Design of Hypersonic Lifting Body Vehicle (Lead Chief Engineer's Report)

Signatures:

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Abstract:

This final report is the key deliverable for the MAE 4351 Capstone Project in fulfillment of ABET objectives, documenting the activities and contributions of the lead engineer of Ascension Aerospace, a team composed of 21 senior-level aerospace undergraduates. After introducing the scope of the project and some historical background, the author sets out to build a business case and define the mission profiles of a vehicle system which launches three variants of the Model 176 Hypersonic Lifting Body by means of two SpaceX reusable launch systems: Falcon B5 and Falcon Heavy. The business case is built on civilian market demand for low-Earth orbital access and point-to-point transportation. The costs and revenues are examined for a nominal mission and operations are assessed against competition in potential markets. Trade studies are conducted to properly design the reconnaissance mission profile to meet geopolitical goals. As the chief, the author reports on the team management and workflow methodology, as well as chronicles the program development of Ascension Aerospace. The methodology is then built for the multi-disciplinary analysis of reverse-engineering the technical aspects of meeting the proposed mission profile. A summary of some of the methods used for this analysis is supported by a literature review near the beginning of the report. The report concludes with an initial engineering-development cost estimate and some suggested future work.

Distribution:

Institution:	Dept.:	Name:
The University of Texas at Arlington	MAE	Dr. Bernd Chudoba

Work Disclosure Statement

The work I performed to document the results presented in this report was performed by me, or it is otherwise acknowledged.

Date: 08/05/2018

Signature: Leonardo Pinero-Perez

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Nomenclature

A	Historical Trend Constant
B	Historical Trend Constant
C	Coefficient (defined by subscript)
D	Drag Force
g	Gravitational Constant at Earth's Surface
ICI	Industrial Capability Index
I_p	Propulsive Index
I_{sp}	Specific Impulse
I_{str}	Structural Index
M	Mach Number
MY	Man-years (engineering)
N	Mission Parameter (integer number, such as crew members)
$\dot{q}$	Heat Convection
R	Geometric Radius
$r_{O/F}$	Oxidizer to Fuel Ratio
S	Reference Area
T	Thrust
W	Weight
V	Velocity
VOL	Volume
β	Mach-related Flight Parameter, $\sqrt{M^2 - 1}$
γ	Flight Path Angle
ε	Emissivity (Blackbody Radiation)
ρ	Density

I. Introduction

A. Project Scope

The following report documents the conceptual design process for a hypersonic and launch vehicle pair conducted by Ascension Aerospace, a team of 21 undergraduate senior-level aerospace engineering students, as the main deliverable for the MAE 4351 Capstone Design Course at the University of Texas at Arlington, Texas. This report is focused on the roles of the author as the lead chief engineer.

This design project is centered around reverse engineering the methodology used to develop the McDonnell Military Model 176. This mission included a launch vehicle and a hypersonic vehicle, and the team is responsible for validating and explaining the underlying physics behind the design to prove that it is capable of achieving mission goals. This validation method is based on extensive literature research on the topics of hypersonic flight, aerothermal heating, mission operations, stability across speed regimes, and fully-integrated sizing methodology. Trade studies are then implemented in this methodology to find further modern-day applications of this design.

There is much research which pertains to the hypersonic vehicle, such as work on the characteristics of lifting bodies and hypersonic aerodynamics. The Model 176 was recently declassified, so these documents provide a catalyst for developing the reverse engineering methodology as well as validation data. This defers from designing a completely novel vehicle, as this team's methods must now produce a physical truth that was developed historically. Design from the ground up does not have any physical validation, so there is significant room for faking erroneous results which could easily pass by review. Since there is a physical basis to verify this design, an inadequate methodology will be quickly discovered in this project's scope. However, this leaves the temptation to develop a trivial methodology which overly relies on the validation data. It must be made clear that the methodology must be derived from an independent knowledge-base of physics, multi-disciplinary analysis, and commonalities with historical vehicle precedents. The combined vehicle, with the Model 176 and launch system, is shown below alongside comparative configurations [1]:

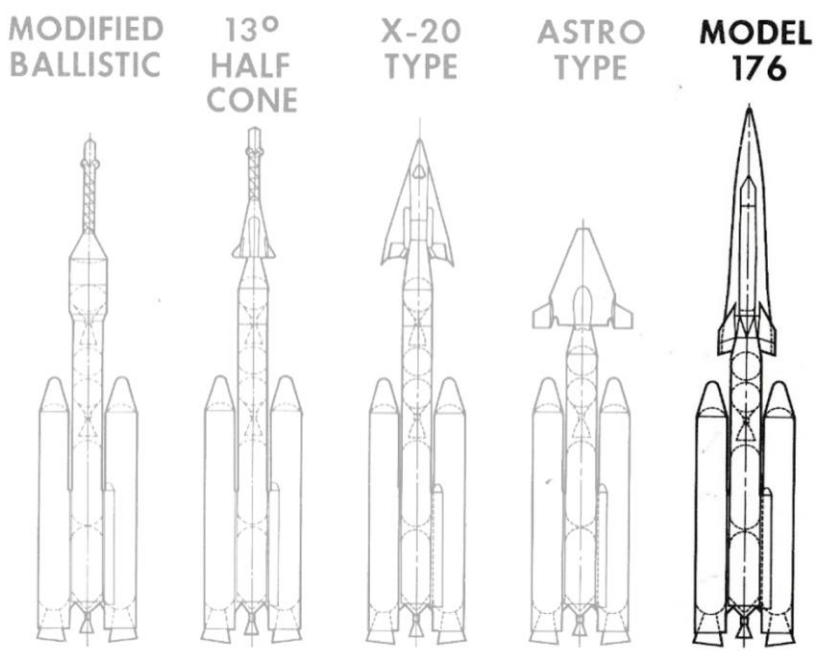


Figure 1. Operationally Comparative Configurations for Hypersonic Re-entry Vehicles *[1]*

For the launch platform, the real-world vehicles used are SpaceX's Falcon 9 B5 and Falcon Heavy. These will be reverse-engineered. One of the main tasks of the launch team is to apply the SpaceX launching platforms (Falcon B5 and Falcon Heavy) as a new launch platform for the Model 176. One of the main advantages of these platforms is that the lowest, largest stages are fully reusable, removes most of the expendable characteristics from the total mission.

The mission requirements for the design in question concern the development of a concept for a re-usable, manned space transport system, which is capable of orbital operations, fuel-efficient orbital maneuvering by atmospheric entry and exit, and point-to-point global transport. Such a vehicle must be integrated into some launch platform, and two sub-teams are created to divide the task of developing the upper stage hypersonic vehicle and launch vehicle.

Additionally, this report includes a business case before disciplinary analysis to first show that this configuration is economically viable in addition to the underlying physics and engineering choices.

B. Global Context and Applications

The United States is falling behind in space warfare, and there has been renewed interest in the rapid development of hypersonic vehicles to counteract Russian and Chinese strategy. One application of a hypersonic craft is the ability to rapidly change the orbital inclination by descending from orbit into a hypersonic upper-atmospheric flight profile and using aerodynamic effectors to alter the trajectory [1]. This application allows for surprise reconnaissance, which is a maneuver that an orbit-only satellite is incapable of doing without expending copious amounts of fuel. The capability to efficiently change the inclination of an orbit at such high speed is what makes this upper stage unique. By doing this, the military craft has complete flexibility in major course corrections, which is especially useful for military reconnaissance. The current situation only permits predictable orbits which can potentially allow an enemy to effectively hide operations by waiting for a known satellite to pass.

The figure below demonstrates the relationship between the momentum change required for an in-orbit inclination change. This plot is qualitatively derived from an observation of the two systems' required mission profiles. It is apparent that for high inclination-change missions such as those required for dynamic reconnaissance, the superior design is that which uses a maneuver synergetic with the momentum imparted on the atmosphere at the expense of energy loss due to drag. The slope of this line is directly related to the ratio between lift and drag. In terms of orbital mechanics, this is analogues to the ratio between to normal or anti-normal thrust and orbital decay.

One other observation is that conventional inclination changes are actually advantageous for lower inclination changes since the initial fuel cost is zero. A synergetic mission profile requires multiple burns to descend into the upper atmosphere and ascend into the new orbit.

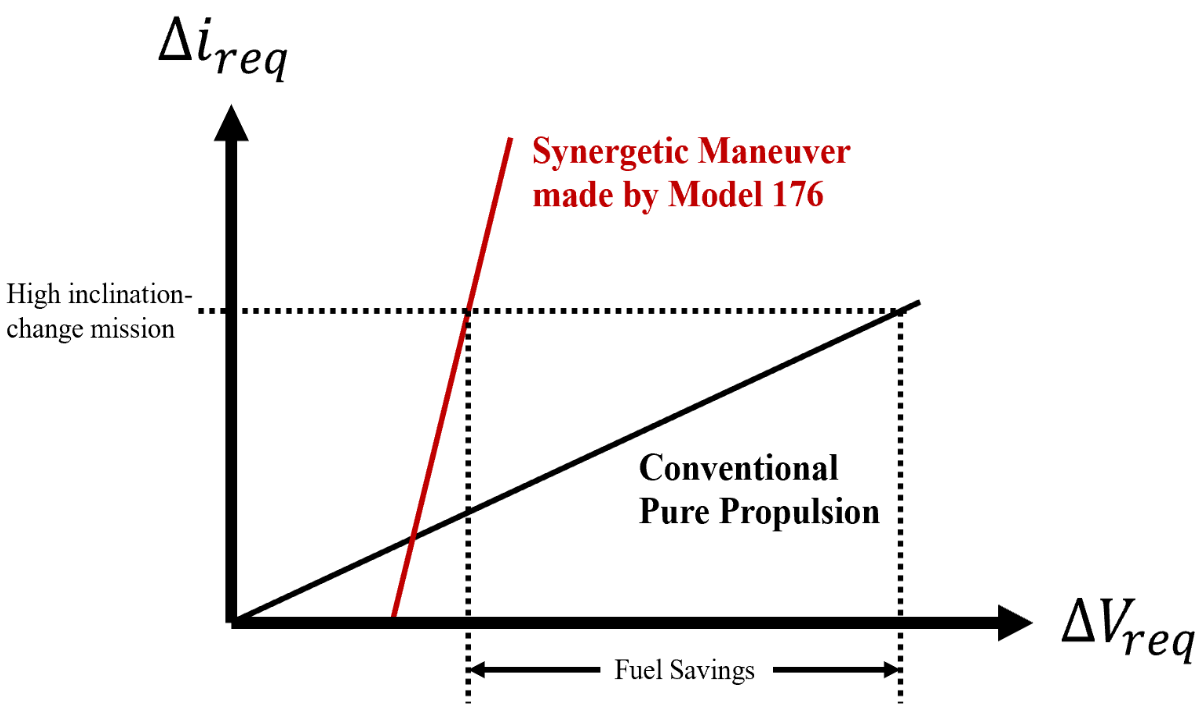


Figure 2. Comparison of Systems for Velocity Change Required to Change Orbital Inclination

Additionally, a high cross range capability is desired so that orbit time can be reduced. It is advantageous to land the craft at any time, which allows for more logistical flexibility. The capability for high lateral range allows for rapid turn-around time, which is a mission performance requirement to allow for the feasibility of the business case presented.

The geography of the Earth's surface is assessed for insight on inclination changes and required surface coverage. The current military powers that pose a threat to American interests, particularly in the realm of orbital capability and space readiness, are China and Russia. Because of China's relatively close distance to the equator (situated between 20 and 55 degrees north of the equator), it is not a critical mission trade to cover by an inclination change. However, full coverage of mainland Russia (situated between 50 and 70 degrees north of the equator), requires a larger inclination change and is assessed.

From the launch pad at Cape Canaveral, Florida, a low-inclination parking orbit is desirable for military applications, since it is the least visible from Earth's surface. Additionally, it is desirable from the perspective of launch system performance requirements, as the surface of the Earth is rotating at a greater velocity tangent to the equator at lower latitudes. Given the compounding effects of the rocket equation, this is a non-trivial contribution from the Earth's angular momentum. The mission trades will consider a military mission starting from a low inclination parking orbit and performing an inclination change to fully cover the Russian land mass. With these military considerations, the mission trades will be conducted assuming a 30-degree inclination as a mission requirement.

Landing safely in the continental United States requires the cross-range capability provided by a lifting body. The lifting capability of this body is also used to assist in the inclination changes without expending large amounts of fuel. With a lift-to-drag ratio of about 3, the vehicle may produce an inclination maneuver for a third of the cost of conventional purely propulsive means. The concept of a lift-to-drag is the "free" momentum change for every non-conservative loss to drag. This concept is verified by a trajectory comparison analysis by the performance discipline. A lift-to-drag ratio of 3 also allows for a cross-range to land anywhere around the world at any time [2].

The military applications of this vehicle are explored first since the initial conception for the Model 176 was to develop military technology for geopolitical Cold War objectives. To develop a profitable business model, the chief engineer of Ascension Aerospace has focused on affordable launch costs based on an increased flight frequency and vehicle reusability. Reusability opens the door to a rapid turn-around time, which becomes a design requirement. This frequent flight rate must be supported by a large civilian market. From this support, the proper infrastructure is put in place for a budget-feasible launch of a military mission. The military missions alone are not expected to bear the cost of the infrastructure and manufacturing, hence the need for a larger civilian market.

Like the vehicle's orbital maneuvers, the business model is also synergetic between the military and global commerce. This seems to be the inverse of the automobile manufacturer Tesla's business model, which produces high-end luxury cars to support the development of the mass production of more affordable ones. Since flight frequency is the key to affordability, the goal of Ascension Aerospace is to produce frequent point-to-point civilian transport to support the infrastructure for a more expensive military vehicle and mission.

C. Historical Background

The following section is organized chronologically and provides the context in which the reverse-engineered vehicle was developed.

Research into high-lift orbital vehicles began with the Silbervogel project, developed for the German Ministry of Aviation during World War II. The Silbervogel, being part of the Amerika bomber program, was meant to carry a weapons payload over the continental United States as part of the war effort. The flight profile included taking the craft up to a sub-orbital path, having been propelled on a rocket sled and its own rocket engines. Upon re-entering the atmosphere, the Silbervogel would then initiate a pull-up maneuver, where its high-lifting characteristics will allow it to skip outside of the atmosphere again. The process would repeat, where some drag losses would reduce the velocity of the vehicle so that each successive skip would reduce in magnitude [3]. However, the aircraft would cover a very high range in the process, allowing it to take off from Germany and land at least as far as the Japanese Empire. This would allow for a rapid turn-around and guaranteed safety of the pilot.

It is interesting to note that the most lasting innovation from this program was the regenerative cooling concept, where cryogenic fuel would be pumped around the nozzle to keep it cool. The cryogenic fuel would be used immediately but has successfully carried heat away from the critically heated sections of the craft.

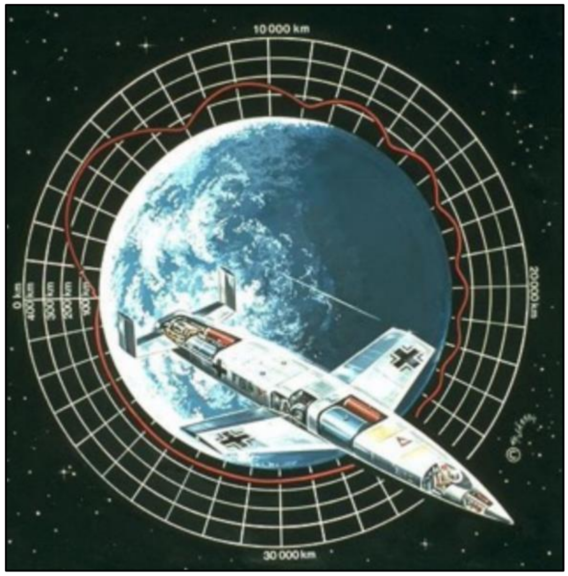


Figure 3. Silbervogel Artwork Showing the Skipping Flight Profile *[4]*

The Kyldesh bomber was a later Soviet variant based on Sänger's work, which mostly differed from the parent design by adding ramjet engines on the wing tips. This bomber design never moved past the conceptual stages. The United States variant moved beyond and developed the X-20 Dyna Soar aircraft. This was intended to continue Sänger's work for a lifting wing-body vehicle but was later cancelled as the need for manned aircraft diminished in favor of unmanned satellites. Much of the groundwork for man-rated boost glide systems was developed in the X-20 program, including the X-15 program [5].

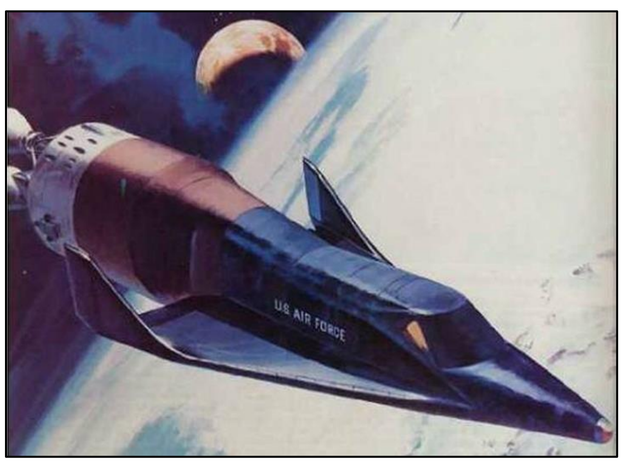


Figure 4. Depiction of X-20 Dyna Soar with Orbital Stage *[6]*

Over half a century ago, there was classified development of a hypersonic lifting body vehicle which could be used to increase the cross-range capability of aircraft. This cross-range capability is the driving reason for using a hypersonic lifting body. As the report will later cover, the design of a lifting body vehicle adds much complexity and weight to the upper stage of a crewed space mission. However, the benefits of such a craft are reduced wait times and operational flexibility, leading to increased flight frequency and reusability. There is great financial cost in supporting the logistics and infrastructure required to maintain a spacecraft, and an increase in frequency will drastically reduce launch cost [2]. In light of this economic and military paradigm shift, increased cross-range capability and ready reusability is worth the increased complexity and weight. As was stated in the business case, the key differentiator of this craft when compared to other vehicles in the civilian market is that of short turn-around times.

The addition of complex geometry and aerothermodynamics incurs a significant risk to the crew and vehicle. Reducing risk by test flights is expensive if done to its full mission extent, so the lifting body geometries are first developed to be tested in the less extreme flight regimes, as well as wind tunnels. Some good resources for the history of testing lifting bodies are saved on the team's literature search. This first section of the "Mission" chapter will justify the key elements and rationale behind the mission of the Model 176.

Figure 5. Historical Aircraft at Dryden Research Center (Artist: Robert McCall) *[5]*

The main utility of these resources is that they provide lessons learned from implementing these designs into the physical world. From these lessons, the design can be guided and important driving variables selected. The three main factors that guided re-entry into the atmosphere by any object are as follows [5]:

- Intense heat generated by friction with the Earth's atmosphere
- High accelerations felt by pilots during rapid loss of orbital speed
- Selection and control of initial entry angle to determine heating and acceleration

One of the lessons that can be deduced from these points is that thermal protection against a rapid heat pulse was out of the question. This is the more conventional type used on ballistic missile systems but would be inappropriate for human transports. A flight path which could be thermally protected by using heat pulse methods would be a flight path that kills the human passengers from intense g-loading [5].

There are two methods for thermal protection systems, and both can be utilized in the same vehicle:

- Active Cooling: cold fluid through a hot area to then be dissipated by a radiator
- Radiative Cooling: special material to radiate as much energy into the atmosphere as taken from convection

The use of conventional ablator technology is not attractive because it may change the shape of the craft's geometry. This is alright for capsules with small lift-to-drag ratios. Even if the geometry was held constant by charred ablator, then the aircraft's lift-to-drag ratio would not be useful, as its time in the air would be limited to the time of ablator use. Additionally, using an ablator material significantly reduces the re-usability of such an aircraft.

II. Literature Review

A. Management

As the lead chief engineer, the author has realized that there is in fact a learning curve to developing the skills necessary to extract maximum productivity from the team. This section will provide some sources for managing the team which led to the project deliverables. This section is relevant to an engineering report since the process for managing directly influences the effectiveness of the engineering methodology described.

There are many books on leadership and business development, so it is important to select relevant sources. References were selected by the ethos of the writers of these books. Writers were selected by their individual accomplishments in the technology industry, since this industry is based on a product which must be engineered (thus relevant to this engineering project).

The writers selected are listed below:

- Peter Theil, founder of PayPal and Palantir Analytics [7]
- Ben Horowitz, CEO/founder of Opsware [8]
- Andrew Grove, cofounder and CEO of Intel [9]

Peter Theil's book *Zero to One* is useful as a primer to the idea of startups and their place in American and global culture. Being a short book, it does not give highly specific methods of management, but many of the chapters are good for insights on the place of our project as an engineering deliverable to the rest of the world and its goals.

Ben Horowitz's book *Hard Things* has been very useful on the details of managing a large company and developing methods of communication and business-side strategy. It is lacking in engineering expertise and product development, although the book seems to have been meant for engineers-turned-managers. Even so, the human element in the author's role is significant, so this book is still relevant. Some useful principles drawn from this book:

When hiring a management team, most startups focus exclusively on IQ, but a bunch of high-IQ people with the wrong kind of ambition won't work.

Perhaps the CEO's most important operational responsibility is designing and implementing the communication architecture for her company.

This communication architecture might include [8]:

- Organizational Design
- Meetings
- Processes
- Email/Yammer
- One-on-one meetings

The source has an entire chapter dedicated to setting up one-on-one meetings with employees to extract as much productivity and direction from the organization as a whole. One-on-one meetings are also referenced in Dr. Grove's book *High Output Management*, which is actually referenced in Horowitz's book.

Not much information has yet been extracted from Andrew Grove's *High Output Management* since this book has not been read completely by the time of this report's writing. However, the book's introduction does explain what type of information the reader can gain from reading it. The performance measures of a manager are stated, and the book explains how a manager can increase that performance measure from a manufacturing point of view (every working organization has a product, such as the intellectual property of an engineering group).

The output of a manager is the output of the organizational units under his or her supervision or influence. [9]

In his book, Dr. Grove explains the concept managerial leverage, which measures the influence a manager has on the productivity of his or her team. The productivity of a manager is determined by the way managerial leverage is exercised. The book goes over how to exercise such leverage effectively and as much as possible. This has to do with task-relevant feedback [9].

B. Cost

A useful resource for numerically determining costs of a program or individual vehicle after the first iteration is the NASA Cost estimating handbook [10] and the older NASA Spacecraft Cost Estimation report [11]. The cost will be calculated when the first iteration is completed, so that future iterations can be optimized for commercial missions. The methodology from this handbook is outlined in the Nassi-Schneiderman diagram below.

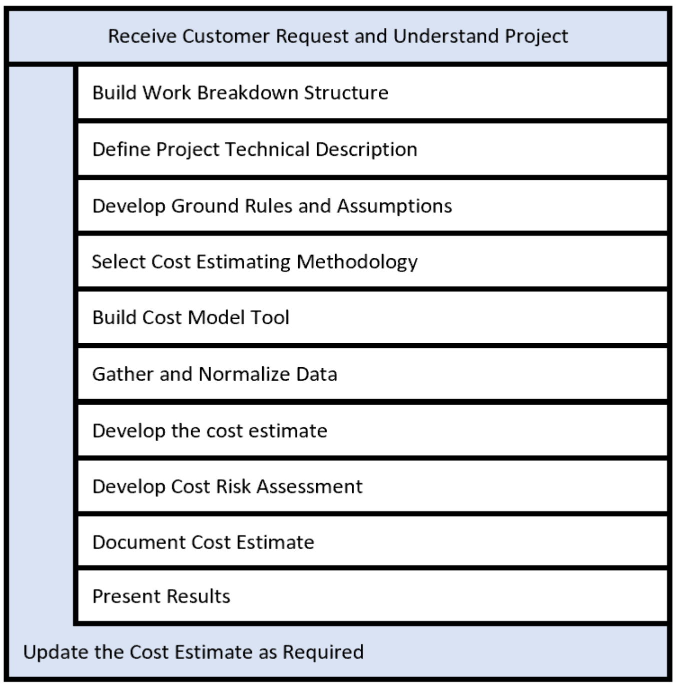


Figure 6. Nassi-Schneiderman Diagram for the NASA Cost Estimating Handbook

The first step requires an understanding of the project, and from here a work breakdown structure can be built.

Data

- What data do you need?
- Are the data readily available?
- If the data are not readily available, what are your alternatives?
- Are the organizations you need to collect the data from cooperative & accessible?
- Are non-disclosure agreements required?

Expectations

- What is your expectation of the estimate?
- What is the expected outcome or usage of the estimate? (based on estimate type)
- What is the customer's expectation of the estimate?
- What is the team expectation of the estimate?
- What is the Agency-wide expectations of the estimate outcome and usage?

Resources

- How many people are required to conduct the estimate?
- How many people are available to conduct the estimate?
- What is the budget required to conduct the estimate?
- What is the available budget to conduct the estimate?

Schedule

- How long have you been given to complete the estimate?
- How long do you need to complete the estimate, given the available resources and data?
- Do you have the resources needed to conduct the estimate with the allotted schedule?
- Do you have the time to collect the required data and analyze the data?

Figure 7. Four Critical Elements Related to Conducting and Understanding a Cost Analysis *[10]*

Below is an example work break down structure (WBS) and will be used as inspiration for developing Ascension Aerospace's completed and anticipated activities.

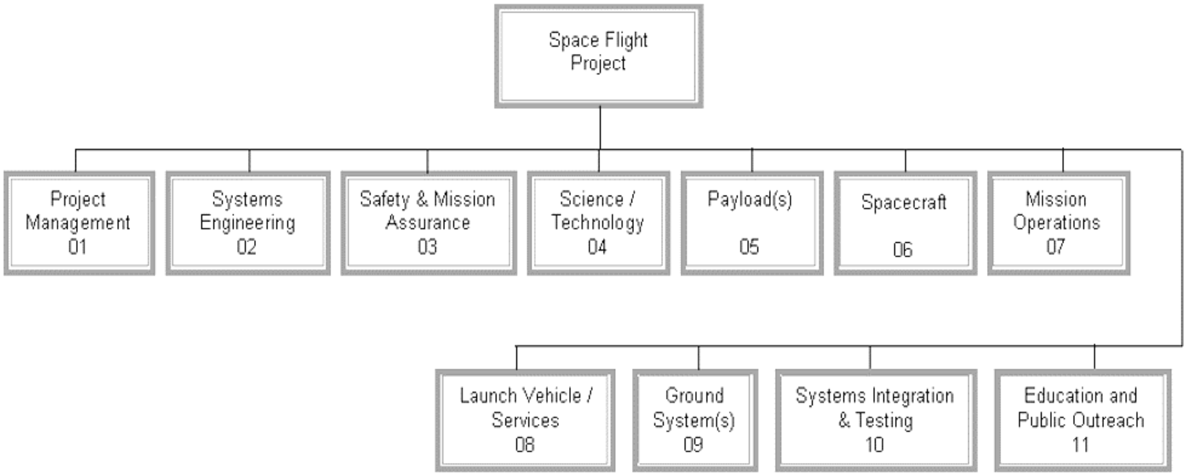


Figure 8. Standard NASA Project Level 2 Work Breakdown Structure *[10]*

After a WBS is made, the project must be defined and estimation ground rules established. Defining ground rules and assumptions may include the following [10]:

- Scope: this is largely defined by the WBS
- Make vs. buy decisions: do you develop an engine or buy off-the-shelf?
- Government Furnished Equipment/Information (GFE/GFI): the government (or other organizations) may provide technical capabilities and assistance or information which could drive down program costs. This assumption allows the free development of technology and products which fall under this category
- Contractors and Subcontractors: similar to buying off-the-shelf, but for research and development
- Budget profile: predicts the effect that a budget will have on the overall cost
- Labor resources and rates: this is dependent on program location and availability of labor for specific skill sets
- Risks and associated risks: if risk mitigation measures are put into place, they will increase the baseline cost estimate but reduce cost overruns
- Production units and quantities: increased quantities, as with flight frequency described later in this section, can drive down costs per unit
- Description of dollars and inflation: this is especially useful for programs that run for several years

A useful plot was found suggesting that operational costs drastically reduce with increasing flight frequency. This is because existing infrastructure can be used multiple times to support increased flight frequency, driving down the cost of LEO transport. [2] The image below will be referenced as a guide for reducing cost. Keeping in mind that the plot is logarithmic, the biggest cost drivers are research and development, reusability, and infrastructure. The latter two can be greatly reduced by designing a reusable craft with rapid turn-around time [2].

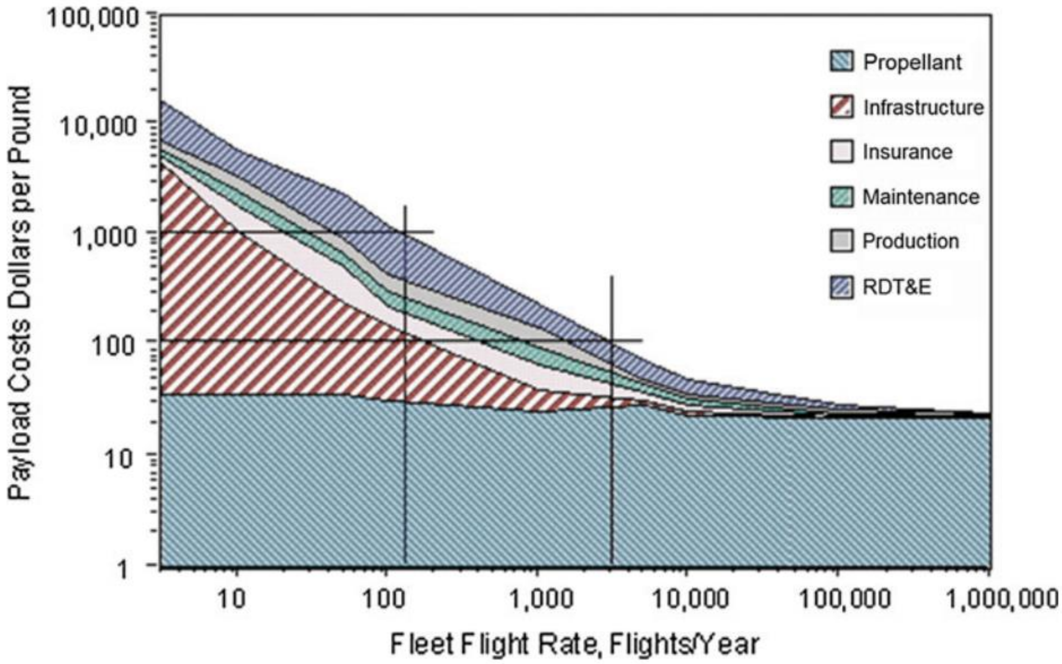


Figure 9. Payload Cost per Pound vs. Flights per Year *[2]*

C. Synthesis

The sizing method currently under consideration is the Czysz method [2], which takes the industry capacity index (ICI) into account to determine which geometries are possible for a given planform and structural ratios. The required ICI is what represents mission requirements such as payload volume and delta V. These mission requirements size the aircraft for a given payload size. Ideally, the most cost-effective design with the lowest ICI in the solution space which can carry the required payload. One such space found in [2] is demonstrated:

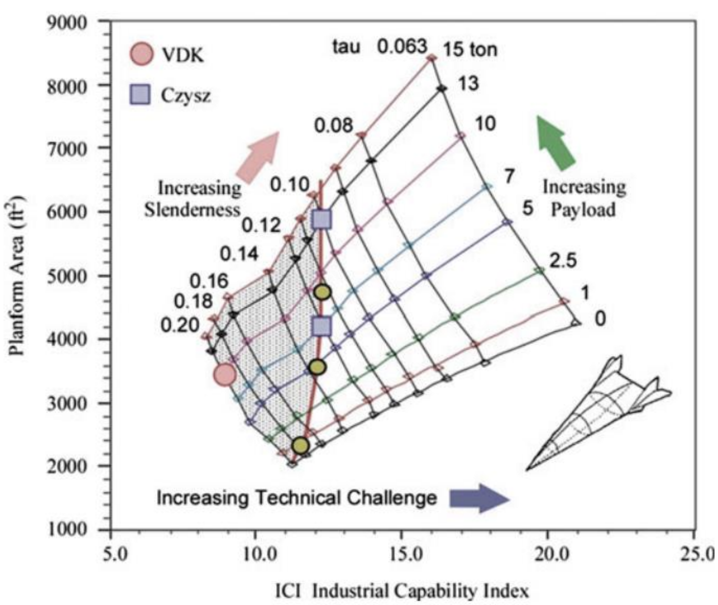


Figure 10. Czysz Sizing Method found in Literature *[2]*

The sizing process begins with defining certain based on disciplinary available technology and mission requirements. The coefficients are defined as follows:

$$ICI = 10 * \frac{I_p}{I_{str}} = 10 * \frac{\left(\frac{\rho_{ppl}}{W_R - 1}\right)}{(W_{str}/S_{wet})}$$

I_p is a function of propulsive technology and delta V required, and I_{str} is dependent on advancement in material and structural design. Together, they formulate the industrial capacity index. The I_p is further described by [2]:

$$I_p = \left[\frac{\rho_{fuel}(1 + r_{O/F})}{1 + r_{O/F} * \frac{\rho_{fuel}}{\rho_{oxidizer}}}\right]\left\{\exp\left[\frac{\Delta V * \frac{T}{D}}{g * I_{sp} * (\frac{T}{D} - 1 - \frac{\sin\gamma}{\left(\frac{T}{D}\right)}}\right] - 1\right\}^{-1}$$

It is planned to use Corning's method as inspiration for the development of a methodology to select key design variables early on and will be used as a testing ground for the methodologies of each of the disciplines when they are asked to provide deliverables. In this book, chapter two outlines this early methodology for the design of re-entry vehicles [12]. The Nassi-Schneiderman diagram of this method has been built to use as a comparison to the methods developed by the author. The process found in Corning's book provided inspiration for the important variables in disciplinary analysis.

