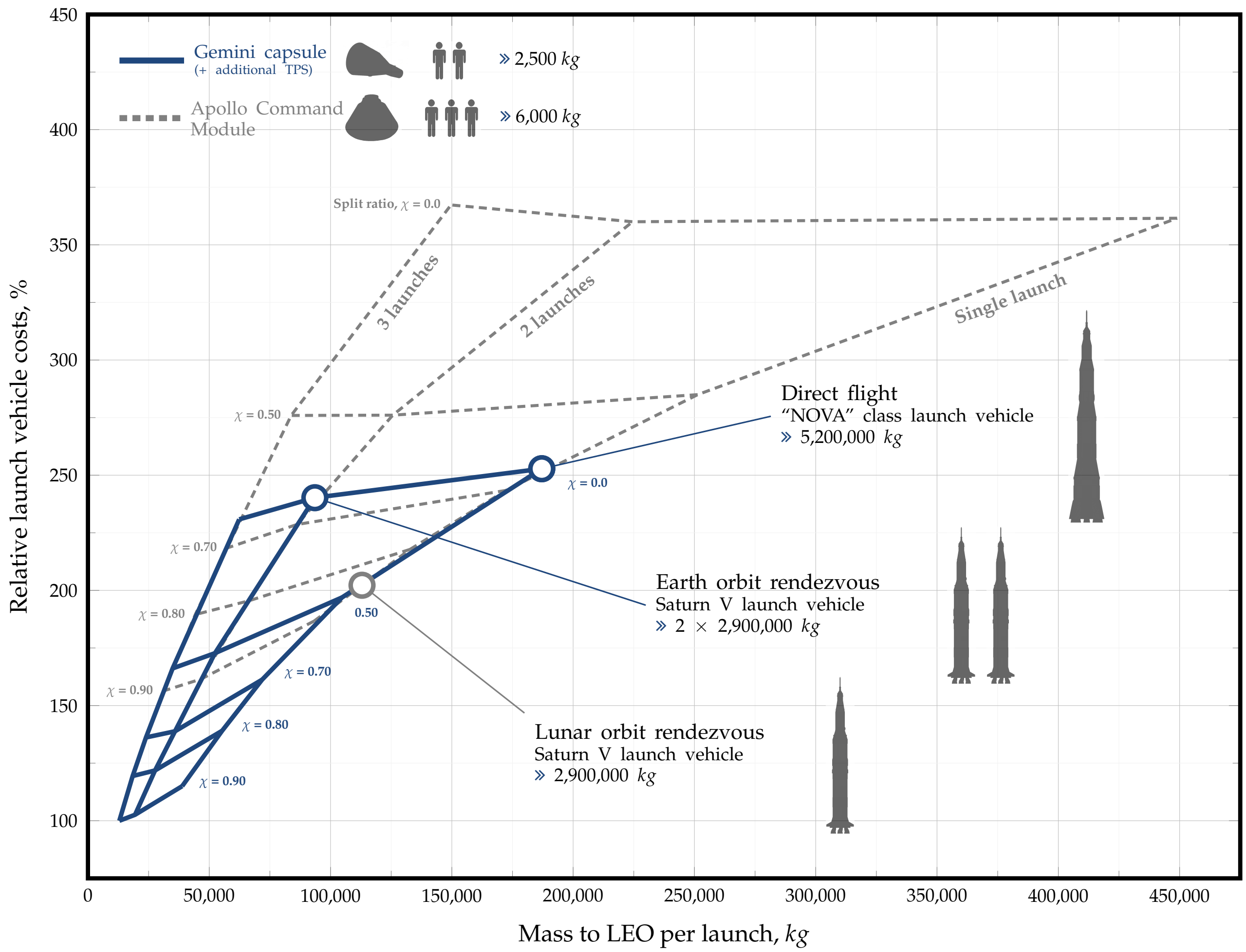

FIGURE 4.23 – Validation data on the solution space of the three competing mission architectures for Project Apollo.

of the program will be discussed again the in final section of this case study on Project Apollo.

4.4 *Comparison of program alternatives leading up to Project Apollo*

The previous two sections of the case study have demonstrated the powerful insights into Project Apollo that can be gained with the proper mentality and tools for analysis. This section is the final part of the case study and considers entire space programs. First, this section expands on the introduction of this chapter and provides some additional information on the national setting leading up to the key decisions for Project Apollo. Two alternative sets of spacefaring goals and implementation strategies are introduced and their resultant space program objectives compared. Next, a desired set of goals are selected, leading to the definition of multiple candidate space programs by sweeping through the possible implementation strategy factors. Finally, multiple trades studies are conducted on the program level to compare alternative program objectives and mission architectures.

THE OBJECTIVES of this final section of the Project Apollo case study are provided below:

» Demonstrate the capability of the *Ariadne* prototype system to appropriately derive space program objectives for alternative sets of spacefaring goals and implementation strategies;

» Visualize the solution space topography of candidate space programs with a constant set of spacefaring goals;

» Visualize and compare the effect that mission architecture selection has on the entire program;

» Compare alternative program objectives with the historical objectives of Project Apollo.

4.4.1 Background of possible program directions

The early planners of the U.S. space program, prior to Kennedy's challenge for the Moon, were considering many different directions for the program. Table 4.12 contains a list of program milestones recommended by the U.S. National Advisory Committee for Aeronautics (NACA), the predecessor to NASA. To say that some of their milestones were ambitious would still be quite an understatement. The list of milestones begins with a 9 *kg* payload, expected to be launched that year (1958), and then includes a 20-man space station by 1966 and the establishment of a permanent moon base by 1974.

Two years later, now better understanding some of the realities of space flight (complexity, costs, *etc.*), the NASA Office of Program Planning and Evaluation published their *Ten Year Plan* [36] which has been summarized in Table 4.13. Notice the still ambitious, although much more reasonable, pace of the program and the included names of launch vehicles that were under development at the time.[22] Also noteworthy are the final two entries in the plan: first launch in manned circumlunar flight and space station effort by 1967, and a manned lunar landing and return sometime *after* 1970.

In early 1961, NASA's appointed Manned Lunar Landing Group, headed by George Low,[23] published their plan for a manned lunar landing program [37]. Figure 4.24 illustrates the converging tracks of launch vehicle development and manned spacecraft development, along with the supporting research areas, as identified by the Low Committee.

[22] Most noticeably, the 2-stage Saturn launch vehicle, the Saturn I. After the Saturn program received top priority from President Eisenhower [6, 16], the Saturn I had its first launch in 1961, ahead of schedule per this *Ten Year Plan*.

[23] The group became known as the Low Committee.

Item	Date	Event	Vehicle Generation
1	Jan. 1958	First 9 *kg* satellite (ABMA/JPL)	I
2	Aug. 1958	First 14 *kg* lunar probe (Douglas/RW/Aerojet)	II
3	Nov. 1958	First recoverable 140 *kg* satellite (Douglas/Bell/Lockheed)	II
4	May 1959	First 680 *kg* satellite	II
5	Jun. 1959	First powered flight with X-15	–
6	Jul. 1959	First recoverable 950 *kg* satellite	II and/or III
7	Nov. 1959	First 180 *kg* lunar probe	II and/or III
8	Dec. 1959	First 45 *kg* lunar soft landing	II and/or III
9	Jan. 1960	First 135 *kg* lunar satellite	II and/or III
10	Jul. 1960	First wingless manned orbital return flight	II and/or III
11	Dec. 1960	First 4,500 *kg* orbital capability	III
12	Feb. 1961	First 1,300/270 *kg* lunar hard or soft landing	III
13	Apr. 1961	First[1,100 *kg* planetary or solar probe	III
14	Sept. 1961	First flight with 6.7-million-newton thrust	IV
15	Aug. 1962	First winged orbital return flight	III
16	Nov. 1962	Four-man experimental space station	III
17	Jan. 1963	First 13,800 *kg* orbital capability	IV
18	Feb. 1963	First 1,590 *kg* unmanned lunar circumnavigation and return	IV
19	Apr. 1963	First 2,500 *kg* soft lunar landing	IV
20	Jul. 1964	First 1,590 *kg* manned lunar circumnavigation and return	IV
21	Sept. 1964	Establishment of a 20-man space station	IV
22	Jul. 1965	Final assembly of first 900 metric-ton lunar landing vehicle (emergency manned lunar landing capability)	IV
23	Aug. 1966	Final assembly of second 900 metric-ton landing vehicle and first expedition to moon	IV
24	Jan. 1967	First 2,300 *kg* Martian probe	IV
25	May 1967	First 2,300 *kg* Venus probe	IV
26	Sept. 1967	Completion of 50-man, 450 metric-ton permanent space station	IV
27	1972	Large scientific moon expedition	V
28	1973/1974	Establishment of permanent moon base	V
29	1977	First manned expedition to a planet	V
30	1980	Second manned expedition to a planet	V

TABLE 4.12 – Milestones of the *Recommended U.S. Spaceflight Program*, July 1958. Adapted from Ezell [34]. Originally from NACA [35].

Note that the Apollo and Lunar programs are listed separately in this figure. This can also be seen in Figure 4.25 by NASA's Space Task Group. The planned Apollo program at the time included two phases: and "A" phase that expanded the manned capabilities of Earth orbital flight, and phase "B" that involved manned circumlunar flight. An orbital laboratory was included in this initial plan, and long term goals included both a manned lunar landing and a space station.

After the successful flight of Yuri Gagarin in April of 1961 by the Soviet Union, President Kennedy wrote a memo to Vice President Johnson asking "Is there any ... space program which promises dramatic results in which we could win? [38]" The Vice President relayed the question to W. von Braun who responded:

> ...we have an excellent chance of beating the Soviets to the first land-

Date	Event
1960	First launching of meteorological satellite First launching of pass-reflector communications satellite First launching of Scout vehicle First launching of Thor-Delta vehicle First launching of Atlas-Agena B (DoD) First suborbital flight by astronaut
1961	First launching of lunar impact vehicle First launching of Atlas-Centaur vehicle Attainment of orbital manned spaceflight, Project Mercury
1962	First launching of probe to vicinity of Venus or Mars
1963	First launching of 2-stage Saturn
1963–1964	First launching of unmanned vehicle for controlled landing on moon First launching of orbiting astronomical and radio astronomical laboratory
1964	First launching of unmanned circumlunar vehicle and return to Earth First reconnaissance of Mars or Venus, or both, by unmanned vehicle
1965–1967	First launching in program leading to manned circumlunar flight and to permanent near-Earth space station
Beyond 1970	Manned lunar landing and return

TABLE 4.13 – The Ten Year Plan of the National Aeronautics and Space Administration. Adapted from Ezell [34]. Originally from NASA [36].

> ing of a crew on the moon (including return capability, of course). The reason is that a performance jump by a factor of 10 over their present rocket is necessary to accomplish this feat. While today we do not have such a rocket, it is unlikely that the Soviets have it. Therefore, we would not have to enter the race toward this obvious next goal in space exploration against hopeless odds favoring the Soviets. With an all-out crash program I think we could accomplish this objective in 1966/68.
>
> ...I do not believe that we can win this race unless we take at least some measures which thus far have been considered acceptable only in times of a national emergency. [38]

Shortly after this, President Kennedy made his address to Congress that was included at the beginning of this chapter. This set the U.S. space program on a path to eclipse the Soviet efforts, leading to arguably the greatest achievement of mankind. In order to accomplish this feat within the decade, all other objectives had to be postponed.[24] There are many additional sources available that walk through the history leading up to his decision in greater depth [38, 41–45].

[24] Some have argued that the race to the moon was to ultimately detrimental to the U.S. space program in the long run [39–41].

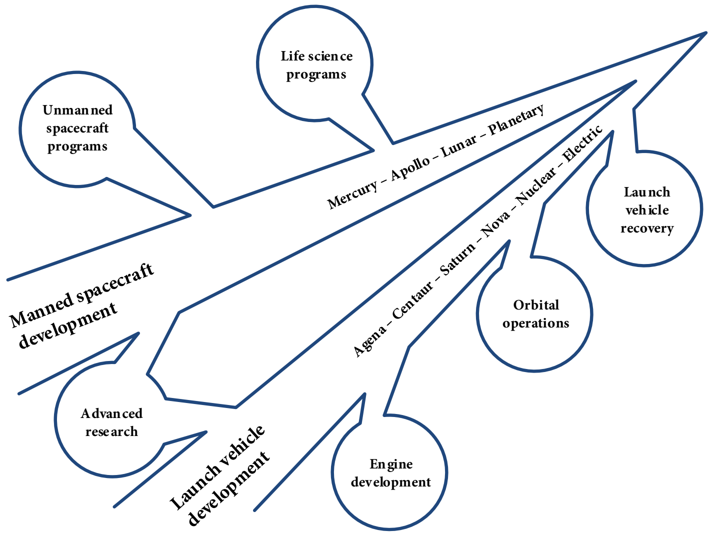

FIGURE 4.24 – Contributing elements of a manned lunar landing program. Reproduced from Low [37].

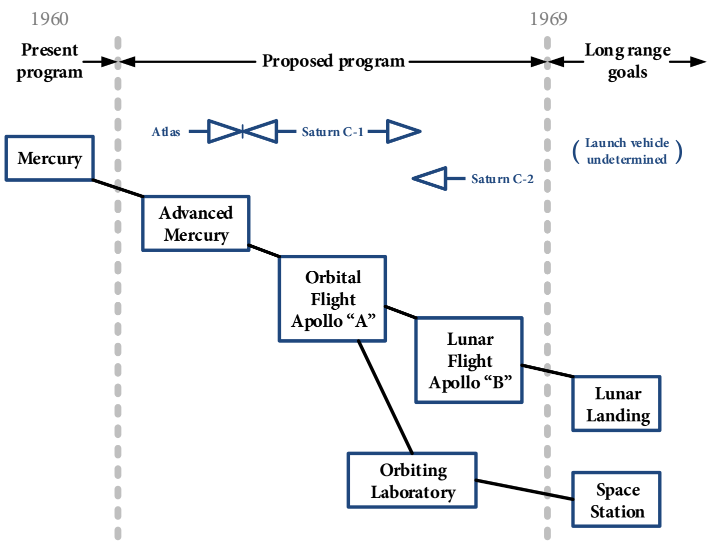

FIGURE 4.25 – Early thoughts on the U.S. manned spaceflight program. Reproduced from NASA's Space Task Group [46].

4.4.2 *Comparison of alternative spacefaring goals*

Project Apollo benefited from a very unique time period in America's history: the cold war with the Soviet Union provided the backdrop for the most unifying and consistent set of goals for the country's space program. Using the AHP framework outlined previously in Chapter 3, two consistent, competing sets of spacefaring goals and strategies have been defined: one set aligned with the historical goals leading up to Project Apollo and the U.S. space program and a second, alternative set of possible goals with a focus on science and little interest in manned activity in space. A lot can be learned by applying the *Ariadne* prototype tool to derive the program objectives for each alternative.

Leading up to Project Apollo » goals and strategies

The following goals and strategies have been compiled from many sources [39, 43, 47–49]. The goal of this section is not to present the definitive set of the "true" spacefaring goals of Apollo, but rather to illustrate the procedure and the transparency it provides.

The prioritized goals used in this example are given in Figures 4.26 – 4.29. Each figure illustrates of the elements of the tier (or sub-tier) being compared with each of the other elements. A numbered line is drawn from -9 to 9 between two goals and a score assigned for the comparison based on the relative importance between the two goals for the space program as a whole.[25]

[25] See Table 2.4 in Appendix A for the definition of each score used in Saaty's pairwise comparisons.

Figure 4.26 depicts an absolute importance of Pragmatism over Science for the Apollo program. This is representative of the U.S. efforts to catch up to the Soviets in manned space flight; yes, science was still done when given the opportunity, but science was not a priority for the manned space program. In this same way, Destiny is given importance over Science as well. Pragmatism is favored over Destiny due to the practicalities of beating the Soviets and all that would mean for the U.S.

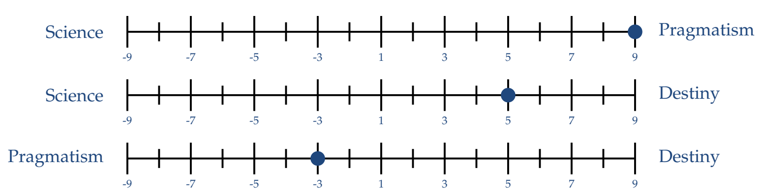

FIGURE 4.26 – The pairwise comparisons of the top tier of the hierarchy of spacefaring goals.

These comparisons can be written in a decision matrix (Equation 4.2) and subjected to the AHP to determine the appropriate weight of each spacefaring goal.

$$\begin{array}{r c} & \begin{array}{ccc} \text{Science} & \text{Pragmatism} & \text{Destiny} \end{array} \\ \begin{array}{r} \textbf{Science} \\ \textbf{Pragmatism} \\ \textbf{Destiny} \end{array} & \begin{bmatrix} 1 & {}^{1}/_{9} & {}^{1}/_{5} \\ 9 & 1 & 3 \\ 5 & {}^{1}/_{3} & 1 \end{bmatrix} \end{array} \qquad (4.2)$$

$$\begin{array}{r c} & \text{Weight} \\ \begin{array}{r} \textbf{Pragmatism, } W_P \\ \textbf{Destiny, } W_D \\ \textbf{Science, } W_S \end{array} & \begin{bmatrix} 0.672 \\ 0.265 \\ 0.063 \end{bmatrix} \end{array}$$

Figure 4.27 details the pairwise comparisons of the Science sub-tier. Overall, science goals tended towards the understanding of Human Beings and the Earth, and the goals were prioritized as such. The

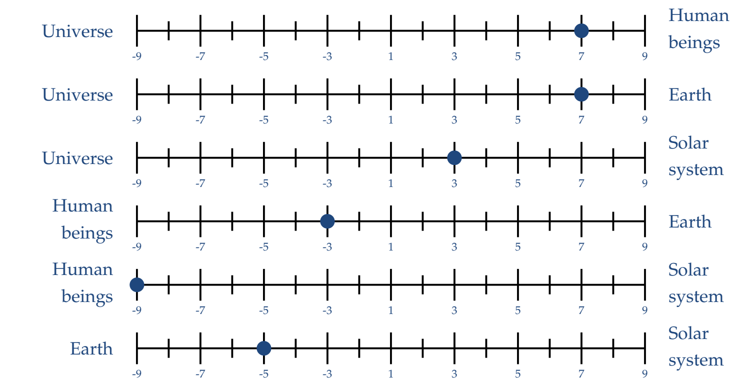

FIGURE 4.27 – The pairwise comparisons of the Science sub-tier of the hierarchy of spacefaring goals.

decision matrix and final weights are provided here.

$$\begin{array}{r c} & \begin{array}{cccc} \text{Universe} & \text{Humans} & \text{Earth} & \text{Solar Sys.} \end{array} \\ \begin{array}{r} \textbf{Universe} \\ \textbf{Humans} \\ \textbf{Earth} \\ \textbf{Solar Sys.} \end{array} & \begin{bmatrix} 1 & 1/7 & 1/7 & 1/3 \\ 7 & 1 & 3 & 9 \\ 7 & 1/3 & 1 & 5 \\ 3 & 1/9 & 1/5 & 1 \end{bmatrix} \end{array} \qquad (4.3) \qquad \begin{array}{r c} & \text{Weight} \\ \begin{array}{r} \textbf{Humans, } W_{S_4} \\ \textbf{Earth, } W_{S_1} \\ \textbf{Solar Sys., } W_{S_2} \\ \textbf{Universe, } W_{S_3} \end{array} & \begin{bmatrix} 0.588 \\ 0.285 \\ 0.081 \\ 0.047 \end{bmatrix} \end{array}$$

Figure 4.28 details the pairwise comparisons of the Pragmatism sub-tier of spacefaring goals. These comparisons capture the fact that De-

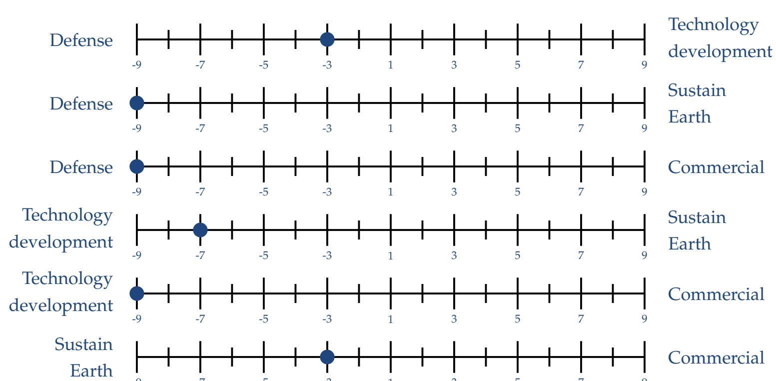

FIGURE 4.28 – The pairwise comparisons of the Pragmatism sub-tier of the hierarchy of spacefaring goals.

fense and Technology development were important goals of the early space program while the space flight was still too new to really provide an opportunity for commercial efforts. The decision matrix and final weights for this sub-tier are listed here.

$$\begin{array}{r} \\ \textbf{Defense} \\ \textbf{Tech. Dev.} \\ \textbf{Sustain} \\ \textbf{Commercial} \end{array} \begin{array}{c} \begin{array}{cccc} \text{Defense} & \text{Tech. Dev.} & \text{Sustain} & \text{Commercial} \end{array} \\ \begin{bmatrix} 1 & 3 & 9 & 9 \\ 1/3 & 1 & 7 & 9 \\ 1/9 & 1/7 & 1 & 3 \\ 1/9 & 1/9 & 1/3 & 1 \end{bmatrix} \end{array} \tag{4.4}$$

$$\begin{array}{r} \\ \textbf{Defense, } W_{P_3} \\ \textbf{Tech. Dev., } W_{P_4} \\ \textbf{Sustain, } W_{P_2} \\ \textbf{Commercial, } W_{P_1} \end{array} \begin{array}{c} \text{Weight} \\ \begin{bmatrix} 0.582 \\ 0.312 \\ 0.068 \\ 0.038 \end{bmatrix} \end{array}$$

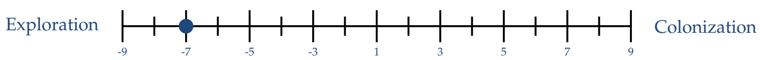

FIGURE 4.29 – The pairwise comparisons of the Destiny sub-tier of the hierarchy of spacefaring goals.

The pairwise comparisons of the last sub-tier, Destiny, are provided in Figure 4.29. Again, due to the infancy of space flight, the main goal was the initial Exploration of space, with any desired or possible Colonization coming much later.

$$\begin{array}{r} \\ \textbf{Exploration} \\ \textbf{Colonization} \end{array} \begin{array}{c} \begin{array}{cc} \text{Exploration} & \text{Colonization} \end{array} \\ \begin{bmatrix} 1 & 7 \\ 1/7 & 1 \end{bmatrix} \end{array} \tag{4.5}$$

$$\begin{array}{r} \\ \textbf{Exploration, } W_{D_1} \\ \textbf{Colonization, } W_{D_2} \end{array} \begin{array}{c} \text{Weight} \\ \begin{bmatrix} 0.875 \\ 0.125 \end{bmatrix} \end{array}$$

The implementation strategies of Apollo that will have an effect on the program objectives include the level of man's involvement in the program (S_{man}), and the level of technology used (S_{tech}).[26] For Project Apollo, the technology factor was selected to be 2 (scaling up a lot of existing hardware) and the level of man's involvement to be 4.

[26] The other two strategy factors will be used in the next section.

Next, a set of alternative goals and strategies is provided to illustrate another possible program that could have emerged at the time.

"SCIENCE IS KING" » ALTERNATIVE GOALS AND STRATEGIES

As an example, consider an alternate Space Race that would have been a race to show off scientific prowess; particularly with knowledge and understanding of the other planets in the Solar System. The space-faring goals of such a program would look quite different and would result in an alternative set of space program objectives. With such a focus on only Science, there is very little interest in sending man up to space, when the required satellites for scientific study can be delivered for much cheaper.[27]

[27] ...especially when the desired destinations are well beyond LEO.

Since the general weighting process has been exercised previously, the figures of pairwise comparisons are omitted and only their results presented in the following decision matrices. The final calculated weight of each goal is listed kin the margin.

$$\begin{array}{r} \\ \textbf{Science} \\ \textbf{Pragmatism} \\ \textbf{Destiny} \end{array} \begin{array}{c} \begin{array}{ccc} \text{Science} & \text{Pragmatism} & \text{Destiny} \end{array} \\ \begin{bmatrix} 1 & 5 & 9 \\ 1/5 & 1 & 3 \\ 1/9 & 1/3 & 1 \end{bmatrix} \end{array} \quad (4.6) \qquad \begin{array}{r} \\ \textbf{Science, } W_S \\ \textbf{Pragmatism, } W_P \\ \textbf{Destiny, } W_D \end{array} \begin{array}{c} \text{Weight} \\ \begin{bmatrix} 0.751 \\ 0.178 \\ 0.070 \end{bmatrix} \end{array}$$

$$\begin{array}{r} \\ \textbf{Universe} \\ \textbf{Humans} \\ \textbf{Earth} \\ \textbf{Solar Sys.} \end{array} \begin{array}{c} \begin{array}{cccc} \text{Universe} & \text{Humans} & \text{Earth} & \text{Solar Sys.} \end{array} \\ \begin{bmatrix} 1 & 3 & 1/5 & 1/7 \\ 1/3 & 1 & 1/5 & 1/9 \\ 5 & 5 & 1 & 1/3 \\ 7 & 9 & 3 & 1 \end{bmatrix} \end{array} \quad (4.7) \qquad \begin{array}{r} \\ \textbf{Solar Sys., } W_{S_2} \\ \textbf{Earth, } W_{S_1} \\ \textbf{Universe, } W_{S_3} \\ \textbf{Humans, } W_{S_4} \end{array} \begin{array}{c} \text{Weight} \\ \begin{bmatrix} 0.589 \\ 0.275 \\ 0.088 \\ 0.047 \end{bmatrix} \end{array}$$

$$\begin{array}{r} \\ \textbf{Defense} \\ \textbf{Tech. Dev.} \\ \textbf{Sustain} \\ \textbf{Commercial} \end{array} \begin{array}{c} \begin{array}{cccc} \text{Defense} & \text{Tech. Dev.} & \text{Sustain} & \text{Commercial} \end{array} \\ \begin{bmatrix} 1 & 1/3 & 1/7 & 1/5 \\ 3 & 1 & 1/3 & 1/3 \\ 7 & 3 & 1 & 3 \\ 5 & 3 & 1/3 & 1 \end{bmatrix} \end{array} \quad (4.8) \qquad \begin{array}{r} \\ \textbf{Sustain, } W_{P_2} \\ \textbf{Commercial, } W_{P_1} \\ \textbf{Tech. Dev., } W_{P_4} \\ \textbf{Defense, } W_{P_3} \end{array} \begin{array}{c} \text{Weight} \\ \begin{bmatrix} 0.525 \\ 0.279 \\ 0.139 \\ 0.057 \end{bmatrix} \end{array}$$

$$\begin{array}{r} \\ \textbf{Exploration} \\ \textbf{Colonization} \end{array} \begin{array}{c} \begin{array}{cc} \text{Exploration} & \text{Colonization} \end{array} \\ \begin{bmatrix} 1 & 9 \\ 1/9 & 1 \end{bmatrix} \end{array} \quad (4.9) \qquad \begin{array}{r} \\ \textbf{Exploration, } W_{D_1} \\ \textbf{Colonization, } W_{D_2} \end{array} \begin{array}{c} \text{Weight} \\ \begin{bmatrix} 0.90 \\ 0.10 \end{bmatrix} \end{array}$$

For the selected implementation strategies for the alternate "Science is King" program, an S_{tech} of 2 (to keep at least one thing similar between the two alternatives) and an S_{man} of 1. This means that this alternative has no desire for manned space flight, which will be reflected in the derived objectives.

Comparing the derived objectives

Each of the alternatives detailed above were processed with the *Ariadne* prototype system which resulted in two sets of derived space program objectives, summarized in Table 4.14.

Immediately noticeable from Table 4.14 is the lack of any priority assigned to each manned program objective.[28] The selected strategy factor for man's involvement of 1 effectively removes those objectives from consideration.

Also, notice that the "Science is King" strategy has fewer objectives that score over the 50 percentile.[29] This means that the pre-Apollo program has more initial possibilities to consider, but also that it will have to find additional ways of sorting through its potential objectives.[30]

Now, these two alternatives are practically polar opposites and thus the resultant objectives are fairly intuitive. An expanded approach applied in the ideal *Ariadne* system would serve to highlight additional

[28] Recall Table 3.3.

[29] The next set of trades for such program would be to prioritize the desired Solar System destinations based on their scientific value, instrumentation required, and time required and assemble and compare the required space programs. Such analysis is beyond the scope of this research.

[30] ...*e.g.*, the goals highlight potential program objectives a lunar base, supplied colony and cis-lunar space station. It is very unlikely that all three objectives could be pursued simultaneously, so additional efforts must be made to determine the best route forward.

Spacefaring goals of Project Apollo			Alternative set of spacefaring goals		
ID	Objective	Priority	ID	Objective	Priority
O_{103}	Earth space station	**1.00**	O_{308}	Science instruments	**1.00**
O_{102}	Earth sat. (manned)	**0.93**	O_{201}	Planet/moon satellite	**0.76**
O_{301}	Launch propulsion	**0.79**	O_{209}	Solar probe	**0.76**
O_{313}	Ground systems	**0.79**	O_{210}	Interstellar probe	**0.76**
O_{206}	Planet/moon surface base	**0.70**	O_{311}	Modeling/simulation	**0.71**
O_{207}	Planet/moon supplied colony	**0.67**	O_{203}	Planet/moon lander	**0.67**
O_{105}	Cis-lunar space station	**0.66**	O_{305}	Communications	**0.62**
O_{307}	Human exploration systems	**0.65**	O_{101}	Earth satellite	**0.44**
O_{204}	Planet/moon lander (manned)	**0.63**	O_{315}	Aeronautics	**0.31**
O_{106}	Earth orbital aux. vehicle	**0.61**	O_{106}	Earth orbital aux. vehicle	**0.20**
O_{205}	Planet/moon space station	**0.61**	O_{104}	Earth orbital power plant	**0.19**
O_{306}	Human health	**0.59**	O_{303}	Power generation	**0.17**
O_{304}	Robotics	**0.58**	O_{312}	Materials	**0.17**
O_{101}	Earth satellite	**0.58**	O_{304}	Robotics	**0.12**
O_{202}	Planet/moon sat. (manned)	**0.52**	O_{309}	EDL systems	**0.10**
O_{303}	Power generation	**0.38**	O_{301}	Launch propulsion	**0.05**
O_{312}	Materials	**0.38**	O_{313}	Ground systems	**0.05**
O_{314}	Thermal	**0.32**	O_{314}	Thermal	**0.04**
O_{201}	Planet/moon satellite	**0.32**	O_{302}	In-space propulsion	**0.03**
O_{209}	Solar probe	**0.32**	O_{310}	Nanotechnology	**0.03**
O_{210}	Interstellar probe	**0.32**	O_{102}	Earth sat. (manned)	–
O_{203}	Planet/moon lander	**0.31**	O_{103}	Earth space station	–
O_{309}	EDL systems	**0.31**	O_{105}	Cis-lunar space station	–
O_{315}	Aeronautics	**0.30**	O_{107}	Earth orbital colony	–
O_{311}	Modeling/simulation	**0.29**	O_{202}	Planet/moon sat. (manned)	–
O_{305}	Communications	**0.28**	O_{204}	Planet/moon lander (manned)	–
O_{302}	In-space propulsion	**0.28**	O_{205}	Planet/moon space station	–
O_{310}	Nanotechnology	**0.28**	O_{206}	Planet/moon surface base	–
O_{107}	Earth orbital colony	**0.21**	O_{207}	Planet/moon supplied colony	–
O_{208}	Planet/moon sufficient colony	**0.16**	O_{208}	Planet/moon sufficient colony	–
O_{104}	Earth orbital power plant	**0.09**	O_{306}	Human health	–
O_{308}	Science instruments	**0.08**	O_{307}	Human exploration systems	–

TABLE 4.14 – Comparing the derived space program objectives of two different sets of spacefaring goals.

non-intuitive combinations of objectives for consideration. This possibility is left for future work and is discussed in Section 5.2 in the next chapter.

4.4.3 *Comparison of candidate programs*

The set of goals resembling the actual state of the program leading up to Apollo have been selected for further analysis in this section. With the spacefaring goals held constant, various sweeps of the strategy factors are made to synthesize a wide range of programs and determine the solution space topography between them. There are four different plots shown below that do explore the programmatic solution space for strategy factor trades, mission architecture trades, and program objective trades.