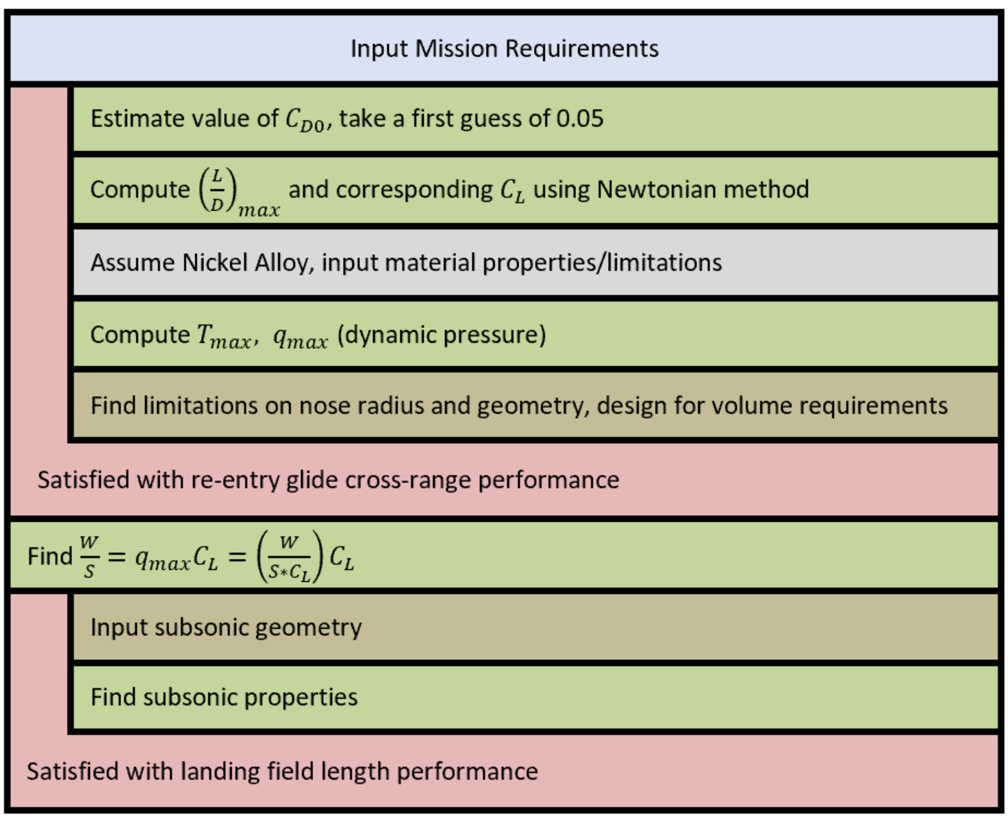


Figure 11. Nassi-Schneiderman Diagram for Corning's Method for Re-entry Vehicle Characteristics

D. Performance

A good source for performance and re-entry trajectories was Gerald Corning's book on Aerospace Vehicle Design [12]. This was an inspiration on where the author could start developing a trajectory-based MDA. The following plots are what drive the design based on mission requirements for cross-range capability. The following plot describes the descent trajectory for a vehicle of given aerodynamic characteristics following a line of constant dynamic pressure (that is what these characteristics solve for). This constant dynamic pressure determined by the lifting characteristics of the body implies that this is a steady state descent, but this plot is not useful to determine the rate of speed decay due to drag. To determine this, the ballistic coefficient must be known, which is based on drag characteristics and a different area of the vehicle. The equation which describes the trajectory according to the key design parameter is written below.

$$V_{vehicle} = \left[\frac{1}{V_C^2} + \frac{\rho}{2\left(\frac{W}{SC_L}\right)}\right]^{-\frac{1}{2}}$$

The variable V_C defines the circular velocity speed, which is the velocity required to maintain a circular orbit at that altitude. This is a function of altitude in a similar way as the function for atmospheric density. It is interesting to note that all of the velocity keeping the vehicle in the air is due to V_C in orbit, where the density is equal to zero. The equation will show this truth with this extreme case: the forces keeping the vehicle afloat are a combination of aerodynamic lift and orbital velocity.

The author plotted this equation in MATLAB, trying to visualize different values for different hypothetical descending craft. Their descents are shown below and the script is found in the appendix.

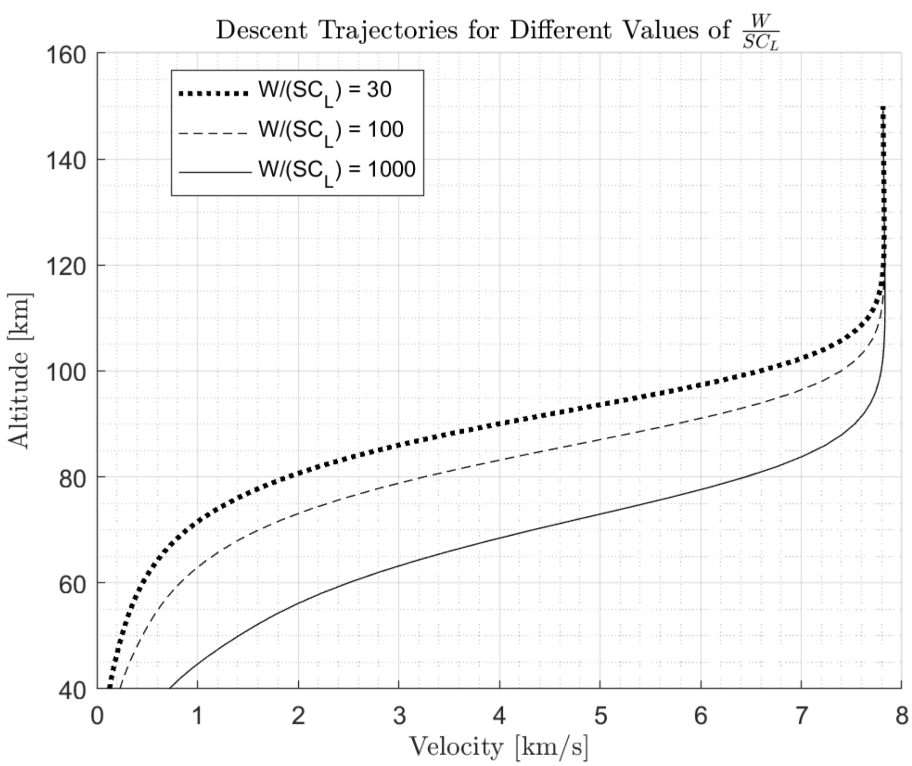


Figure 12. Descent Trajectory based on Changing Design Parameters

The following trajectory plot can be used to bleed speed or prove that it is easy to escape the atmosphere solely by aerodynamic maneuvering. As was described in the historical background, this has a precedent in the Silber Vogel design and allows for increased range by skirting through several sub-orbital paths. This was theorized in the Amerika Bomber in German designs to increase the range of military aircraft before and during World War II [13].

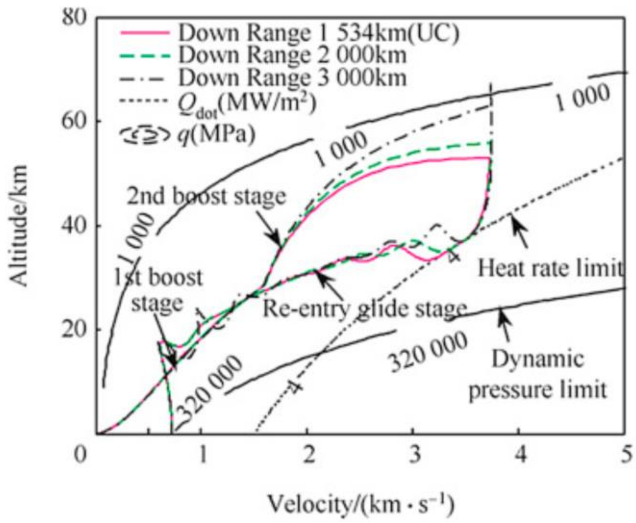


Figure 13. Dynamic Soaring Trajectory Plot *[14]*

A summary of the lit review found by the Performance discipline is described below [15]:

Convective heating for vehicle geometry and velocity:

$$\dot{q}_{conv} = 15\left(\frac{\rho_\infty}{R_0}\right)^{0.5}\left(\frac{V_\infty}{1000}\right)^3(\cos\Delta)^{1.5}$$

Stefan-Boltzmann's equation (for outgoing radiation heating rate):

$$\theta_w = \left[\frac{\dot{q}_{conv}}{\varepsilon \upsilon_{SB}}\right]^{\frac{1}{4}}$$

E. Aerodynamics

The aerodynamics database can be found in Section A of the Appendix and a more detailed literature search in their respective reports. A summary of this discipline's literature research is shown below [16]:

Useful References:

- Anderson, J. D., Hypersonic and high-temperature gas dynamics, Reston: AIAA, 2006.
- Anderson, John D., Jr., "A Survey of Modern Research in Hypersonic Aerodynamics," AIAA Paper 84-1578, June 1984.
- Anderson, John D., Jr., "Hypersonic Viscous Flow over Cones at Nominal Mach 11 in Air," ARL Rept. 62-387, Wright-Patterson Air Force Base, OH, July 1962.
- Draper, A. C., and Sieron, T. R., "Evolution and Development of Hypersonic Configurations 1958-1990", Final Report for Period July 1990 to March 1991, Flight Dynamics Directorate, Wright Laboratory, Air Force Systems command, Wright-Patterson AFB, OH, Sep. 1991.
- Dunbar, B., "NASA Dryden Fact Sheet - Lifting Bodies," NASA Available: https://www.nasa.gov/centers/armstrong/news/FactSheets/FS-011-DFRC.html.
- Nicolai, L. M., and Carichner, G. E., Fundamentals of Aircraft and Airship Design: Aircraft Design; Volume I, Reston, VA: AIAA, 2010.

The Anderson books are used to develop the methodology for hypersonic aerodynamic variables. Draper and Dunbar are historical references on hypersonic geometry development. Nicolai's design book will be used to develop the subsonic-supersonic methodology for Wing & Airfoil Selection and Drag Build-up.

Estimation Equations [16]:

$$\left(\frac{L}{D}\right)_{max} = \frac{A}{M}\,(M + B)$$

$$C_{D0} = \frac{0.05772\,.\exp(0.4076)}{\beta} = \frac{0.087}{\sqrt{M^2 - 1}}$$

$$C_D = C_{D0}(1 + B)$$

Parameters A and B are regression values to a trendline depending on what is considered state-of-the art. These values improve over time, ranging from 3 in the 1960's to 4 in the foreseeable future [2].

F. Propulsion

The propulsion database can be found in Section A of the Appendix and a more detailed literature search in their respective reports.

Literature Sources [17]:

- Dr. Chudoba's Project descriptions used to find volume of propellant
- Rocket and Spacecraft Propulsion parameter outputs and propellant densities
- Design of liquid propellant engines for actual ranges of liquid propellants
- Space Propulsion Design and Analysis book

Methods:

- Humble Method
- Huzel and Huang Method
- Building trendlines in Excel from historical engines

Historical Examples for engines:

Engine	Propellants	O/F	Thrust v. (kN)	Thrust sl	I_{sp} (v.) (s)	I_{sp} (sl) (s)	Mass (kg)	D_E (m)	P_C (bar)	R_{exp}	C_F (v.)	C_F (sl)
Vulcain	LO_2/LH_2	5.2	1,075	815 kN	431	310	1,300	2.0	105	45	1.87	1.44
Vulcain 2	LO_2/LH_2	6.1	1,350		434	318	2,040	2.15	116	58.5		
SSME	LO_2/LH_2	6.0	2,323	1,853 kN	455	363	3,177	2.4	204	78	1.91	1.53
RS 68	LO_2/LH_2	6.0	3,312		420	365	6,597	2.46	96	21.5		
HM7 B	LO_2/LH_2	5.14	62		445	310	155	0.99	36	83		
Vinci	LO_2/LH_2	5.8	180		465		550	2.15	60	240		
RL 10	LO_2/LH_2	5.0	68	0.16 k	410	10	131	0.90	24	40	1.76	0.09
RL 10A-4-1	LO_2/LH_2	5.5	99		451		168	1.53	39	84		
J-2	LO_2/LH_2	5.5	1,052		425	200	1,438	2.1	30	28		
F-1	LO_2/Kerosene	2.27	7,893	6,880 kN	304	265	8,391	2.0	70	16	1.82	1.59
RS 27	LO_2/Kerosene	2.25	1,043	934 kN	295	264	1,027	1.1	48	8	1.60	
XLR 105-5	LO_2/Kerosene		370	250 kN	309	215	460	3.1	48	25	1.74	1.22
11D-58	LO_2/Kerosene		850		348		300	1.2	78	189	1.82	
RD 170	LO_2/Kerosene	2.63	8,060	1,925 kN	337	309	9,750	4.2	245	37		

Figure 14. Propulsion Team Historical Engine Database [17]

G. Structures

The structures team has compiled several equations regarding thermal loads and useful equations for determining peak temperature as a function of velocity and altitude, based on the aircraft's geometry and material properties.

A summary of the literature search conducted by the structures team [18]:

Literature Sources (PDF's and Technical Papers):

- Performance study of two-stage-to-orbit Reusable launch vehicle propulsion alternatives.
- Comparative Analysis of two-stage-to-orbit Rocket & Airbreathing Reusable Launch Vehicles for Military Applications
- Launch Vehicle Design Process: Characterization, Technical Integration, and Lessons Learned.
- Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design
- Development of a Conceptual Design Weight Estimation Library

Literature Sources (Textbooks):

- Space Vehicle Design: Second Edition
- Elements of Spacecraft Design
- Design Methods for Space Transportation
- Future Spacecraft Propulsion Systems: Enabling Technologies for Space Exploration

Equations:

- Elements of Spacecraft Design- Chapter #7: Set of equations for basics of preliminary Aerothermodynamic analysis
- Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design: Set of empirical equations for weight estimations of a spacecraft

Historical Examples:

- Weight parametrization of the Hermes Space vehicle
    - Statistical and parametrical methods for weight estimation with an example of Hermes.
- Elements of Spacecraft Design:
    - Method for Parametric analysis for Aerothermodynamics with a few example problems.

H. Stability and Control

A summary of the literature search conducted by the stability and control discipline team [19]:

Literature Sources:

- *Aircraft Flight Dynamics*, a lecture series by Stengel provides S&C equations for supersonic and subsonic regimes
- *Equations and Charts for the Evaluation of the Hypersonic Aerodynamic Characteristics of Lifting Configurations by the Newtonian Theory* by Clark and Trimmer provides S&C equations for hypersonic regimes
- *Flight Determined Stability and Control Characteristics of the M2-F3 Lifting Body* and *A Correlation Between Flight-Determined Derivatives and Wind-Tunnel Data for the X-24B Research Aircraft* are both NASA technical reports by Sim. These both contain stability and control documentation for the two major types of lifting bodies, the flat body X-24B and the round bodied M2-F3. These are used for verification of the methods.
- *Investigation of the Low-Subsonic Stability and Control Characteristics of a 1/3-Scale Free-Flying Model of a Lifting-Body Reentry Configuration* by Hassell is a NASA technical document that contains S&C data for the subsonic range of capsule reentry vehicles. This will be used to investigate if S&C has any major benefits using a lifting body over a capsule.
- *Airplane Flight Dynamics and Automatic Flight Controls, Vol 1 & 2* by Roskam includes estimations for control derivatives which should be viable even in the hypersonic regime.

Methods and equations:

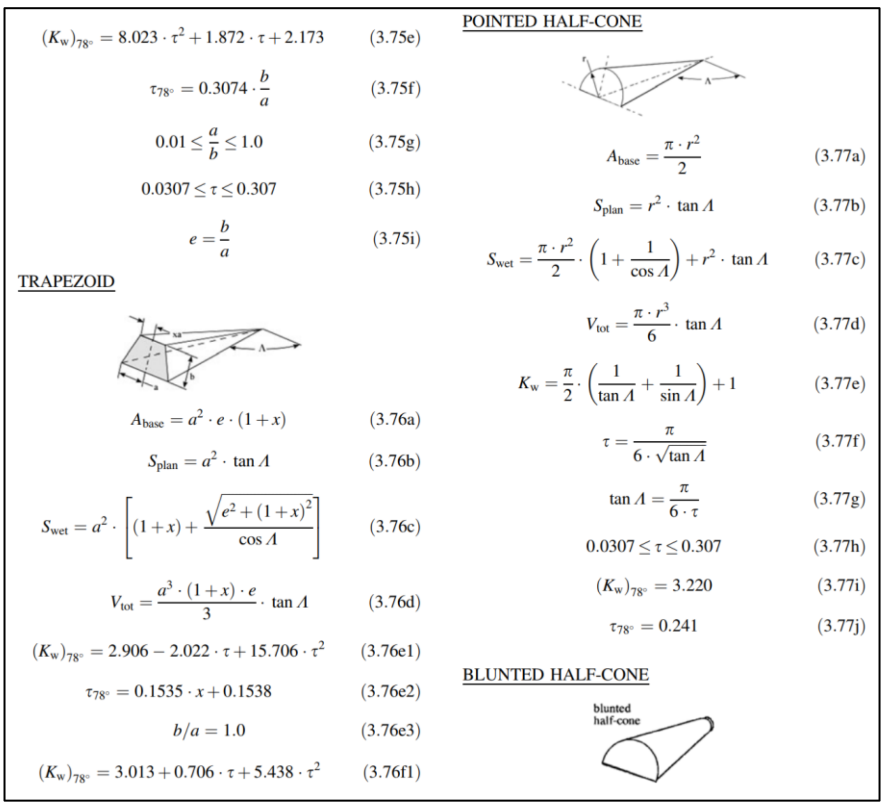
- *S&C MATLAB Code evaluating the equations for a set geometry and inputs, performing an analysis of the stability and controllability of the craft for all flight conditions.*
- $C_{n_{\delta_r}} = -\frac{(y_{e1}+y_{e2})(F_T+\Delta D)}{\delta_r q S b}$

Historical examples:

- "More than once, vehicle disturbances occurred that were followed by an oscillation sustained by damper augmentation... [which] could produce an unstable closed loop vehicle" (Sim's analysis of the M2-F3 round lifting body)

I. Geometry and Layout

A good resource for developing new geometry from sizing parameters, the following geometric definition section lends itself useful during trade studies for different vehicle geometries [2]. For now, the team intends to use the trapezoidal shape.



$$(K_w)_{78^\circ} = 8.023 \cdot \tau^2 + 1.872 \cdot \tau + 2.173 \quad (3.75e)$$

$$\tau_{78^\circ} = 0.3074 \cdot \frac{b}{a} \quad (3.75f)$$

$$0.01 \le \frac{a}{b} \le 1.0 \quad (3.75g)$$

$$0.0307 \le \tau \le 0.307 \quad (3.75h)$$

$$e = \frac{b}{a} \quad (3.75i)$$

TRAPEZOID

$$A_{base} = a^2 \cdot e \cdot (1+x) \quad (3.76a)$$

$$S_{plan} = a^2 \cdot \tan\Lambda \quad (3.76b)$$

$$S_{wet} = a^2 \cdot \left[(1+x) + \frac{\sqrt{e^2+(1+x)^2}}{\cos\Lambda}\right] \quad (3.76c)$$

$$V_{tot} = \frac{a^3 \cdot (1+x) \cdot e}{3} \cdot \tan\Lambda \quad (3.76d)$$

$$(K_w)_{78^\circ} = 2.906 - 2.022 \cdot \tau + 15.706 \cdot \tau^2 \quad (3.76e1)$$

$$\tau_{78^\circ} = 0.1535 \cdot x + 0.1538 \quad (3.76e2)$$

$$b/a = 1.0 \quad (3.76e3)$$

$$(K_w)_{78^\circ} = 3.013 + 0.706 \cdot \tau + 5.438 \cdot \tau^2 \quad (3.76f1)$$

POINTED HALF-CONE

$$A_{base} = \frac{\pi \cdot r^2}{2} \quad (3.77a)$$

$$S_{plan} = r^2 \cdot \tan\Lambda \quad (3.77b)$$

$$S_{wet} = \frac{\pi \cdot r^2}{2} \cdot \left(1 + \frac{1}{\cos\Lambda}\right) + r^2 \cdot \tan\Lambda \quad (3.77c)$$

$$V_{tot} = \frac{\pi \cdot r^3}{6} \cdot \tan\Lambda \quad (3.77d)$$

$$K_w = \frac{\pi}{2} \cdot \left(\frac{1}{\tan\Lambda} + \frac{1}{\sin\Lambda}\right) + 1 \quad (3.77e)$$

$$\tau = \frac{\pi}{6 \cdot \sqrt{\tan\Lambda}} \quad (3.77f)$$

$$\tan\Lambda = \frac{\pi}{6 \cdot \tau} \quad (3.77g)$$

$$0.0307 \le \tau \le 0.307 \quad (3.77h)$$

$$(K_w)_{78^\circ} = 3.220 \quad (3.77i)$$

$$\tau_{78^\circ} = 0.241 \quad (3.77j)$$

BLUNTED HALF-CONE

Figure 15. Geometric Properties of Hypersonic Shapes *[2]*

Below is a summary of the literature research that the CAD discipline has conducted [20]:

Useful references:

- Bornemann, W. (1980). *Aerodynamic Design Data Book: Orbiter Vehicle STS-1.* Rockwell International.
- C.F. Ehrlich, F. G. (1986). *Preliminary Design and Experimental Inveistgation of the FDL-5S Unmannedd High L/D Spacecraft.* Wright-Patterson Air Force Base.

- E. Conbeer, D. C. (n.d.). Space Express: Hypersonic Aircraft Design Concept. Princeton: Princeton University.
- G.M. Gregorek, D. D. (n.d.). *The Design of Four Hypersonic Reconsaissance Aircraft.* The Ohio State University.
- Lowther, S. (n.d.). *Model 176 Art (1).* Aerospace Projects Review.
- Lowther, S. (n.d.). *Model 176 Art (2).*

Methods:

- OpenVSP
- SolidWorks

Historical Examples:

- *FDL-5*
- *Space Shuttle STS-1*

III. Business Case

A. Market Potential

The business case begins with the assessment of the global market. After reading through several documents of market research, the author believes there is global interest in sustaining a frequent launch rate, as was desired in the introduction.

The first market need the design can fulfill is rapid point-to-point transportation. The flight profile of such a service will include trans-Pacific and trans-Atlantic flight. The vehicle's ability to access this market is limited by launch sites, but one-way trips are still available from a launch site to a runway of suitable length. This required runway length is later assessed by the aerodynamic and performance disciplines.

The market forces for this mission profile are not unlike those of supersonic transport, since the starting flight must cross over the ocean for safety reasons (instead of supersonic boom noise regulations). Additionally, the passengers are expecting a short travel time. The author deems the comparison reasonable since the onboarding and offboarding procedures coupled with travel to a launch site are comparable to a longer travel time with a potential supersonic transport. The global market for high-velocity transport is shown below [21].

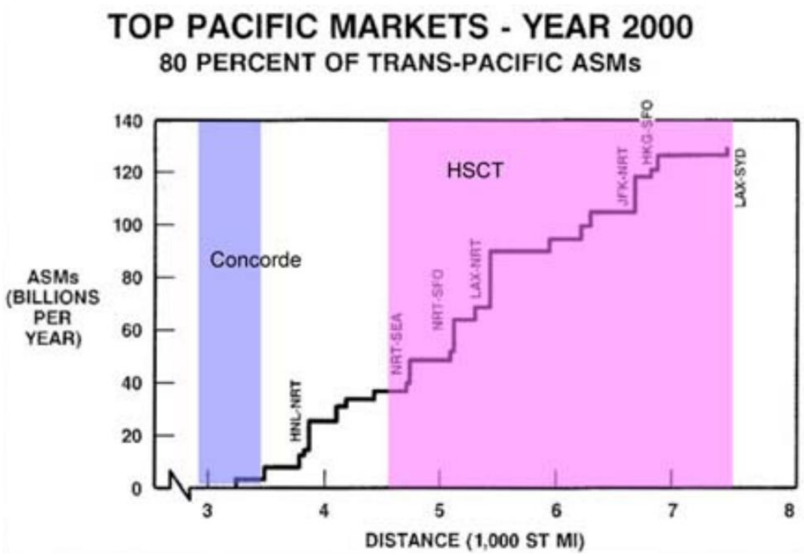


Figure 16. Global Market for High-Velocity Point-to-Point Transport *[21]*

The amount of the market which a vehicle can capture is shown as a function of range. An orbit-capable craft can capture the entire market since its cross-range has global access upon reaching orbital velocity above the atmosphere. However, the orbit-capable vehicle is limited by the available launch sites. To capture as much of the market as possible, it is planned to transport passengers by conventional subsonic means to the nearest launch site before embarking across the world. By doing this, the total time of travel is expected to approximate a supersonic or hypersonic flight, where all high-velocity aircraft will drastically reduce flight time for a premium when compared to conventional air travel available today.

In regards to market capture, currently the best available transport service for important employees and wealthy individuals is a first-class ticket aboard a conventional subsonic aircraft. By these means, travelling from Washington D.C. to Sydney, Australia will take about 22 hours of flight time, not including the time to drive to the airport and make it to the seat of the aircraft. This best available transport today will only allow for extra comfort and accommodations during the flight to allow for a better and more productive transition, but little else can be done to increase logistical productivity when transporting employees or travelling for leisure.

However, purchasing a ticket for high-speed flight will allow for a short flight duration (about an hour), minimizing passenger fatigue and increasing productivity. Important employees such as executives likely cost their employers considerable money for their time, who must be compensated for during a less-productive flight. Additionally, this salary they earn contributes little to the output of the employee during the actual transport, even if the aircraft is enabled with an internet connection. Essentially, the company would much rather have an executive at their destination rather than in-flight. These incentives from a customer company will provide a better argument for paying for the expectedly high price per ticket.

Realistically, due to weight sensitivity and volume constraints, the in-flight accommodations aboard an orbital point-to-point transport will be less than that of a fully-equipped first-class ticket on a larger subsonic vehicle. However, this can be more easily tolerated since the actual flight time is very short and comparable to a daily commute.

The market estimates for supersonic markets are conservative since orbital transport is potentially faster than a hypersonic flight (depending on the onboarding and offloading procedure). An additional conservatism is that customers are more willing to fly on a space vehicle for the cultural prestige and exhilarating experience associated with such a mode of transport. For these reasons, companies and individuals may be willing to pay a premium for tickets and the potential market for space flight is actually much larger than the market for atmospheric supersonic and hypersonic flight.

In addition to the point-to-point market, the civilian vehicle may dock with orbital space hotels. The ticket price for such a capability covers orbital propellant and increased logistics to perform a successful docking with a space station such as an orbital hotel.

Figure 17. Space Hotel Concept by the Orion Span Company *[22]*

The idea is that other third-party companies will develop the technology for a space hotel if a feasible, reliable mode of transportation were developed. To use an earth-bound analogy, conventional surface hotels are in a symbiotic relationship with the airlines that take tourists to their location, and tourists use airlines to arrive to these hotels. In the same way, a space hotel business could help further increase flight frequency, where multiple hotels will be required to keep up with the logistical capability of a rapid turn-around reusable vehicle.

There has been market research gaging public interest and willingness to spend proportions of individual salaries to go on a trip to a space hotel. The market for paying for transport to low earth orbit is larger than the author expected and allows for enough revenue to support the program proposed by Ascension Aerospace. The tabulated values of market interest are shown below [23].

Ticket Price	US Passengers/Year	World Passengers/Year	Revenue/Year ($B)
$72,000	7,500	150,000	$10.80
$24,000	137,500	550,000	$13.20
$12,000	237,500	950,000	$11.40
$6,000	600,000	2,400,000	$14.40
$2,000	1,250,000	5,000,000	$10.00

Figure 18. US and Global Demand for Orbital Transportation *[23]*

It is worth noting that these numbers are conservative given that the lack of such a service makes market estimates speculative. The source compensated for this by assuming that only 25% of positive respondents would actually purchase a ticket should a service for orbital transport appear. This is a conservatism and does not account for the increased demand after the presence of affordable orbital transport.

The data from this market demand is plotted to visualize how the ticket price will affect the number of flights per year needed to capture the entire global market and generate the potential revenue. The number of flights is based on the expected number of passengers per flight. The Model 176 can carry 16 passengers [1].

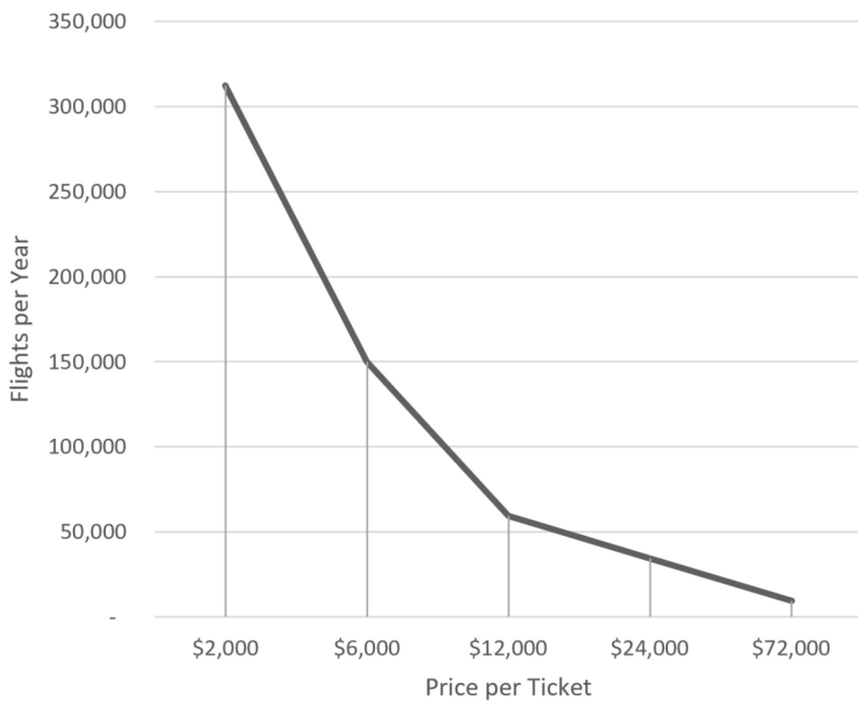


Figure 19. Global Demand Curve for LEO Transport

Ticket prices above 72,000 USD are considered too high to allow for a predictable operation and regularly scheduled flights. One of the organizational goals of Ascension Aerospace is to give the general public access to space, and any price above this maximum is deemed too infrequent by the source.

The above demand curve will aid in the development of the annual budget based on potential revenue. This is the starting point for building the civilian business case.

One of the main design missions for the vehicle is military operations. This can fall under two categories: command and control (C & C) or surprise reconnaissance. The first layout will house equipment needed for an extended stay in orbit and will be expected to make frequent, scheduled flights for long-duration military objectives, such as space presence and dominance. The opportunity for premium, reliable contracts exists, particularly given recent presidential initiatives, such as the formation of the Space Force.

The reconnaissance mission is not expected to be nearly as reliable as the civilian market or C & C needs, given that this would only be used for short duration missions for tense, time-sensitive situations. The expected situation is to have a contract with the military to purchase available vehicles whenever they are needed.