CONSTANT SPACEFARING GOALS » SWEEP OF PROGRAM SCALE

For the first example, S_{time} is held constant at 2 and S_{scale} is swept

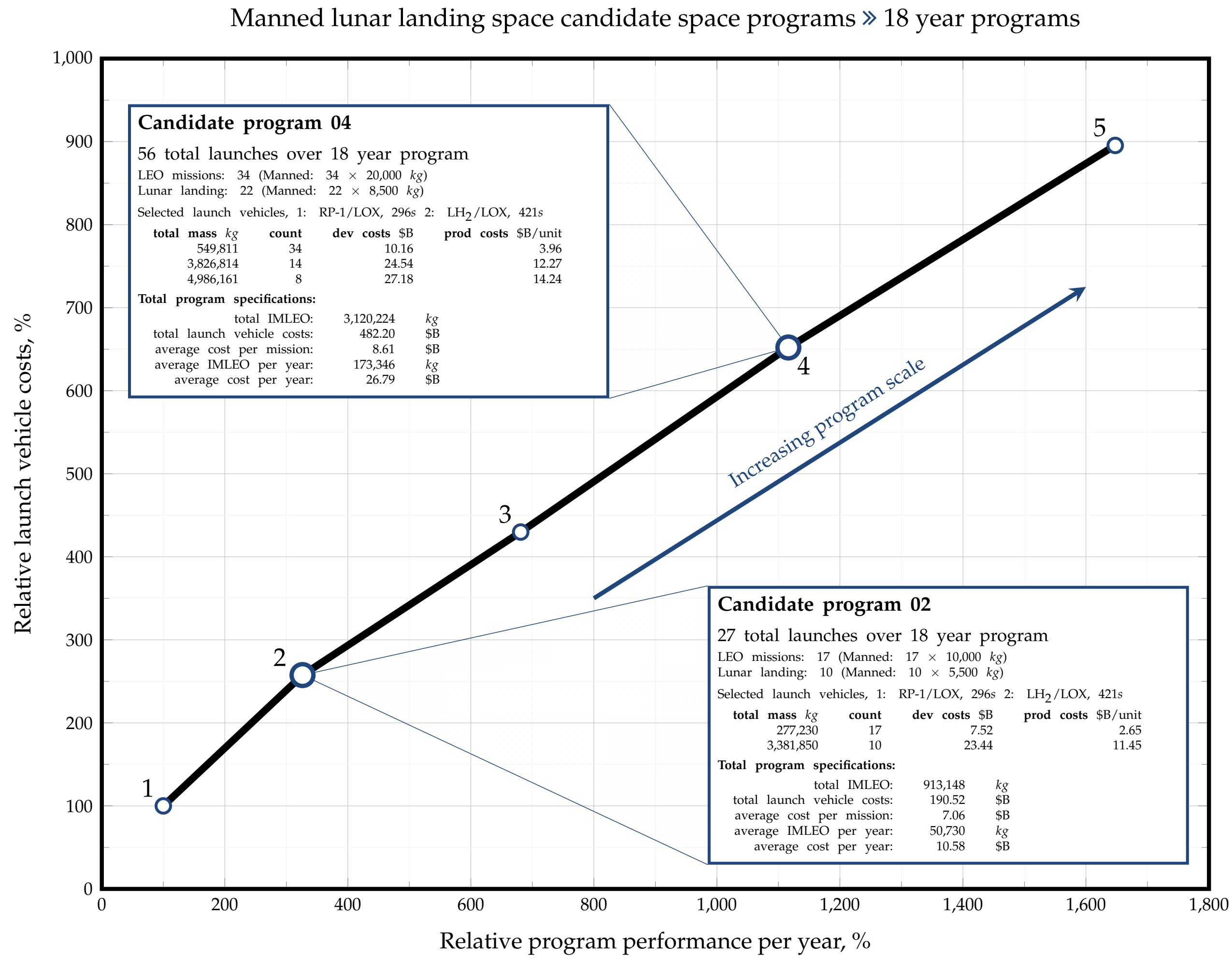

FIGURE 4.30 – The trend of manned Lunar landing candidate space programs as S_{scale} increases

from 1–5. Based on the derived objectives for this set of spacefaring goals, there are several possible, high-priority objectives available for consideration including: a space station in LEO, manned satellites in LEO, a Lunar/planetary base, circumlunar space station, and a manned Lunar landing. Since this case study is concerned with Apollo, the manned satellites and manned lunar landing are selected and candidate programs are assembled based on the recommended typical payloads and launch rates. The results are five 18–year candidate programs of increasing scale that are ready to be analyzed by the prototype *Ariadne* system. For these candidates, the previously discussed LOR mission architecture with $\chi = 0.84$ has been selected as the means of completing the manned Lunar landing and return. Also, the sized launch vehicles are all "Saturn-like" with two stages and similar propulsion characteristics.[31] Figure 4.30 contains the results of this initial analysis.

[31] There are undoubtedly many excellent opportunities for optimization here: number of stages, propellant type, number of engines per stage, *etc.* For the prototype, these are all held constant and relative costs are used for comparison.

Each data point in Figure 4.30 represents an entire program capable of completing its assigned missions, with the appropriate number of launch vehicles developed to minimize costs. The y-axis represents the relative total costs of the launch vehicles (including development and production costs), normalized to a reference program cost as shown in Equation 4.10:

$$\text{Total relative launch vehicle costs} = \frac{C_{total,candidate}}{C_{total,ref}} \times 100\,, \qquad (4.10)$$

where $C_{total,candidate}$ represents the *TransCost*–calculated total cost of developing and producing the candidate program's required launch vehicles and $C_{total,ref}$ is the calculated total cost of a selected reference program. In this example, $C_{total,ref}$ is the total launch vehicle costs of Candidate program 01, S_{scale} = 1 in Figure 4.30. The use of a relative cost comparison enables a viewer to quickly compare candidates. For example, looking at Figure 4.30 it can be seen that launch vehicle requirements of Candidate program 04, with S_{scale} = 4, would cost about 2.5 times the amount of Candidate program 02. The actual total estimated costs of candidates 02 and 04 (190 $B and 480 $B, respectively), have many assumptions from the *TransCost* implementation built in and could easily be reduced with further optimizations of the selected launch vehicle characteristics. By normalizing the costs to a designated program, programs can be directly compared with the assumption that similar cost optimizations could be made for each candidate.

The x-axis of Figure 4.30 represents the relative performance of the program. A candidate program's performance is defined as

$$P = \frac{\text{IMLEO}}{\text{program length}}\,. \qquad (4.11)$$

Like the relative costs metric, the performance of a specific candidate is best observed when normalized to a reference candidate as can be seen in Equation 4.12:

$$\text{Relative program performance per year} = \frac{P_{candidate}}{P_{ref}} \times 100\,, \qquad (4.12)$$

where $P_{candidate}$ is the performance of a given program and P_{ref} is the performance of the selected reference program. This metric is intended to be a neutral metric and can be interpreted in a couple of different ways:

» First, it provides a rough estimate to the cost of the in-space elements.[32]

» Second, it represents how active a candidate program is. Staying active is important for a commercial program, and continual activity could help a public program secure future funding.

[32] Due the limitless payload options, there is no cost estimation method implemented in *Ariadne* for in-space costs.

Mission	Count	Payload	Actual launch vehicles	
Low Earth orbit	10	9,000 *kg*	510,000 *kg*	» *Saturn I*
	5	18,600 *kg*	587,900 *kg*	» *Saturn IB*
Lunar landing	12	5,600 *kg*	2,896,200 *kg*	» *Saturn V*
» *Ariadne*–calculated launch vehicle requirements[a]				
15 × 551,605 *kg*				
12 × 3,215,242 *kg*				
» *Ariadne*–calculated program metrics for 11 year program				
Total IMLEO				1,197,700 *kg*
IMLEO per year, P_{ref}				**108,900** *kg*
Total launch vehicle costs, $C_{total,ref}$				**234.18 $B**
Launch vehicle costs per year				21.29 $B

[a] Assuming 2-stage launch vehicles:
1st stage: propellant = RP-1/LOX, I_{sp} = 296 *s*, 4 engines
2nd stage: propellant = LH_2/LOX, I_{sp} = 421 *s*, 2 engines

TABLE 4.15 – Reference Apollo data and the results of analyzing the program with the *Ariadne* prototype system.

Referring back to Figure 4.30, Candidate program 04 places over 3.6 times the amount of IMLEO of Candidate program 02. This could be considered a positive argument for only requiring 2.5 times the launch costs, but if the same organization is behind the development and production of both the launch vehicle and in-space payload, the complete costs of the program could quickly become prohibitive.

Constant spacefaring goals » sweep of scale and time

The same sweep of S_{scale} is repeated for the other possible program lengths (S_{time}). This completed solution space of converged space programs can be found in 4.31. Note that the bold line in Figure 4.31 is the same line from the previous figure, except in this figure, it has been normalized to reference Apollo data. The reference Apollo data has been summarized in Table 4.15. The estimated costs found in Table 4.15 are substantially higher than the actual totals from Project Apollo.[33] With all of the assumed launch vehicle attributes (number of stages, propellant selection, number of engines, *etc.*) and the factors omitted from *TransCost* (organization experience factor, mass production cost reduction factor, *etc.*), the resultant cost discrepancy is understandable. Normalizing to this cost will still enable meaningful comparisons between programs whose costs have been estimated under identical assumptions. In Figure 4.31, the launch vehicle costs for each program at a given scale are roughly the same with only slight variation. This makes sense, as they are each completing the same missions and placing the same total amount of payload in LEO. Assuming that a candidate program must pay its total cost in the length

[33] According to Sforza, the total launch vehicle budget for Project Apollo was approximately 53 $B [50].

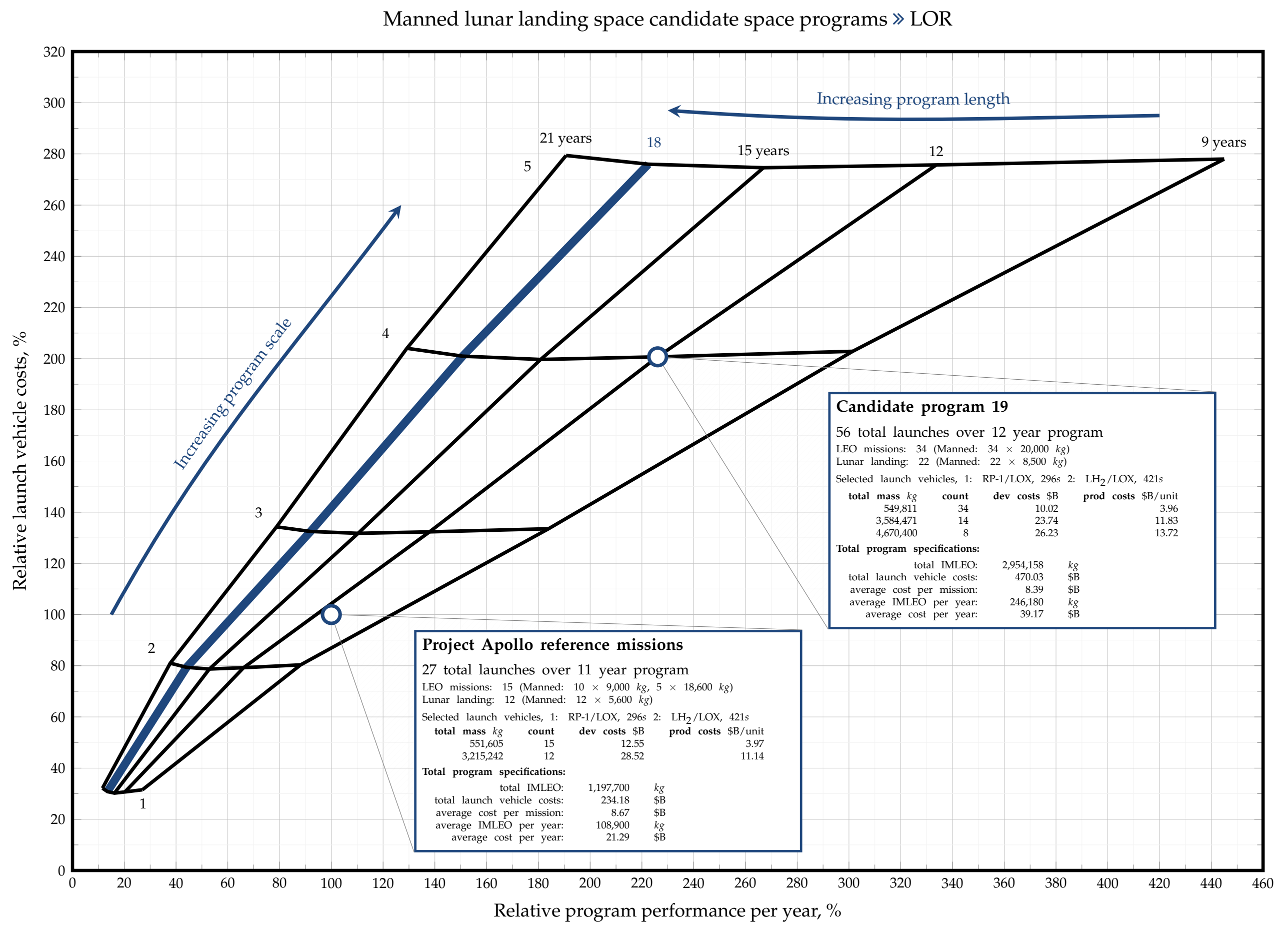

FIGURE 4.31 – The solution space of manned lunar landing programs for different strategy factors, S_{time} and S_{scale}.

of the program, the average cost per year of the quick pace programs will be much higher. The 21 year programs will benefit the most from spread out costs, though there are plenty of challenges for long term programs as well.

Comparing mission architectures » LOR vs Direct

Figure 4.32 contains the solution spaces for two similar programs with the same missions defined. The only difference is the selected lunar landing mission architecture. Figure 4.32 further captures the effects of selecting either the Direct or LOR architecture that was explored in the previous section of this case study. Note the increased payloads in LEO and launch vehicle size for the same missions. If there is nothing planned beyond these missions, this increase in relative performance is incredibly inefficient. If, however, there were follow on plans that called for the same extreme heavy-lift capability, the follow-on program might benefit from the "over-sized" launcher. After Kennedy's speech, Apollo was on a strict deadline and the LOR

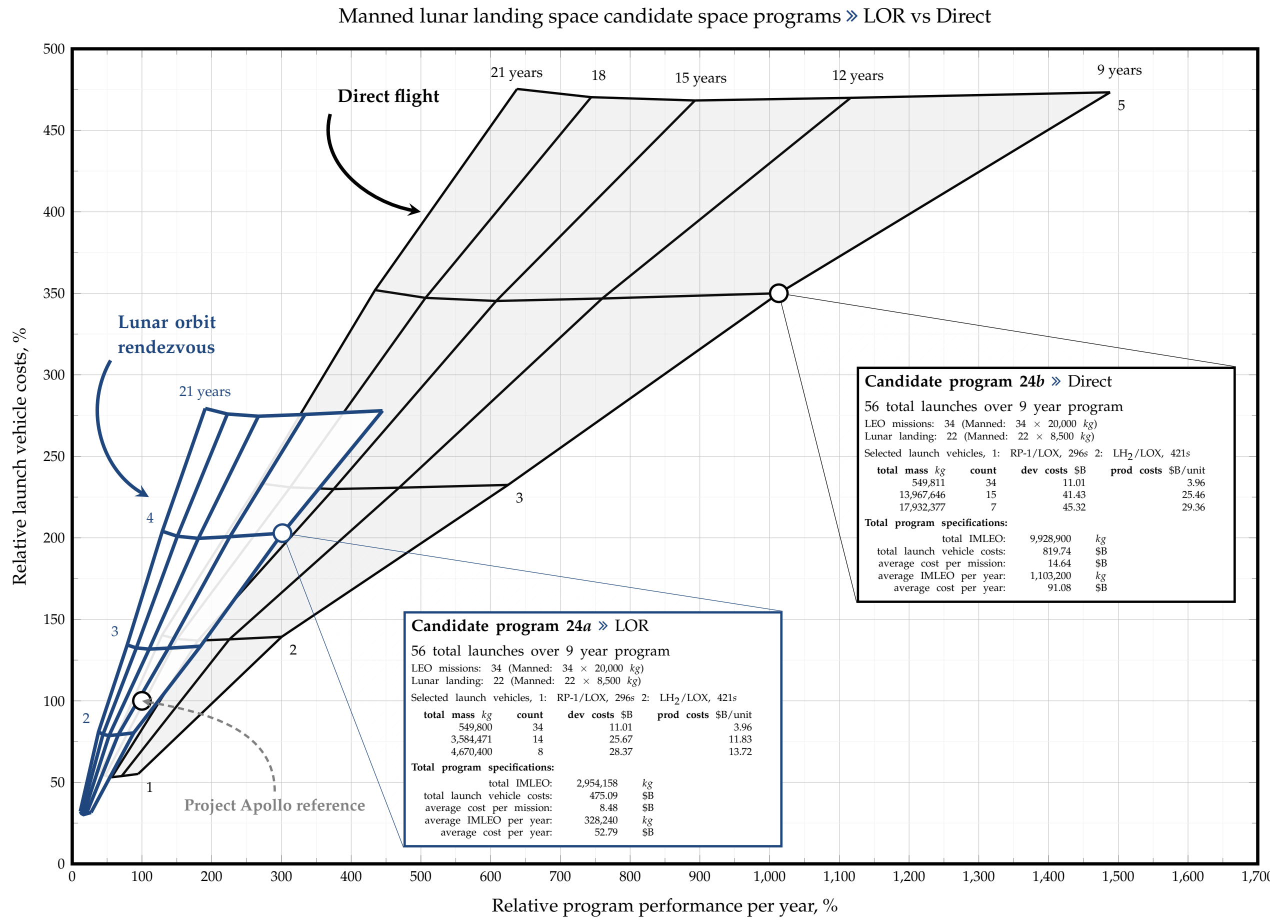

FIGURE 4.32 – A comparison of the solution space for identical manned Lunar landing program objectives. The only difference between the two sweeps is the selected mission architecture » LOR vs. Direct.

approach provided the cheaper, faster option.

Comparing program objectives » space station

If the reader will recall Figure 4.25, the early plans for the U.S. space program included both manned lunar landing and a space station in LEO. Maier says the following about the origin and eventual postponement of a space station for the U.S. space program:

> The space station program can trace its origins to the mid-1950s. By the early 1960s it was a preferred way station for traveling to and from the moon. But when, for reasons of launch vehicle size and schedule, the Apollo program chose a flight profile that bypassed any space station and elected instead a direct flight to lunar orbit, the space station concept went into limbo until the Apollo had successfully accomplished its mission. [51]

Figure 4.33 compares the solution spaces for two alternative sets of program objectives: a manned lunar landing program[34] and a manned lunar landing program with an added space station in LEO. The two highlighted programs illustrate the possibility of scaling down and

[34] This is the same LOR-based manned lunar landing program discussed in the previous two figures, Figure 4.31 and 4.32.

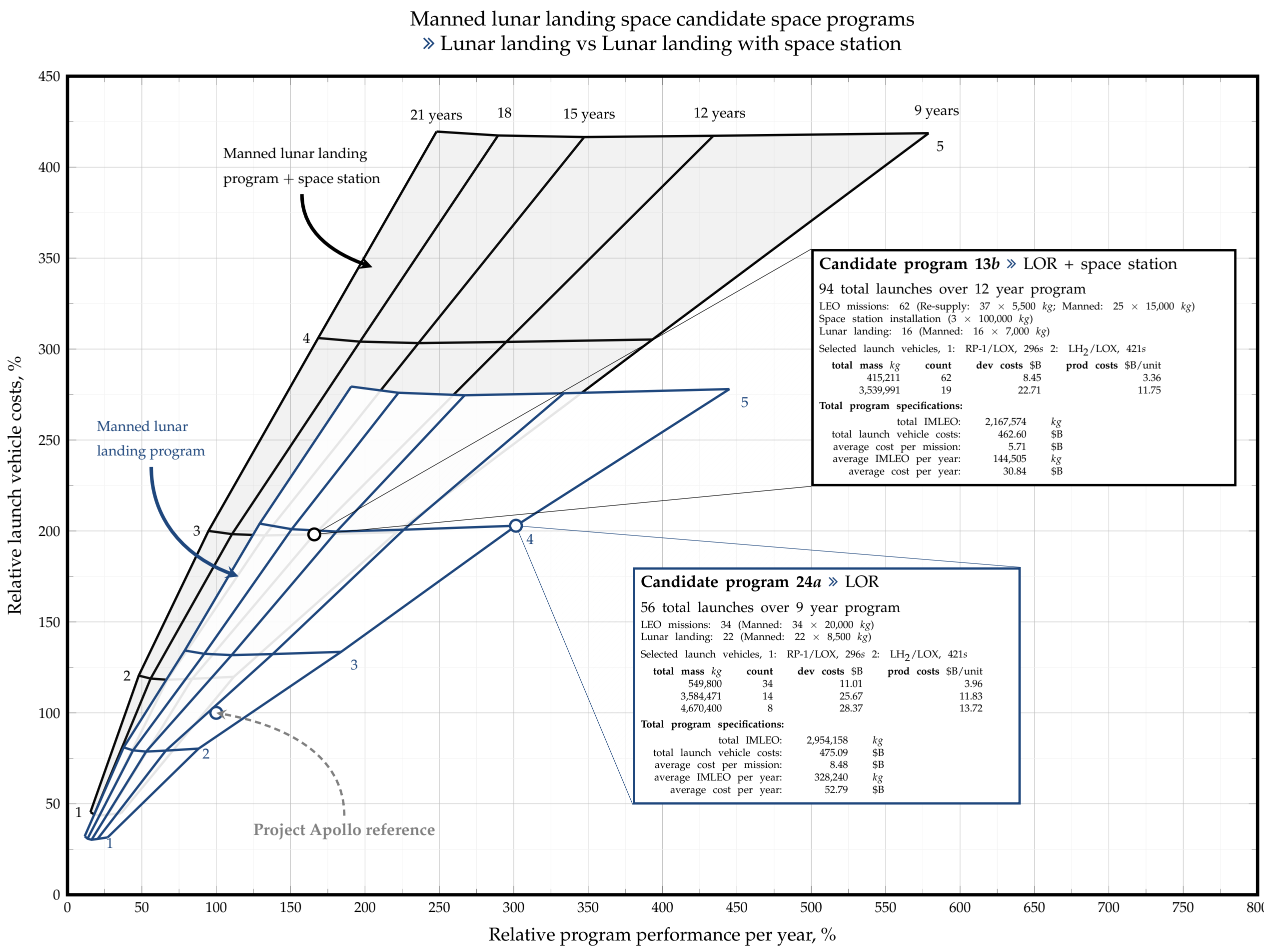

FIGURE 4.33 – Comparison of the solution space for two different sets of program objectives » 1) a manned Lunar landing, and 2) a manned Lunar landing with a LEO space station.

lengthening the program in order to accomplish both a manned lunar landing program and the installation of a substantial space station in LEO for only a 14% increase in costs.

4.4.4 *Conclusions*

The four figures above demonstrate the abilities of the *Ariadne* prototype system to quantify and visualize a wide range of candidate space programs. In this way, the case study has fulfilled the objectives laid out for it at the beginning. First, the *Ariadne* system demonstrates the ability to derive alternative sets of recommended program objectives based on the selected spacefaring goals and implementation strategies. Second, the *Ariadne* system can successfully provide the data required to visualize the solution space of candidate architectures. Finally, alternative sets of programs can be compared to inform the strategic planner on the relative merits and requirements of different program objectives and mission architectures.

The ultimate deliverable of this system is a better informed decision-maker. President Kennedy said the following about Project Apollo:

> We have given this program a high national priority, even though I realize that this is in some measure an act of faith and vision, for we do not know what benefits await us. [42]

The fully applied *Ariadne* prototype allows prospective program level decisions to be made transparent and parametrically compared against each other consistently, reducing the risks of making decisions based on 'acts of faith.' With the time and resources required by any modern effort, no organization can afford to pursue a space program for long based solely on faith of its eventual payoff.

4.5 Chapter summary

This chapter applied the *Ariadne* prototype system to various aspects of the early U.S. space program, specifically Project Apollo. Apollo provided an ideal case study due to the available validation data and numerous pivotal decisions that could be re-assessed in an effort to better inform the decision-makers.

Such a decision aid will be an invaluable in today's planning efforts, which face similar challenges to those faced by Apollo. John Aaron, the "steely-eyed missile man" in mission control during Apollo 12, said it best when he said:

> I talk to people who say, "Gosh, John, all we gotta do is think back twenty-five years ago we can go to Mars the same way." I say, "No, you can't. It was a unique set of circumstances that lined up all those dominoes."[52]

The logic behind the *Ariadne* system is adaptable and easily expandable to face today's challenges and objectives.

THE FOLLOWING RESEARCH OBJECTIVES have been successfully demonstrated via the Project Apollo case study:

» The development of a parametric planning methodology to augment top-level decisions-makers and enable informed decisions concerning the overall direction of the program;

» Evaluate the effects of a given program objective or mission architecture on the feasibility synthesized space programs.

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Chapter 5

CONCLUSIONS

The previous chapters have introduced, supported, and proven the original hypothesis of this research:

> There is a disconnect between the top-level decision-makers of a space program and the designers and specialists of individual missions, hardware, and technologies that can be parametrically remedied to better support decisions being made at all levels of a program.

The disconnect has been confirmed in Chapter 2 with a survey of several hundred previously proposed mission architectures and program plans. The survey has revealed the overall lacking of published plans, but also that even the higher quality plans were failing to convince the decision-makers of their feasibility. Later in Chapter 2, a review of available tools and methodologies for space program planning has shown that few processes have attempted to bridge the disconnect between the top-level goals and strategies with the technical details of the lower levels of the program. Existing processes primarily focus only on the technical levels (mission architectures, hardware, and technology) and if the goals and strategies are included, it is largely in a qualitative manner.

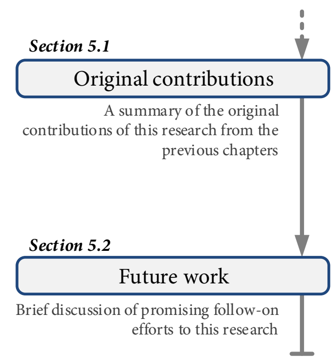

FIGURE 5.1 – Outline of Chapter 5.

An ideal solution concept, designated *Ariadne*, has been developed in Chapter 3 based on derived specifications from the previous chapter. These specifications[1] has proved too lofty for a single researcher, so a reduced list of specifications has been derived that focus on the truly original components of the solution concept.[2] The functioning algorithms of the prototype have been presented including the original development of a method to quantify and connect spacefaring goals, implementation strategies, and space program objectives.

To demonstrate the capabilities and prove the utility of the *Ariadne* prototype system, Project Apollo has been selected as a fitting case study. Apollo provides a well-documented, historical example involving critical decisions concerning the launch vehicle, mission architecture selection, primary program objectives, and program pacing and scale. With this case study the *Ariadne* prototype system has demonstrated its ability to inform decision-makers not only of a single candidate program of interest, but also of the surrounding solution space topography. This enables a decision-maker to see other opportunities and begin to quantitatively compare desirable alternatives.

[1] The complete list of ideal system specifications can be found compiled in Table 2.9.

[2] Similarly, the reduced set of objectives are presented in Table 3.1.

Two of the original research objectives, listed below, have been successfully completed with the *Ariadne* prototype and the Project Apollo case study:

» The development of a parametric planning methodology to augment top-level decisions-makers and enable informed decisions concerning the overall direction of the program;

» Evaluate the effects of a given program objective or mission architecture on the feasibility synthesized space programs.

The remainder of this chapter summarizes the original contributions made throughout this research and recommends opportunities for future efforts relating to this research.

5.1 *Summary of original contributions*

The following is a summary of the original contributions made by this research in the previous chapters:

» A thorough investigation of over 300 previously proposed mission architectures and space program plans;

» A representative review of available space program planning methodologies;

» Parametric mission architecture modeling for manned lunar landing missions;

» A parametrically connected, space program synthesis prototype, *Ariadne*. The prototype accounts for the parametric influences from spacefaring goals, implementation strategies, mission architectures, and launch vehicle development;

5.2 *Recommended future work*

The final research objective, to determine the possible impacts of forecasted technologies on the entire program as a whole to better inform research investments into enabling technologies, was deemed beyond the scope of this effort and is left as the primary recommendation for future work.

Throughout this research, additional opportunities for expanding the *Ariadne* prototype have been mentioned. The primary focus moving forward is the expansion to include new concepts for comparison, not refine the sizing estimates of the existing implementation. For example, mission architectures for Mars could be modeled with the inclusion of time in the sizing process to allow comparisons between the existing lunar architectures given in the Apollo case study. This should be done before any efforts are made to refine the sizing prediction for a specific lunar architecture. Similarly, additional launch vehicle concepts[3] can be modeled and included to quantify their possible benefits on the space program as a whole.

Beyond other straightforward expansions of the system that were discussed with the ideal solution concept,[4] two unique opportunities are proposed and discussed below.

[3] *E.g.,* single or multiple stages to orbit and including air-breathing or fully-reusable stages.

[4] See Section 3.1.2. Also, a more involved implementation of the *TransCost* process, more accurate mission modeling, *etc.*

5.2.1 Real-time dashboard user interface

The current initial feedback for the *Ariadne* prototype system is a static dashboard of information derived from the selected spacefaring goals, implementation strategies, and available space program objectives as seen previously in Table 3.7. The next generation of the *Ariadne* system could provide an interactive dashboard that contains a database of results from previous executions of the system. Many of the assumptions in the prototype would simply be adjustable inputs on the dashboard, with visual feedback to the user in real time as the up-front inputs are varied. This immediate feedback would accelerate the understanding of the user on the effects of a given goal or strategy.

The dashboard could also be custom-tailored to the desired user. The top-level decision-maker dashboard and the in-space propulsion specialist dashboard would look very different but, with the same integrated framework in the background, could both be used to enable more informed decisions.

5.2.2 Refining the connection between goals, strategies, and objectives

The identification of primary connections between the spacefaring goals, implementation strategies, and space program objectives was provided in Table 3.3. The developed method of combining the AHP weighted goals with their connections to the objectives proved its utility for the prototype system. For the next generation of the *Ariadne* system, the entire connection process could be expanded. Rather than the simple binary system currently in place, where a goal is either connected to an objective or its not, a deeper method could be applied where the degree of connection could be incorporated. Such a system could alleviate some of the effort required of any users of the *Ariadne* prototype system in distinguishing between some of the similar program objectives.

Appendix A

COMPLETE SURVEY OF MISSION ARCHITECTURES AND PROGRAM PLANS

The literature review found in Chapter 2 contains an extensive survey of previously proposed space mission architectures and program plans in an effort to familiarize the author to *what* all has been planned including the quality of the efforts and the approaches used to try and convince decision-makers. This appendix contains the full list of reviewed sources and the scores awarded and calculated for each.

TABLE A.1 –
Complete list of surveyed mission architectures and program plans.

ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
1	1953	von Braun - *The Mars Project*	3	3	0	3	0	0	**2.69**	[1]
2	1958	Ehricke - *Instrumented Comets*	1	3	1	4	1	1	**2.02**	[2]
3	1960	Gilruth - *Advanced Manned Space Vehicle Program*	3	2	0	1	1	1	**2.17**	[3]
4	1960	U.S. Army - *Lunar Soft Landing Study*	3	2	2	4	2	1	**2.80**	[4]
5	1961	Houbolt - *Lunar Orbit Rendezvous*	3	3	0	3	2	3	**2.89**	[5]
6	1961	Koelle - *Lunar and Martian Mission Requirements*	4	3	0	3	0	1	**3.20**	[6]
7	1961	U.S. Air Force - *LUNEX*	3	2	1	2	1	1	**2.37**	[7]
8	1962	NASA - *Project Apollo*	2	3	1	5	1	2	**2.72**	[8–12]
9	1962	Joy - *Future Manned Space and Lunar Base Programs*	1	1	0	1	0	0	**0.90**	[13]
10	1963	Hammock - *Mars Landing*	1	3	0	2	0	2	**1.63**	[14]
11	1964	Bono - *ICARUS*	2	1	1	2	0	1	**1.62**	[15]
12	1964	Lockheed - *Manned Interplanetary Missions*	3	4	0	3	1	2	**3.05**	[16]
13	1966	Ehricke - *Future Missions*	3	3	0	3	0	2	**2.77**	[17]
14	1966	Bellcomm - *Manned Flybys of Venus and Mars*	2	2	0	3	0	0	**1.97**	[18]
15	1966	Woodcock - *Manned Mars Excursion Vehicle*	1	2	0	4	0	1	**1.72**	[19]
16	1967	Auburn University - *Jupiter Orbiting Vehicle for Exploration*	4	4	3	4	4	2	**3.90**	[20, 21]
17	1968	NASA - *Advanced Mars Orbiter and Surveyor*	2	4	2	4	4	2	**2.92**	[22]
18	1968	Gardner - *Earth Orbital Program Strategy*	0	1	0	0	0	0	**0.24**	[23]
19	1968	Ginzburg - *Interplanetary Spaceflight Missions*	3	3	0	2	3	0	**2.63**	[24]
20	1968	Aerospace Group - *Integrated Manned Interplanetary Spacecraft*	3	4	2	5	2	4	**3.58**	[25–31]

Continued on next page

TABLE A.1 – continued from previous page

ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
21	1969	Rockwell - *Extended Lunar Orbital Rendezvous (ELOR)*	0	3	1	1	2	2	**1.08**	[32, 33]
22	1969	McDonnell Douglas - *Integral Launch and Reentry Vehicle System*	5	4	1	5	2	2	**4.42**	[34–38]
23	1969	NASA - *An Integrated Program of Space Utilization and Exploration*	3	2	0	1	1	1	**2.17**	[39, 40]
24	1969	Bellcomm - *Integrated Space Program*	3	3	0	3	2	2	**2.85**	[41]
25	1969	von Braun - *Integrated Space Program*	2	2	0	2	0	1	**1.83**	[42]
26	1969	von Braun - *Manned Mars Landing*	3	3	0	2	2	2	**2.67**	[43]
27	1970	Grenning - *Integrated Manned Space Flight Program Traffic Model*	2	2	0	1	0	2	**1.69**	[44]
28	1971	Anderson - *Evolutionary Interim Earth Orbit Program*	2	3	1	4	2	3	**2.62**	[45]
29	1971	North American Rockwell - *Lunar Base Synthesis Study*	0	5	1	4	4	3	**2.22**	[46–49]
30	1972	Martin Marietta - *Astronomy Sortie Mission*	5	3	2	5	2	3	**4.25**	[50–52]
31	1972	Demoret - *An International Space Station Program*	1	2	1	0	1	0	**1.02**	[53]
32	1974	Various - *Mars Surface Sample Return*	4	4	2	4	4	3	**3.91**	[54–59]
33	1975	Boeing - *Future Space Transportation Systems Analysis Study*	5	4	0	4	2	3	**4.26**	[60–62]
34	1976	Martin Marietta - *Titan Exploration Study*	0	3	1	4	2	2	**1.63**	[63]
35	1978	Rockwell International - *Space Industrialization Study*	1	2	1	0	2	1	**1.10**	[64]
36	1979	Boyland - *Manned Geosynchronous Mission*	2	3	1	1	0	3	**1.99**	[65]
37	1985	Howe - *Nuclear Rocket for a Manned Mars Mission*	0	3	0	0	0	2	**0.80**	[66]
38	1986	Hamaker - *Manned Mars Mission Cost Estimate*	1	1	0	0	0	0	**0.71**	[67]
39	1986	Mulqueen - *Manned Mars Mission*	0	1	0	4	0	1	**1.00**	[68]
40	1987	Nock - *Mars Transportation System*	2	2	0	2	0	1	**1.83**	[69]
41	1988	Nolan - *Manned Mars Explorer (MME)*	3	2	0	2	1	2	**2.39**	[70]
42	1988	Goldstein - *Space Transportation Architecture Study (STAS)*	3	1	0	1	0	1	**1.89**	[71]
43	1989	NASA - *1988/1989 Case Studies*	5	2	2	3	2	2	**3.61**	[72–74]
44	1989	NASA - *90 Day Study*	3	2	1	2	2	2	**2.45**	[72, 73, 75, 76]
45	1989	Rockwell International - *Integrated Space Plan*	3	2	0	1	1	1	**2.17**	[77–79]
46	1990	Griffi - *Infrastructure for Early Lunar Development*	1	0	0	1	0	1	**0.70**	[80]
47	1990	Katzberg - *Assembly vs Direct Lauch*	3	2	0	1	0	1	**2.13**	[81]
48	1990	Martin Marietta - *Manned Mars System Study (MMSS)*	3	4	1	3	3	2	**3.15**	[82, 83]
49	1990	DoD - *National Space Launch Strategy*	3	0	0	1	0	2	**1.69**	[84]
50	1990	Woodcock - *Economical Space Exploration*	1	3	0	2	0	1	**1.59**	[85]
51	1991	Zubrin - *Mars Direct*	3	3	0	2	0	2	**2.59**	[72, 86–90]
52	1991	Synthesis Group - *Space Exploration Initiative*	5	5	2	5	2	2	**4.69**	[72, 73, 91–95]
53	1991	Boeing - *Space Transfer Vehicle (STV)*	5	5	1	5	2	3	**4.70**	[96–103]