B. Business Strategy

As was mentioned in the introduction, flight frequency is the key to affordable space flight. This strategy was employed after literature research on costing and operations of a space program. An existing analogy for the competitive advantage provided by a rapid turn-around reusable vehicle is that of a roller coaster. If a roller coaster and all its infrastructure and operations were used to support one ride per week, the roller coaster would not be economically viable. The only reason any single roller coaster can stay in business is because a single ride can process several thousand passengers a year.

In the same way, the Model 176 must be used several times per day if it is to remain economically viable. This business need translates to an engineering requirement, where high cross-range is needed as well as increased design robustness for reliability. The cost analysis in the following section will go over how flight rate affects the cost from researched cost data.

C. Cost

The cost of the engineering development of this hypersonic vehicle is further explored in the Cost Analysis chapter. The operational and logistical costs associated with running the business are explored here.

The cost analysis for a general reusable vehicle is conducted by the Aerospace Corporation which explains the assumptions and methods used to arrive at the results below [23].

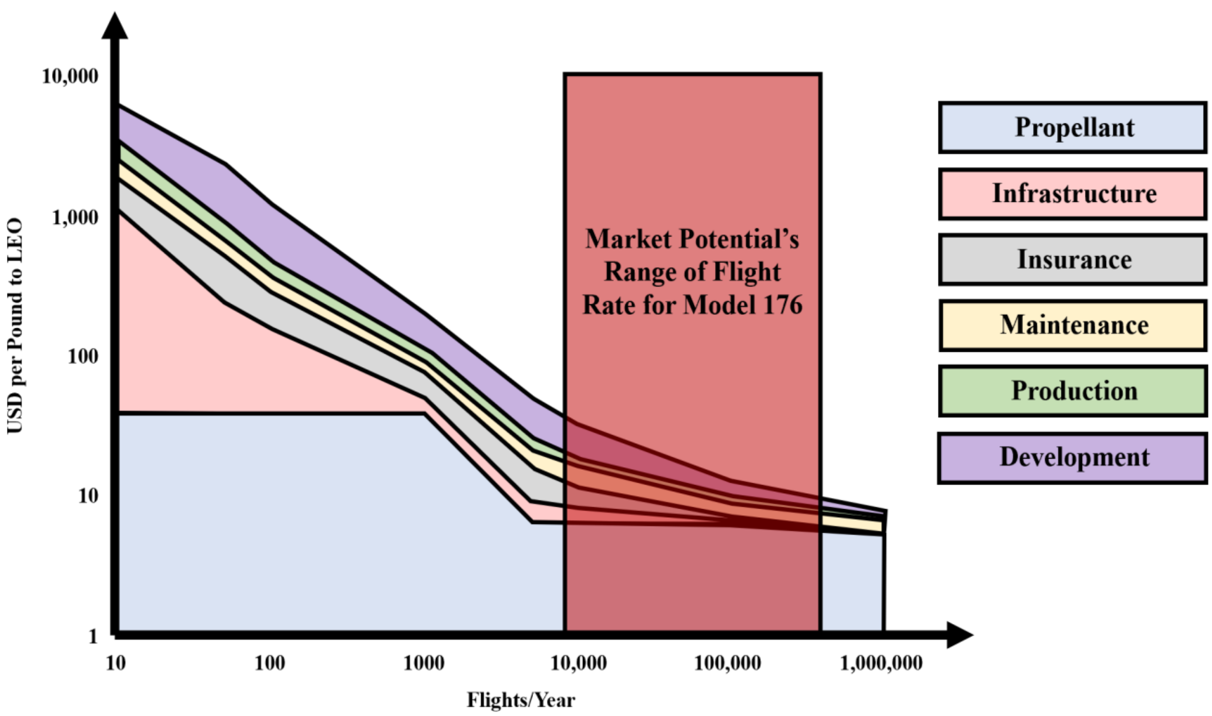


Figure 20. Cost of Payload to LEO as a function of Flight Frequency *[23]*

For the given range of flight, the propellant used for the vehicle is kerosene and oxidizer. This can be fine-tuned according to the fuels given by propulsion analysis, but since the launch system uses kerosene, it is deemed an appropriate approximation.

Infrastructure expenses are a function of the yearly flight rate. Fortunately, the infrastructure price per pound in orbit decreases with increasing flight rate since the relationship between flight rate and total infrastructure required is not linear. The cost of infrastructure is tabulated below [23]. Maintenance works in a similar way, with manhours per flight decreasing with increasing flight rate.

Flight Rate/Year	10	50	100	1,000	5,000	10,000	100,000	1,000,000
Yearly Infrastructure Cost ($M)	300	300	350	400	400	600	1,200	3,000
Cost/Flight ($)	30,000,000	6,000,000	3,500,000	400,000	80,000	60,000	12,000	3,000

Figure 21. Infrastructure Costs as a Function of Flight Rate *[23]*

The insurance cost is based on the lost value of a single vehicle, which is set at a value of 1.5 to 2 billion USD in the graph above. However, as traffic increases, the reliability of the system is expected to increase as there is more opportunity to develop a solid, safe procedure. Conventional flight today is among the safest modes of transportation, despite being immensely complex and transporting large amounts of people in one flight. This can be attributed to a high industry standard and a long history of many flights per day.

Production and development costs are initial costs which are fairly independent from the number of flights. These up-front costs are amortized among the many flights they were designed to initialize.

D. Profit

Now that the market has been assessed and the costs have been computed, the yearly profits can now be evaluated for a 16-passenger orbit-capable transport. Using the results from previous sections, the revenues and costs of a scheduled space program are shown below.

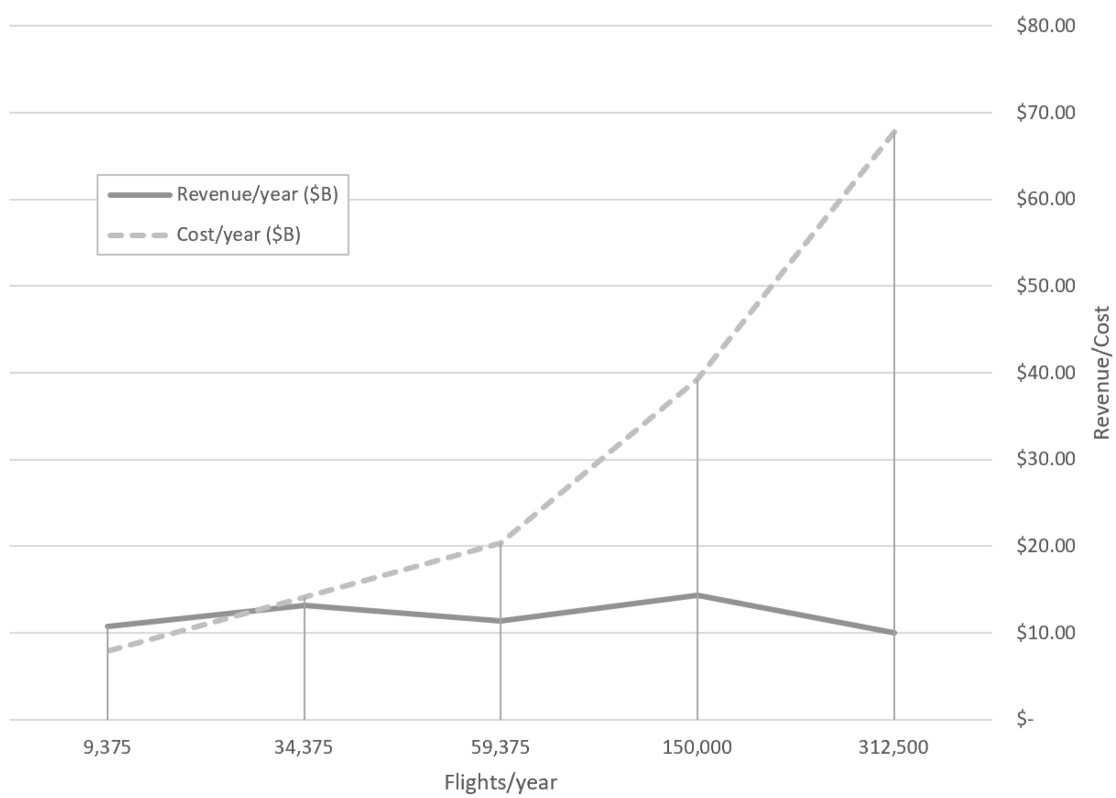


Figure 22. Revenues and Costs as a Function of Flight Rate

The business model is not profitable for higher launch rates since the total revenue does not match the increased costs of running such a large operation. While the price per payload pound to LEO is cheaper for such high flight rates, the global market is not large enough to pay for it. As previously mentioned, if the flight rate were decreased by increasing the ticket price, there comes a point where the demand is too low for regularly scheduled flights, which would not utilize the vehicle's high turn-rate capability.

When the flight rate is decided, the ticket price can be determined and the target market within the global market can be determined and catered to. The plot below shows that the flight rate must be kept at a minimum to maximize profits while maintaining regularly scheduled flights with the Model 176. The market size is what constrains the number of flights per year. If the conservative estimates under-represent the true market size (this will shift the revenue curve upwards), the operations can always be scaled up to maximize profits.

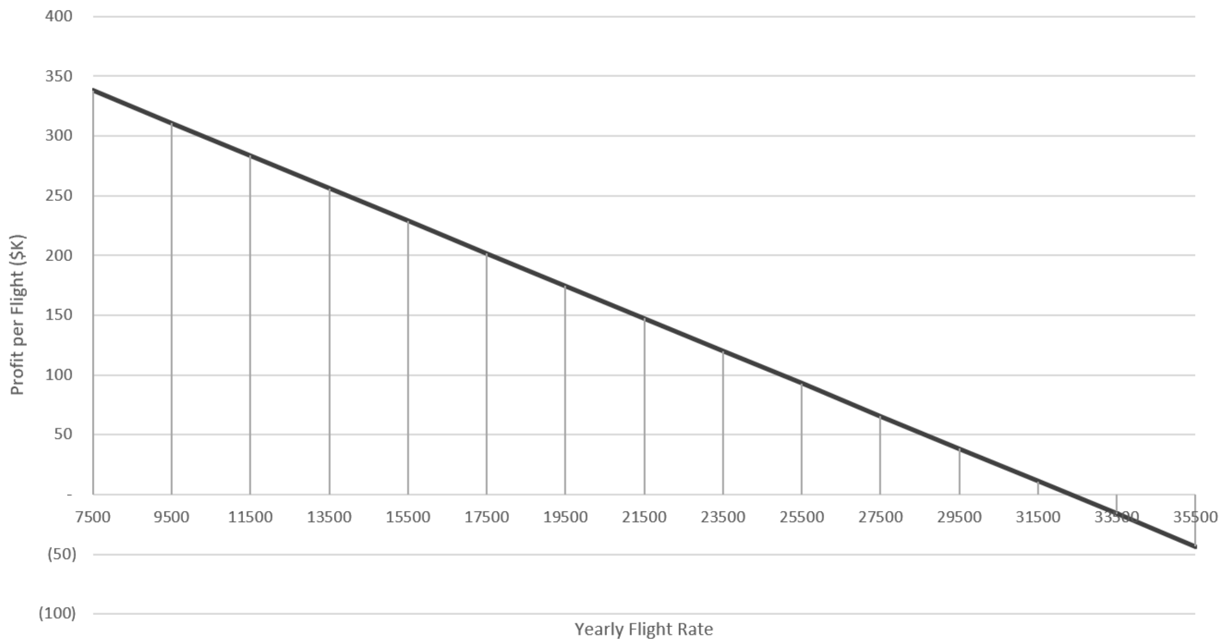


Figure 23. Profit per Flight based on Flight Rate

The cross-over point for a profitable operation is at around 32,000 flights per year. Again, this cross-over point will shift upwards as the market demand increases.

E. Selected Scale

The selected flight rate is chosen to be 8,760 per year since this corresponds to one flight per hour year-round. This is also close to the lower limit of regularly schedules flight rates for a vehicle of this size. The price per ticket can be determined as an extrapolation of the market data and is found to be 73,200 USD. By extrapolating the demand curve, the decrease number of flights allows the operation to slightly increase the ticket price. This is helpful since a decreased flight rate also increases the price per flight, as shown in previous sections.

The expected turn-around time is dependent on the size of the fleet. If the turn-around time for these vehicles were once a day (which is the optimistic limit), then a fleet of 25 vehicles can be maintain operations. At the conservative estimated turn-around time of 48 hours, a fleet of 50 vehicles is required to meet market demand. The extra vehicles are to allow slack for maintenance and emergencies. It is quite possible that the initial flight rate is not attainable due to constant learning and troubleshooting at the early stages of operation.

The fleet will also include military variants paid for by the Department of Defense, but this cannot be predicted or scheduled in the same way as market demands. The monetary value of these contracts will need to be assessed separately. However, the civilian supported infrastructure will allow for a lower operational cost for these special vehicles.

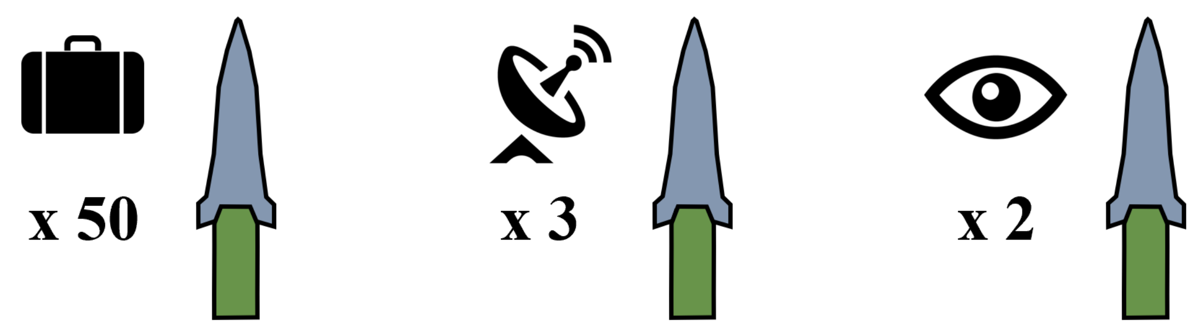


Figure 24. Expected Fleet Size by Layout (Shown Left to Right): Civilian PtP, C&C, and Reconnaissance

F. Operations

Regular operation will include launching the vehicle from one site to another, which will require an up-front cost in infrastructure investment. The sites will be placed strategically around the globe to capture as much of the global market as possible. As previously mentioned, one-way trips allow for more flexibility since such a mission profile only requires a runway at the end of the flight. However, transporting the upper stage back to a launch site may take more resources and turn-around time.

While launching from site to site allows for quick recuperation, it is worth noting that half of the trips on the way back will be going against the Earth's rotation. This will increase the delta V loads on the launch system. This is not expected to be a design concern, however, since the launch system is designed to carry the reconnaissance layout. Additionally, the way back can still access orbit the conventional way (flying in the same direction as the Earth's rotation) and reach any destination from here. This will be a slightly longer flight since the vehicle is going the long way around, but orbital speeds are high enough to make this slight increase in distance trivial in contribution to the total flight time. The direction of flight depends on the surface geography as the vehicle cannot launch over land due to safety concerns.

The potential launch sites are designed to capture the global market, where many are existing today. Acquiring the permissions required to launch to and from foreign nations is expected to be a lengthy process, and may limit the market to trans-Atlantic, Australian and Japanese flights. It may be easier to build a new retro-grade launch site (launched against the Earth's orbit) in Ireland.

There is also a dearth of launch sites closer to the South Pole. Their associated high-inclination orbits can be accessed by the high cross-range capability of the vehicle or synergetic maneuvers. In the same way, vehicles launched from here can access other parts of the globe. Unfortunately, the inner parts of the continents are not available for access due to safety concerns of a potential failed launch. The Russian launch site is an exception since the land below the launch path (launching easterly) is sparsely populated.

Figure 25. Existing Launch Sites and Planned Launch Sites for Point-to-Point Transport

Using a Russian launch site may prove to be difficult due to geopolitical reasons. The idea of orbit-capable vehicles (potential missiles) being launched from the Russian landmass to the United States or vice versa is not something either nation is likely to tolerate. China does not have available civilian launch sites for this reason.

Military operations are likely to be less frequent (especially if Command and Control is staying in space for an extended period of time), but they will be supported by the civilian infrastructure. The pricing will be higher than a civilian variant, since the development amortization costs are much higher for the military fleet size.

For the reconnaissance variant, the expectation is that the mission will be unexpected and of short duration. This is to allow the military to respond to a fast-paced situation and will likely use a scheduled flight as the fastest means to take-off. The civilian flight will be cancelled, but the passengers will be compensated for the delay (since there are 24 flights per day, this shouldn't be very long). The military will have a contract to cover the development and deployment of a unique variant on such short notice.

G. Competition Analysis

A direct competitor to the point-to-point variant is the airline industry. This currently captures the majority of the human transportation market for global coverage. It is the fastest mode of transportation available for business travelers and tourists, and it is the market Ascension Aerospace intends to disrupt.

The current going rate for a round-trip first-class ticket is close to 10,000 USD, and takes about a day per flight, with one stop. The Point-to-Point variant costs 73,200 USD for a one-way, non-stop flight and will take about an hour. The price of the point-to-point ticket is a major disadvantage, but it should be noted that the time of flight also costs money to business travelers. For a highly compensated executive, the difference in travel time may be worth the increased ticket price. If the difference between flight times is 20 hours, then the value obtained from not having an executive travelling can be worth the extra expense.

In 2013, the median chief executive salary was an annual 11.4 million USD [24]. If a 60-hour work-week is assumed for executives, then money lost in productivity for a 20-hour difference in travel time can easily be determined. The value of that time is 76,000 USD, which is a little bit more than the price of a ticket and easily covers the difference in price for a one-way trip (68,000 USD). Considering the round-trip doubles these differences. This is a very rough estimate, as the compensation of an executive is largely determined by returns and aren't exactly paid by hour. However, time is very valuable with that level of compensation, as executives are expected to produce more than that value for their shareholders per year. The value of an executive's time may fluctuate according to the state of his or her company, but these median values give a rough idea of how a faster travel time may have monetary value that allows for orbital ticket purchases. It is worth noting that if the executive can be productive on an airliner, much of the difference in value with respect to travel time can be reduced. However, there will always be more value in appearing in person for the reason of travel than conducting business on an aircraft. There are also emotional benefits to a reduced travel time.

This comparison does not include the excitement and perk of travelling to space while working, of which an adventurous executive may be willing to help pay for. The market analysis in the first section of this chapter should cover a lot of public enthusiasm for leisure travelling as well.

SpaceX is offering a service most similar to the point-to-point variant. Since the service is mostly under proprietary development, the following analysis is deduced from the author's speculations and SpaceX's promotional video [25], where the hope is that a rational assessment of both of the vehicle's capabilities is considered sufficient supporting evidence.

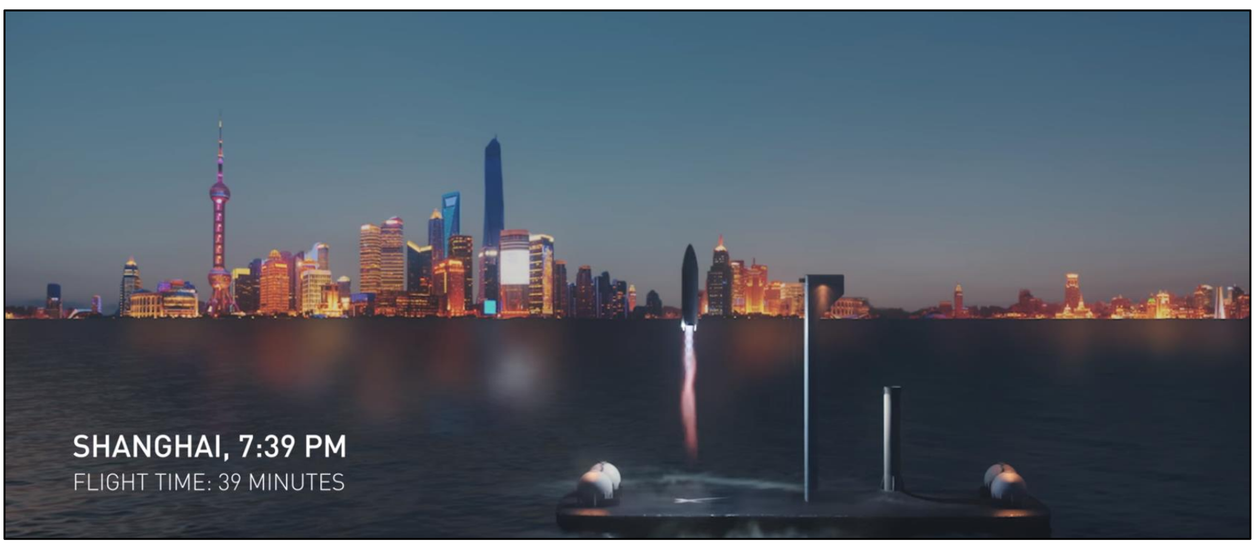


Figure 26. SpaceX Earth-to-Earth Transport via Big Falcon Rocket Promotional Video *[25]*

The ballistic trajectories used by SpaceX's BFR (Big Falcon Rocket) are unlikely to be considered safe to the destination cities. Regional government approval for a vehicle approaching their coastlines is difficult to obtain. They must also land on specialized launch pads, which passengers may not deem as safe or comfortable as a runway landing. These launch pads are accessed by a sea vehicle (adding to travel time). Clearly, landing as a conventional aircraft increases the destination flexibility and its receptibility (can travel further inland). It is acknowledged that the passengers for the Point-to-Point variant proposed by Ascension Aerospace are supposed to travel to the launch site, but since this site is based on land, it is accessible by a commuter airline or ground transportation.

Blue Origin is not proposing to build a point-to-point transportation system around their vehicle (New Shephard), but they do share the market for space tourism. The company is making progress towards man-rating the vehicle and both the first stage and capsule is recoverable. However, the performance of this vehicle is significantly lower than the Model 176, particularly in its range capacity. This vehicle will also not have access to an orbital space hotel, though the New Glenn will (still in development). Each flight will take 6 tourists, whereas the Model 176 will seat 16.

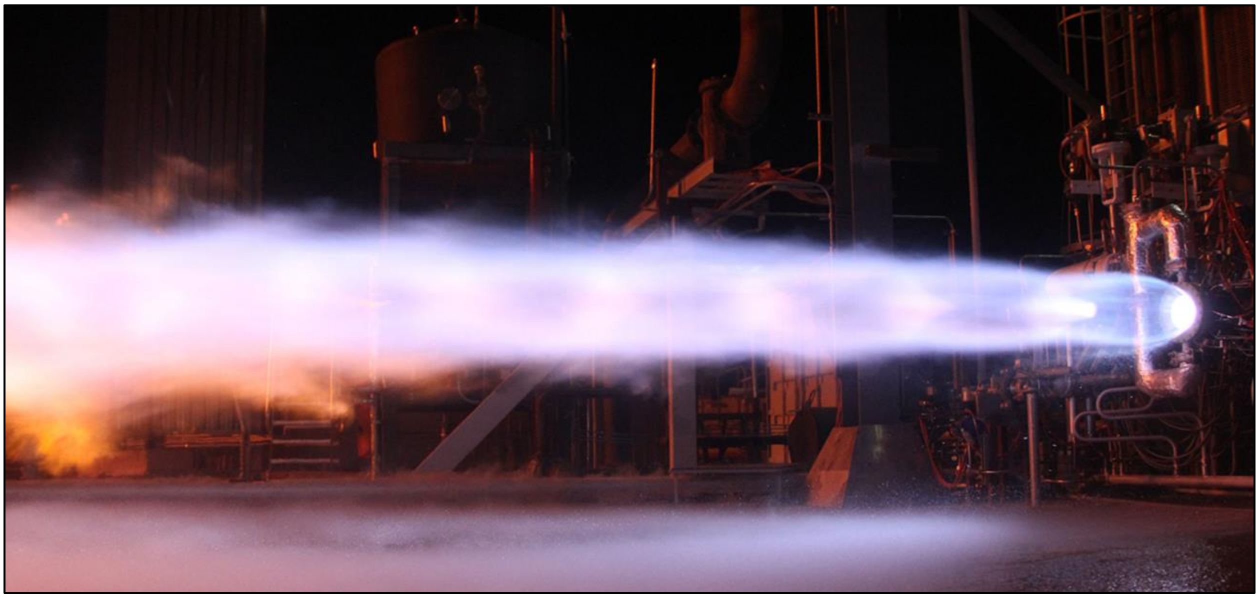


Figure 27. Blue Origin Test Firing their BE-4 Engine for the New Shephard *[26]*

IV. Mission

A. Key Mission Design Parameters

In Dale Reed's Lifting Body history book, the cross-range advantages of different re-entry vehicles (with different aerodynamic properties such as lift-to-drag ratio and slenderness) are shown to reveal the "race-horses" of lifting bodies. There is an obvious strong positive correlation. This relationship is analogous to the lift-to-drag ratio found in the atmospheric Breguet Range equation. However, the range is greatly increased and more sensitive to lift-to-drag ratios when the trajectory is combined with orbital velocities, where the curvature of the Earth reduces the lift required to stay afloat.

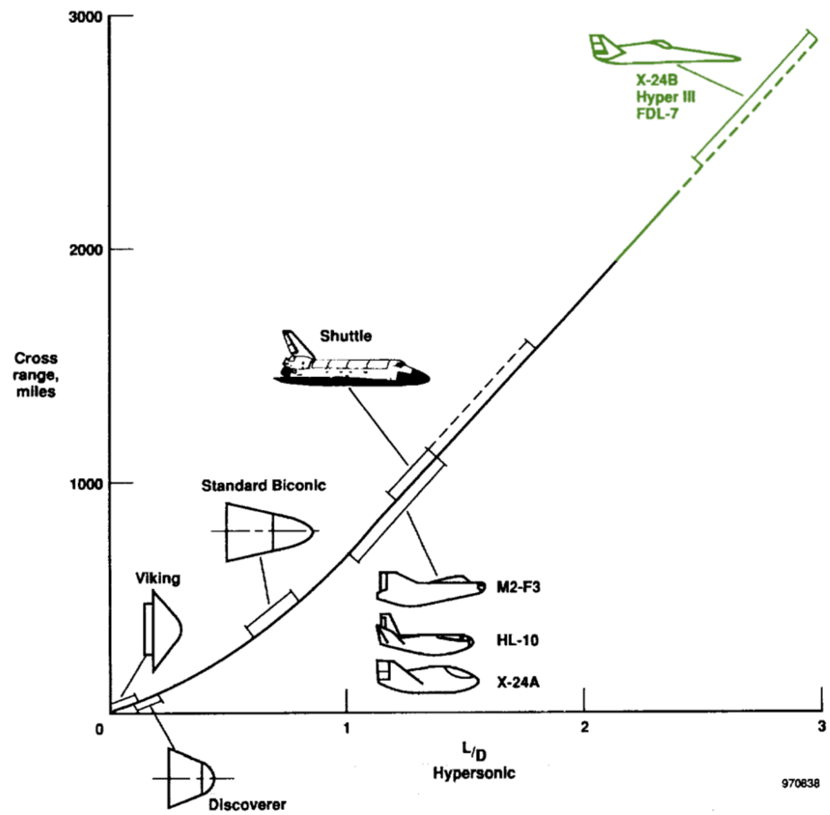


Figure 28. Lifting Body Race-Horses Defined by Cross-Range Capability *[27]*

This correlation can be found to be defined as a function of cross-range in the Spacecraft Propulsion Integration book [2]:

$$\text{Lateral Range [nmi]} = \text{LR} = 1.667 + 68.016\left(\frac{L}{D}\right) + 706.67\left(\frac{L}{D}\right)^2 - 91.111\left(\frac{L}{D}\right)^3$$

$$\text{Down Range [nmi]} = \text{DR} = 4866.6 + 4.70417(LR)$$

These equations found in ref. [2] are plotted in the design script. It is useful to compare this plot with the figure above to find the region for which these trend equations are valid. This comparison script can be expanded to calculate the required lift-to-drag for a given down-range coverage percentage. Below is a plot of the lift-to-drag that a 30% circumferential coverage would require.

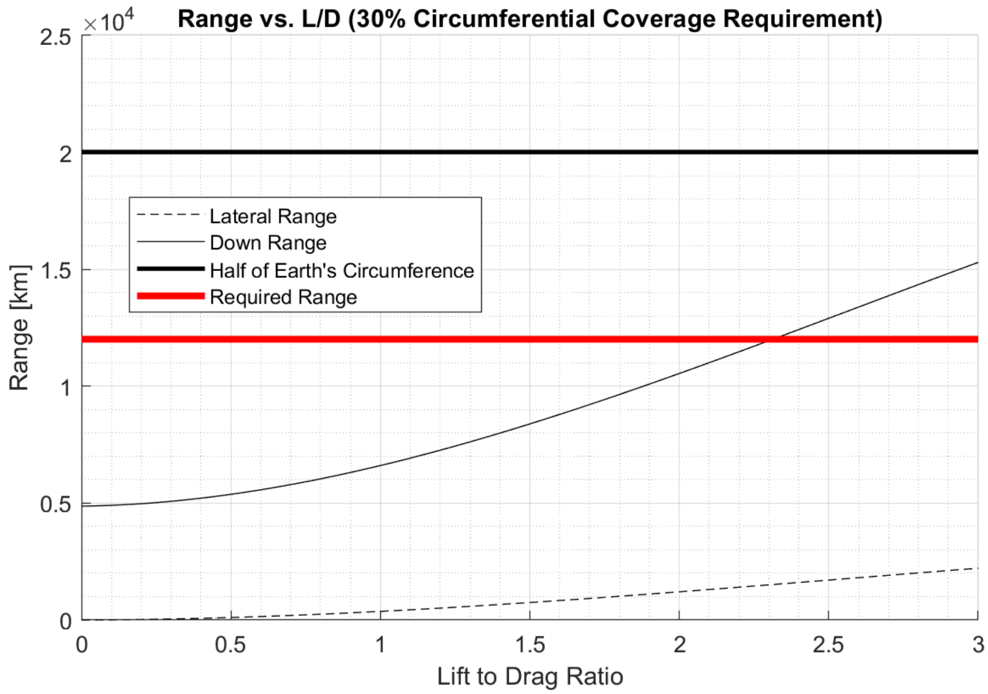


Figure 29. Lift-to-drag Requirement for 30% Circumferential Coverage

While these lift-to-drag characteristics are exhibited in hypersonic flight, the aircraft still needs a different geometry which provides a low-enough wing-loading for safe runway landing. A demonstrator is shown below [27]. This vehicle was unpiloted and is different from the Model 176 in its deployment method. This can be a good trade study for systems, structures, and aerodynamics. There must be a design choice between switch blade wings and single piece pivot wings.