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TABLE A.1 – continued from previous page

ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
54	1991	Martin Marietta - *Space Transfer Vehicle (STV)*	3	5	2	5	2	3	**3.77**	[104–108]
55	1991	Boeing - *Space Transfer Concepts and Analysis for Exploration Missions*	3	5	1	5	3	3	**3.79**	[72, 109–115]
56	1992	Botbyl - *Systematic Transfer of Near Earth Resources (Project STONER)*	3	4	1	2	2	2	**2.93**	[116]
57	1992	Amirine - *Project Ares*	3	3	1	4	4	2	**3.14**	[117]
58	1992	NASA - *Mars Exploration Architecture*	3	3	1	3	1	2	**2.83**	[118–121]
59	1992	NASA - *Moon to Stay*	3	3	1	3	1	2	**2.83**	[118, 119]
60	1992	NASA - *Ten-year Space Launch Technology Plan*	3	1	0	1	0	1	**1.89**	[122]
61	1992	O'leary - *International Manned Missions to Mars*	2	2	0	2	0	2	**1.87**	[123]
62	1992	University of Michigan - *Project APEX*	3	4	2	4	4	2	**3.40**	[124, 125]
63	1992	MIT - *Project Columbiad*	4	4	2	4	4	2	**3.87**	[126, 127]
64	1992	NASA - *Science Emphasis for the Moon and Mars*	3	3	1	3	2	2	**2.87**	[118, 128]
65	1993	NASA - *Space Resource Utilization*	3	3	2	3	1	2	**2.85**	[118, 129]
66	1993	Rockwell International - *Advanced Transportation System Study (ATSS)*	4	5	1	2	0	2	**3.56**	[130]
67	1993	NASA - *Human Transportation System (HTS) Study*	3	3	1	2	1	3	**2.69**	[131–133]
68	1993	Various - *Mars Semi-Direct*	3	3	2	2	0	1	**2.59**	[72, 89, 134]
69	1993	Penn State - *Project Arma*	2	4	1	4	3	1	**2.82**	[135]
70	1994	Penn State - *Project Firefly*	2	4	1	4	3	1	**2.82**	[135]
71	1994	NASA - *Access to Space Study (94)*	3	3	0	0	0	1	**2.18**	[136]
72	1994	NASA - *Strategic Plan*	1	1	0	1	0	0	**0.90**	[137]
73	1995	University of Maryland - *Orion*	4	4	0	4	3	1	**3.74**	[138]
74	1995	Borowski - *Lunar/Mars Exploration*	1	5	0	3	0	1	**2.25**	[139]
75	1995	Various - *First Lunar Outpost (FLO) - 1993*	1	2	0	3	0	2	**1.58**	[73, 140]
76	1995	Koelle - *Post-Apollo Earth-Lunar Space Transportation Systems*	3	4	0	1	0	1	**2.60**	[141]
77	1996	Landis - *Footsteps to Mars*	0	1	0	1	0	1	**0.46**	[142]
78	1996	Koelle - *Lunar Base Facilities*	1	2	0	1	0	0	**1.13**	[143]
79	1996	Koelle - *Lunar Development Programs for the 21st Century*	0	2	0	0	1	2	**0.60**	[144]
80	1996	Koelle - *Lunar Laboratory*	1	3	0	0	0	1	**1.23**	[145]
81	1997	Koelle - *Lunar Settlement*	2	2	0	0	0	1	**1.47**	[146]
82	1997	NASA - *Design Reference Mission 3.0 (DRM 3.0)*	3	3	0	3	1	2	**2.81**	[72, 147–149]
83	1997	Gaubatz - *Space Transportation Infrastructure*	1	1	1	1	1	1	**1.00**	[150]
84	1997	Koelle - *Launch Vehicle Comparison for Cargo to LEO and the Moon*	3	3	0	0	0	1	**2.18**	[151]
85	1997	Koelle - *Lunar Factory*	2	2	1	0	0	0	**1.45**	[152]
86	1997	Koelle - *Integration of Moon and Mars Programs*	2	2	0	0	1	1	**1.51**	[153]
87	1998	Koelle - *Initial Lunar Base*	3	3	0	2	1	1	**2.59**	[154]
88	1998	Hawthorne - *Combination Approach*	2	1	0	0	0	0	**1.19**	[155]
89	1998	Gulkis - *Mission to the Solar System*	2	3	0	3	1	1	**2.29**	[156]
90	1998	Brothers - *Human Mission to Mars (HMM)*	2	3	1	3	3	2	**2.44**	[148, 157]

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ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
91	1998	International Academy of Astronautics - *International Exploration of Mars*	3	1	0	3	0	1	**2.25**	[158]
92	1998	Koelle - *Human Exploration of Mars*	3	3	0	0	2	1	**2.27**	[159]
93	1999	Koelle - *Return to Moon to Stay*	3	3	1	1	2	1	**2.47**	[160]
94	1999	Baumann - *Interplanetary Mass Transit System*	1	1	0	1	0	0	**0.90**	[161]
95	1999	Rokey - *CloudSat*	1	2	1	2	1	2	**1.46**	[162]
96	1999	NASA - *Design Reference Mission 4.0 (DRM 4.0)*	1	3	0	2	0	0	**1.55**	[72, 163]
97	1999	Koelle - *Reusable Heavy Lift Launch Vehicle*	3	3	0	0	0	1	**2.18**	[164]
98	1999	Koelle - *Near Term Lunar Laboratory*	1	2	0	1	0	2	**1.22**	[165]
99	1999	Koelle - *Space Solar Power Development*	2	2	0	0	0	1	**1.47**	[166]
100	1999	Koelle - *Lunar Crew Operations*	1	3	0	0	0	0	**1.19**	[167]
101	1999	The Mars Society - *Mars Society Mission (MSM)*	2	2	0	2	0	1	**1.83**	[168]
102	1999	NASA - *Mars Surveyor*	1	1	0	1	1	1	**0.98**	[169]
103	1999	Marshall - *Hyperspectral Imaging architecture*	1	3	3	3	3	3	**2.05**	[170]
104	1999	Wertz - *Large-scale Lunar Colony*	1	1	0	0	0	1	**0.76**	[171]
105	2000	Young - *Human Mars Mission*	1	0	0	3	0	0	**1.02**	[172]
106	2000	Various - *Aladdin*	0	2	0	2	0	1	**0.88**	[173, 174]
107	2000	Various - *Combo Lander*	1	2	0	2	0	2	**1.40**	[72, 73]
108	2000	NASA - *DPT Mars Long-stay Mission*	3	2	0	1	0	2	**2.17**	[175]
109	2000	NASA - *DPT Mars Short-stay Mission*	3	3	0	3	0	2	**2.77**	[176]
110	2000	Various - *Dual Lander*	1	2	0	2	0	2	**1.40**	[72, 73]
111	2000	NASA - *LUNOX*	1	2	0	2	0	2	**1.40**	[73]
112	2000	NASA - *Exploration in the Earth's Neighborhood*	1	1	0	2	0	1	**1.12**	[177]
113	2000	Golombek - *Mars Exploration*	0	1	0	0	1	0	**0.28**	[178]
114	2000	Hansen - *Mars Exploration Discovery Program*	1	1	0	1	0	1	**0.94**	[179]
115	2000	Anon - *Human Lander Return*	1	2	0	2	0	2	**1.40**	[73]
116	2000	NASA - *Interstellar Probe (ISP)*	1	3	0	2	3	2	**1.76**	[180]
117	2000	Juerwicz - *Mars Sample Return without Landing*	0	2	0	1	0	1	**0.70**	[181]
118	2000	Koelle - *Moonbase 2015*	2	2	0	0	0	1	**1.47**	[182]
119	2000	Various - *Mars Sample Return (MSR)*	3	3	2	2	2	3	**2.76**	[57, 183–193]
120	2000	Morgenthaler - *Mars Transportation Architecture*	3	3	0	1	0	2	**2.41**	[194]
121	2000	NASA - *New Millennium Program (NMP)*	0	3	0	0	0	1	**0.76**	[195]
122	2000	Paige - *Mars Exploration Strategy*	0	1	0	1	0	1	**0.46**	[196]
123	2000	NASA - *Space Elevators*	3	0	1	2	0	1	**1.85**	[197]
124	2000	Orbital Sciences Corporation - *Space Transportation Architecture*	2	2	0	1	0	0	**1.61**	[198]
125	2001	Aldrin - *Mars Cycler*	3	4	0	3	0	1	**2.96**	[199]
126	2001	Koelle - *Integrated Lunar and Mars Exploration*	3	1	1	0	1	2	**1.81**	[200]
127	2001	Georgia Tech - *Moon-based Advanced Reusable Transportation Architecture (MARTA)*	1	2	0	1	0	2	**1.22**	[201]
128	2002	Cooke - *Exploration Requirements*	3	3	0	2	0	2	**2.59**	[202]
129	2002	ESA - *European Mars Missions*	2	3	1	1	1	1	**1.95**	[203]
130	2002	Gray Research - *Integrated In-Space Transportation Plan*	3	4	0	2	0	0	**2.74**	[204]

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ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
131	2002	Koelle - *Roadmap to a Lunar Base*	3	3	0	0	0	4	**2.31**	[205]
132	2002	Bradenburg - *Mars X*	2	2	1	0	0	1	**1.49**	[206]
133	2002	NASA - *NASA Exploration Team (NEXT)*	2	2	0	3	0	2	**2.05**	[207]
134	2003	NASA - *MicroMaps*	2	3	2	4	2	1	**2.56**	[208]
135	2003	ESA - *Aurora*	0	1	0	0	0	1	**0.28**	[209, 210]
136	2003	Burleigh - *The Interplanetary Internet*	0	0	1	0	2	0	**0.11**	[211]
137	2003	Campbell - *The Impact Imperative*	0	2	2	0	2	0	**0.60**	[212]
138	2003	Cooke - *Innovations in Mission Architectures*	0	2	0	2	0	0	**0.84**	[213]
139	2003	Dostal - *Mission to Mars*	3	4	0	3	1	2	**3.05**	[214]
140	2003	Hormingo - *Moon and Mars Colonies*	1	1	0	1	2	0	**0.98**	[215]
141	2003	Kemble - *Small Satellite to Jupiter*	1	3	0	3	0	1	**1.77**	[216]
142	2003	Morgenthaler - *Integrated Architectures*	3	2	0	1	0	2	**2.17**	[217]
143	2003	NASA - *Mars Exploration Strategy*	0	1	0	0	0	1	**0.28**	[218, 219]
144	2003	von Scheele - *A Low-cost Mission to Mars*	1	1	0	1	0	0	**0.90**	[220]
145	2004	Aleksandrov - *Manned Mission on Mars*	1	2	0	1	0	0	**1.13**	[221]
146	2004	Augros - *Aurora Mars Manned Mission*	3	3	0	1	0	0	**2.32**	[222]
147	2004	Andews Space - *Concept Area 1 (CA-1)*	1	2	0	1	1	3	**1.30**	[223, 224]
148	2004	Boeing - *Concept Area 1 (CA-1)*	3	2	1	2	1	1	**2.37**	[225, 226]
149	2004	Lockheed Martin - *Concept Area 1 (CA-1)*	2	1	1	1	1	1	**1.48**	[227]
150	2004	MIT - *Concept Area 1 (CA-1)*	1	1	0	0	0	1	**0.76**	[228]
151	2004	Northrop Grumman - *Concept Area 1 (CA-1)*	3	1	0	1	1	1	**1.93**	[229]
152	2004	Orbital Sciences Corp - *Concept Area 1 (CA-1)*	3	3	1	1	1	2	**2.47**	[230, 231]
153	2004	Raytheon - *Concept Area 1 (CA-1)*	1	1	0	0	0	1	**0.76**	[232]
154	2004	SAIC - *Concept Area 1 (CA-1)*	1	1	0	0	0	2	**0.80**	[233]
155	2004	Schafer Corportation - *Concept Area 1 (CA-1)*	1	1	1	1	1	0	**0.96**	[234]
156	2004	SpaceHAB - *Concept Area 1 (CA-1)*	1	2	1	2	0	2	**1.42**	[235]
157	2004	t Space - *Concept Area 1 (CA-1)*	1	1	0	1	1	1	**0.98**	[236]
158	2004	Crocker - *Go Horizontal*	3	0	0	1	0	2	**1.69**	[237]
159	2004	NASA - *Design Reference Mission 1.0 (DRM 1.0)*	1	2	0	2	0	0	**1.31**	[72]
160	2004	Toribio - *Galileo*	0	1	1	1	1	0	**0.48**	[238]
161	2004	MacDonald - *GeoSail*	1	5	0	3	0	0	**2.21**	[239]
162	2004	MacDonald - *Geostorm*	1	5	0	3	0	0	**2.21**	[239]
163	2004	Hopkins - *The Lunar Transportation Trade Space*	2	1	0	3	0	2	**1.82**	[240]
164	2004	International Space University - *Human Missions to Europa and Titan*	1	3	0	3	0	2	**1.82**	[241]
165	2004	Leisman - *CEV Architectures*	2	3	0	2	0	1	**2.07**	[242]
166	2004	NASA - *Lunar Design Reference Mission 2 (LDRM-2)*	2	4	3	5	3	3	**3.12**	[243]
167	2004	Engle - *LunAres*	1	1	0	1	1	1	**0.98**	[244]
168	2004	Bonin - *Luna and Mars for Less*	4	4	0	4	0	1	**3.62**	[245, 246]
169	2004	MacDonald - *Polar Observer*	1	5	0	3	0	0	**2.21**	[239]
170	2004	Price - *Mini Mars Mission*	1	2	0	2	0	1	**1.36**	[247]
171	2004	Hastings - *The Space Systems, Policy, and Architecture Research Consortium (SSPARC)*	5	4	0	2	1	3	**3.86**	[248, 249]

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ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
172	2004	NASA - *The Vision for Space Exploration*	1	1	1	1	1	1	**1.00**	[250, 251]
173	2004	Woodcock - *Reusable Launch Architecture*	2	2	0	2	0	0	**1.79**	[252]
174	2005	Barker - *Lunar Exploration and Development*	0	3	1	2	1	1	**1.18**	[253]
175	2005	Bennett - *Commercially Funded Robotic Expedition*	0	1	0	0	0	0	**0.24**	[254]
176	2005	Cohen - *Model-based Trade Space Exploration*	1	1	1	0	5	5	**1.15**	[255]
177	2005	NASA - *Exploration Systems Architecture Study (ESAS)*	5	5	1	5	2	3	**4.70**	[256–261]
178	2005	Hofstetter - *Modular Building Blocks for Manned Spacecraft*	3	3	0	2	0	1	**2.55**	[262]
179	2005	Hofstetter - *Affordable Human Moon and Mars Exploration*	1	4	0	2	0	1	**1.83**	[263]
180	2005	ESA - *Human Lunar Architecture*	2	2	0	1	0	1	**1.65**	[264]
181	2005	NASA - *Strategic Roadmap*	1	3	0	1	1	1	**1.45**	[265, 266]
182	2005	NASA - *Exploration Architecture*	3	3	0	1	0	2	**2.41**	[267]
183	2005	Phipps - *Entry-level Mars Orbiter*	3	3	1	3	2	1	**2.83**	[268]
184	2005	Sanders - *ISRU Capability Roadmap*	1	3	0	2	1	2	**1.68**	[269–271]
185	2005	MIT - *Paradigm Shift in Design for NASA's SEI*	1	3	0	1	0	2	**1.45**	[272]
186	2005	Wooster - *Mars-back Approach*	2	3	0	0	0	2	**1.75**	[273]
187	2006	Del Bello - *AGILE*	0	1	1	0	1	2	**0.38**	[274]
188	2006	Yazdi - *Asteroid Sample Return (ASR)*	0	1	0	2	0	2	**0.68**	[275]
189	2006	Braun - *ARES*	3	2	1	3	3	3	**2.72**	[276]
190	2006	Cooke - *Exploration Strategy and Architecture*	0	1	0	0	1	0	**0.28**	[277]
191	2006	Dale - *Exploration Strategy and Architecture*	1	1	0	0	1	1	**0.80**	[278]
192	2006	Epps - *Design of Spacecraft for the Moon and Mars*	4	4	0	4	0	2	**3.66**	[279]
193	2006	Gily - *EXOMARS*	1	2	0	1	1	1	**1.22**	[280]
194	2006	Kosmann - *A Recommended Lunar Exploration Architecture*	2	2	1	2	0	2	**1.90**	[281]
195	2006	NASA - *Lunar Robotic Architecture Study (LRAS)*	1	2	1	2	2	1	**1.46**	[282]
196	2006	Taraba - *Project M3*	3	4	0	4	2	2	**3.27**	[283]
197	2007	Bennet - *The Skyclimber*	0	2	0	0	0	1	**0.52**	[284]
198	2007	Cooke - *Lunar Architecture*	0	3	0	1	0	2	**0.98**	[285]
199	2007	NASA - *Design Reference Mission 5.0 (DRM 5.0)*	3	3	2	3	3	3	**2.98**	[286–291]
200	2007	NASA - *Exploration Blueprint*	3	3	1	3	1	2	**2.83**	[292]
201	2007	Germain - *Utilizing Lunar Transportation Elements for Mars*	2	3	0	3	0	2	**2.29**	[293]
202	2007	Hofstetter - *The Intermediate Outpost*	1	5	0	0	0	1	**1.71**	[294]
203	2007	Hofstetter - *Lunar Outpost Strategies*	2	1	0	1	0	2	**1.45**	[295, 296]
204	2007	Landau - *Mars Habitation*	1	2	0	4	0	3	**1.80**	[297]
205	2007	Reaction Engines - *Project Troy*	2	4	0	2	0	2	**2.35**	[298]
206	2007	Wooster - *Mission Design Options for Mars*	0	0	0	5	0	1	**0.95**	[299]
207	2008	Bhasin - *Integrated Network Architecture*	1	1	1	1	3	2	**1.12**	[300]
208	2008	Czysz - *Low Earth Orbit Infrastructure*	3	3	1	1	1	1	**2.43**	[301]
209	2008	ESA - *Architecture Trade Report*	3	3	2	3	2	3	**2.94**	[302]
210	2008	ESA - *Space Exploration Architecture*	2	1	0	1	0	1	**1.41**	[303]

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ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
211	2008	ESA - *Integrated Exploration Architecture*	2	3	0	1	2	3	**2.05**	[304]
212	2008	ISECG - *Global Exploration Roadmap*	1	1	0	1	1	2	**1.02**	[261, 305–308]
213	2008	Hofstetter - *Launch and Earth Departure Architectures*	2	4	0	1	0	0	**2.09**	[309]
214	2008	Beaty - *International Mars Sample Return*	1	2	1	1	1	1	**1.24**	[310]
215	2008	MarsDrive - *MarsDrive DRM 2.5*	0	2	0	2	0	1	**0.88**	[311]
216	2008	The National Academies - *A Constrained Space Exploration Technology Program*	1	1	0	0	0	1	**0.76**	[312]
217	2008	DARPA - *Orbital Express*	0	1	0	1	0	2	**0.50**	[313]
218	2008	Smitherman - *Lunar and Mars Mission Analysis*	3	4	0	3	0	1	**2.96**	[314]
219	2008	Balint - *Venus ISRU*	0	2	0	0	0	1	**0.52**	[315]
220	2008	Wingo - *Lunar Exploration Architecture*	2	3	0	1	0	1	**1.89**	[316]
221	2008	Yazdi - *Lunar Exploration Architecture*	2	2	0	2	0	0	**1.79**	[317]
222	2009	Amade - *Mars Rapid Round Trip Design*	0	2	0	3	0	2	**1.10**	[318]
223	2009	Price - *Austere Architecture*	2	2	0	1	0	2	**1.69**	[319, 320]
224	2009	NASA - *Constellation*	2	2	0	1	0	2	**1.69**	[321, 322]
225	2009	ESA - *Reference Architecture*	1	1	0	1	1	0	**0.94**	[323]
226	2009	Heinonen - *Manned Mars Mission by 2019*	1	1	0	1	0	1	**0.94**	[324]
227	2009	Korbalev - *Russian Plans for Venus and Mars*	1	3	0	0	1	1	**1.27**	[325]
228	2009	Lafleur - *F6*	5	2	2	0	2	0	**2.98**	[326]
229	2009	Landau - *Human Mars Missions*	3	4	0	4	0	2	**3.19**	[327]
230	2009	Augustine - *Seeking a Human Spaceflight Program Worthy of a Great Nation*	3	2	0	1	0	1	**2.13**	[328, 329]
231	2009	Steinfeldt - *High Mass Mars EDL*	3	4	0	3	0	1	**2.96**	[330]
232	2010	Akin - *In-space Operations*	1	3	0	2	0	2	**1.63**	[331]
233	2010	Aliakbargolkar - *Architecting Families of Space Systems for Super Heavy Lift*	3	1	0	1	0	0	**1.85**	[332]
234	2010	Burke - *Interplanetary Mission Design Handbook*	4	0	0	4	0	0	**2.63**	[333]
235	2010	Corbet - *SICSA Mars Project*	3	1	0	1	0	2	**1.93**	[334]
236	2010	NASA - *Europa Jupiter System Mission*	2	1	0	2	2	1	**1.68**	[335]
237	2010	NASA - *Ganymede Orbiter*	2	3	0	2	2	0	**2.11**	[336]
238	2010	Guest - *Near Earth Object Exploration*	5	5	0	3	0	1	**4.15**	[337]
239	2010	Schmidt - *Human Exploration using Real-time Robotic Operations*	1	1	0	2	0	1	**1.12**	[338]
240	2010	NASA - *Human Space Exploration Framework*	2	1	1	1	1	1	**1.48**	[339]
241	2010	ISECG - *Lunar Reference Architecture*	0	1	0	0	0	1	**0.28**	[340]
242	2010	NASA - *Neptune-Triton-KBO Study*	3	3	1	3	3	3	**2.95**	[341]
243	2010	NASA - *Mars Polar Climate Concepts*	0	3	0	1	0	1	**0.94**	[342]
244	2010	Wilhite - *Near Term Space Exploration*	3	2	0	0	1	2	**2.03**	[343]
245	2010	NASA - *Encladus*	3	3	2	3	2	3	**2.94**	[344, 345]
246	2010	NASA - *Titan*	3	3	1	2	3	3	**2.77**	[346]
247	2010	Spudis - *Affordable Lunar Return*	3	2	1	1	2	1	**2.23**	[347]
248	2010	Thronson - *Human Exploration Beyond LEO*	2	3	0	3	0	2	**2.29**	[348]

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ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
249	2010	NASA - *Io*	3	3	3	2	2	3	**2.78**	[349]
250	2010	Whitmore - *Interim Access to Space*	4	2	1	4	0	3	**3.25**	[350]
251	2011	Adler - *Rapid Mission Architecture Trade Study*	2	2	1	2	1	1	**1.90**	[351]
252	2011	Braun - *Investments in our Future*	1	1	0	0	0	0	**0.71**	[352]
253	2011	AVD - *Manned GEO Servicing*	3	5	0	1	0	3	**2.92**	[92, 353]
254	2011	Salotti - *Concept 2-4-2*	2	3	0	1	0	1	**1.89**	[354]
255	2011	Cooke - *Plans for Human Exploration*	2	2	0	0	0	0	**1.43**	[355]
256	2011	Culbert - *ISECG Mission Scenarios*	0	1	0	3	0	2	**0.86**	[356]
257	2011	DARPA - *Horizontal Launch Study*	3	0	2	1	0	1	**1.70**	[357]
258	2011	NASA - *Human Space Flight Architecture Team*	2	1	0	2	0	2	**1.63**	[358]
259	2011	NASA - *NASA Strategic Plan*	1	1	1	1	1	0	**0.96**	[359]
260	2011	NASA - *Interim Report on NASA Technology Roadmaps*	1	1	0	1	0	1	**0.94**	[360]
261	2011	Olson - *Sustainable Human Exploration*	0	1	0	1	0	1	**0.46**	[361]
262	2011	Strickland - *Access to Mars*	2	2	0	3	1	1	**2.05**	[362]
263	2011	Suarez - *Integrating Spacecraft and Aircraft*	1	3	1	1	1	2	**1.52**	[363]
264	2012	Hoffman - *Capability-Driven Framework*	1	1	0	2	0	3	**1.20**	[364]
265	2012	Dejanseo - *Integrated Space Plan*	1	1	0	0	0	1	**0.76**	[365]
266	2012	Dorney - *Possible Scenarios for Manned Mars Exploration*	3	4	0	2	0	1	**2.78**	[366]
267	2012	Drake - *Alternative Strategies for Exploring Mars and the Moon*	2	5	0	5	0	1	**3.09**	[367]
268	2012	Elhman - *Mars Rover of Ancient Mars*	0	2	0	1	1	0	**0.70**	[368]
269	2012	Grover - *Red Dragon*	2	2	0	2	0	2	**1.87**	[369]
270	2012	Mackwell - *Concepts and Approaches for Mars Exploration*	1	1	0	1	1	1	**0.98**	[370]
271	2012	Benton - *Mars Exploration Vehicle*	5	3	0	3	0	3	**3.76**	[371, 372]
272	2012	Bartel - *Mars One*	2	4	0	1	0	1	**2.13**	[373, 374]
273	2012	Connolly - *Mars Rover Sample Return*	1	1	0	1	0	1	**0.94**	[57]
274	2012	McElrath - *Earth and Mars Tugs*	0	3	0	2	0	2	**1.16**	[375]
275	2012	Amah - *Mining and Acquisition of Valuable Extraterrestrial Resources for Industrial Commercialization*	2	2	0	2	0	1	**1.83**	[376]
276	2012	NASA - *Lunar Surface Exploration*	2	1	0	2	1	2	**1.68**	[377]
277	2012	NSS - *Milestones to Space Settlement*	1	1	0	1	0	0	**0.90**	[378]
278	2012	Smitherman - *Lunar Exploration Architecture*	3	3	0	3	0	3	**2.81**	[379, 380]
279	2013	Mazanek - *Asteroid Retrieval Mission*	2	3	0	0	0	2	**1.75**	[381]
280	2013	Barton - *Capability Driven Framework*	1	0	0	0	0	0	**0.48**	[382]
281	2013	ESA - *Strategic Framework*	1	1	0	0	0	0	**0.71**	[383]
282	2013	Tito - *Inspiration Mars*	2	2	0	5	0	1	**2.37**	[384]
283	2013	Jain - *Phobos-Deimos Mission Architecture*	3	1	0	1	0	1	**1.89**	[385]
284	2013	Jain - *Space Architecture*	3	3	0	3	2	2	**2.85**	[386]
285	2013	French - *Private Sector Lunar Return*	3	1	1	1	1	3	**2.03**	[387]
286	2013	Raferty - *Affordable Mission to Mars*	3	1	0	3	0	0	**2.21**	[388]
287	2013	Rarick - *ISS Operations*	1	2	2	1	2	2	**1.34**	[389]

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TABLE A.1 – continued from previous page

ID	Date	Architecture / Plan	Launch	Space	Ground	Orbit	Comm.	Op.	Score	Source(s)
288	2013	NASA - *SCaN*	3	4	2	4	2	0	**3.23**	[390]
289	2013	Trost - *Modular Space Architecture*	3	3	0	3	1	2	**2.81**	[391]
290	2014	Boeing - *Mission to Mars in Six (not so easy)Pieces*	2	1	0	1	0	1	**1.41**	[392]
291	2014	NASA - *Evolvable Mars Campaign*	3	3	0	2	0	2	**2.59**	[393–400]
292	2014	Zacny - *Mars 2020*	0	2	0	0	0	0	**0.48**	[401]
293	2014	NASA - *Human Path to Mars*	1	1	0	1	0	1	**0.94**	[402]
294	2014	Watson - *Launch Vehicle Control Center Architectures*	1	0	1	1	1	1	**0.76**	[403, 404]
295	2014	Wilhite - *Plan B*	3	3	0	1	0	2	**2.41**	[405]
296	2015	ESA - *Exploring LEO Together*	1	1	1	1	1	1	**1.00**	[406]
297	2015	The Planetary Society - *Humans Orbiting Mars*	1	2	0	1	0	2	**1.22**	[407]
298	2015	NASA - *Journey to Mars*	3	1	0	1	2	1	**1.97**	[408]
299	2015	NASA - *Technology Roadmaps*	3	3	3	1	3	1	**2.56**	[409]
300	2015	NASA - *Humans to Mars Orbit*	1	2	0	1	0	2	**1.22**	[410]
301	2015	Purdue University - *Project Aldrin-Purdue*	2	4	4	4	4	2	**2.97**	[411]
302	2015	Moonspike - *Project Moonspike*	2	2	1	4	2	3	**2.38**	[412]

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Appendix B

PROCESS LIBRARY

B.1 Introduction

This appendix serves as the final deliverable of an effort to create a database of space program planning processes. Space programs are exposed to a very unique set of problems that require the planner to operate in a nearly-overwhelming scope, while not forgetting some of the details that can ruin a program down the line.

B.2 Nassi-Shneiderman charts

A common way of depicting the internal logic for each reviewed process is required. The author has decided to use a method created by Nassi and Shneiderman, and applied in a previous lab members' own process library for aircraft design [1–3].

Nassi-Shneiderman diagrams, also referred to as N-S diagrams or structograms, provide a consistent means of looking at the logic within a given process. As can be seen in Figure B.1, N-S diagrams provide a means of depicting many of the common concepts traditionally depicted with flow charts.

In the next chapter, a N-S diagram has been included with many of the key processes, allowing a clear representation of the critical methods and approaches. This allows a common comparison between processes.

B.3 Process library

The following processes have been selected as those representative of the concepts and trends uncovered in this process survey. Also included are those with milestone contributions.

A particular focus was placed on understanding those processes that have attempted to include the higher levels of the hierarchy, an area that is particularly neglected as can be seen in Figure 2.20.

Each process page includes: the year published, a figure illustrating the tiers of the exploration hierarchy addressed by the process, the strengths and weaknesses of the process, and finally a N-S structogram depiction of the process.

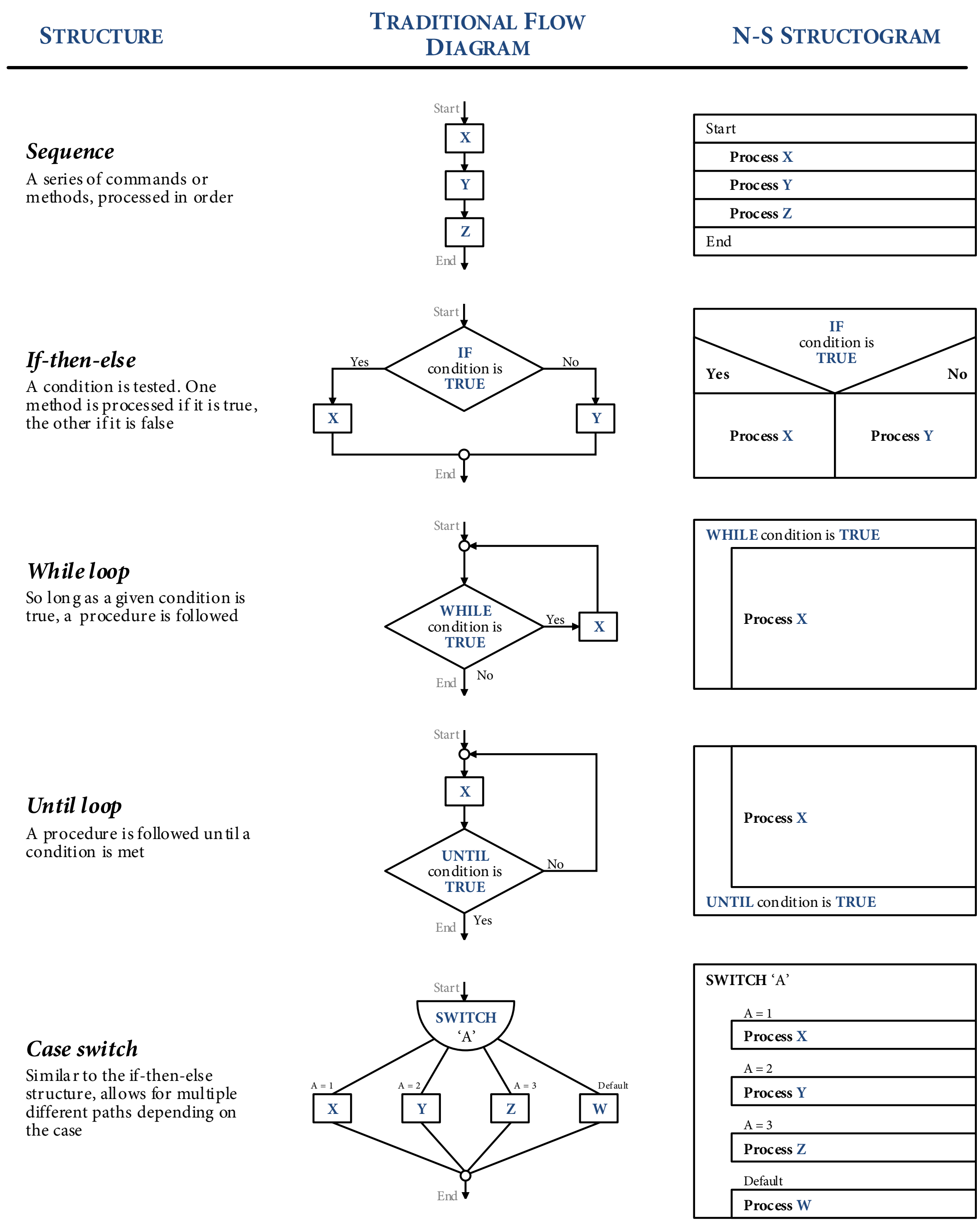
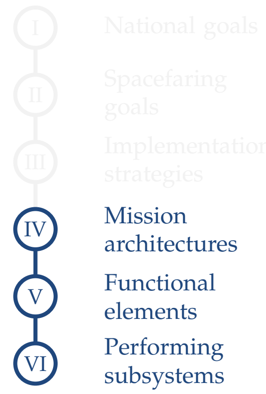


FIGURE B.1 – Traditional flow diagram concepts, their purpose, and their analogous N-S structogram representation.

B.3.1 *The Mars Project*



Published date: 1953
Type: Book

> "My basic objective during the preparation of *The Mars Project* had been to demonstrate that on the basis of the technologies and the know-how then available (in 1948), the launching of a large expedition to Mars was a definite technical feasibility." (von Braun, preface to the 1962 edition of *The Mars Project*, [4])

Author/organization(s):
W. von Braun

Process description:
In an effort to promote the feasibility of human exploration, von Braun presents his plan for a manned trip to Mars and his defense of why it is possible. His plan calls for a flotilla of ships making the transit with over 70 crew on board. This process leans more towards a point design architecture but with the methods to prove the portions that seemed infeasible at the time, namely the launcher. von Braun's plan called for a 3 stage reusable launcher, revealing his keen intuition on the need for reusability that has dominated many of the recent launcher efforts.