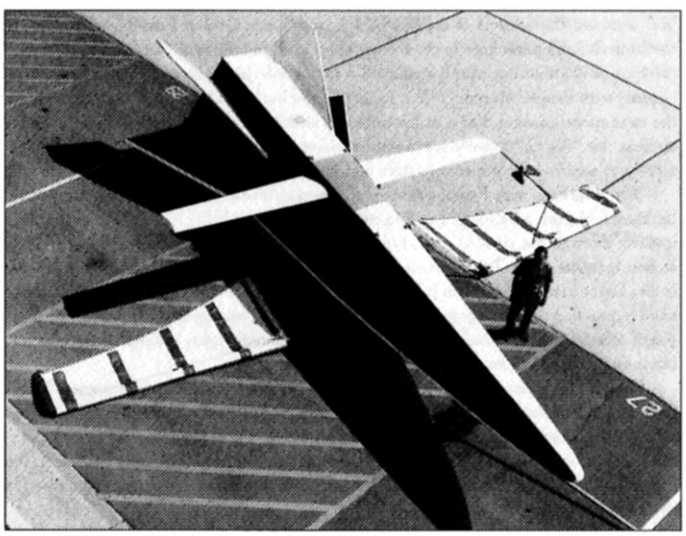
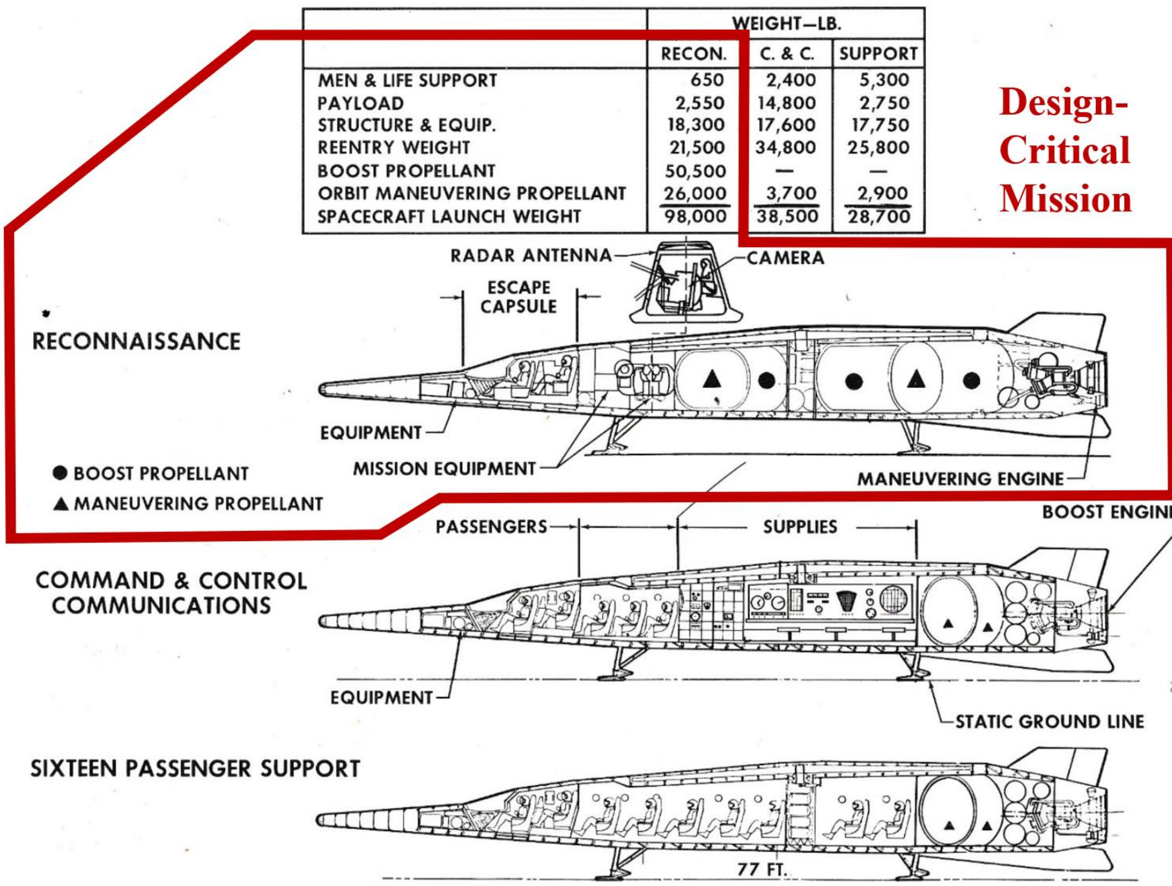


Figure 30. Hyper III Demonstrator with Single-Piece Pivot Wing Installed *[27]*

One of the main historical sources for this design project is the Model 176. Many declassified specifications make the initial weights and geometries known. This allows the launch and hypersonic vehicle teams to work independently. The starting document also gives the missions the vehicle was designed for. The reconnaissance mission is considered the design-critical mission from which the two other mission alternatives can be built upon. In this way, the same design can be minimally reconfigured to fit multiple roles in military operations. A section of the project introduction document is shown below, with the reconnaissance portion outlined in red. The first methodology will be built around accomplishing this mission.



	WEIGHT—LB.		
	RECON.	C. & C.	SUPPORT
MEN & LIFE SUPPORT	650	2,400	5,300
PAYLOAD	2,550	14,800	2,750
STRUCTURE & EQUIP.	18,300	17,600	17,750
REENTRY WEIGHT	21,500	34,800	25,800
BOOST PROPELLANT	50,500	—	—
ORBIT MANEUVERING PROPELLANT	26,000	3,700	2,900
SPACECRAFT LAUNCH WEIGHT	98,000	38,500	28,700

Figure 31. Mission Descriptions of the Model 176 (Critical Mission Highlighted) *[1]*

The model 176 had two in-flight configurations, featuring a switchblade wing to decrease wing-loading to allow for slower, safer landing speeds. An objective of aerodynamics will now be to determine the necessity of such a device and explain why the vehicle would have failed mission requirements without it.

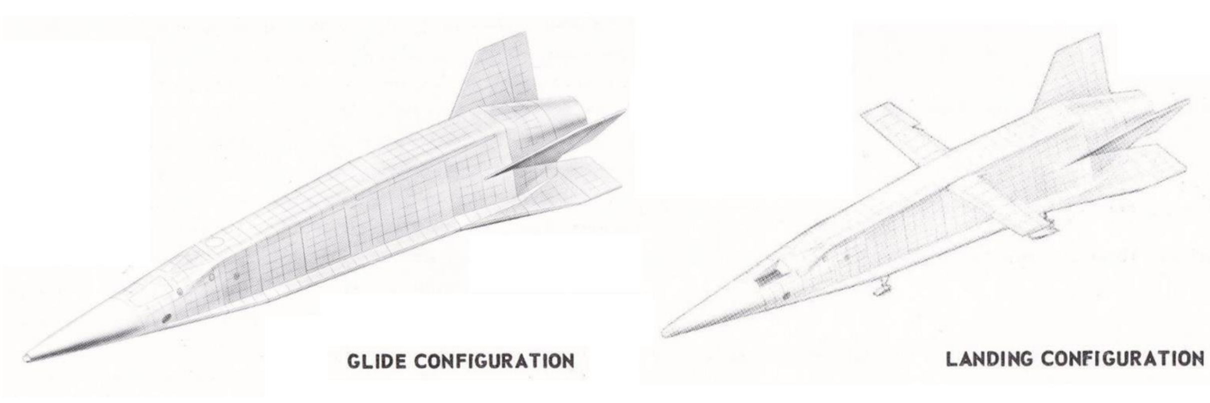


Figure 32. Both of the In-flight Configurations of the Model 176 *[1]*

B. Point-to-Point Transport

The point-to-point variant will be the most common layout, as it will support the civilian market. The layout, as shown in the previous section, includes 16 passenger seats. This variant is the lightest of the three, and rightly so, as these missions will generate the least revenue in lower ticket prices to capture as much of the market as possible.

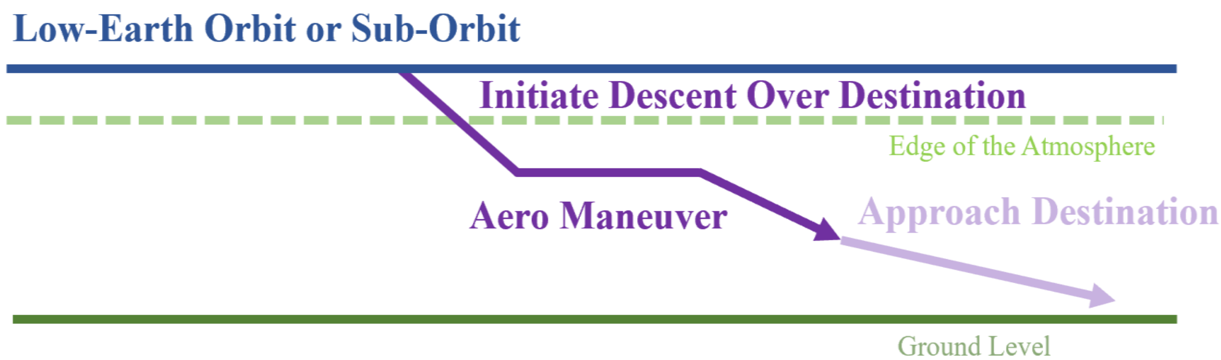


Figure 33. Point-to-Point Variant Upper Stage Mission Profile

During descent, the vehicle may initiate an aerodynamic maneuver to increase its lateral ground-track. Doing this will allow the craft access to more airports outside of its direct orbital path. Much of the purpose of the point-to-point mission was described in the business case, where the main idea is to have as much access to as many destinations as possible to create the demand to support a sustained flight rate, lowering costs. These lowered costs allow for greater flight rate, and so on and so forth. This translates into a mission parameter in the sense that the craft must have a large cross-range and lateral range, which is dependent on aerodynamic characteristics (see previous section in this chapter).

Additionally, the sustained flight rate is made cheaper by reducing the number of vehicles in the fleet. Keeping the fleet size at 50 (as described in the business case) requires a turn-around time. In this sense, this vehicle has a special economic advantage because of its reusability and rapid turn-around time. A comparison of current space-launch operations with what they could be by reducing payload costs is shown below. This is the predicted market-change result from an increased flight-frequency by increasing cross-range capability [2].

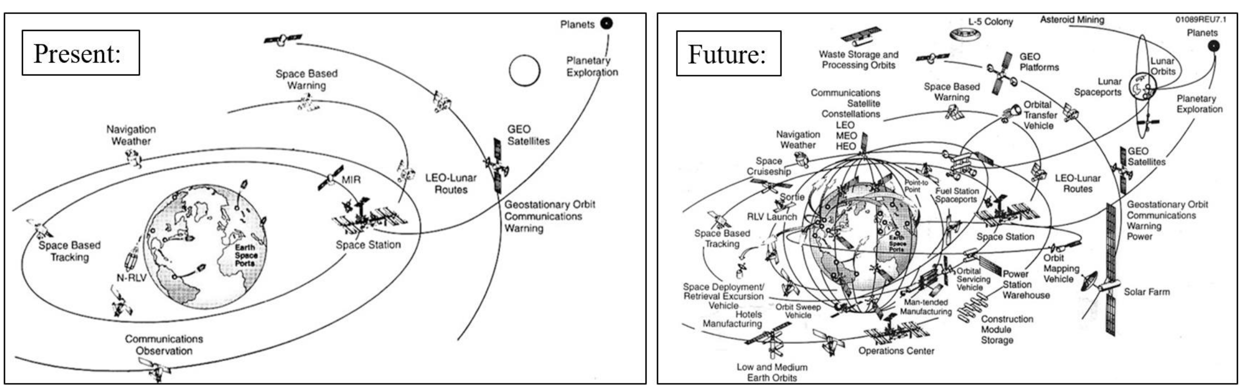


Figure 34. Space Activity Before and After Reduced Launch Costs from Operational Flexibility *[2]*

The effects of cross-range capability on orbital wait time are shown below. For a given inclination, the runway must be perfectly aligned with the orbital path if there is bad cross-range capability. A reduction in wait time greatly reduces operational costs. If a vehicle could go in and out and be rapidly recovered, the logistical and business benefits to run such an operation are obvious.

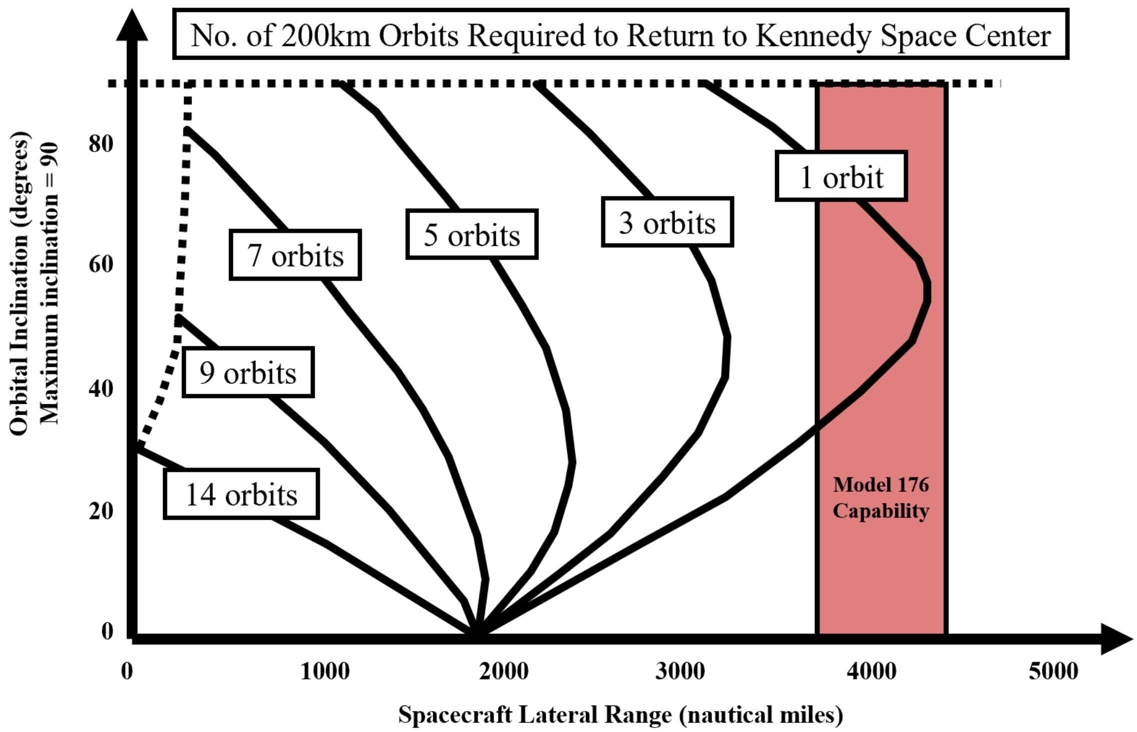


Figure 35. Lateral Range Capability and Inclination's Effects on Orbital Wait Time *[2]*

The Model 176 has a lateral range capability which allows for a turn-around time within the orbit. This is required if the vehicle will be ready the next day. The overall mission profile of the point-to-point variant is shown in Figure 36. This profile can provide access to low-earth orbit as well as one-way global coverage of destinations.

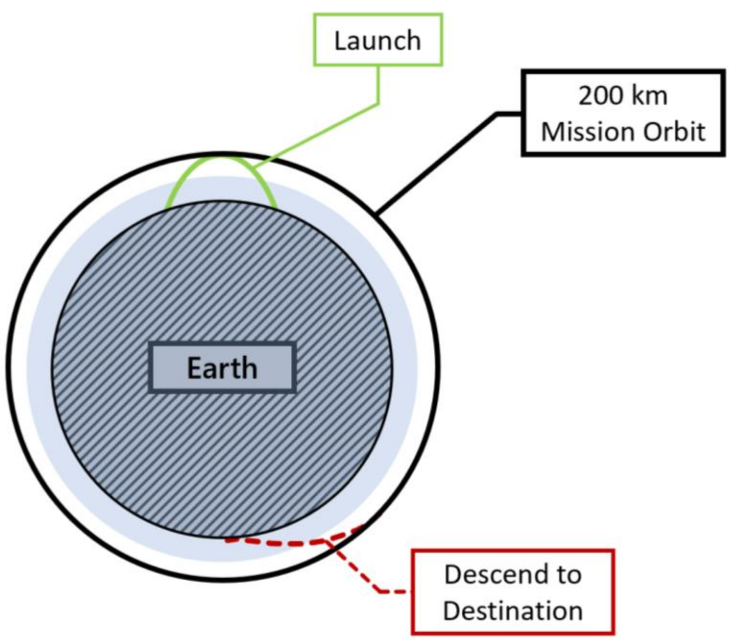


Figure 36. Point-to-Point Variant Overall Mission Profile

C. Command and Control

The Command-and-Control variant is an orbit-capable craft designed to conduct military operations independently in space. This can include satellite repair, bringing supplies and equipment to orbit, or coordinating space activities with human presence in low-Earth orbit. Whatever the operation may be, this variant is designed with a higher payload capability for life-support to sustain up to 5 astronauts. Additionally, the craft retains the capability to land safely on continental U.S. soil at any time.

The life support requirements for this variant are found below. In the weight equation, "lifesupport()" is a MATLAB function (found in the appendix) developed by the author and CAD team to determine the amount of food, water, and air a crew will need for a given mission duration. This was found in Corning's Aerospace Vehicle Design book [12].

$$VOL_{crew} = 1.247 N_{crew}^{0.136} N_{days}^{0.150} = 1.247(5)^{0.136}(7)^{0.150} = 2.08\, m^3$$

$$W_{pay} - W_{equipment} = W_{crewsupport} = lifesupport(N_{crew}, N_{days}) = lifesupport(2,3) = 831\, kg$$

A simplified diagram of the mission profile is shown below. If the fact that this variant is designed for a week of operation with extra payload is ignored, this is profile is similar to the other two minus their respective added capabilities.

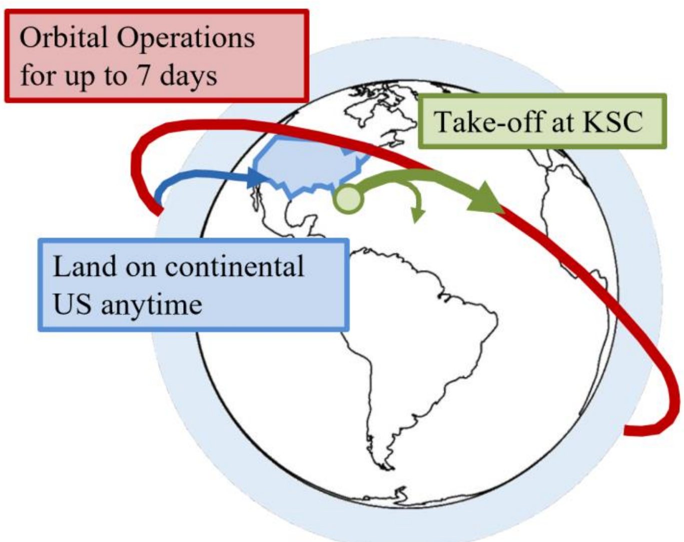


Figure 37. Command-and-Control Variant Mission Profile

The overall mission profile for the command and control variant is shown in Figure 38. This represents a need to maintain a 600 km orbit to reduce the ground-track velocity. The spacecraft will inject into a 200 km orbit as a requirement for the launch vehicle. From here, the orbital propulsion capabilities of the Model 176 will conduct a Hohmann transfer to achieve the desired orbit altitude. This altitude will be maintained for up to a week according to life support requirements, where, as demonstrated by Figure 37, the spacecraft must be capable of landing in the continental United States at any point in time. This capability is determined by the lateral range, as shown in Figure 35.

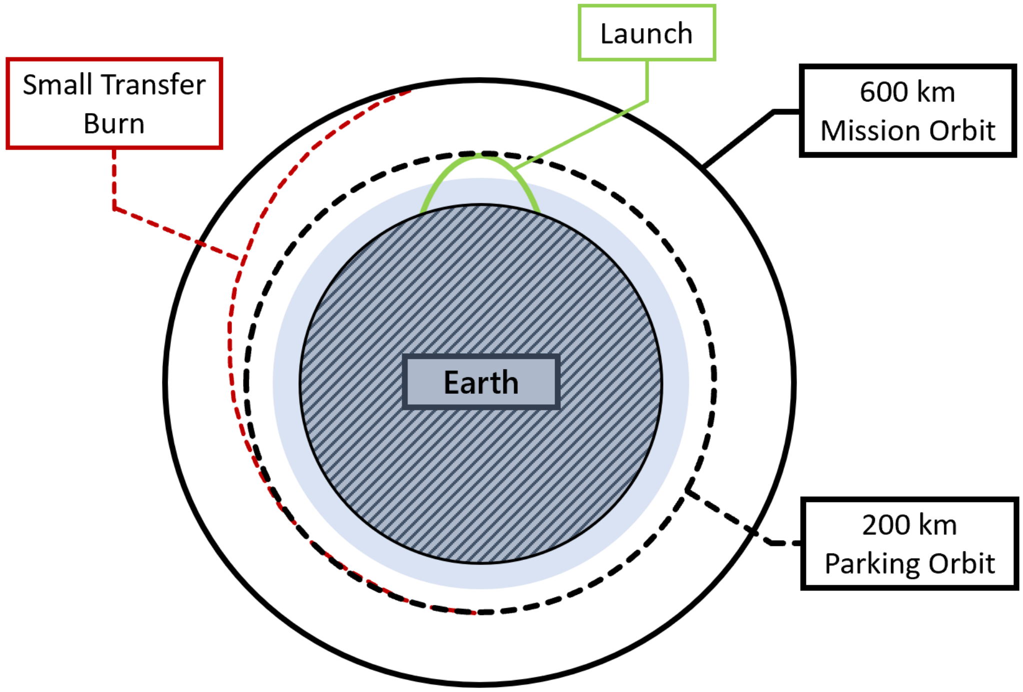


Figure 38. Command and Control Variant Overall Mission Profile

D. Reconnaissance

The design-critical mission takes advantage of the lifting qualities of a hypersonic lifting body in another way. The orbiting stage can use its engine to descend into the atmosphere and make a turn by using the lift to perturb the orbit so that it can change inclination. This new orbit allows for the surprise reconnaissance of any location on earth while using minimal fuel for a trajectory change. The only fuel required is used to decelerate to descend into the upper atmosphere and then to increase speed back to orbital velocity. Minor orbital maneuvering can be used to alter inclination slightly or rendezvous with another body in Low-earth Orbit (LEO). A diagram of this maneuver is shown below.

By adding this maneuver, the reconnaissance variant experiences the design-critical mission profile which is most demanding on the vehicle sizing in terms of fuel and volume required. The mission profile for the upper stage can be summarized by the figure below.

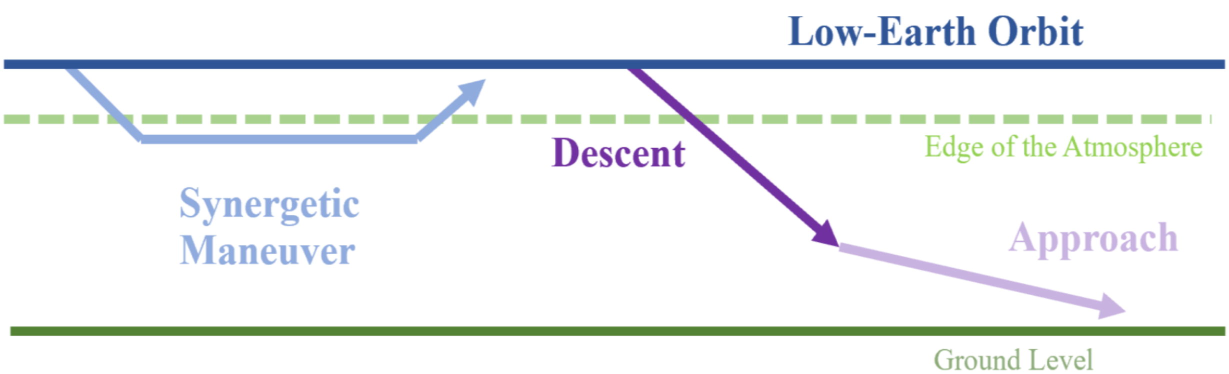


Figure 39. Reconnaissance Variant Upper Stage Mission Profile

The maneuver requires three burns (calculated as four velocity increments): descent, boost, and re-circularization. The descent and re-circularization should be the same velocity increment if the initial and final orbit differ only by inclination. The boost burn is calculated as two velocity increments: one will add the velocity lost during the maneuver (function of lift-to-drag ratio) and the other is the increment required to accelerate back up to the new orbit. Vehicle parameters such as the lift coefficient, weight, and lift-to-drag determine the altitude of turn, the length of turn, and the achievable inclination change with a given delta V for a vehicle are determined inside a NASA report detailing Nyland's analytical methods [28].

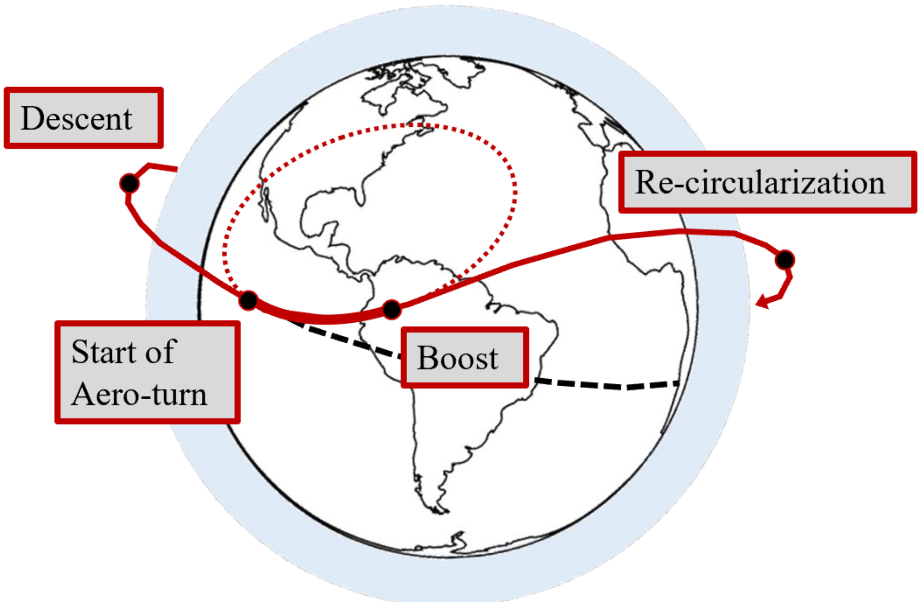


Figure 40. Diagram of Synergetic Maneuver for Reconnaissance Mission

The combination of this upper stage with the launch system provides the full mission profile. There is a delta V requirement at launch to insert itself into orbit, as the historical Titan III launch system was not able to carry the upper stage. This is the heaviest variant, since it needs to carry fuel for the burns shown in the figure above.

Synthesis has laid out a diagram of the critical mission combined with the launch vehicle below [29].

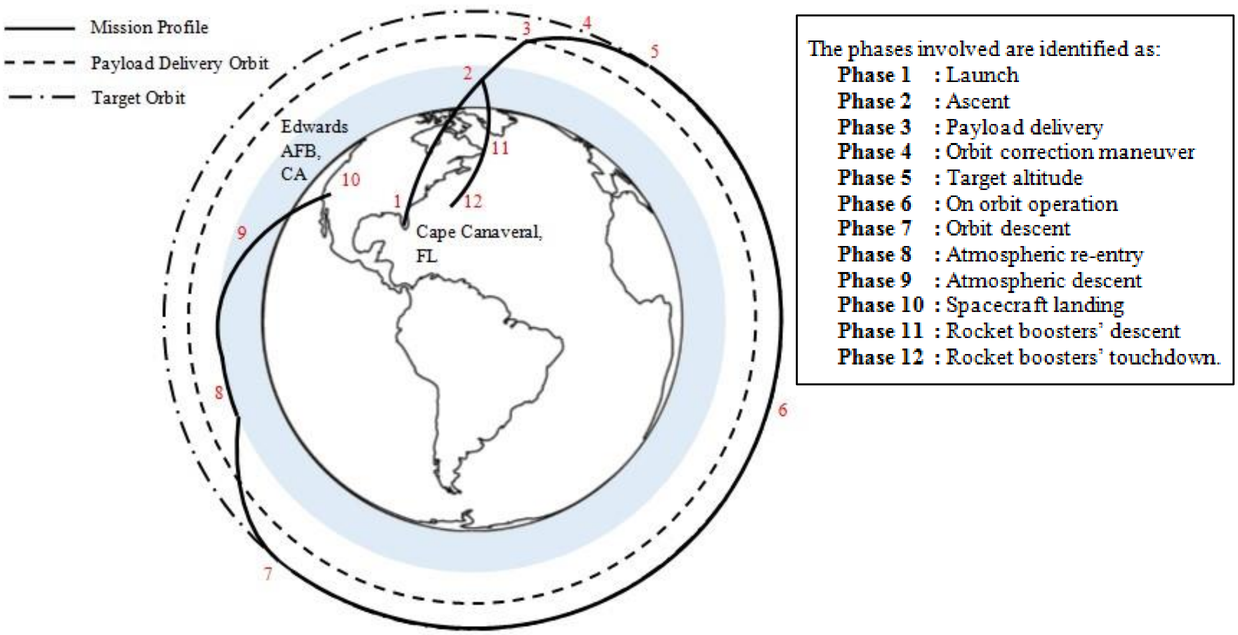


Figure 41. Critical Mission Sketch *[29]*, *[30]*

In order to ensure full Russian coverage at an inclination of 70 degrees, it was determined that the launch vehicle should place the hypersonic vehicle at an inclination of 40 degrees. At this low inclination parking orbit, the hypersonic vehicle cannot be easily seen. When it is time to begin the reconnaissance pass, the hypersonic vehicle is required to complete a 30-degree inclination change by means of aerodynamic maneuvering.

The number of crew required for such an operation is three. For a vehicle to remain in orbit for three days, the volume of the payload can be defined by a trendline from various studies on human tolerances [31]:

$$VOL_{crew} = 1.247 N_{crew}^{0.136} N_{days}^{0.150} = 1.247(2)^{0.136}(3)^{0.150} = 1.62\ m^3$$

The number of crew members for a given mission time will also affect the amount of life support carried. The CAD discipline wrote a small MATLAB function (found in the appendix) to tabulate how much of a weight contribution this is to the mission payload. The mission payload also includes the equipment for the reconnaissance mission. The methods are found in the appendix, but the called function appears as the following:

$$W_{pay} - W_{equipment} = W_{crewsupport} = lifesupport(N_{crew}, N_{days}) = lifesupport(2, 3) = 248\ kg$$

This is in line with the historical value of 266 kg for this mission's crew and their life support.

A technical summary for the critical mission profile is listed below:

- Logistics:
  - Manned
  - Reusable
  - Reconfigurable
  - Adapted to two SpaceX launch vehicles
- Performance:
  - Design Aerodynamic Speed: Mach 7 (Possibly Mach 12 – 15)
  - Ability to sustain Mach 0 – 25
  - Atmospheric Endurance: 30 – 45 min
- Operations:

- Aerodynamically Assisted Inclination Change
- Orbit Capable
- Orbital Endurance: 3 days
- Max Payload: 7000 kg

Specifically, for the hypersonic stage, the vehicle state upon staging with the launch team has been determined in the class script. This work domain is iterated upon in light of the deliverables from propulsion and performance. Increasing the performance demands on the hypersonic stage decreases those demands of the launch vehicle, and vice versa. It is imperative that the weight-critical mission converge on a good staging point along the combined vehicle's trajectory.

The general mission profile for the Reconnaissance variant is shown in Figure 42, including the transfer burn to a slower ground-track, higher altitude orbit to increase proficiency in Reconnaissance. After launch and during orbital operations, the spacecraft can perform its synergetic maneuver to change its inclination.

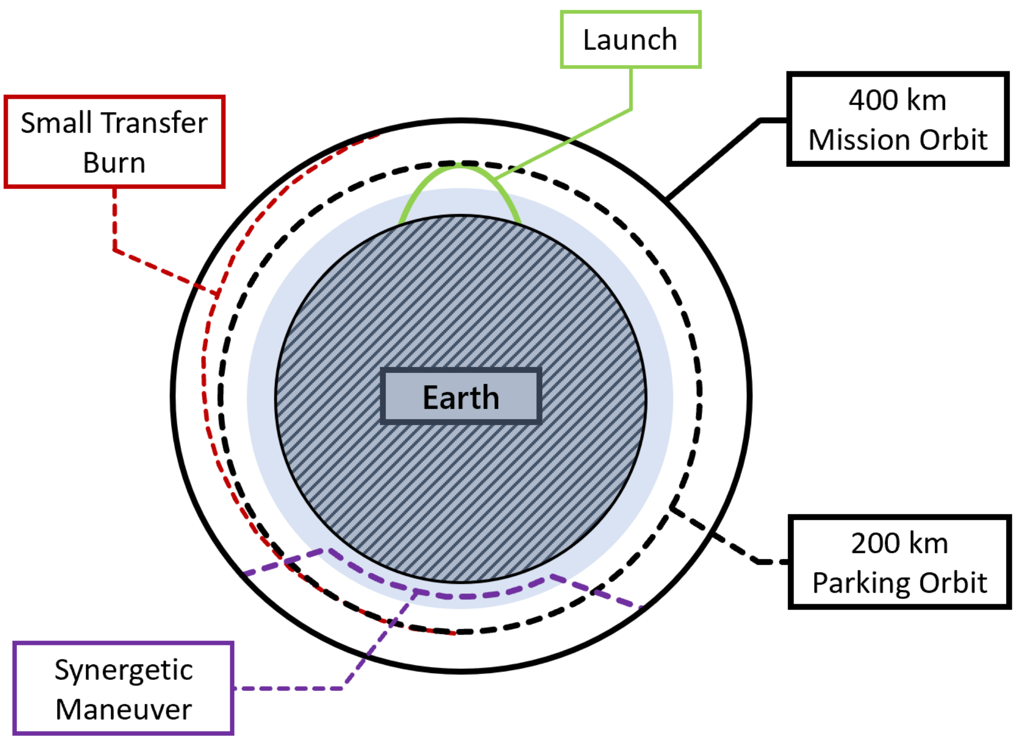


Figure 42. Reconnaissance Variant Overall Mission Profile

E. Mission Trade Studies

The mission trades begin with the launch site minimum inclination, which is the inclination which is most advantageous for visibility from the surface (therefore the stealthiest parking orbit) as well as velocity change required from the launch vehicle. However, this reduces the potential coverage by any incidence angle changes made by the hypersonic vehicle and increases performance demands. For example, a low-inclination starting orbit will be easier on the launch vehicle but will require two incidence angle turns (at the 30-degree requirement) to effect global reconnaissance coverage. This is advantageous if global coverage is required but will not cover the entirety of Russia. Moreover, covering such high latitudes is not expected to be significant given the low amounts of human activity at the poles. Even if the 2nd turn is unlikely to be initiated, a one-turn version of this mission can be utilized for China, Europe, and Southern Russia (covering most of the population centers as well as the main seaports).