Strengths:
von Braun clearly highlights the primary variables in his efforts to prove the feasibility of his concept.

Weaknesses:
The Mars Project process is very linear and meant to answer the only question von Braun was trying to prove in this work. It does not attempt to visualize the total design space, or even trade the gross payload options. It does not lend itself to application elsewhere as a whole, though several portions of the process may be applied as individual methods to solve other problems.

Status:
The Mars Project has been re-published multiple times, including under the title *Project Mars: a Technical Tale* [5] which also includes a science fiction story along with it, allegedly written by von Braun (though this is apparently up for debate).

Reference: See [4, 5]

The Mars project process

Assume departure orbit: T, altitude
Determine terminal velocity of 3rd stage: ΔV
Launcher performance analysis: MR, ΔV
Analyze return of 3rd stage – trajectory and aeroheating analysis
Launcher propulsion sizing – nozzle exit areas and stage geometry
Launcher weight estimation
Determine velocity changes required for trans-Mars injection: ΔV
Calculate mass ratios of transfer spacecraft
Determine favorable transfer windows
Estimate payload weights: $m_{payload}$
Estimate size of entry vehicles

Figure B.2 – The process is very linear, with many assumptions along the way. It does not include iteration but does increase the fidelity in areas where it was needed to prove feasibility at the time (propulsion).

B.3.2 *Mission Velocity Requirements and Vehicle Characteristics*

I National goals
II Spacefaring goals
III Implementation strategies
IV Mission architectures
V Functional elements
VI Performing subsystems

PUBLISHED DATE: 1961
TYPE: Textbook

> "...the preliminary determination of velocity requirements for individual missions and the characteristic data of the vehicle which is required to carry out the mission of interest" (p. 25–14 [6])

AUTHOR/ORGANIZATION(S):
H. H. Koelle and H. F. Thomae

PROCESS DESCRIPTION:
"The stage specific impulses and the velocity requirement result in an over-all effective mass ratio. Propellant fraction and payload ratio result in an average structural factor. This and the total mass ratio result in the optimum number of stages required, and give a first estimate of the growth factor (take-off weight/payload weight). Combining this with the take-off weight, obtained by dividing the take-off thrust by the take-off acceleration, fives a first estimate of the total weight-carrying capability (weight of instrumentation, guidance and control components, payload containers, and net payload), defined as the dry gross payload.

A preliminary optimization of the propellant loadings of the stages follows, which in turn allows more detailed weight estimates of the subsystems and components. Adding all these weights results in a preliminary vehicle weight breakdown, which then is used for performance check calculations. (p. 25–14)"

MISSION VELOCITY REQUIREMENTS AND VEHICLE CHARACTERISTICS

Calculate energy required for mission: ΔV
Assume specific impulses of stages: I_{sp}
Calculate overall mass ratio: MR
Assume total propellant fraction and payload ratio: $\xi_{p,T}$, λ_{gd}
Calculate structure ratio: ε_n
Calculate optimum number of stages and first estimate of the growth factor: n_{opt}, N_{gd}
Assume launch performance – take-off thrust and acceleration: F_0, a_0
Calculate take-off weight
Stage-mass ratio optimization and weight distribution
Performance check – calculate the cutoff velocity and altitude of each stage: v, y

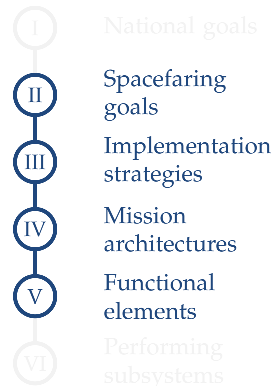
FIGURE B.3 – Koelle and Thomae present a very straightforward approach with many rules-of-thumb and appropriate assumptions given.

STRENGTHS:
Clear identification of the primary drivers in both energy requirements and the launch vehicle's properties. Provides a flow diagram that also demonstrates where the process fits into the larger design process.

WEAKNESSES:
The methods and variables involved use assumed values for other variables and any figures and equations are then based on this assumption. Slightly limits the applicability if the user needed to trade other variables.

STATUS:
The process is still applicable, though no direct follow-up efforts have been observed.

REFERENCE: See [6]



B.3.3 A Comprehensive Analytical Basis for Long-Range Planning Decisions in Future Manned Space and Lunar-Base Programs

PUBLISHED DATE: 1962
TYPE: Journal article

> "This paper is concerned with the development of a formalized systems approach to decision making for second-generation, post Apollo manned space programs." (p. 1 [7])

AUTHOR/ORGANIZATION(S):
D. P. Joy and F. D. Schnebly
Lockheed Missiles & Space Company

PROCESS DESCRIPTION:
The authors understood the importance of making the best decisions possible for the future of the space program. They attempted to quantify and compare assumed candidate programs based on three metrics: (1) performance, (2) schedule, and (3) success probability.

STRENGTHS:
The process correctly targets the full program scope and provides a means of comparing missions and projects in the context of the overall program.

WEAKNESSES:
The authors provide fixed, and at times very specific, mission, hardware and technology assumptions. While this enables their top-level decision-aid process, it does limit its applicability.

STATUS:
The theoretical process is applied to a single case study in the report. No sign of its application elsewhere has been found.

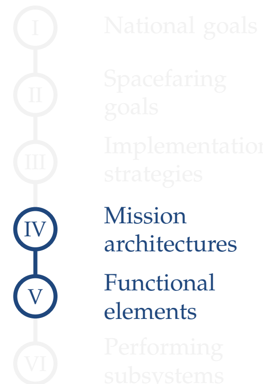
REFERENCE: See [7]

ANALYTICAL BASIS FOR LONG-RANGE PLANNING DECISIONS

- **Define mission concepts** – maneuvers and hardware required
- **Combine mission concepts into candidate programs**
- For each candidate program
  - For each mission
    - **Analyze performance**
    - **Analyze schedule**
    - **Analyze success probability**
  - **Estimate total program cost**
- **Compare program alternatives**

FIGURE B.4 – A structogram representation of Joy and Schnebly's planning process.

B.3.4 *Mass Ratio Design Process*



PUBLISHED DATE: 1964
TYPE: Textbook

> "The problem facing the designer is to balance the energy requirements of the mission with the total weight of the spacecraft and launch vehicle system within the operational limits or constraints which are placed on him. This balance is most effectively demonstrated by a consideration of the spacecraft mass ratio, that is, the ratio of the total weight to the weight in the unfueld or empty condition. This ratio, in effect, yields the total fuel percentage. The mass ratio required to perform a given mission is determined uniquely by the mission energy requirements expressed in the form of a characteristic velocity budget, the specific impulse of the propulsion system, and the number and size of the propulsion stages." (Lunar Missions and Exploration, p. 24 [8])

AUTHOR/ORGANIZATION(S):
W. R. Laidlaw

PROCESS DESCRIPTION:
The basic concept that seems to find itself as a component in many of the other processes reviewed here. The process revolves around the Tsiolkovsky rocket equation [SOURCE] which includes many of the primary variables needed for preliminary planning efforts.

STRENGTHS:
The process is based on first principles and can applied in many different ways. It is a balancing act, allowing the user to hold certain variables steady while trading the others. This, while not mentioned by Laidlaw, leads to the visualization of the solution space.

WEAKNESSES:
The process is logically sound but, as presented, does not formalize the best way of connecting to other planning areas. It's generic nature can be seen as both strength and a weakness.

STATUS:
The process can be found applied in many of the other processes in this database. It continues to serve its role in the early planning process.

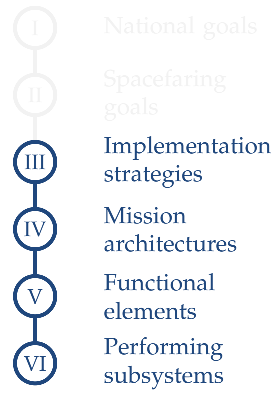
REFERENCE: See [9]

MASS RATIO DESIGN PROCESS

Calculate mission energy requirement: ΔV
Define stage propulsion performance: I_{sp}
Performance analysis – calculate mass ratio required: MR_{req}
Define payload weight: m_{pl}
Increase gross weight: m_0
Calculate propellant weight and structure weight: m_p, m_s
Calculate actual mass ratio: MR_{actual}
Until: $MR_{req} = MR_{actual}$

FIGURE B.5 – The process describes itself as a balancing act between the energy required, propulsion performance and the masses of the payload, propellant and structures.
Many combinations of processes can be depicted with certain parameters held constant while others are traded. This figure illustrates one such process.

B.3.5 Space Technology Analysis and Mission Planning



Published date: 1964
Type: Technical report

> "The original intent of the program that resulted in the analytical Model was to simulate the operation of those transportation systems which could possibly play a role in future interplanetary travel. Estimates of the cost, schedule, feasibility, manpower requirements, size of the vehicle, number of vehicles, and subsystem requirements were to be calculated." (p. I-1 [10])

Author/organization(s):
G. W. Morgenthaler
R. Novosad and M. Capehart
Aerospace Division of the Martin Marietta Corporation

Process description:
The process, developed at Martin Marietta and seemingly championed by Morgenthaler, was developed as part of a NASA contract. NASA was interested in finding a process to quantify their decision making for an exploration program.

Strengths:
Can compare missions and vehicles within the context of an entire program to determine their value.

Weaknesses:
The entire program, desired missions, objectives and schedules are an input into the process. There does not seem to be any methodology in place for the guidance of created this initial input. Therefore, any impacts from the goals and strategies of the program are purely dependent on the initial planner.

Status:
Submitted to NASA in 1965, the developed process was then used and discussed in a journal article in 1966. Any future development beyond this point has not been found.

Reference: See [10–18]

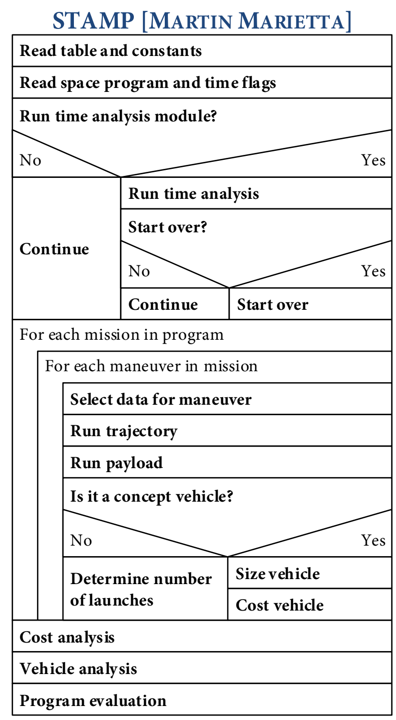


FIGURE B.6 – The STAMP (Martin Marietta) process.

B.3.6 *Space Technology Analysis and Mission Planning*

Published date: 1964
Type: Technical report

> "The development of a Basic Planetary Transportation System Model (BPTSM) has the purpose of assisting NASA in the formulation and evaluation of plans for the manned exploration of space. These plans require long lead times and involve a variety of complex projects and programs in supporting roles, such as the development of appropriate Earth launch vehicles, adequate propulsion systems, unmanned deep space probe programs, planetary landing and launch vehicles, orbital laboratories and others." (p. 613 [19])

Author/organization(s):
K. A. Ehricke
General Dynamics

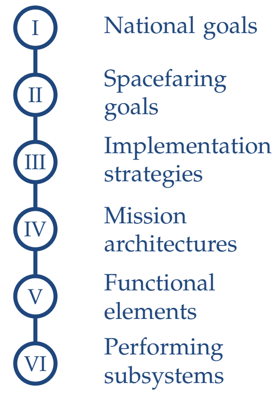
Process description:
Developed under the same NASA contract as the Martin Marietta STAMP model, this process had a similar purpose, aiding exploration program decisions, and was championed by Ehricke.

Strengths:
The most extensive process uncovered thus far. The process concerns itself with all but the highest tier level, which is per definition outside its scope.

Weaknesses:
Provides a methodology for tiers 2 and 3, but does not connect them directly with the lower three tiers.

Status:
As with the Martin STAMP model, General Dynamics's attempt appears in a journal not long after, but no sign of its development is seen after that.

Reference: See [19–23]



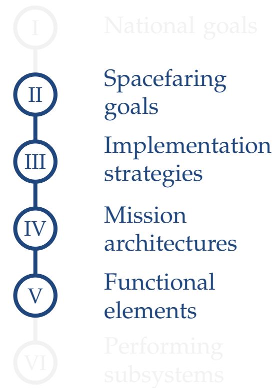
STAMP [General Dynamics]

- **Define general program objectives**
- **Determine objective weights** – process uses a survey of company employees
- **Calculate *National Space Program Value***
- **Formulate *General Space Programs***
- For each *General Space Program*
  - For each program objective
    - **Determine utility factor** – the program's effectiveness in accomplishing the objective
  - **Determine *Space Program Quality*:** cost-effectiveness, operational effectiveness, ability, and growth
  - **Calculate utility of *General Space Program***
- **Compare *General Space Programs*** – compare against the *National Space Program Value*, as well

- **Define the operational achievements and technology milestones**
- **Define mission objectives**
- For each mission
  - **Calculate performance requirement:** ΔV
  - **Payload analysis**
  - **Propulsion analysis**
  - **Vehicle analysis**
  - **Mission performance analysis and weight calculation**
  - **Vehicle-mission integration**
- **Synthesize projects and sub-programs**
- **Assemble operations program**
- **Assemble development program**
- **Synthesize *National Space Program***

Figure B.7 – This process was a large, multi-year, company-led effort. Various drafts, changes, and future plans that were never realized are evident throughout.

B.3.7 *Program Analysis and Evaluation Process*



Published date: 1965
Type: Technical report

> "The "Program Analysis and Evaluation Procedure" is the formalization of a methodology to serve as a management tool for program integration. It is a device through which alternative space program plans, while observing the constraints selected by the manager or analyst, can be simulated, evaluated, and analyzed in an integrated fashion. The procedure will permit the study of a great number of alternative courses of action within the basic structure of a national space program, particularly the effectiveness of individual launch vehicles or spacecraft, as well as the relative worth of adding, changing, or deleting an individual space project within any particular space program formulation." (p. 1 [24])

Author/organization(s):
H. H. Koelle and R. G. Voss
Future Projects Office, Marshall Space Flight Center

Process description:
Koelle and Voss developed a process to compare the worth and effectiveness of alternate program options.

Strengths:
The Program Analysis and Evaluation Process addresses the correct scope and attempts to do something that is rarely done: quatify the worth of a program. This difficult task is approached correctly and should be very useful moving forward.

Weaknesses:
As an input, similar to others, the user must provide the vehicles, payloads and schedules. This relies solely on the experience of the planner.

Status:
The process has not been seen used outside of this technical document.

Reference: See [24]

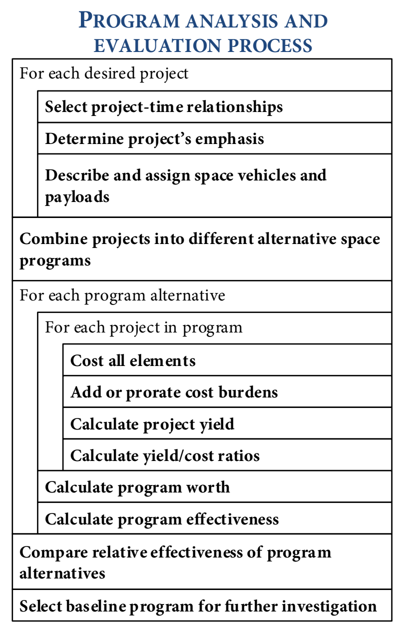
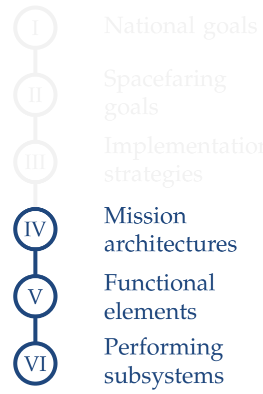


FIGURE B.8 – The program analysis and evaluation process.

B.3.8 The Space Planners Guide



PUBLISHED DATE: 1965
TYPE: Handbook

> "The fundamental aim of this Guide is to provide a rapid means of both generating and evaluating space mission concepts. Fulfillment of this objective should eliminate the need for many funded concept studies. Although the Guide may assist in the definition of detail design studies, it cannot substitute for them." (Space Planners Guide, p. I-1 [25])

AUTHOR/ORGANIZATION(S):
E. D. Harney
The United States Air Force

PROCESS DESCRIPTION:
The Space Planners Guide employs a handbook methodology to empirically size a vehicle, or even entire mission concepts, by iteratively stepping through a series of quality nomographs. The objective of this tool has been to obtain first order estimates to many of the system's elements' size, weight, range, power required, etc.

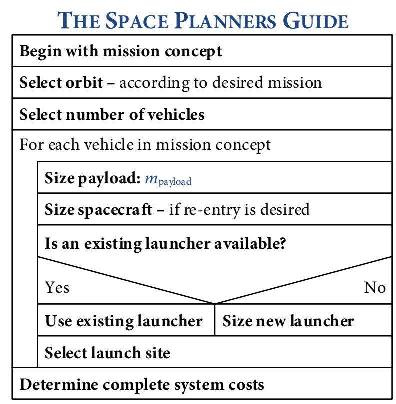


FIGURE B.9 – The Space Planners Guide process.