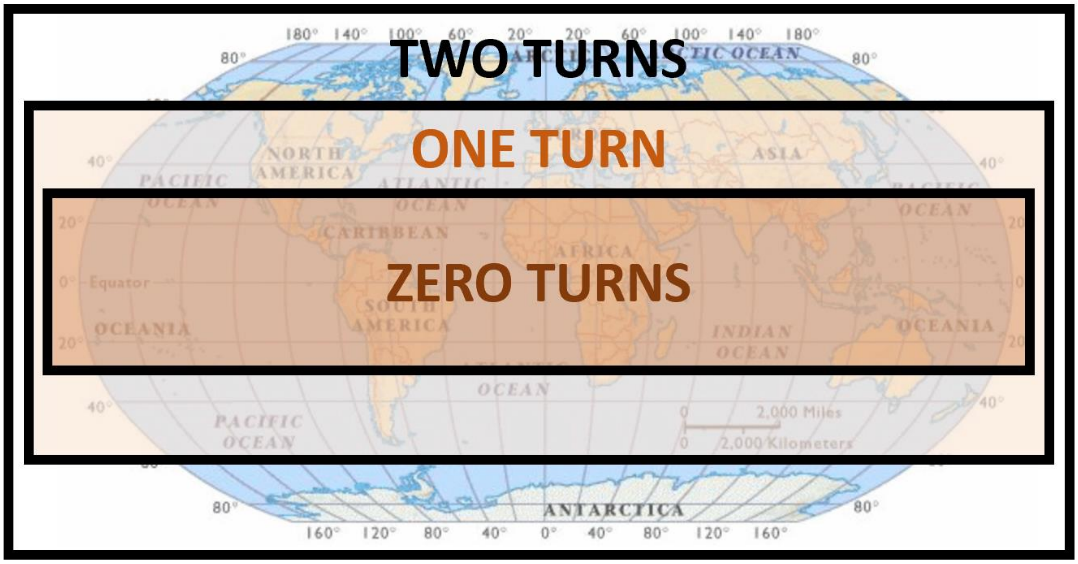


Figure 43. Two-turn Mission with Full Global Coverage

This mission type is useful for civilian transport, as launch costs can be reduced by going into an "easy" low-inclination orbit, then using a turn to arrive at a population center. This is using the cross-range capability of the vehicle to the fullest advantage.

Another possible mission requires one turn from a higher-inclination orbit to allow for full coverage of Russia. This is considered a critical mission for the team as it requires a vehicle aerodynamic turn and higher performance requirements from the launch vehicle.

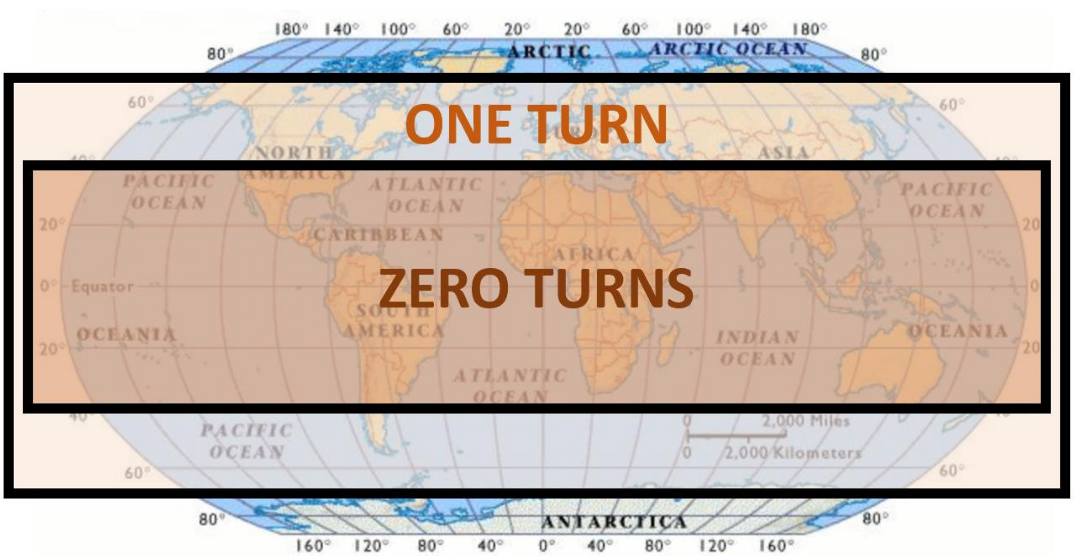


Figure 44. Critical Mission which Allows for full Russian Coverage with One Turn

The next mission type requires zero aerodynamic turns (therefore a much lighter hypersonic vehicle weight), but it has a high inclination at launch (therefore the highest delta V requirement for the launch vehicle). This mission is intended to dock with the international space station (inclination of 51.6 degrees), where no surprise orbital changes are required. The crewed vehicle can then use its cross-range capability to quickly land anywhere on the globe with a sufficient runway.

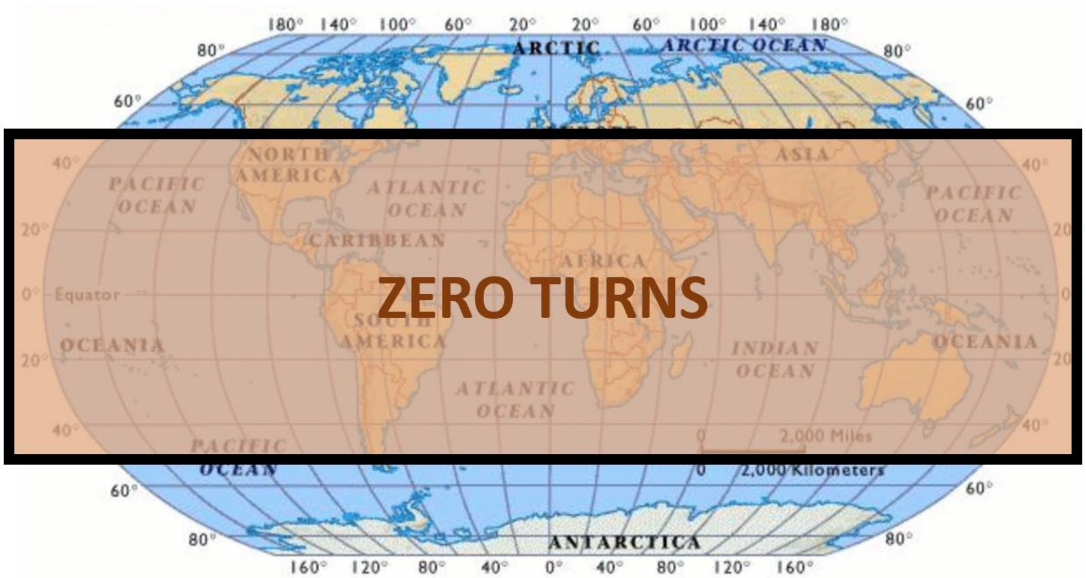


Figure 45. High Inclination Launch to Dock with Space Station

A lunar fly-by was considered for the tourism market, which pays about 175 million USD per ticket according to what competitors are selling [32]. While the delta V requirements were found to be sufficient for a free-return trajectory, this mission is determined to be unfeasible due to the re-entry velocity. There is not enough fuel to slow down the return vehicle by means of propellant before using the atmosphere to destructively slow down the space craft. A thermal protection system to combat this type of loading will be too heavy to carry on a lunar mission, and reduce delta V. The margins for delta V required for a lunar fly-by are just too low.

The Model 176's mission performance can be assessed for different planets. In conjunction with the senior project, the author has developed a script that is able to conduct performance analysis in different planetary environments and different aerodynamic capabilities [33], and finds it appropriate for a mission trade study. Additionally, this analysis serves to provide numerical evidence for the advantages of a synergetic maneuver over a propulsive one. Essentially, this is a more detailed look into Figure 2 by using Nyland's analysis from reference [28].

To consider the nominal mission for the reconnaissance variant, the following process is implemented as outlined in Figure 46:

Input Planetary Environment: $\boldsymbol{\rho_0}, \boldsymbol{\beta}, \boldsymbol{R_0}, \boldsymbol{g_0}$
Establish Mission: $\Delta \boldsymbol{i}_{\boldsymbol{desired}}$
Input Model 176 Parameters: $\boldsymbol{L/D}$
Run Nyland Analysis for synergetic turn: $\Delta \boldsymbol{V}_{\boldsymbol{required}}$
Run purely propulsive analysis for inclination change: $\Delta \boldsymbol{V}_{\boldsymbol{required}}$
Compare trades numerically and visualize

Figure 46. Methodology for Mission Profile Trade Study *[33]*

Figure 2 is replicated here according to the analysis shown in Figure 46, where the lift to drag ratio is 3.

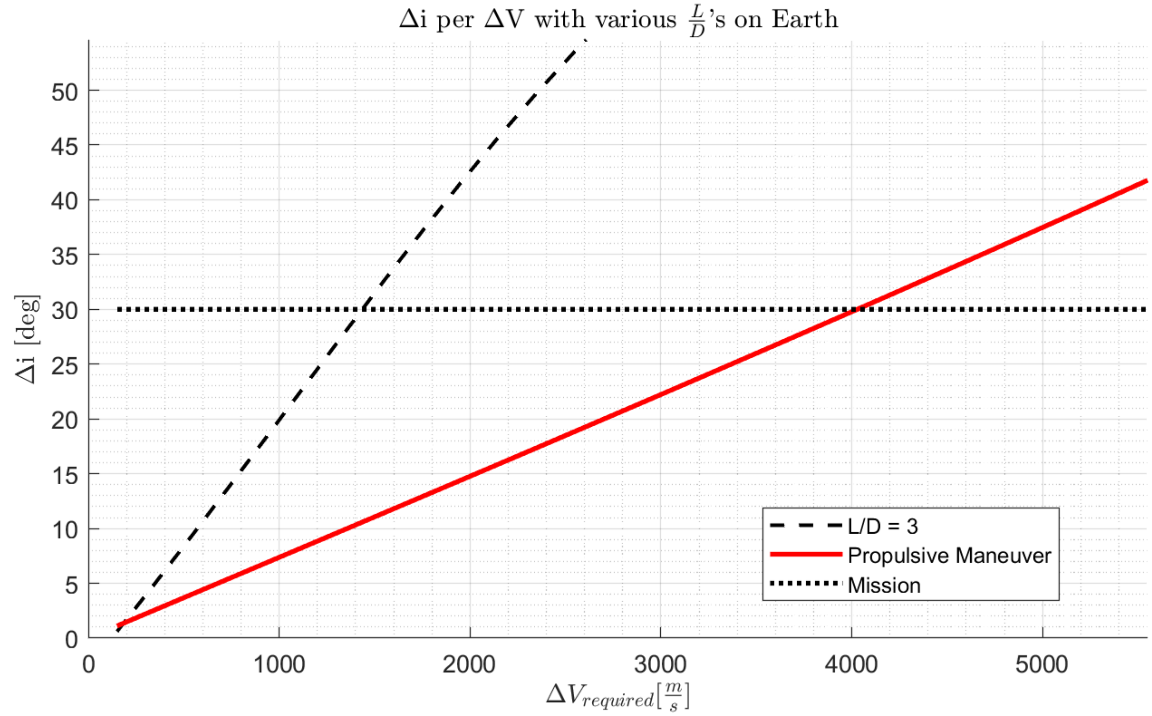


Figure 47. Synergetic vs. Purely Propulsive Plane Change for Model 176 Mission

Clearly, the aircraft can save a tremendous amount of fuel (about 2.6 km/s of delta V) to achieve the same mission requirement of a 30-degree inclination change. However, at the lowest inclination change requirements, it may be advantageous to simply use propulsive means to achieve the inclination change. This crossover point is visible in Figure 47 at around 1 degree of inclination change. This makes sense, as the reader will find it a waste of effort to have the craft's profile enter the atmosphere for a minor inclination change like docking with a space station. However, this is only the case for very small inclination changes as changing the large orbital velocity vector is expensive in terms of fuel costs.

The next question the author wanted to look into was how dependent the mission profile choice was on the aerodynamic capability of the aircraft. This is more of a sensitivity analysis to see how the aerodynamic capability "buys its weight" on this craft. Different lift-to-drag requirements are tested, as shown in Figure 48, to see much of a fuel cost is incurred by lowering the aerodynamic capability.

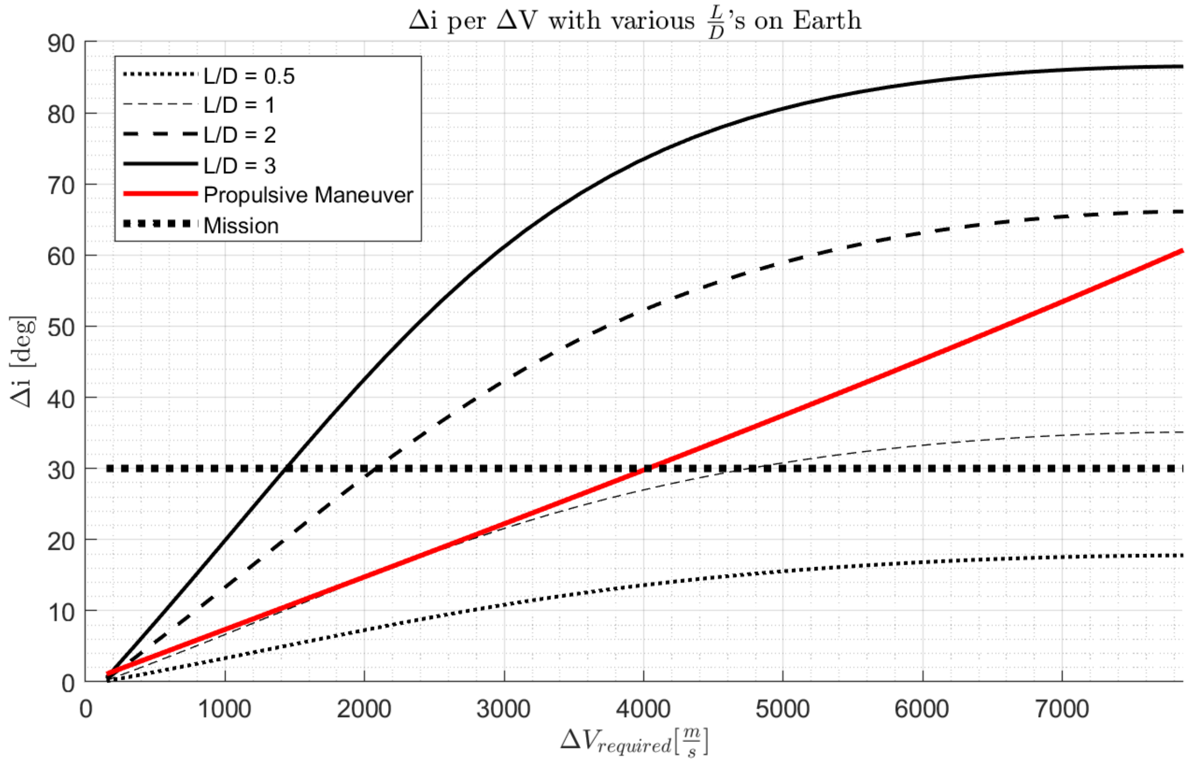


Figure 48. Vehicle Mission Capability for Different Aerodynamic Capability

This is one of the most interesting analyses in this report, as it demonstrates the advantages gained by increasing aerodynamic performance of the spacecraft. It seems as though a synergetic maneuver with an associated average lift-to-drag ratio of 1 is the lowest which can substitute a propulsive inclination change at around 16 degrees. However, is not suitable for the mission. Additionally, lower lift-to-drag ratios are not capable of completing the mission without completely re-entering the atmosphere. It is almost certainly not desired to enter the atmosphere with the intention of changing orbital inclination with such low aerodynamic performance.

The lower inclination changes are more closely examined in Figure 49 to determine the decision cross-over point between various aerodynamically performing aircraft and the conventional propulsive maneuver.

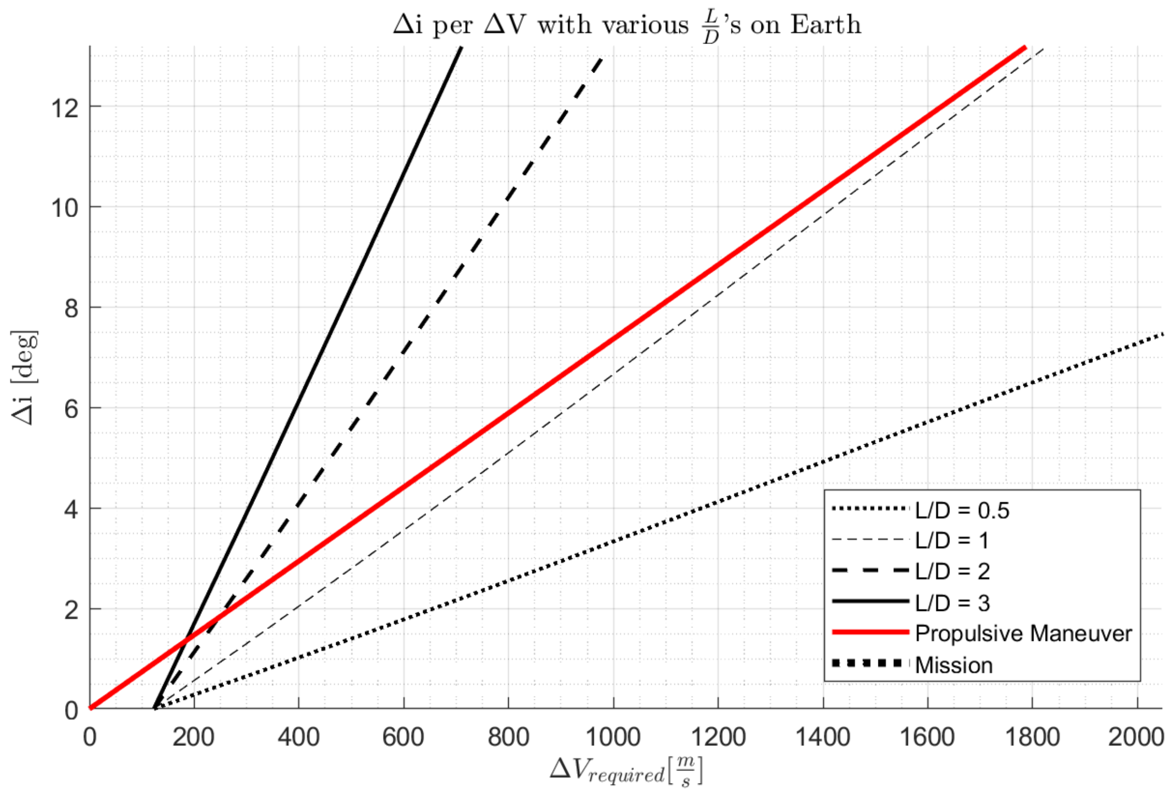


Figure 49. Closer Look at the Small Inclination Changes Expected for Orbital Rendezvous

The crossover point (if there is one) is increasing as the aerodynamic capability drops. This makes sense, as the lower performing vehicles conduct an aerodynamic maneuver with higher delta V costs. It is interesting to see that the shape of this plot resembles the author's reasoning as shown in Figure 2.

Lastly, the author wanted to change the planetary environment in which the Model 176 performed to see how much the performance of the aircraft was dependent on the atmospheric environment and its respective gravitational constant. Considering that even on Earth these parameters fluctuate across different seasons and latitudes, it was deemed a relevant trade study for the scope of this project.

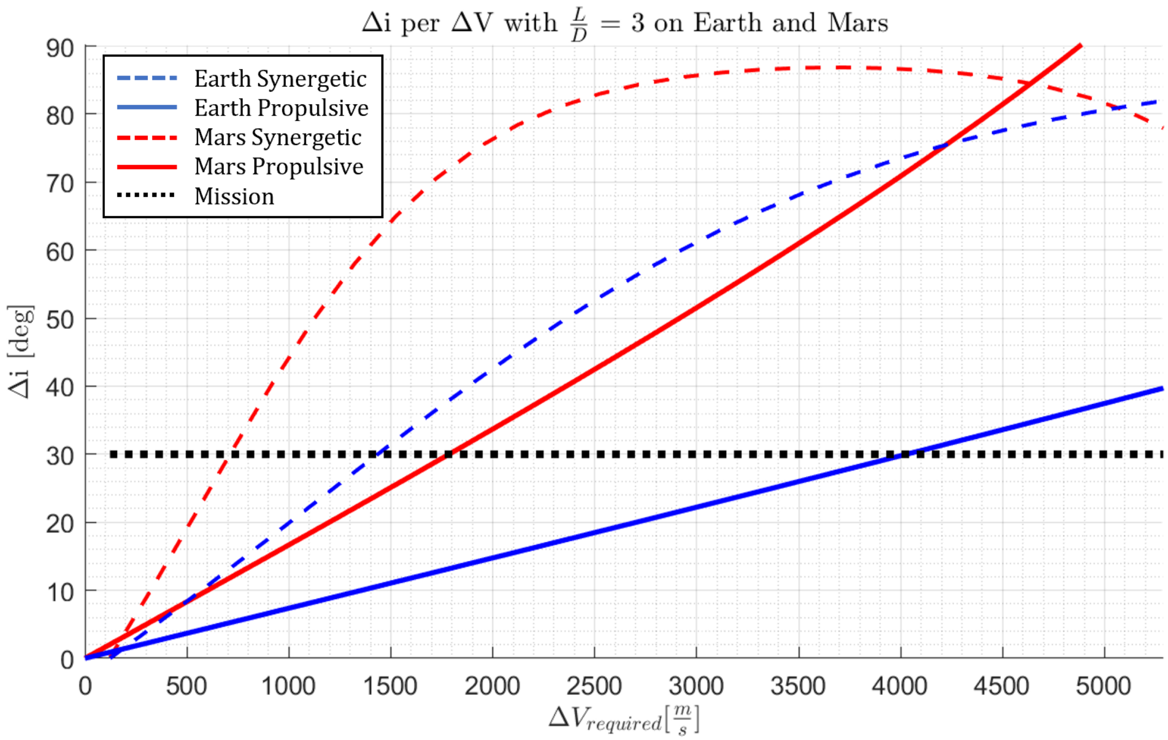


Figure 50. Mission Performance According to Different Planetary Environments

The relationship between synergetic and propulsion remains the same, but scaled to different fuel requirements for different gravitational fields. This is not scaled linearly, as different atmospheric scales require different burns to initiate a turn. For example, the turn for Earth is initiated at 67 km above the surface, and the turn for Mars is initiated at 34 km above the surface due to its thinner atmosphere. It is interesting to see that the crossover points between the two mission profiles for each planet scale as a function of the required orbital velocity for a 200 km orbit. This makes sense, as orbiting something like an asteroid would take minimal delta V to change the direction of the orbit since the angular momentum required to maintain an orbit is so low.

V. Team Management

A. Team Structure

The team is split into two main groups: one designs the hypersonic vehicle, the other designs the launch system. Both teams are headed by the author, who is the chief engineer for the hypersonic vehicle. Both chiefs are involved in their respective vehicle's synthesis and are to determine costs and the business case for a mission. The structure and domains of work for the rest of the team are shown below, which is divided by discipline.

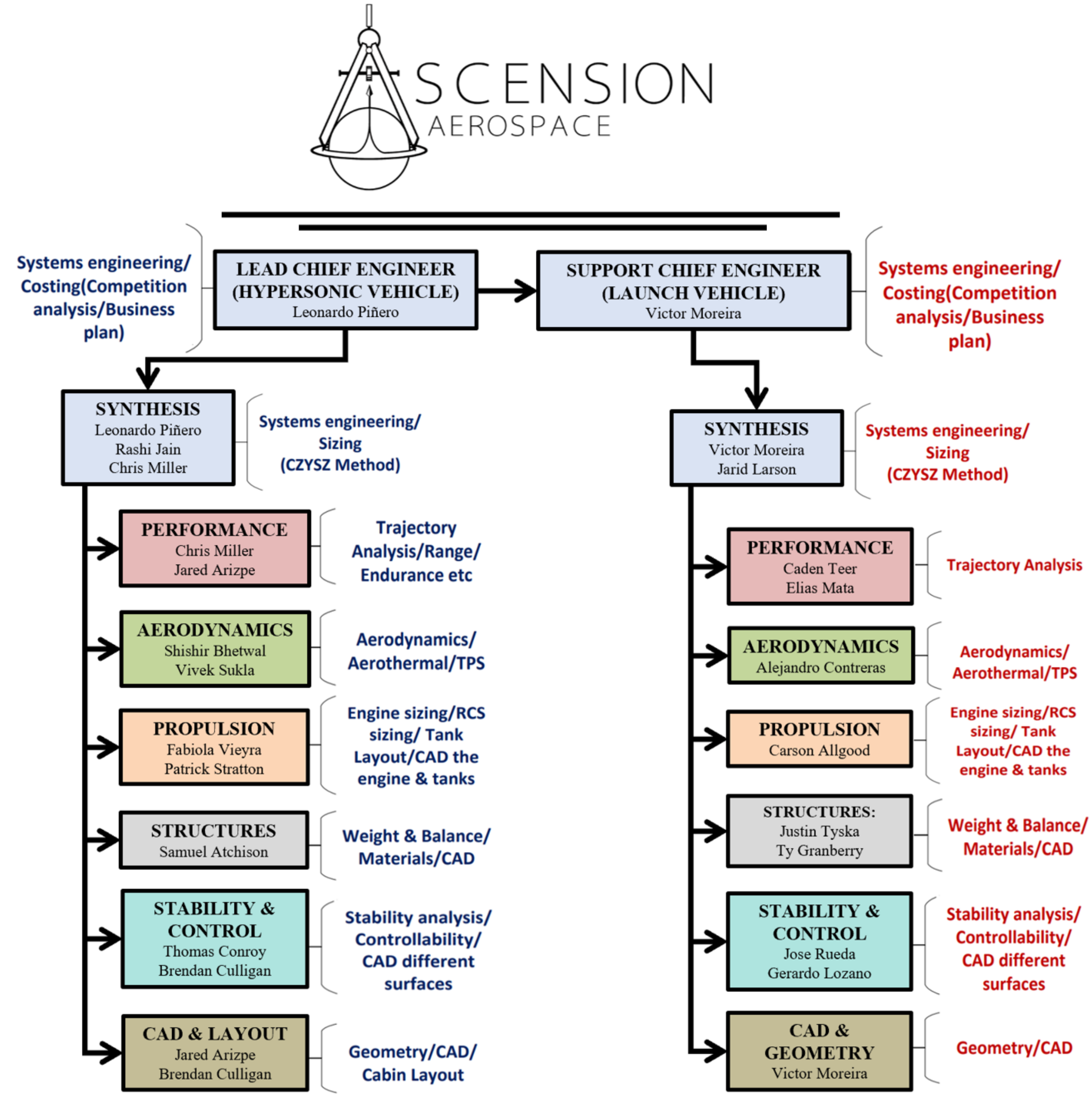


Figure 51. Team Structure and Responsibilities

The author's main work is to direct and bring together the many aerospace disciplines in the class team into one cohesive design. This includes writing the team report, one synthesized script, and developing a sizing methodology to provide key design parameters that meet the mission. The team structure is shown below, outlining all disciplines and their main task in developing a conceptual design for this vehicle.

The author must also develop a business case for the mission to meet military and market needs. This position is unique in that it must be familiar with the roles of all disciplines to build the bigger picture of the purpose of design.

The methodology and early decision-making in conceptual design is usually what determines the success of a program. For this reason, the role of developing the multi-disciplinary analysis and methodology bears a large responsibility.

Additionally, the role of chief engineer contains a significant human element. It is the responsibility of the author to make sure team deliverables are completed on schedule and every group member is contributing to the team effort. The chief is also the arbiter of disagreements over deliverables and domains of work. This requires a balance between what is realistic to complete in the summer semester timeframe and what must be done to produce an exceptional project [33].

There are many individuals who are in multiple disciplines according to preferences and needs, which facilitates inter-disciplinary know-how and interaction. For example, many stability derivatives are from geometric values, so a CAD engineer is placed in the stability and control discipline.

B. Near-term Timeline

For day-to-day tasks before reports, the chief lays out what each discipline must complete to keep the team on schedule for deliverables expected for the next report. This timeline used to be organized for each individual, but was updated to show day-by-day progress, with individuals responsible for the task marked by each task. Incomplete tasks are bolded, and un-bolded when complete. Below is a sample of this timeline, taken at the end of June 11th, 2018:

6/11/18

-post lit research tables for team consistency (Leonardo)

-evaluate each group member's progress (Leonardo)

-lit research on sizing (Leonardo)

-lit research on cost (Leonardo)

-work on MDA (Leo, Rashi, Chris)

-materials for different conditions (time-dependent?) TPS (Sam)

-IDA/MDA for aerothermal/structures sub-team, flowchart format (Sam, Shishir, Vivek)

-turn in S&C IDA (Brendan)

6/12/18

-work on developing trajectory equations to make atmospheric incidence angle turn (Chris)

-find delta V required to exit atmosphere for "n" incidence turns, as a function of L/D and other inputs (Chris)

-turn in CAD IDA (Brendan, Jared)

-compile and review IDA's (Leonardo)

-put together mission proposal (Leonardo, Rashi, Chris)

-prepare a compendium of possible load cases as sections of pseudo-code on MATLAB script, take a look at our flight phases in group chat and email pseudocode to Leo (Sam)

-find function to determine heating based on altitude and velocity, I found some sample data for a design (Sam)

-Propulsion IDA (Patrick, Fabiola)

-send Leo an update on aero variables you're filling in to table, I will add/modify data categories (Shishir, Vivek)

-turn in current progress on lit research (Vivek)

Figure 52. Sample of Near-Term Timeline at the end of June 11th, 2018

C. Semester Timeline

The semester timeline follows the main deliverables for key points such as the midterm presentation and team report (being the chief's responsibility to put together, much of the team report is developed in his individual reports). The timeline shown below is a tentative one for the summer semester following the class syllabus and will be continuously updated according to team performance. By midterm, the timeline has been followed without delays, which is a testament to lessons learned in MAE 4350. The timeline is shown below:

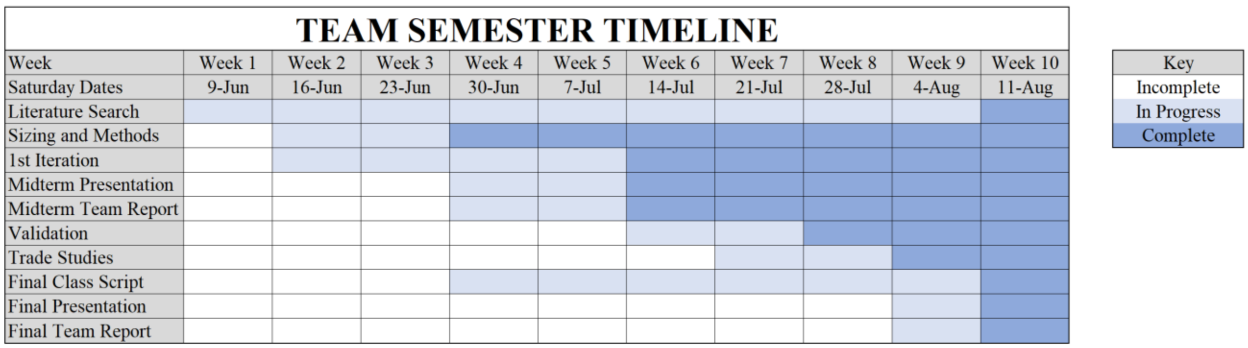


Figure 53. Semester Timeline

D. Team Member Performance

So far, every group member has been able to keep to their deadlines. Any delays are discussed prior to the deadline and readjusted in the near-term timeline within reason so that the team is not unexpectedly held back.