STRENGTHS:
Provides nomograms for all of its methods, allowing anyone with a straight edge to use them to obtain meaningful results. The Guide is properly focused on providing most of the solution quickly, allowing the rapid screening of different options before moving on to more detailed studies.

WEAKNESSES:
The Guide does not provide the methods behind the nomographs or even detail all of the assumptions that went in to creating them. This limits the ability to adapt the methods of the process to anything beyond that for which they were originally designed.

STATUS:
Has not been maintained beyond its original printing. Some of the nomographs were digitized and incorporated into Acquisition Deployment and Maneuvering (ADAM) the Space Game in 1987 by J. E. Heier of Air University at Maxwell AFB [26].

REFERENCE: See [25, 27]

B.3.9 *A Space Mission Success Evaluation Model*

Published date: 1966
Type: Technical report

> "The approach presented shows a technique for determining a quantitative measure of success, a procedure for evaluating this measure both a priori and posteriori, a systematic technique for collecting and displaying the necessary input information, and a method for determining the optimal allocation of resources." (p. 1 [28])

Author/organization(s):
R. G. Chamberlain
Jet Propulsion Laboratory

Process description:
Chamberlain presents a process to analyze the probability of a given mission's success and the effect that success might have on future missions.

Strengths:
By numerically determining the value of a success, it is possible to evaluate how the success will affect future missions in the pursuit of the program's objectives. An experienced planner could then adjust his objectives for the most beneficial program.

Weaknesses:
The connection between tiers 3 and 4 is entirely dependent on the experience of the planner. If he does not take the selection of objectives into account, then this process is focused solely on a mission, without even diving into the specific hardware.

Status:
The process does not appear anywhere else, though Chamberlain's research and efforts can be seen throughout this survey.

Reference: See [28]

B.3.10 *A Methodology to Compare Policies for Exploring the Solar System*

PUBLISHED DATE: 1970
TYPE: Journal article

> "This paper describes procedures to process part of the information affecting decisions about unmanned exploration of the solar system." (p. 593 [29])

AUTHOR/ORGANIZATION(S):
R. G. Chamberlain and L. Kingsland, Jr
Jet Propulsion Laboratory

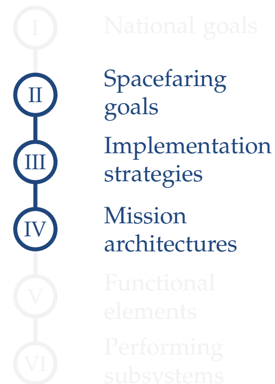
PROCESS DESCRIPTION:
The process is the first found that attempts to model the effects that changes in goals and strategies can have on the overall program.

STRENGTHS:
Finally quantifies the connection between tiers 2 and 3 with 4. Makes use of Monte Carlo simulation to account for success probabilities throughout the various missions.

WEAKNESSES:
The process is focused on unmanned exploration, so many of the objectives that are defined can only apply to this topic. Lacks and of the deeper details about actual hardware or new technologies and the effects they can have.

STATUS:
No signs of development beyond this report.

REFERENCE: See [29]



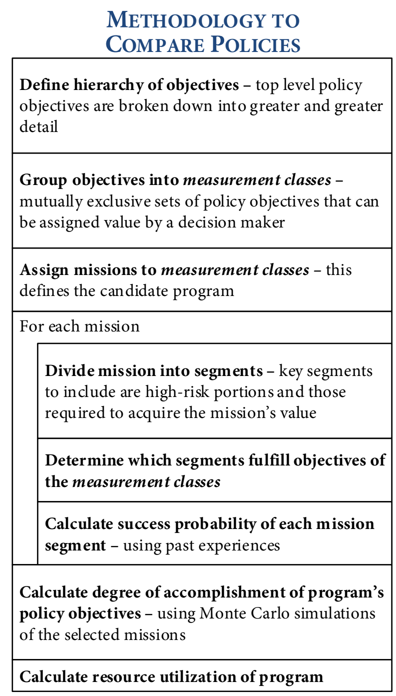


FIGURE B.10 – Chamberlain's policy comparison process.

B.3.11 STARS: The Space Transportation Architecture Risk System

I National goals
II Spacefaring goals
III Implementation strategies
IV Mission architectures
V Functional elements
VI Performing subsystems

Published date: 1997
Type: Technical report

> "Because of the need to perform comparisons between transportation systems that are likely to have significantly different levels of risk, both because of differing degrees of freedom in achieving desired performance levels and their different states of development and utilization, an approach has been developed for performing early comparisons of transportation architectures explicitly taking into account quantitative measures of uncertainty and resulting risk. The approach considers the uncertainty associated with the achievement of technology goals, the effect that the achieved level of technology will have on transportation system performance and the relationship between transportation system performance/capability and the ability to accommodate variations in payload mass." (p. ii [30])

Author/organization(s):
J. S. Greenberg
Princeton Synergetics, Inc.

Process description:
The STARS process uses a Monte Carlo simulation approach to quantify the risk for given technology, hardware and missions.

Strengths:
The Monte Carlo methods are very appealing for determining the overall risks of such a complex system such as space program.

Weaknesses:
The process does not look at the overarching scope. It would be very valuable to see how sensitive a program is to failed missions.

Status:
There is no evidence of further development of the STARS process.

Reference: See [30]

B.3.12 Bringing Policy into Space Systems Conceptual Design: Qualitative and Quantitative Methods

I National goals
II Spacefaring goals
III Implementation strategies
IV Mission architectures
V Functional elements
VI Performing subsystems

PUBLISHED DATE: 2002
TYPE: Thesis

> "The goal of this thesis research is to enable the creation of policy robust system architectures and designs through making policy an active consideration in the engineering systems architecting and design process. Qualitative and quantitative analysis methods are brought to bear on the problem using space systems as the application domain, and a process is set down through which policy can become an active consideration instead of a static constraint." (p. 3 [31])

AUTHOR/ORGANIZATION(S):
A. L. Weigel

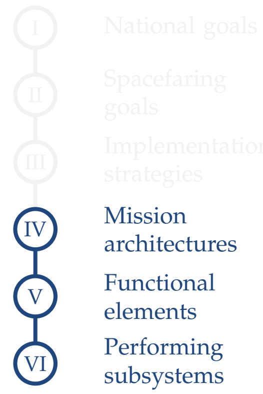
PROCESS DESCRIPTION:
Weigel's process highlights the difficulty in quantifying the effects of goals and strategies on a design. She provides qualitative approaches, semi-quantitative, and quantitative approaches.

STRENGTHS:
The qualitative methods are useful for sanity checks and the quantitative methods are supported by many years of decision analysis application.

WEAKNESSES:
The process, like many others that investigate the connection between tiers 2, 3, and 4, stops short of and sizing methods. The process works more like a logistics approach in this sense.

STATUS:
No further developments have been observed.

REFERENCE: See [31]



B.3.13 AVD^{sizing}

PUBLISHED DATE: 2010
TYPE: Technical report

> "AVD^{sizing} is a constant mission sizing process capable of first-order solution space screening of a wide variety of conventional and unconventional vehicle configurations." (p. 17 [32])

AUTHOR/ORGANIZATION(S):
B. Chudoba, et al.
The Aerospace Vehicle Design Laboratory

PROCESS DESCRIPTION:
The AVD^{sizing} process is largely based on the *Hypersonic Convergence* process developed by the McDonnell Aircraft Company in the 1970s. It has been adapted in such a way that has enabled its application to other types of hardware: in-space elements, re-entry capsules, and even launch systems.

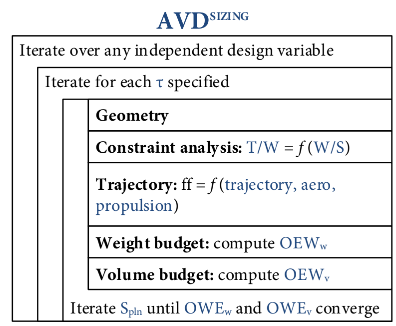


FIGURE B.11 – The AVD^{sizing} design process

STRENGTHS:
The process emphasizes *convergence* of each vehicle that is restricted by various constraints. It can deliver a solution space of appropriately converged vehicles, allowing the decision-maker to see where a given design resides in relation to other design options.

WEAKNESSES:
AVD^{sizing} operates with a constant defined mission and any large alterations to its mission require a lot of effort to re-apply the process. This limits its connections to any of the higher scopes.

STATUS:
AVD^{sizing} is still under-development and its currently in the middle of a extensive overhaul that will expand its capabilities well beyond its current state [33–35].

REFERENCE: See [3, 32, 36]

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Appendix C

ADDITIONAL VALIDATION OF THE SPACE PLANNERS GUIDE SIZING PROCESS

To validate the use of the Space Planners Guide sizing process, eight launch vehicles (including the Saturn IB) were sized and compared with their actual reported values: four launch vehicles from the era of the Space Planners Guide, three more recent launch vehicles, and one future concept. These studies provided an opportunity to see where the Guide excels and where it falls short. The final conclusion of the studies is a great deal of confidence in the Guide's process in sizing the overall weight. Beyond the overall weight, there are simply two many variations and goals for optimization to expect any single process to excel for such a wide range of launch vehicle types.

Launch vehicle	**Saturn IB**		**Atlas LV-3B**		**Saturn V**[a]		**Saturn V**[b]	
	Actual	**% Error***	**Actual**	**% Error***	**Actual**	**% Error***	**Actual**	**% Error***
Number of stages	2	– / –	2	– / –	2	– / –	3	– / –
Velocity capability, m/s	9,450	– / 0.01	9,450	– / -0.15	8,750	– / 0.01	12,800	– / 0.12
Payload mass, *kg*	18,600	– / –	1,360	– / –	135,000	– / –	45,800	– / –
Escape tower mass, *kg*	4,000	– / –	580	– / –	4,000	– / –	4,000	– / –
Third stage								
Stage mass, *kg*	–	–	–	–	–	–	116,000	-32 / 96
Propellant mass, *kg*	–	–	–	–	–	–	104,300	-31 / 98
Structure mass, *kg*	–	–	–	–	–	–	11,700	-40 / 73
Total mass, *kg*	–	–	–	–	–	–	161,800	-23 / 69
Second stage								
Stage mass, *kg*	116,500	-27 / 2	114,600	-91 / -92	481,400	54 / 106	481,400	-56 / 148
Propellant mass, *kg*	105,300	-27 / 2	111,200	-92 / -93	439,600	54 / 106	439,600	-56 / 148
Structure mass, *kg*	10,600	-18 / 11	3,400	-68 / -73	41,800	56 / 109	41,800	-52 / 151
Total mass, *kg*	135,100	-23 / 2	116,000	-90 / -91	616,400	47 / 88	616,400	-45 / 138
First stage								
Stage mass, *kg*	452,300	5 / 18	4,800	1,756 / 2,468	2,233,800	81 / 0	2,233,800	-74 / -39
Propellant mass, *kg*	414,000	7 / 20	1,400	5,807 / 8,080	2,066,100	84 / 1	2,066,100	-74 / -38
Structure mass, *kg*	40,500	-22 / -13	3,400	89 / 157	167,700	50 / -16	167,700	-77 / -48
Total mass, *kg*	**587,900**	**-1 / 14**	**120,800**	**-17 / 10**	**2,896,200**	**71 / 17**	**2,896,200**	**-68 / -2**

All *actual* values are approximate and have been compiled from [1–9]
* For the sake of brevity, the percent error for the initial and final estimates made by the Guide are given in the format **(initial % / final %)**

[a] Two-stage Saturn V converging on a lower velocity requirement
[b] Three-stage Saturn V converging on the velocity required for a Trans-Lunar injection

TABLE C.1 – Comparison of the Guide's sizing process with launch vehicles from the Guide's era.

The results of the Space Planners Guide era launch vehicles are shown in Table C.1. Note that the approximate actual value for each mass is reported, along with the percent error of the first estimate, and the percent error of the final estimate. Along with the previously explored Saturn IB, the Atlas LV-3B, and two versions of the Saturn V

Launch vehicle	Titan II23G		Soyuz U		Falcon 9 v1.1		SpaceX ITS[a]	
	Actual	**% Error***	**Actual**	**% Error***	**Actual**	**% Error***	**Actual**	**% Error***
Number of stages	2	– / –	3	– / –	2	– / –	2	– / –
Velocity capability, m/s	9,450	– / -0.11	9,700[b]	– / -0.10	9,450	– / -0.01	9,450	– / 0.03
Payload mass, *kg*	3,175	– / –	6,900	– / –	13,150	– / –	550 tons	– / –
Escape tower mass, *kg*	–	– / –	–	– / –	–	– / –	–	– / –
Third stage								
Stage mass, *kg*	–	–	25,000	-35 / -15	–	–	–	–
Propellant mass, *kg*	–	–	21,400	-30 / -7	–	–	–	–
Structure mass, *kg*	–	–	2,400	-49 / -38	–	–	–	–
Total mass, *kg*	–	–	31,000	-27 / -11	–	–	–	–
Second stage								
Stage mass, *kg*	29,000	-23 / -40	105,400	-48 / -56	96,600	-20 / -30	2,100 tons	10 / -4
Propellant mass, *kg*	27,000	-25 / -41	95,400	-48 / -56	92,700	-24 / -33	2,010 tons	6 / -8
Structure mass, *kg*	2,800	-25 / -40	6,900	-28 / -38	3,900	74 / 55	90 tons	103 / 77
Total mass, *kg*	32,800	-22 / -37	137,300	-43 / -46	109,800	-18 / -26	2,650 tons	8 / -3
First stage								
Stage mass, *kg*	122,000	47 / 21	44,500	314 / 308	418,800	26 / 1	6,975 tons	72 / 31
Propellant mass, *kg*	118,000	42 / 17	39,200	337 / 331	395,700	25 / 0	6,700 tons	67 / 27
Structure mass, *kg*	4,800	138 / 100	3,800	239 / 235	23,100	53 / 23	275 tons	188 / 119
Total mass, *kg*	**154,000**	**33 / 10**	**310,000**	**-15 / -17**	**505,800**	**22 / 0**	**10,500 tons**	**42 / 11**

All *actual* values are approximate and have been compiled from [7, 8, 10–12]
* For the sake of brevity, the percent error for the initial and final estimates made by the Guide are given in the format **(initial % / final %)**

[a] Proposed values for the spaceship varient, not the tanker
[b] Adjusted velocity required due to launching from a higher latittude. Room for improvement.

TABLE C.2 – Comparison of the Guide's sizing process with more recent (even future) launch vehicles.

were studied.

The Atlas LV-3B, the launch vehicle used to send John Glenn into orbit, provided an excellent look at a two-stage launch vehicle that operated differently than what is typical. The first stage of the Atlas mainly consists of only an engine, sharing fuel with the upper stage. This allowed the Atlas to use one engine that would perform well at sea-level, then drop it once a sufficient had been reached and use a more efficient engine for the rest of the launch. The Space Planners Guide process has no way of accounting for this, thus its mass estimates for the first stage are wildly inaccurate. However, it is more important to note that the final estimate of the total launch vehicle is still within the target accuracy. This means that, early on before other design details have been determined, a planner could still provide a rough estimate of the launch capability, and with some other methods, the cost of the launch vehicle. The results of more recent and future launch vehicle studies are shown in Table C.2. The Titan II23G, the Soyuz U, Falcon 9 v1.1 and the future SpaceX International Transport System have been studied.

The Titan II23G final estimate was again, well within the expected accuracy of the Guide. It was however, the first to reveal another trend, namely, the estimates for the structures have become less and less reliable. It seems the Guide can account for propulsion advancements though the input specific impulse and propellant density of each stage, but the estimates for the structure factors are hard coded into nomographs that seem to be outdated.

The Soyuz U, Russia's reliable launch vehicle, explored a number of constraints with the process. First, the Soyuz U consists of two stages, with an initial cluster of rockets around the typical first stage. When the boosters have run out, they fall away leaving the two stages to advance. For the Guide, this needed to be treated as a three-stage vehicle, though again the optimized stages masses will not be correct. Second, the Soyuz typically launches from the Baikonur Cosmodrone often into a different orbit than many of those that launch from Florida. This results in a necessary re-estimation of the velocity required for the 185 *km* orbit. A slight modification was made to see that the resulting estimation did in fact improve to within the expected accuracy though no concentrated effort was made. The author is confident that a proper effort to estimate the velocity required will result in a much more accurate estimation.

The SpaceX Falcon 9 v1.1 was selected as a very recent and topical study. Version 1.1 was selected since, as it was just recently retired, the author had more confidence in its reported masses and performance capabilities. The Guide's final estimate struggled with the second stage, though it was spot on for the first stage (aside from the structure mass) and the total mass of the launch vehicle.

Finally, the International Transport System, SpaceX's vision for the transport to colonize Mars was studied to see how well the Guide's process would handle a launch vehicle of that magnitude. The answer is: surprisingly well. Once again, several of the nomographs were not large enough to account for such massive stages so the approximations could be improved. However, the process still sized the entire launch vehicle to within fifteen percent of the total launch mass. Aside from outdated structure masses, the process did well sizing the other stage mass components as well.

In summary, the sizing process of the Space Planners Guide has been digitized and its logic programmed into a twice-converging script, enabling the early estimation of a launch vehicle's total lift-off mass. The process is very straight forward, allowing a planner to follow each step and trace its inputs from beginning to end.

Several case studies were performed to determine the applicability of the sizing process for a modern early planning effort. The results show that the process is capable of providing the total launch mass of a variety of different launch vehicles for altered missions as well, within the target confidence levels stated within the Guide, $\pm 20\%$. With care, the process can also be adapted and/or augmented for more reliability in the estimates of stage masses and component masses.

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