An emphasis on creating MATLAB scripts has been made in week 4, as every discipline is now expected to write out their methods in code. By doing this, the "rubber meets the road", and disciplines are required to declare variables (with needed inputs using dummy values at the introduction) and see how they connect across equations. Additionally, scripts have a top-down flow that is paramount to streamlining methods and IDA's. This is also a concrete deliverable that will be extremely useful in gaging discipline progress, as well as developing a cohesive synthesis script in preparation for the midterm presentation.

The author has noticed that there is a decrease in attendance which is negatively affecting team performance. After the midterm, the lead chief will begin to fill out an attendance report for each meeting to track performance on this avenue. It is deemed a necessary counter-measure to bring productivity back up.

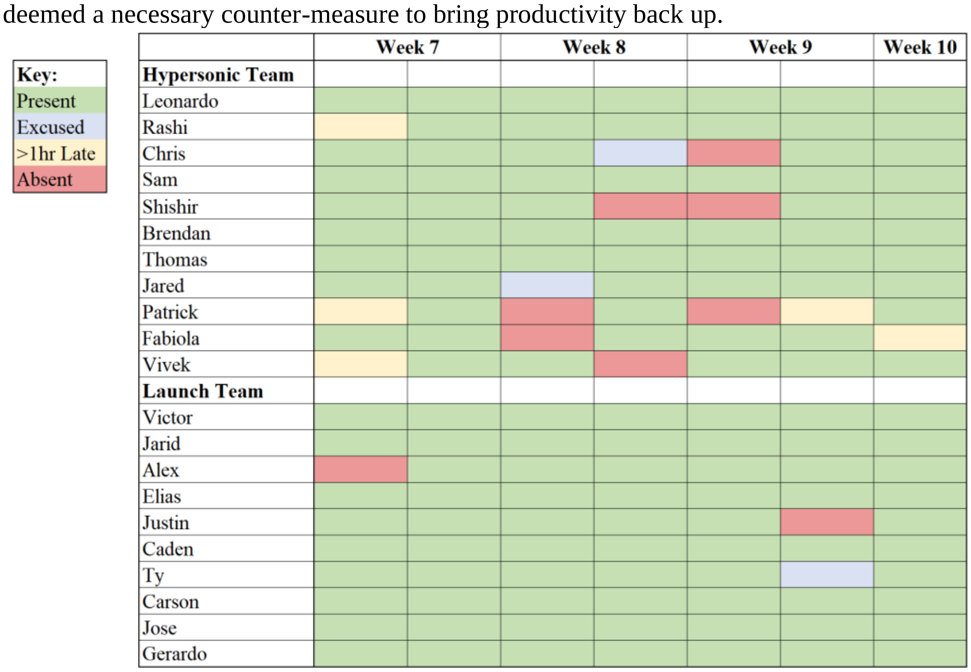


Figure 54. Team Attendance Sheet

The author has found that Aerodynamics is not providing their deliverables this late in the semester. Stability and Control requires inputs from Aerodynamics (Shishir Bhetwal) concerning the lift-curve slopes for the craft. This discipline has not provided the required deliverables for Stability and Control as needed. If he has provided deliverables, it was not in a timely manner nor of sufficient quality (especially considering this is a primary deliverable). Consequentially, this sort of performance is exemplified by the attendance sheet. Fortunately, the Stability and Control team is able to conduct analysis with substitute data, but by lacking data from other disciplines, they are not as attached to the multi-disciplinary analysis as they would like to be. This discipline is among the best-performing in the hypersonic team, as all their deliverables have been met on time or ahead of schedule, so it is not from a lack of analysis on their part. This statement is also exemplified by the attendance sheet.

The performance team has not been able to provide angle-of-attack data vs. Mach to stability and control as well; however, this comes from a gridlock of not having lift-curve slope data. Performance has been in communication with Stability and Control and has provided whatever he can.

Fortunately, the aerodynamics discipline was able to provide deliverables on time before the final presentation, but it was not in time for stability and control to include the new aerodynamic parameters in their methodology for the presentation. The performance engineer was able to conduct his analysis.

VI. Methodology

A. Derivation of Methods

To build the team multidisciplinary analysis, the chief first talks with the disciplines to assess progress on their respective literature reviews. From this, the methods which require inputs from others come into view. A synthesis discipline meeting determined the flight phases to be examined within the design-critical mission. These identified some design drivers (focused on the hypersonic vehicle) which would influence the team MDA and IDA's. The flight phases examined are as follows:

- Launch Pad
  - Structures: stationary load case
  - Synthesis: weight pressure on interface with rocket stage (launch team structural load case)
- Launch
  - Performance: obtain trajectory from launch team
  - Structures: load case for maximum dynamic pressure
  - Controls: neutral point and moment contributions to overall vehicle
- Engine Burn in Space
  - High thrust for more efficient orbital maneuvers
  - Controls: keep spacecraft pointed in the right direction
  - Performance: insertion trajectory and trajectory beginning
- Operations in Space
  - CAD: mission hardware, mass distribution, life support, etc.
  - Controls: RCS performance
  - Performance: orbit decay, trajectory visualization
  - Propulsion: engine restart capability
  - Structures: radiation protection (heating), pressurized vessel
- Atmospheric Inclination Change
  - Controls: hypersonic stability at vehicle orientation, RCS usage, control surface sizing
  - Performance: time to turn, periapsis, trajectory change, energy loss and delta V required
  - Aerodynamics: aerodynamic properties during hypersonic turn
  - Structures: TPS required for phase
- Final Re-entry from Orbit
  - Structures: TPS with empty weight
  - Performance: trajectory based on aerodynamic variables, keep track of cross-range
  - Controls: stability across all speed regimes
  - Aerodynamics: flight variables for all speed regimes and angle of attack
- Subsonic Approach
  - Performance: meeting field requirements
  - CAD: determine how a switchblade wing would fit and mechanized
  - Control: stability and control sizing for approach and landing
  - Aerodynamics: subsonic flight variables
- Landing and Ground Roll
  - CAD: landing gear
  - Performance: landing
  - Structures: empty weight-on-wheels load case

B. Discipline Inputs and Outputs

The following is a compilation of inputs and outputs written by each discipline, used to derive a reasonable MDA based on each team members' literature review.

Synthesis [29]:

Relevant Flight Phases:
- Launch Pad
- Launch
- Engine Burn in Space
- Operations in Space
- Atmospheric Inclination Change
- Final Re-entry from Orbit
- Subsonic Approach
- Landing and Ground Roll

Inputs:
- Mission Requirements (Chief Engineer, Synthesis and Performance)
- Propulsion Parameters
  - I_{sp} values, Mass Ratios for different flight maneuvers and phases
- Aerodynamic Coefficients (Aerodynamics)
  - Coefficient of lift/drag values at different altitudes and different Mach Number

Outputs:
- Market and Orbit Analysis (to come up with mission requirements)
- Max Take off gross weight
- Generic Weight Sizing
  - Max Take-off weight
  - Fuel weight
  - Empty weight
  - Wing Loading
  - Thrust Loading
- Generic Volumetric Sizing
  - Length
  - Width
  - Height
  - Scaling Factor due to a change in mission requirements

Methods:
- Methods
  - Hypersonic Convergence
  - Cszyz Method

Performance [15]:

IDA Construction: The performance team's IDA is structured around verification of the Model 176's capabilities by performing an analysis of four distinct flight phases using data provided by other teams. These four flight phases are vacuum operations, aerodynamic orbital maneuvering, re-entry, and low-speed landing. As each phase has considerably different requirements and occurs in drastically different environments, four separate analysis methods must be performed. Each method will determine the capabilities of the vehicle during those specific flight phases, and the calculated trajectories (with variables such as Mach number, altitude, and peak vehicle temperature as functions of time) will be used as outputs to various teams. Additionally, the vehicle's capabilities as found by these analyses will be compared to a provided set of mission requirements to ensure that they are possible.

Relevant Flight Phases:
- Engine Burn in Space
- Atmospheric Inclination Change
- Final Re-entry from Orbit
- Subsonic Approach
- Landing and Ground Roll

Inputs:
- Aero: drag polar, $\frac{L}{D_{max}}$, C_{L_α}
- Thermal: T_{max} at nose (sanity check), max total heat absorbed (if applicable)
- Structures: full/no fuel weight (W_{max}, W_{NF}), load factor limits
- Propulsion: max engine thrust, I_{sp}
- Synthesis: mission requirements (desired orbit characteristics, required orbital maneuverability, reentry cross range capability and desired load factor)
- Rocket Team (trajectory): flight condition post separation from Falcon

Outputs:
- To Aero(thermal): $M(t), n(t), h(t), T_{max}$
- To Propulsion/Structures: ΔV_{req}
- To Synthesis: cross-range capability, max reentry load factor, trajectory optimization plots

Methods:
- Aerodynamic Plane Change: Minimum-Fuel Aerodynamic Orbital Plane Change Maneuvers by Joosten
  - Important Variables: $\frac{L}{D_{max}}$, desired inclination change, T_{max}, Q_{max}
- Re-entry: 1st order analysis described in Aerospace Vehicle Design by Corning
  - Important Variables: $\frac{W}{SC_L}, \gamma_{entry}, T_{max}, Q_{max}$
- Vacuum Maneuvers – Space Mission Analysis and Design by Larson and Wertz
  - Important Variables: desired orbit characteristics, engine parameters, fuel weight available
- Low-Speed Landing – using code from previous semester to estimate landing speed and field length
  - Important Variables: $\frac{L}{D_{max}}$, $C_{L_{max}}$

Aerodynamics [16]:

IDA Construction: Vehicle geometry, weight, and flight conditions are provided to the analytical tools. The results from the analysis will be validated using the database collected from the literature review. The validated tools will then be used to analyze and re-engineer Model-176. The results i.e. Aerodynamic Loads, Coefficients, and Stability Derivatives obtained from the analysis are provided to S&C and Structures. Output from this analysis as well as inputs from other disciplines become the inputs for TPS and its design. The convergence criteria lie in the thermal and structural properties of the material used.

Relevant Flight Phases:
- Operations in Space: L/D, Atmospheric Conditions, Geometry, Weight
- Atmospheric Inclination Change: L/D, Atmospheric Conditions, Geometry, Weight
- Final Re-entry from Orbit: L/D, Atmospheric Conditions, Geometry, Weight
- Subsonic Approach: Lift, Drag, Weight
- Landing and Ground Roll: Lift, Drag, Weight

Inputs:
- CAD: Aircraft Geometry
- Synthesis/Structure: Weight, Volume

Outputs:
- S&C: Aerodynamic Coefficients, Stability Derivatives ($C_{L\alpha}$, $C_{M\alpha}$, etc.)
- Structures: Aerodynamics Loads

Methods:
- Wing Planform & Airfoil Selection: Aircraft Weight, Geometry, Flight Conditions (i.e. Landing)
- Drag Estimations: Geometry, Flight Conditions
- Aerodynamics Characteristics: Geometry, Flight Conditions
- ANALYTICAL DETERMINATION OF THE THERMAL LOADS by Anna Kolbe (Technical paper)
- Elements of Spacecraft Design: by Charles Brown
- Design Methods for Space Transportation

Stability and Control [19]:

IDA Construction: The major factors in building the S&C IDA was to determine what flight phases require unique methods to investigate. Through literature search it was determined that there are 4 major regimes that methods must be made for. Orbital RCS, Hypersonic, Supersonic, and Subsonic flight phases. The primary derivatives listed are the major variables to determine stability on a conceptual level. Control sizing has to be assessed at different critical flight phases and of interest are conditions with the lowest dynamic pressure due to a lack of control power and where the most elevator control power is needed at landing.

Relevant Flight Phases:
- Engine Burn in Space: What can be done if engine isn't perfectly center
- Operations in Space: RCS rates and requirements
- Atmospheric Inclination Change: Max bank and angle of attacks, low dynamic pressure control sizing
- Final Re-entry from Orbit: Low dynamic pressure control sizing
- Subsonic Approach: Wing aileron sizing
- Landing and Ground Roll: Elevon sizing

Inputs:
- Mission Requirements (Trajectory and Flight Conditions) (Performance)
- CG and Inertias (W&B/CAD)
- $C_{L\alpha}$, Aero estimated $C_{m\alpha}$, NP (Aero)

Outputs:
- Stability assessment (S&C Cmα Cyβ Cnβ Clβ estimated (Synthesis)
- Control surface requirements (CAD/Aero)
- RCS requirements (Locations, fuel required) (W&B/CAD)

Methods:
- Stengel Method (Geometry/aero input) (Subsonic/Supersonic)
- Clark and Trimmer Method (Geometry input) (Hypersonic)
- Phillips/Nicolai Method (Geometry input) (Air based control sizing)
- Wie Method (W/B input) (Orbital RCS sizing)

Propulsion:

IDA Construction: Design of rocket propulsion history for already built engines. Outputs of engines already made will be used to create the best efficient engine made. Weights of vehicle and engine volume will be needed inputs from other disciplines to size the engine for performance. Main outputs consist of specific impulse, thrust-to-weight ratio, mixture ratio, chamber pressure and velocity at exhaust.

Relevant Flight Phases:

- Engine Burn in Space: What can happen if engine doesn't start.
- Operations in Space: RCS, OMS and Main Engine Burns for Mission Requirements.
- Atmospheric Inclination Change: RCS and OMS propulsion system
- Final Re-entry from Orbit: Engine with Max Thrust & least Propellant Consumption.

Inputs:

- Synthesis/Structure: Weight (W) for Thrust-to-Weight Ratio
- CAD: Engine Volume (V_E)

Outputs:

- Performance: Thrust (T), Specific Impulse (I_{sp}), Engine Weight (W_{engine})
- Mix Ratio (MR) & Fuel Density

Methods:

- Modeling technique but also including historical modeling methodologies
- Engine Selection, Engine Weight, Flight Condition (Space)
- Excel & MATLAB

CAD [20]:

IDA Construction: The CAD IDA was constructed by analyzing the necessary parameters to create the CAD models of the aircraft first. Then, determining which methods (software packages) were going to be used, before determining what is needed from the CAD discipline by other members of the team in order to finish their analysis. Finally, the deliverables were included as an output to the IDA, before adding the feedback iteration loop to demonstrate the nature of the process by which the model is refined. The IDA is simplified in that CAD as a discipline is largely independent of flight phases, with the exception of the wings being deployed at subsonic speed. This allows for a short and streamlined IDA which is easy to read and understand.

Relevant Flight Phases:

- Engine Burn in Space
- Operations in Space
- Atmospheric Inclination Change
- Final Re-entry from Orbit
- Subsonic Approach
- Landing and Ground Roll

Inputs:

- Geometry from research
- Size and weight requirements based on propulsion and aerodynamic needs

Outputs:

- Geometric model to determine aerodynamic characteristics to aerodynamics

- Geometry and inertias to determine stability coefficients for stability and control
- CAD model for 3D print deliverable
- Visualization of the interior cabin of each iteration of the spacecraft

Methods:

- OpenVSP
- SolidWorks

C. Class MDA

The following MDA was devised as a temporary placeholder for the first two weeks, based on the two sub-team MDA diagrams and inter-team communication.

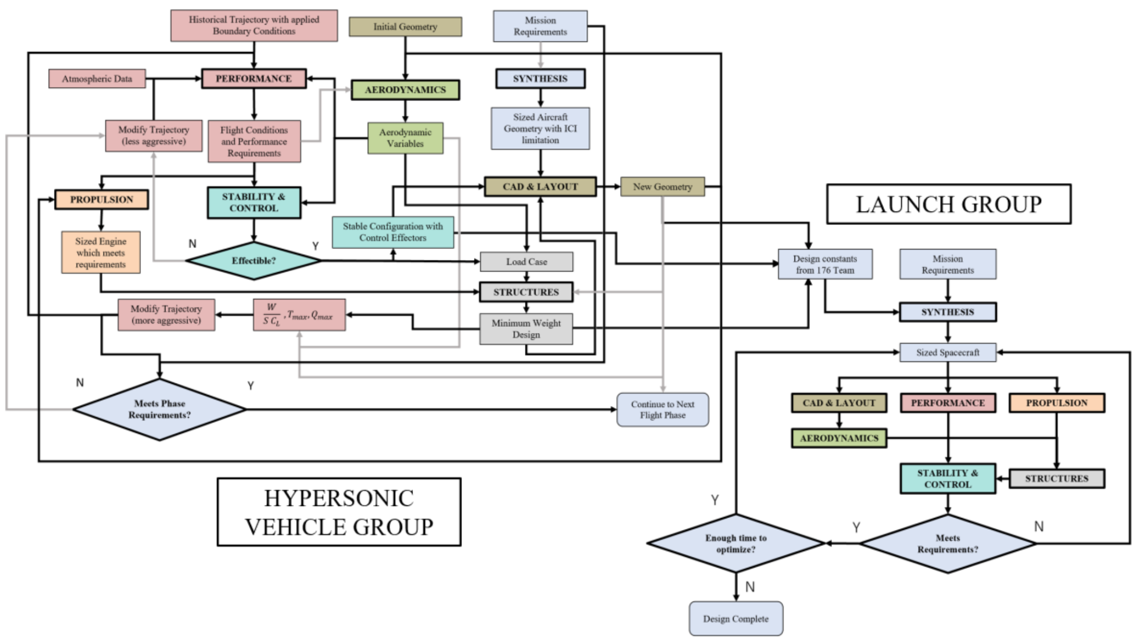


Figure 55. Preliminary Class MDA

After presenting to Dr. Chudoba, the above preliminary class MDA was found to be in need of serious revision. The arrows were flowing in many directions and splitting off, and there was a clear need

The first version shows a simplified version of a Nassi-Schneiderman diagram. Within the class MDA, the two sub-teams will form their MDA's to be iterated. The reason for having two levels of hierarchy is due to a premonition about iterating the sub-team MDA's throughout the semester. In this way, the upper-level class MDA may remain constant with respect to set mission and MAE 4351 Capstone requirements while there is an allowance for sub-team iteration at the lower level of hierarchy.

The second version of the revision shows the previously simple class MDA in a more detailed way. Both versions are kept as the first version is useful for quick explanation of the class process.

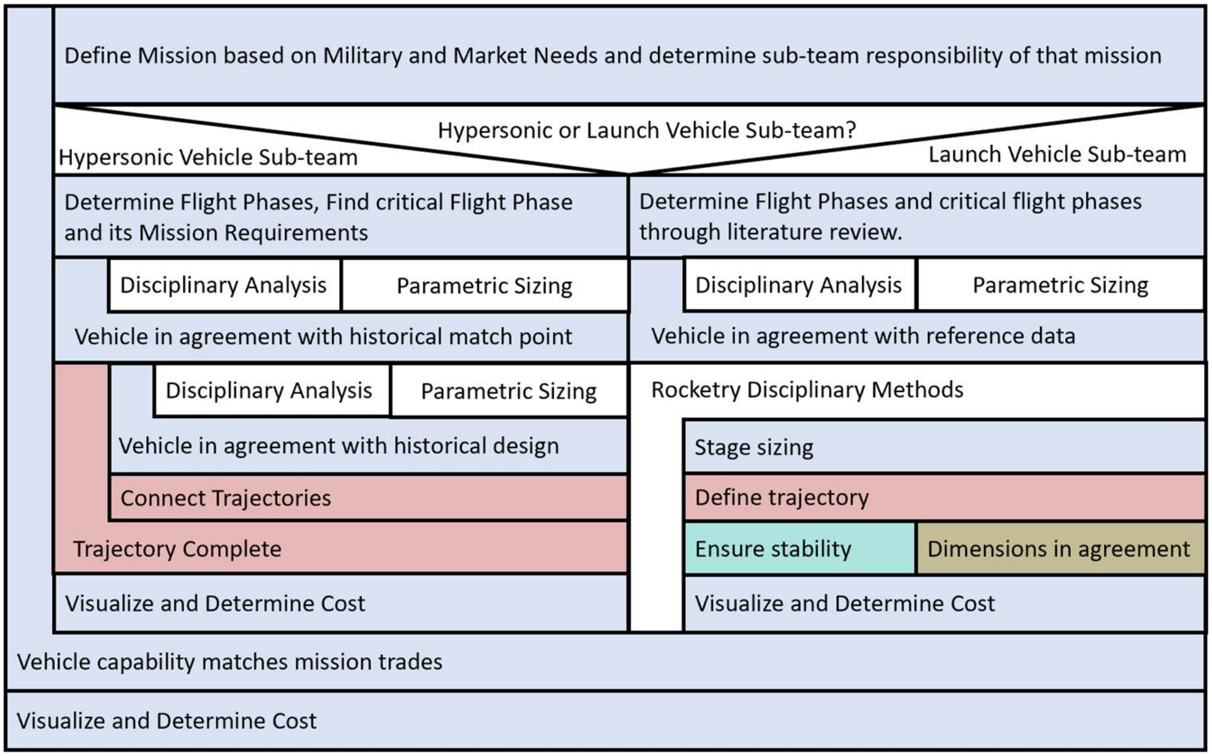


Figure 56. Simplified Class MDA

The class starts with the given mission profile and environment. This includes surface geography, launch sites, and last trajectory of the combined vehicle. With this last trajectory, the class can determine the domains of flight and analysis for each vehicle. This is where the teams split. Flight phases are then determined. The Hypersonic sub-team identifies the design critical flight phase and begins its disciplinary methods inside that loop. The hope is that most of the need for sub-team MDA iteration can be caught in this loop. The next loop continues through the next flight phases until the trajectory is complete. The large outer loop allows for mission trades and/or re-defining of domains (for example, that last known combined vehicle trajectory).

A more detailed look at the class MDA is shown below. Here the critical first step for the class can be examined more thoroughly.

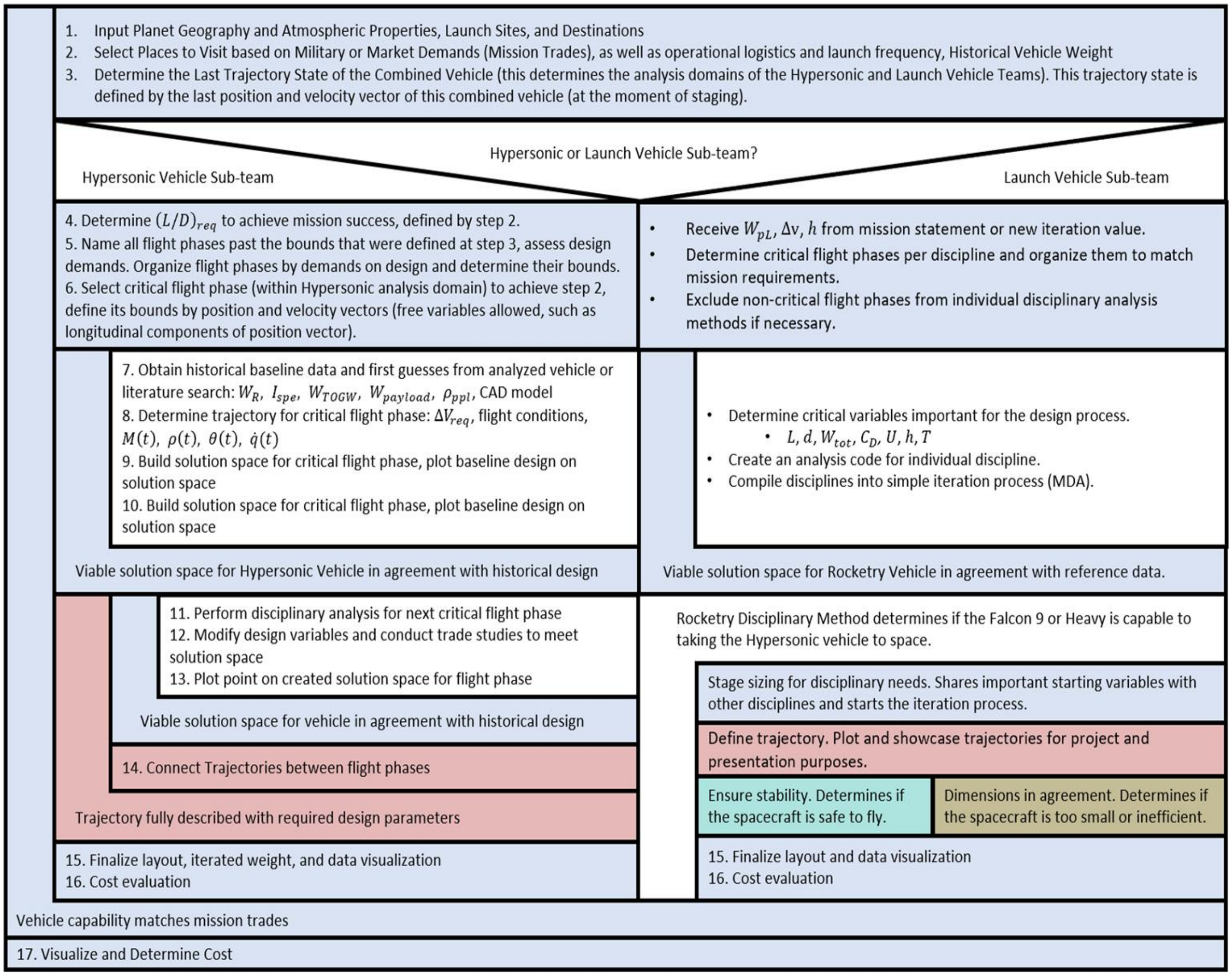


Figure 57. Detailed Class MDA

The main objective of the following steps (and upper-hierarchy class MDA in general) is to generalize input variables by evaluating first principles. From here, the team (and user) can easily recognize the fundamental assumptions made during the design process. By implementing a two-tiered MDA system, the class MDA can be more robust to sub-team MDA iterations by containing them inside logical loops.

The first step is to populate the planetary environment in which the vehicle operates and resides in by adding:

- Surface geography (ex: latitude range of mainland Russia or China)
- Key orbits to access (ISS inclination and altitude)
- Planet size and gravitational field (drives cross-range requirements for global access)
- Launch Sites and Landing Zone (Kennedy Space Center at Cape Canaveral, Florida)
- Atmospheric data

This gives us some locations to conduct mission trade studies.

The second step is an input of potential human needs which this vehicle could fulfill. Military as well as civilian market needs are examined (explained in Mission chapter):

- Surprise Reconnaissance in space above military rival for armed readiness and unpredictability

- Logistically favorable transport to a space station
- Point-to-Point transportation
- Escape Trajectory from planetary sphere-of-influence (not feasible without infrastructure on other bodies with atmosphere to complete mission)

The third step is likely to be a major point of iteration between the two teams, so it is worth some discussion. After the two previous mission selection steps, this is where the two teams decide on the last trajectory that both vehicles will share. This is a non-trivial decision since it was found that neither the Space X Falcon 9 B5 nor the historical Titan launch systems could bring any of the Model 176 stages to LEO orbit. For this reason, the shared trajectory will most-likely be suborbital. This will need to be iterated upon to reverse engineer the trajectory that the Model 176 would have taken.

The agreed upon combined vehicle trajectory reaches a ballistic sub-orbit at an apogee of 200 km from the launch pad. The apogee of this sort of orbit is supported as the stable altitude for an orbit which decays slowly enough to allow for orbital operations to take place (as well as the boost circularization maneuver). The following plot shows the relationship between orbit altitude and length of time for a stable orbit before the small atmospheric drag will bring it back down to Earth [34].

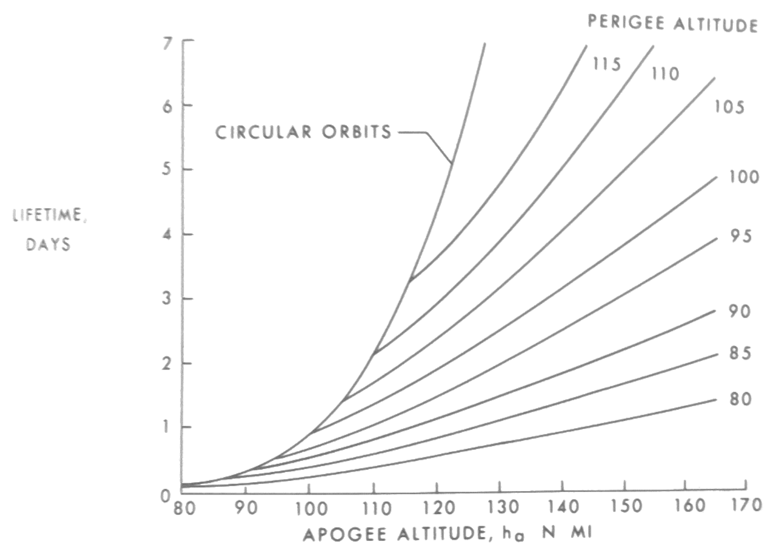


Figure 58. Orbit Lifetime vs. Apogee Altitude, Including Circular Orbits *[34]*

From here, we at least know that the starting point is to determine the domains of both groups: to have the *combined vehicle* achieve a certain state, as defined by position and velocity in an earth-fixed system. This can be user defined in the synthesis code. The hypersonic vehicle is then expected to circularize this sub-orbit. Any failure to meet mission requirements calls for a re-negotiation of flight domains between the sub-teams.

For the next step, the required lift-to-drag ratio is determined from mission requirements. This will guide the decisions for cross-range capability and inclination change maneuvers. The flight phases for both sub-teams are determined and are explained as they relate to the disciplines in the first section of this chapter as part of the derivation of this MDA. The design-critical flight phase is then selected to be the main test run for the multi-disciplinary approach. The selection criteria involve physical strain on the structure, disciplines required, and unique mission relevance.

Before officially beginning disciplinary analysis, some historical variables are given as part of the project introduction, and they are input here. A CAD Model is also constructed from the historical vehicle.

The multi-disciplinary analysis for both sub-teams is featured in the next two sections.

There are two regions for sub-team multi-disciplinary analysis. As previously stated, the hope is that most iterations were hammered out in the first loop before continuing with into the larger loop which iterates the same process as the first loop, but by connecting the boundary conditions of the trajectories. For this reason, performance is the only other discipline other than synthesis visible on the hypersonic vehicle side of the class MDA.

D. MDA for Hypersonic Vehicle

The first version of the Hypersonic Vehicle sub-team's MDA begins with the design-critical flight phase: hypersonic plane-change maneuver. These are characterized by hypersonic thermal loading and high dynamic pressure loading, respectively. For descent, the idea is that we start with a conservative performance trajectory from a heavier vehicle. The loads are calculated for this trajectory based on initial hypersonic lifting body geometry. Disciplines work to produce a load case for structures in the effort to replicate this conservative trajectory from performance. This load case with thermal and dynamic pressure loads will be used to find a minimum weight design from the structure and CAD disciplines' selection of materials, structure, and geometry. The minimum weight design is then iterated back to performance to generate a more advantageous trajectory.

From this, the author has developed the following preliminary MDA which could be generalized to each flight phase:

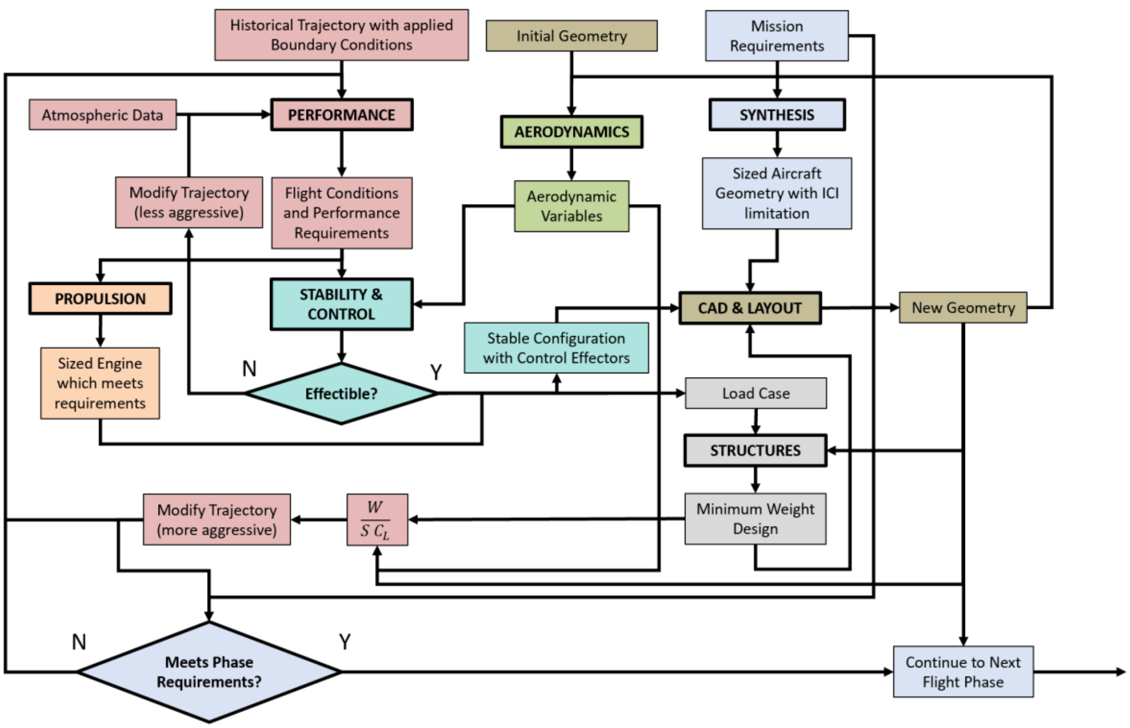


Figure 59. Preliminary Sub-Team MDA for Hypersonic Vehicle Flight Phases

The above MDA is not complete but is rather an expression of the brainstorming process for each phase. This is converted into something more procedural in the time domain, going from top to bottom, as was demonstrated by the class MDA in the previous section. This sub-team MDA will represent the disciplinary analysis introduced as loops in the class MDA.

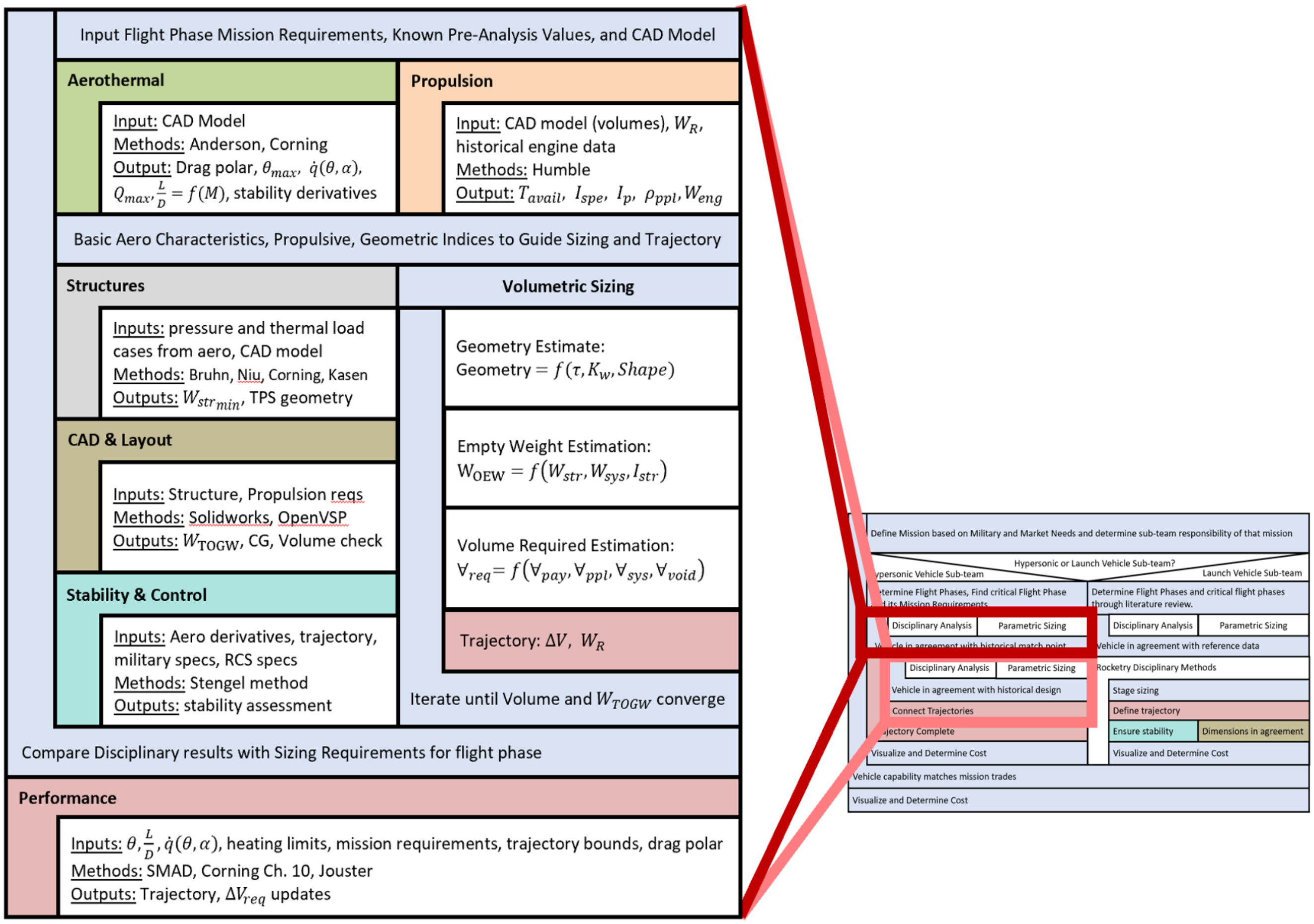


Figure 60. Hypersonic Sub-Team MDA (1st Version)

The disciplinary analysis draws from the data found in the previous steps. Additionally, the aerothermal and propulsion disciplines provide initial data for sizing the vehicle. They take disciplinary literature review and the CAD model to develop outputs for the structural aerothermal load case. The TPS is developed by structures in parallel with the sizing methodology. CAD then takes the structural weight and design to develop the total weight of the vehicle, center of gravity location, and volume distribution to set the stage for the stability & control discipline to assess stability for all flight regimes and flight phases based on methods found from literature search.

The parallel sizing methodology iterates until convergence with the historical design given aerodynamic parameters, mission, and new integrated structural and propulsive design. Performance does a trajectory and weight ratio check based on the new design. When sizing is complete, the results are compared to the disciplinary values. If they don't match, the initial assumptions based on geometry and literature research are re-evaluated. When there is discipline agreement with the solution space, performance builds the trajectory for the analyzed flight phase. After this MDA is complete, the loop highlighted in red situated in the class MDA is complete. The design may now proceed to the next step found at the upper hierarchy level.

After a flow diagram developed by the Synthesis lead (R. Jain), the author decided to reconfigure the hypersonic sub-team MDA to more closely match it, since it seemed more straightforward and made more sense to the synthesis team. This sort of iteration is expected, especially in this for flexible inner loop. However, it is important to note that this sort of iteration can be done within the existing framework of the larger class MDA, which has the benefit of removing a subconscious resistance to change do to its potential negative effects on the class workflow.

Performance was shifted to be the interpreter of the mission requirements to develop a flight profile for propulsion and aerodynamics. The following disciplinary analysis was adjusted to conclude with the CAD and layout discipline to provide a basis for checking the volume and weight budget.

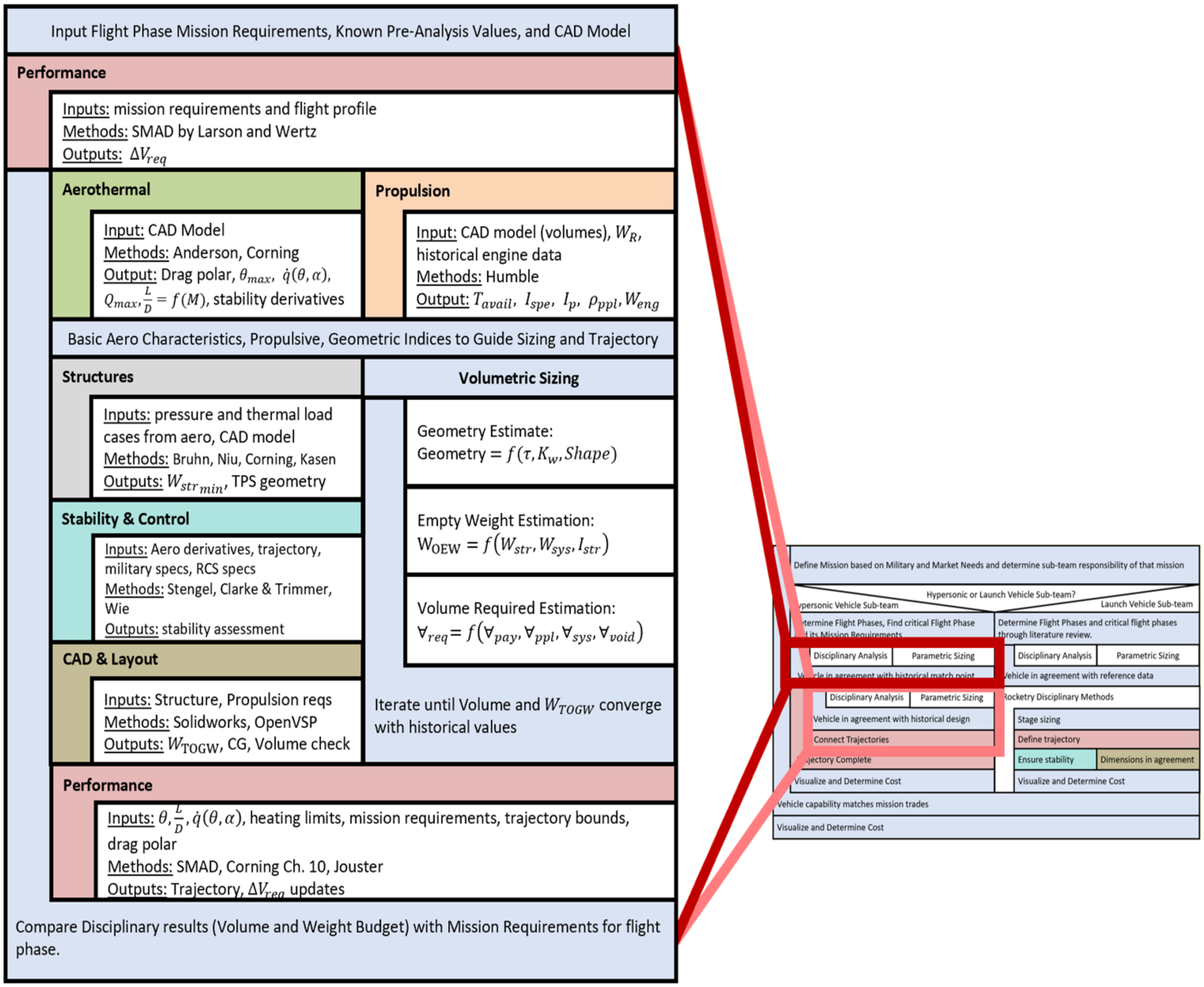


Figure 61. Hypersonic Sub-Team MDA shown as Two Logic Loops in Simplified Class MDA

E. MDA for Launch Team

The preliminary sub-team MDA made by the launch sub-team's chief for the launch team is displayed. This was also later iterated to fit the class MDA.

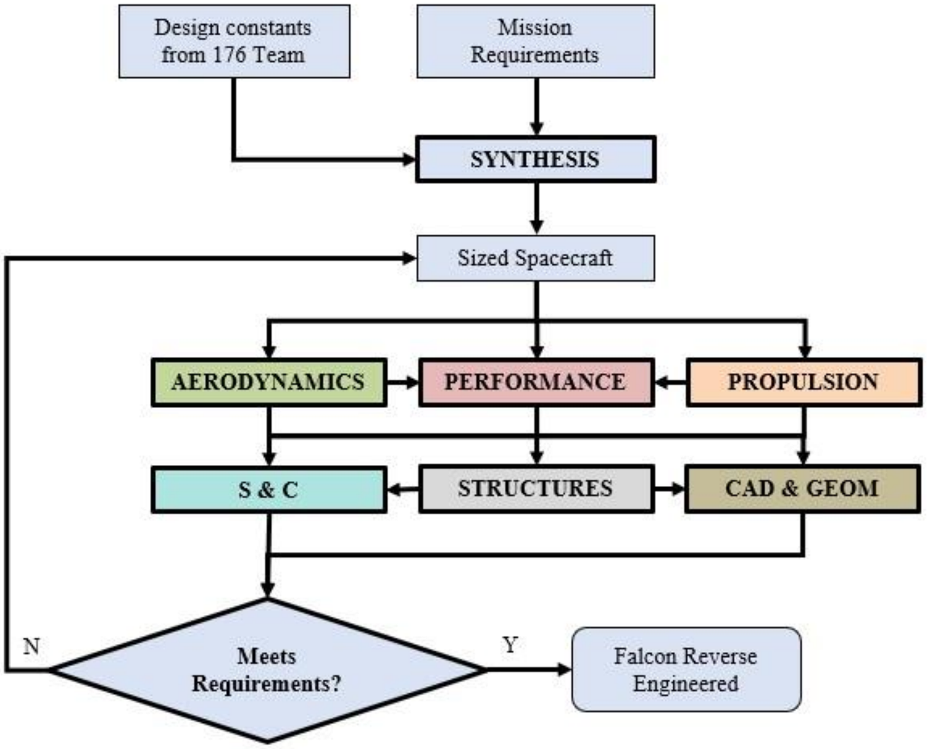


Figure 62. Launch Team Preliminary MDA *[30]*

The latest iteration of the Launch team MDA is shown below. There is one main logical loop in the class MDA where this analysis resides in.

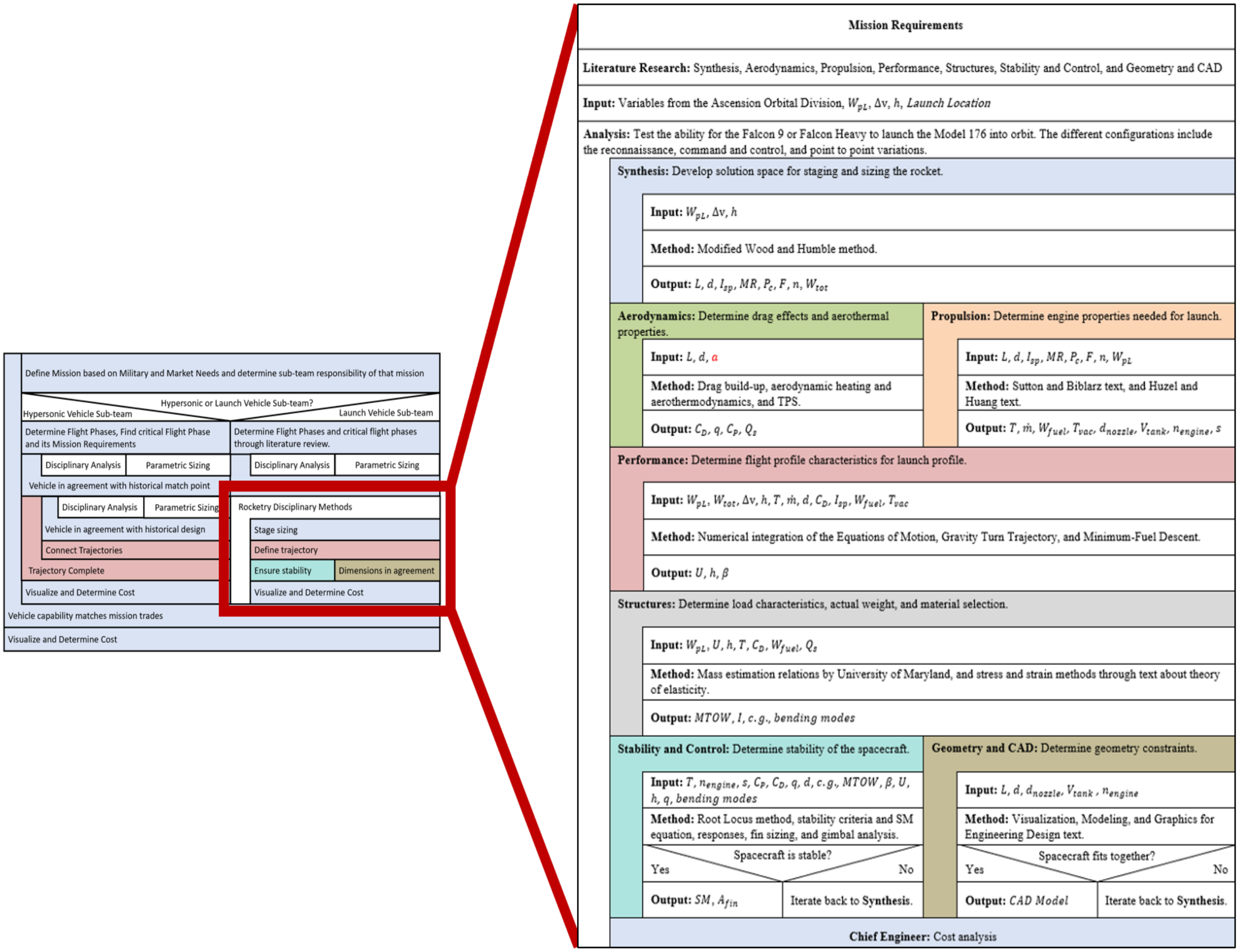


Figure 63. Launch Sub-Team MDA shown as One Logic Loop in Simplified Class MDA *[30]*

F. IDA for Hypersonic Vehicle Disciplines

The disciplines have made a first attempt at their individual disciplinary analysis (IDA), which will be used to finalize the multi-disciplinary analysis. They have all been converted to Nassi-Schneiderman format to reflect the team's progress on methodology development.

Synthesis IDA [29]:

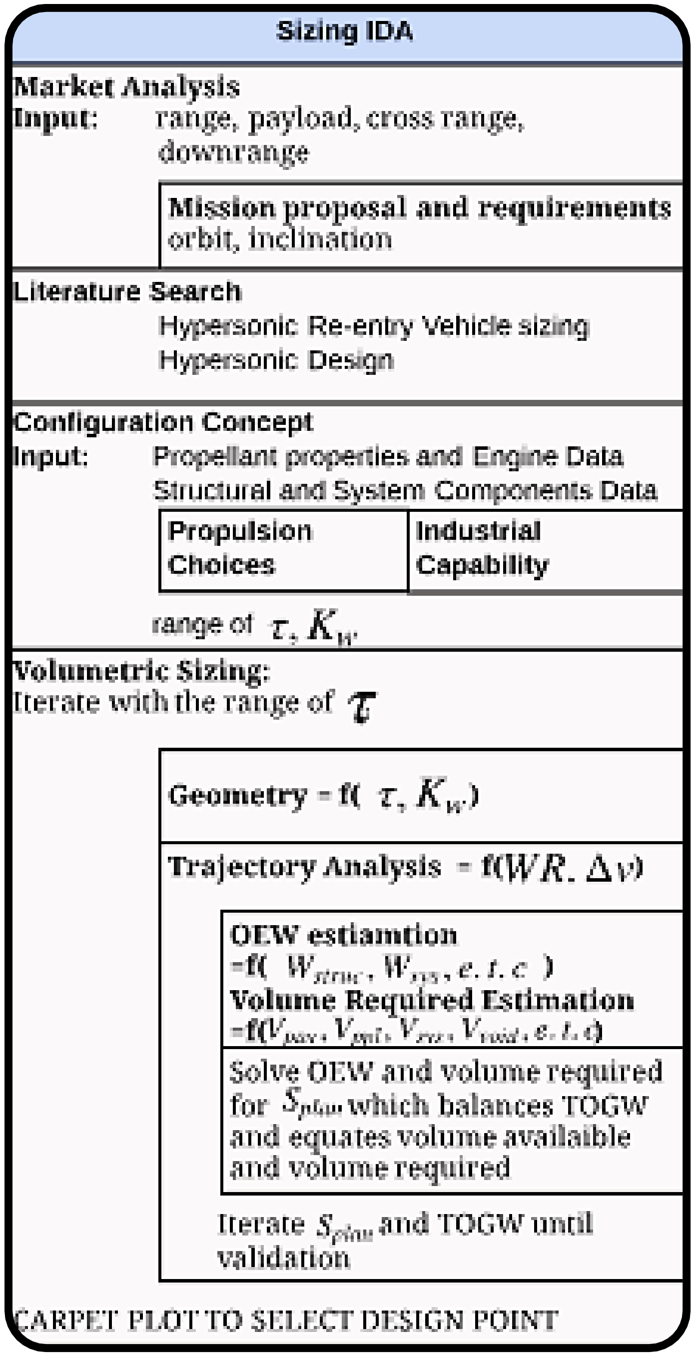


Figure 64. Synthesis IDA *[29]*

Performance IDA:

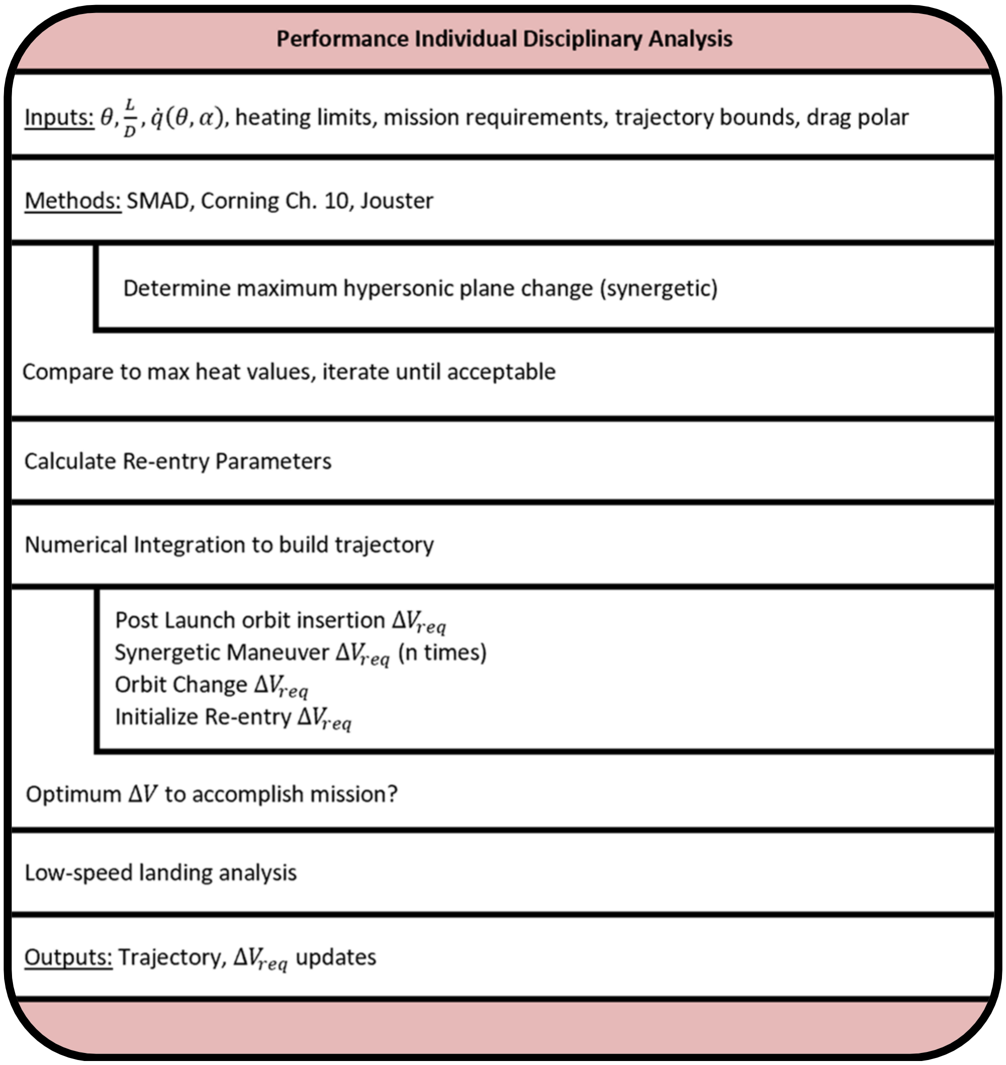


Figure 65. Performance IDA

The IDA for performance differs from the other disciplines in that it is organized by a sequence of flight phases. This is due to the fact that a large role of performance is to stitch together all the trajectories into one, thus fulfilling the outer loop requirement found in the class MDA.

Aerodynamics IDA:

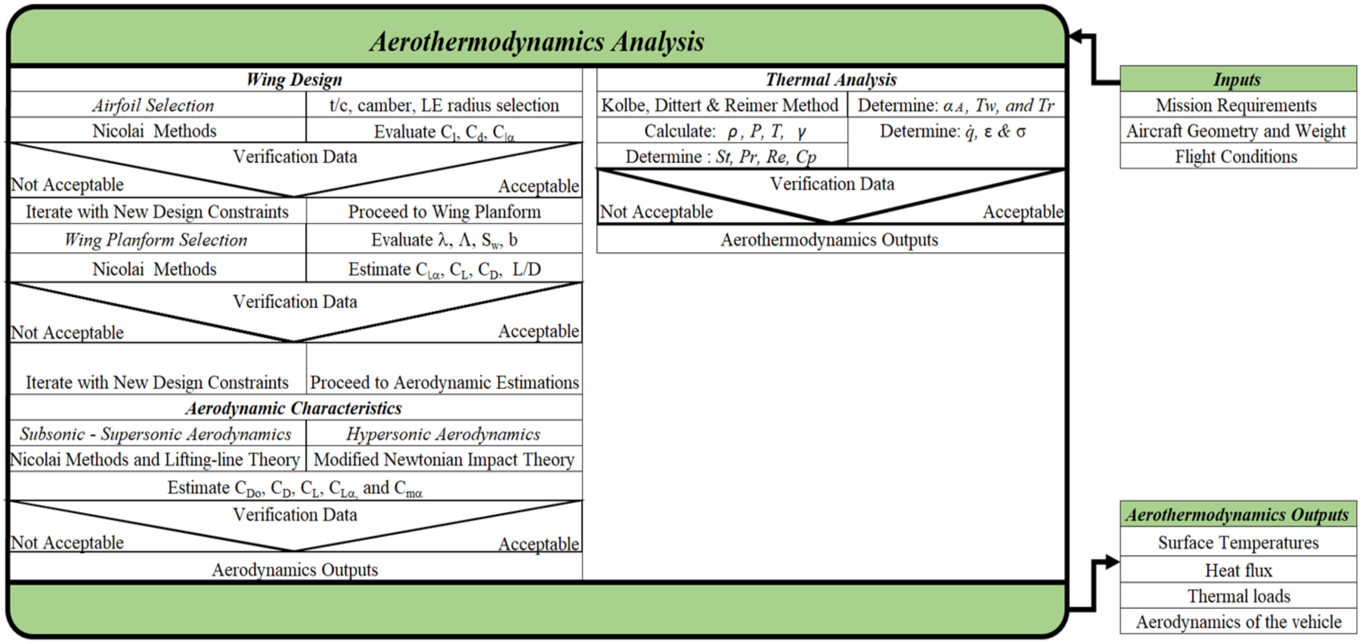


Figure 66. Aerodynamics IDA *[16]*

Propulsion IDA:

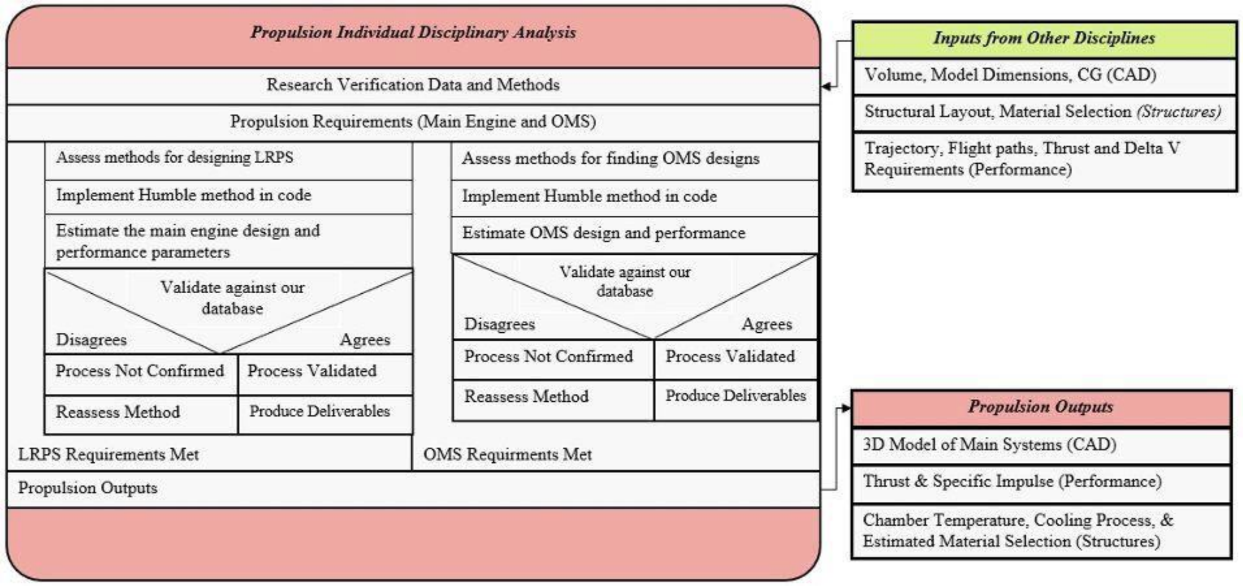


Figure 67. Propulsion IDA *[17]*

Propulsion methodology expansion on the Humble Method:

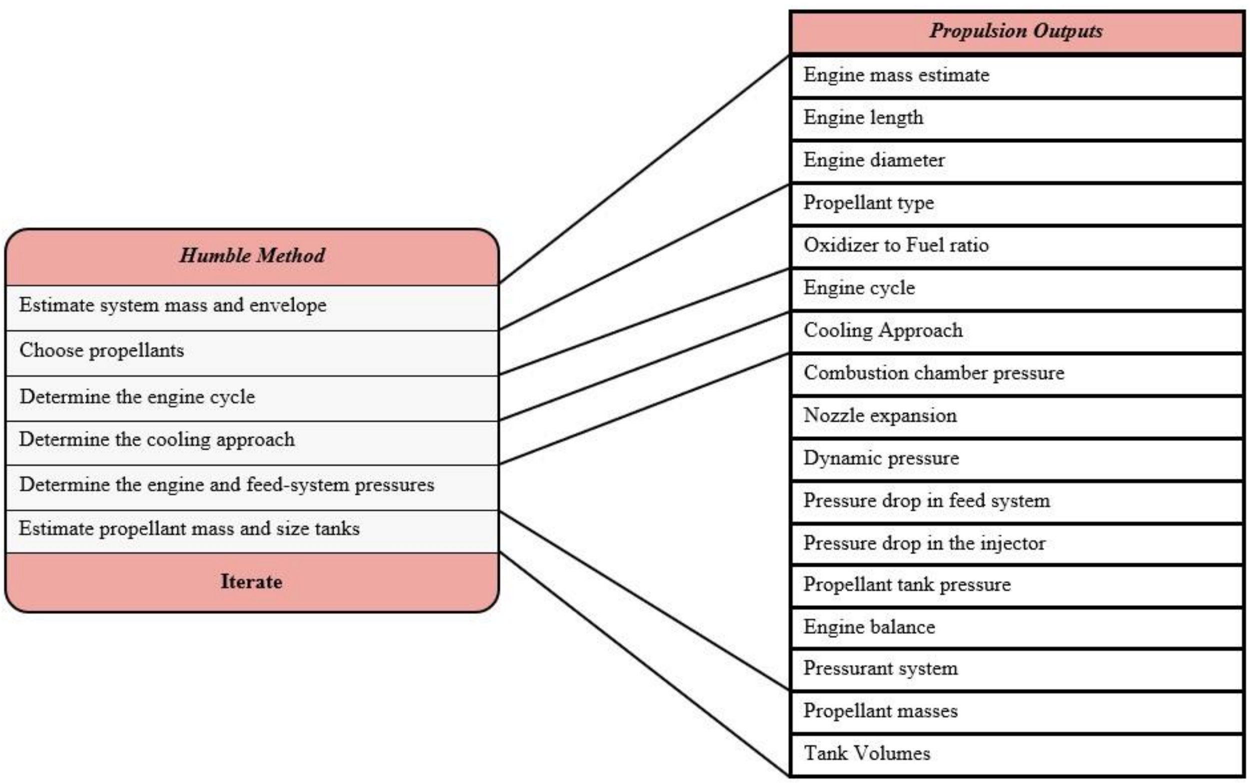


Figure 68. Humble Method for Propulsion Discipline *[35]*

Structures IDA:

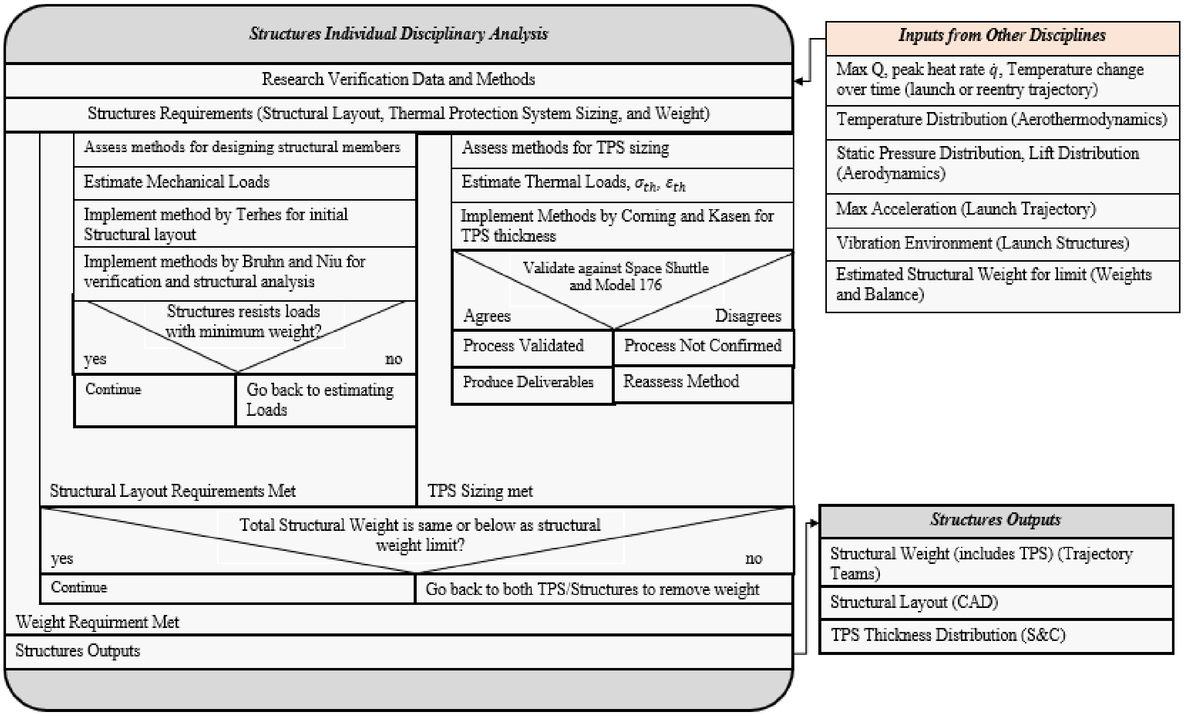


Figure 69. Structures IDA *[18]*

Stability and Control IDA:

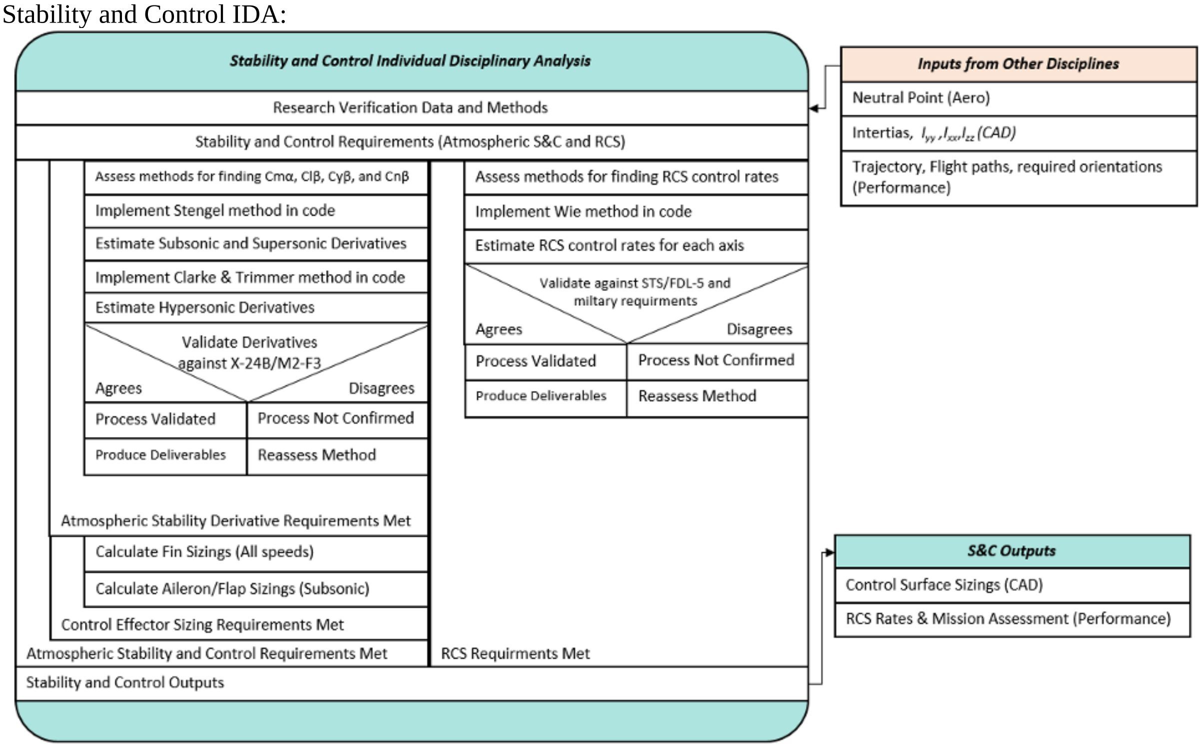


Figure 70. Stability and Control IDA *[19]*

CAD IDA:

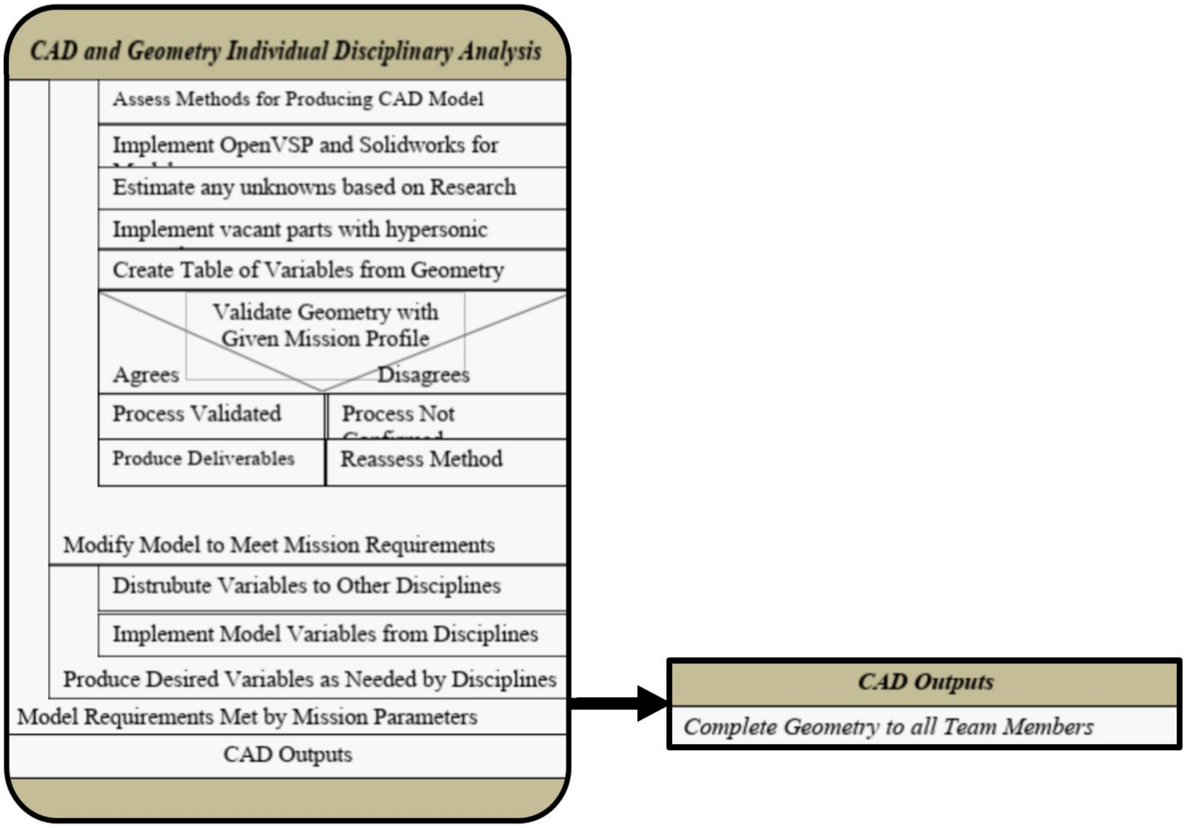


Figure 71. CAD IDA *[36]*

VII. History of Program Development

The following chapter chronicles the team's progress and key issues from a chief engineer's perspective. It includes multidisciplinary insights and future plans for improvement from the class work and meetings the author has had with other team members. The goal of providing this section is to document lessons learned and gage performance of the team during the time of writing. This chapter was introduced in the 4th week of the project, so earlier weeks are not as well documented since they are based on longer-term memory and available records.

A. Week 1

This is the kickoff meeting for the semester project, and Dr. Bernd Chudoba, the senior design professor, introduces the mission and vehicle to be reverse-engineered with a design methodology. The team structure must be defined among three sub-teams according to the three vehicles examined: Falcon 9 B5, Falcon Heavy, and the Model-176. Chief engineers are selected for each sub-team: Caden Teer, Victor Moreina, and Leonardo Piñero-Pérez. The team structure was changed according to a class re-organization, with one lead engineer and on sub-chief for one of the sub-teams. The class decided that the author be the chief engineer for the class (while focusing on the hypersonic team), and Victor Moreina be the chief for the launch team.

Actual disciplinary work was done as literature search and organizing group messaging platforms according to sub-team and function. The chief has asked each team to summarize the results.

B. Week 2

Discussion about the mission with the synthesis team leads to a critical mission choice. Preliminary MDA organization is developed by the author. A lot of preliminary work is assigned, such as the construction of a CAD model of the 176.

C. Week 3

The author presents his preliminary MDA to Dr. Chudoba and the class and is critiqued on the organization style. The flowchart structure is not a top-down approach and needs revision according to the Nassi-Schneiderman approach. Mission is numerically defined at specific latitudes; mission trades are set up.

D. Week 4

This week involved radically extending reports and finalizing the methodology.

Stability and Control is finalizing their stability assessment script, author directed lead to find verification data and get it ready for presentation.

Author directed CAD discipline to develop a geometry script to allow user to interface with script by selecting shapes and reporting geometric parameters.

E. Week 5

The class had a discussion regarding the weekly submission of reports after the author emailed Dr. Chudoba on behalf of the team's concerns over the frequency of reporting progress. The email listed what the team perceived to be pros and cons for three options: keeping weekly reports, changing to bi-weekly reports, or changing to bi-weekly reports with oral reports on alternating weeks. The team had suggested the third option, but Dr. Chudoba elected to keep the weekly reports on account of report quality and maintaining a discipline reflective of industry standards. This discussion was necessary to convince the class of the need for weekly reports and cleared any disagreement with the curriculum requirements. However, in light of the United States' national independence holiday, it was decided that Report 4 will be cancelled, where the next report due is the midterm report.

This chapter is introduced after a conversation with Dr. Chudoba regarding areas of improvement for the author's report. It is intended to document the work of the chief in facilitating disciplinary work and product synthesis.

The methodology is shown in summary to Dr. Chudoba, where he approves of the two-tiered hierarchy of the class structure. This is in accordance with an overall assessment that the class is exponentially improving on their ability to perform and produce meaningful work.

The author had a private meeting with the chief of the launch team, who had concerns that the author was micromanaging and expecting too much from the class team. The author is assigning tasks in the context of the expectation of at least 40 hours of work per week from each team member, regardless of their personal situations. Though the author listened to the launch chief's case for why a lead chief's role should be laissez-faire until the end of the semester when the work was integrated, the author disagreed on the reasoning that this leadership style would be unfair to those team members who are contributing significantly throughout the semester. The author has personally elected to ramp up his involvement in setting expectations for team members.

Author directs the class to prepare for the team midterm presentation. Submission of slides demonstrating preliminary results are due at 06:00 on 13 July 2018. The sub-team chiefs will put together their sections and prepare the presentation.

Conversations with disciplines involve progress on building functions which encapsulate the various design methodologies that the disciplines use. An MS Word template is handed out for disciplines to develop a user guide for each of their methodology MATLAB functions, and these can be viewed in the second section of the Synthesis chapter.

Performance is having trouble with building a trajectory for the hypersonic vehicle during the inclination change. This is not from a lack of work-ethic, as this problem is complex and involves 3-dimensional coordinates and high-level mathematics. The sole performance engineer, Chris Miller, for the hypersonic team has already made several attempts at developing a script to plot this flight phase, which are documented in his report. The main goal this week is to build the trajectory based on vehicle parameters, as that is his last flight phase to construct.

Propulsion will have a preliminary sizing methodology finished with results for the midterm presentation and has already encapsulated this in a function. Patrick Stratton is writing the MATLAB script for this, and Fabiola Vieyra will be concerned with verification.

Structures has developed a methodology to check for vibration analysis and is expected to encapsulate this as a function for the structural analysis.

Stability and Control has developed a function for assessing stability and verifying it with test data.

F. Week 6

At this time, it is expected that all disciplines have finished their MATLAB scripts except for CAD due to the nature of the problem their script is trying to solve. The CAD script is having some difficulties, so it was determined to extend their script deadline to wait for more concrete requirements from sizing, structures, and propulsion. This is more consistent with the MDA. For now, the tools that CAD uses to obtain geometric values (SOLIDWORKS, OpenVSP) will be used as a stand-in when running the code this week.

One key resolution was to determine the interface between the orbital mechanics of performance and the thrust requirements of propulsion. The solution was to base the thrust requirement around re-circularizing the orbit after an aerodynamic inclination change. This is a time-sensitive maneuver, so the thrust requirement would be highest here. After sizing this, the launch team will find it useful to lower the delta-V requirements for the Falcon B5 booster, since the hypersonic upper stage now has more available thrust to generate a larger delta V within the time to apogee.

Much of the author's work has been to coordinate the construction of consistent functions to go into the main script for the hypersonic sub-team. Some group members did not know how to build functions in MATLAB so the author also took care of that.

The 1st workday of the week entailed much coordination between script variables, particularly on nomenclature. The main work in the script was to direct the CAD discipline to initialize the script with the 176 geometric variables in a particular order and grouping according to function. The geometry variable type is denoted as GEOM:

- GEOM: [Planform Area (S_plan), Wetted Area (S_wet), length of vehicle (l), vehicle width (w), trapezoidal base (A_base), body aspect ratio (AR), leading edge sweep (sweepLE), trailing edge sweep (sweepTE)]
- GEOMsub: [Planform Area (S_plansub), Wetted Area (S_wetsub), length of vehicle (l), vehicle width (w), trapezoidal base (A_base), subsonic wing width (b_sub), switchblade wing chord (c_rootsub)]
- GEOMfins: [upper fin area (A_finupp), lower fin area (A_finlow), MAC upper fin (MAC_finupp), MAC lower fin (MAC_finlow), dihedral of the upper fin (Dihed_finupp), dihedral of the lower fin (Dihed_finlow)]
- GEOMextended: [Weight estimation inputs]
- GEOMall: [GEOM; GEOMsub; GEOMfins; GEOMextended] matrix of geometry (preallocate with a zero matrix of a large enough size to avoid matrix dimensional issues)

The author spent much of the available time helping other team members with their respective code and providing guidance on expected deliverables. An example of this is training people on writing functions and ensuring that variables can be condensed and stored as vectors.

G. Week 7

Most of this week was concerned with developing the midterm presentation. The author put forth an outline for the presentation which described disciplinary analysis by flight phase. There was disagreement regarding the format of the presentation, where most in the group preferred the more conventional discipline-by-discipline presentation format. The author elected to go with the group's format since this would be most time-efficient in building the presentation. Additionally, the case was made to make the midterm more technical as a proof of successful analysis to lay down the technical foundation to present the more artistic and managerially-inclined formats at the end of the semester. The author believes that selling an engineer's work in a way which can be received by both the technical and non-technical public is needed for organizational success. However, for the midterm presentation, the disciplinary segregation is most practical, especially since most of the team is not on board with a flight phase approach.

The presentation was intended to take place on 16 July 2018, but the date was postponed by two days since the professor of the design class was ill. The class took this opportunity to run through the presentation and receive constructive criticism from his graduate students. This allowed for further iteration before the midterm presentation date.

The midterm presentation lasted 1 hour and 10 minutes, covering the analysis by discipline. While it was considered to be on the right track to a good final presentation, there were things which needed to be ironed out. For example, the format was too technical and disorganized to be suitable for a general audience.

H. Week 8

This week involved reiterating on our methods and building the presentation format ahead of time. The two chiefs have worked on a new approach which is ordered by flight phases. The agreed upon presentation outline is as follows:

- Introduction
- Business Case
- Mission
- Launch Synthesis
- Launch Flight Phases
    - Ascent
    - Separation
    - Descent
- Hypersonic Synthesis
- Hypersonic Flight Phases
    - Orbital Operations
    - Synergetic Maneuver
    - Hypersonic Descent
    - Subsonic Landing
- Conclusion

Each flight phase will have an "objectives" slide regarding the disciplines, as well as a visual of where the spacecraft is located along the flight profile.

Disciplines are expanding on their own methods and have taken the critique from the midterm presentation to build visuals and better analysis. There was also a need for clearer verification data to validate the disciplinary methods. The timeline for this has been updated accordingly.

After building the business case, the author wanted to look into the need for a second stage on the Falcon Heavy variant to save on production costs. After checking the trajectory code, it was found that the second stage is required due to the fact that most of the velocity change from the launch pad is from the space-grade second stage. Removing it will be a detriment to performance despite the weight of the second stage being removed. In the original configuration, the second stage is responsible for about 5 km/s of velocity change, which is about 65% of the required delta V. This is formalized in a later chapter.

I. Week 9

During this time, trade studies are finalized and the final presentation is being put together and rehearsed according to discipline. This week found a problem with the sizing methodology and how behind the synthesis team is. The author had expected Rashi Jain to complete the sizing while the methodology and business case was developed by the chief. However, the sizing task was much larger than expected and something will have to come together. There seems to be some major misunderstandings between Rashi and the author, particularly regarding the mission profile, and that needs to be resolved as soon as possible.

J. Week 10

The final presentation is given in this week, and the disciplines finish their main deliverables in anticipation of this. The author has prepared by posting flyers (designed by the launch chief) around the UT Arlington campus, mainly for the incoming audience or curious engineering students.

During this week, most of the chiefs' work entails organizing the presentation and its format throughout the previous weekend, while continuously updating the master copy as the presentation is rehearsed. This rehearsal happens throughout the weekend and the morning of presentation day.

The Air Force research division livestreams the presentation throughout for their view, and the attending audience is composed of several dozen family members, friends, and academic colleagues. The chiefs design the presentation such that they introduce each flight phase in order and bring the audience up to speed to allow for context for the disciplinary analysis. The length of the presentation was approximately two hours.

After the presentation, the main priority is to coordinate the construction of a poster and team report.

VIII. Synthesis

A. Sizing

The sizing method used is dependent on the industry capability index (ICI), which is constructed by structural weight, mission requirements, and slenderness. Payload and fuel volume fractions available are also drivers of sizing. This index will size the planform area of the vehicle according to mission requirements.

The ICI is a measure of available technology and is defined such that performance can increase with increasing structural or propulsion capabilities [2]. This is very useful for sizing high-performance aircraft where failure of a concept is often due to a lack of available technology, as it was for many spaceplane concepts after the Apollo program [1]. As shown in the MDA, sizing begins with an understanding of the mission requirements, and technological capability will be built off of 176 data. The ICI is calculated as follows:

Propulsive Index:

$$I_p = \left(\frac{\rho_{ppl}}{W_R - 1}\right) = \left(\frac{\rho_{ppl}}{\frac{W_{TOGW}}{W_{OWE}} - 1}\right) = \left(\frac{1200\ kg/m^3}{\frac{45{,}500\ kg}{9750\ kg} - 1}\right) = 327\ kg/m^3$$

Structural Index:

$$I_{str} = \frac{W_{str}}{S_{wet}} = \frac{8300\ kg}{560\ m^2} = 14.82\ kg/m^2$$

Industrial Capability Index (ICI):

$$ICI = 10 * \frac{I_p}{I_{str}} = 10 * \frac{327\,\frac{kg}{m^3}}{14.82\,\frac{kg}{m^2}} = 22\ m^{-1}$$

To obtain a starting value, the values for the Model-176 are used in this equation, provided by the disciplines. Other geometric values allowed the team to find sizing parameters:

$$\tau = \frac{V_{total}}{S_{planform}^{1.5}} = \frac{107\ m^3}{65.7^{1.5} m^3} = 0.20$$

$$K_w = \frac{S_{wet}}{S_{planform}} = \frac{560\ m^2}{65.7 m^2} = 8.52$$

We now have the ICI for the 1964 Model-176. This provides a starting point for iteration. Producing a carpet plot with variables of planform area versus industrial capacity will now prove useful. It is desirable for the hypersonic vehicle design to be as small and light as possible, since that will drive down launch costs immensely (this is an upper stage so weight reduction here greatly compounds in the rocket equation).

Sizing is conducted from the mission profiles and completed by synthesis engineer Rashi Jain. The two military variants converged to a proper planform size at the time of presentation, but the civilian variant has yet to converge. Fortunately, this variant is not design critical given its low delta-V mission profile and light payload for a geometry and volume budget equivalent to the other two military variants.

B. Disciplinary MATLAB Functions

In order to condense the synthesis script, the disciplines have encapsulated their methods into MATLAB functions. Disciplines have multiple functions called, so user guides are documented in this section. This is an important high-level method which will allow the team to debug any inconsistencies within the hypersonic vehicle's design script.

Performance Functions [15]:

Function Name: subsonic_perf.m
Brief Description:

- Uses basic aero variables and atmospheric properties to calculate the range, optimum glide trajectory, and landing field characteristics for the vehicle (uses Raymer's method, very similar to xb-70 method)

Inputs: subsonic_perf(LDmaxsub,CLsub,CLmaxsub,Ssub,Wsub,hsub);

- LDmaxsub = subsonic maximum lift drag ratio
- CLsub = CL associated with max lift-drag ratio
- CLmaxsub = overall maximum possible subsonic CL
- Ssub = reference area with straight wings deployed
- Wsub = probably vehicle empty weight
- hsub = initial altitude for flight phase

Constants:

- g = 9.81 (gravity in SI)
- atmos function

Outputs: [Range, Vapp, Sg, gammaa] , plot: altitude vs. velocity for optimum glide

- Range: subsonic glide range
- Vapp = approach speed for landing
- Sg = landing field length
- gammaa = approach flight path angle

Function Name: Dvcalc.m
Brief Description:

- This function will input the current vehicle state (altitude, velocity) and calculate the required delta-v's to insert the vehicle into a circular orbit at the desired altitude.

Inputs: Dvcalc(hi,Vi,h_des)

- hi = initial vehicle altitude (usually an output from the Rocket team such as at stage sep.)
- Vi = initial vehicle velocity vector (in plane, so 2d)
- h_des = desired vehicle circular orbit altitude

Constants:

- G = universal gravitational constant (6.67408e-11 m^3 kg^-1 s^-2)
- Re = radius of earth (6371 km)
- me = mass of earth (5.972e24 kg)

Outputs: dVs

- dVs: two-member vector. Each member represents an impulsive delta-v applied at a given point on orbit (first at initial orbit apogee, second at transfer orbit apogee) to achieve the desired orbit. Negative values indicate a retrograde burn instead of prograde.

Function Name: AeroInc.m
Brief Description:

- This function will take vehicle hypersonic aerodynamic properties and current orbit properties and calculate the delta-V required for a 30-degree plane change effected via a synergetic maneuver as well as the altitude and speed regimes that will be encountered during the maneuver

Inputs: AeroInc(W,S,CL,LD,h,sel)

- W: current vehicle weight
- S: hypersonic vehicle reference wing area (NOT frontal area)
- CL: lift coefficient for max hypersonic L/D
- LD: max hypersonic L/D
- h: initial circular orbit altitude
- sel: selector variable, 0 for Hohmann atmospheric insertion, 1 for ballistic (currently ballistic is wonky and not recommended)

Constants:
For this function all the constants are in freedom units, the inputs and outputs are all converted to proper units as necessary

- R0 = radius of earth in feet
- B = atmospheric density exponential approximation coefficient, 1/ft
- u0 = earth skimming orbit velocity (25940 ft/s)
- d0 = sea level air density in lbm/ft^3
- pi

Outputs: dV,V,hret

- dV: vector of four delta-Vs required to execute the maneuver: atmospheric entry orbit insertion, atmospheric exit, transfer to desired final orbit, circularize final orbit
- V: vector of velocities encountered in atmospheric flight, provides max and min velocity.
- Hret = minimum atmospheric flight altitude, the vast majority of the maneuver takes place here

Function Name: Thermmap.m
Brief Description:

- Calculates a sample reentry trajectory based on an important design variable (W/(S*CL)) and plots it on top of a thermal map with user-specified properties.

Inputs: Thermmap(W,S,CL,R0,Tmin,Tmax,Tint,points)

- W: vehicle weight
- S: wing reference area, hypersonic
- CL: re-entry lift coefficient, typically that for L/Dmax
- Tmin: user-specified minimum temperature for thermal map
- Tmax: user-specified maximum temperature for thermal map
- Tint: interval between lines on thermal map
- Points: # of data points for each trajectory and line on thermal map, typically 100

Constants:

- g = 9.81 (gravity in SI)
- atmos function

Outputs: Plot of thermal map. Does not currently output any of the variables (although could be very easily modified to do so)

Aerodynamics Functions [16]:

Function Name: subsonic.m
Brief Description:
- This function will take vehicle geometry, wing loading, and weight from the CAD and synthesis disciplines to calculate subsonic aerodynamics of the vehicle.

Inputs: subsonic(WL_sub, l, A_base, b_sub, S_wetsub, S_refsub)
- WL_sub: Wing Loading after switchblade to calculate C_L and C_Lmax
- l: length of the vehicle
- S_wetsub: wetted area (after switchblade wing deployment)
- S_refsub: Wing/reference area (not the same as S_plan due to switchblade wing)
- b_sub: wing span (after switchblade wing deployment)
- A_base: base area (trapezoid)

Constants:
- e: Oswald's efficiency factor (~.85), subsonic estimation
- pi = 3.1415
- 73 degrees, vehicle sweep angle

Outputs: [C_L, C_D, L_D, L_Dmax], Drag Polar, C_Lalpha,

Function Name: hyper.m
Brief Description:
- This function will take vehicle geometry and weight from CAD and structures team to calculate hypersonic aerodynamics of the vehicle. Flight conditions are determined for various phases. The modified Newtonian method is utilized to determine the aerodynamics of the vehicle.

Inputs: hyper(l, A_base, Sw, Sref, W_e)
- l: length of the vehicle
- W_e: Empty Weight of the vehicle
- S_wet: wetted area
- Sref: Base/Reference area
- A_base: base area

Constants:
- pi = 3.1415
- 73 degrees, vehicle sweep angle
- M= [5.0 25.0]

Outputs: [C_L, C_D, L_D, L_Dmax], Drag Polar, C_Lalpha,

Function Name: neutral_point.m (l)
Brief Description:
- This function will take vehicle geometry, from the CAD to calculate n.p. location of the vehicle.

Inputs: neutral_point (l)
- l: length of the vehicle

Constants:

- M: Mach for given flight conditions

Outputs: [np, np_sub], neutral point locations for each speed regime as a percent of length

Propulsion Functions [35]:

Function Name: PropulsionSizingFunction.m

Brief Description:

- This function will take variables from performance, stability and control, and CAD. It will perform the humble method to determine engine mass, engine length, engine diameter, propellant type, oxidizer to fuel ratio (chosen based upon the maximum specific impulse for specified fuel to oxidizer ratio), vacuum specific impulse as function of O/F, flame temperature as function of O/F, exhaust gas molecular mass as function of O/F, exhaust gas isentropic parameter at the throat as function of O/F, engine cycle (trade study), cooling approach (trade study), chamber pressure, nozzle expansion ratio, pressure of tanks, pressure drops, engine balance, pressure system, propellant masses and volumes. (FOR BIPROPELLANT SYSTEMS)

Inputs: PropulsionSizingFunction(T_req)

- T: Thrust

Constants:

- g0 = 9.81 (gravity in SI)

Outputs: [T2W_eng, D_eng, L_eng, m_eng]

- T2W_eng: thrust to weight ratio for performance
- D_eng: nozzle exit diameter for CAD
- L_eng: length of engine for CAD
- m_eng: mass of engine for W&B and CAD

Stability and Control Functions:

Function Name: StaCon_Function.m
Brief Description:

- This function takes variables from Aerodynamics, Performance, and Geometry, and produces Cma, Cnb, Cyb, Cyl vectors with corresponding Mach regime (Sub/supersonic and hypersonic will be separated into two sets of vectors). In addition, it produces control surface sizing and a Boolean assessment for stability.

Inputs: StaCon_Function(StaConINPUT) (Currently)

- All inputs can be found in order in Stability.xlsx
- StaConINPUT, in order: body length, maximum body width, reference area, maximum CX area, volume, wing root chord, wing span, CG location from tip, yMAC, MAC, NP subsonic (from tip), NP supersonic (from tip), parasitic drag, LE sweep, TE sweep, AR of the body, upper fin area, lower fin area, MAC upper fin, MAC lower fin, dihedral of the upper fin, dihedral of the lower fin

Outputs: [Mstacon, Cma, Cyb, Cnb, Clb, aoaH, Cmah, Cybh, Cnbh, Cybh, Cndr, Cmde, Cydr, Clda], Each output vs M or aoa plots, T/F for long/lat stability

- Mstacon: Flight regime (used for graphs)
- aoaH: Angle of attack regime

- Normal derivative series: Stengel derivative outputs
- h derivative series: Clarke and trimmer outputs
- d derivative series: control powers

CAD and Layout Functions [20]:

Function Name: GeomSizing.m
Brief Description:

- This function will act as the CAD check and is a MATLAB stand-in for the discipline's tools: SOLIDWORS and OpenVSP.

Inputs: myFunction(S_plan, S_wet, GEOM, VOL)

- Planform Area
- Wetted Surface Area
- GEOM: vector with agreed aircraft parameters
- VOL: vector with required disciplinary volumes

Constants:

- none

Outputs: [void], plotter

- Surface Volume Continuum Plot

Structures Functions [18]:

Function Name: materials.m

Brief Description:

- This function will take inputs from aerothermodynamics, material database, and performance with the flight profile

Inputs: materials(Nose, WLeading, Leeward, Windward, VTLeading, VT, HTLeading, HT, YieldM, YieldC, rhoM, rhoC, EM, EC, TempM, TempC, NameM, NameC, TExM,TExC)

- Nose = Max nose temperature
- WLeading: Max temperature at wing leading edge
- Leeward: Max temperature on top of spacecraft
- Windward: Max temperature on bottom of spacecraft (if not separate surfaces yet, leeward = windward)
- VTLeading: Max temperature at vertical tail leading edge
- VT: Max temperature on vertical tail
- HTLeading: Max temperature at horizontal tail leading edge
- HT: Max temperature on horizontal tail
- YieldM: Yield stress of metal materials
- YieldC: Yield stress of ceramic materials
- rhoM: density of metal materials
- rhoC: density of ceramic materials

- EM: Young's Modulus of Metal materials
- EC: Young's Modulus of Ceramic materials
- TempM: max temperature of metal materials
- TempC: max temperature of ceramic materials
- NameM: list of characters for names of metal materials, used to user knows which materials work
- NameC: same as NameM, but for ceramic materials.
- TExM: thermal expansion of material materials
- TExC: thermal expansion of ceramic materials

Outputs: [Mat]

- Mat: vector of useable materials that will go into structures layout, will eventually be just one function

C. Synthesis Script: Product Disciplinary Development

The synthesis script will be written by the author as a contribution to the synthesis discipline, and will take the database as an input, as well as a spreadsheet with user-guided parameters to fulfill the mission. The script will import a database first to reduce computation time (this section can be commented out once a mature database has been imported). The script written to date can be found in section B of the appendix.

Early in the project, the stability script was developed for the supersonic and subsonic speed regimes, heavily modified from the previous semesters' script written by the author. Additionally, load cases were written out in pseudocode as part of structure's literature research. This function was the first to be integrated into the main synthesis script. Plotting features are suppressed and called outside the function in the main synthesis script.

A weight/balance script was written, modified from a CAD engineer's old MATLAB code. There has been significant progress in developing performance trajectory plots. The script for the hypersonic trajectory turn is still in development. CAD has also begun their script to build shapes and determine geometric properties from sizing parameters and user shape selection. This will aid in iteration, volume comparison, and trade studies.

Aerodynamics is also implementing methods to find angle of attack as a function of Mach number by keeping the craft at maximum lift-to-drag ratio as computed by the speed regime. The author aided in the development of this script, and the resulting plots are shown below. These plots are used as inputs to stability assessment.

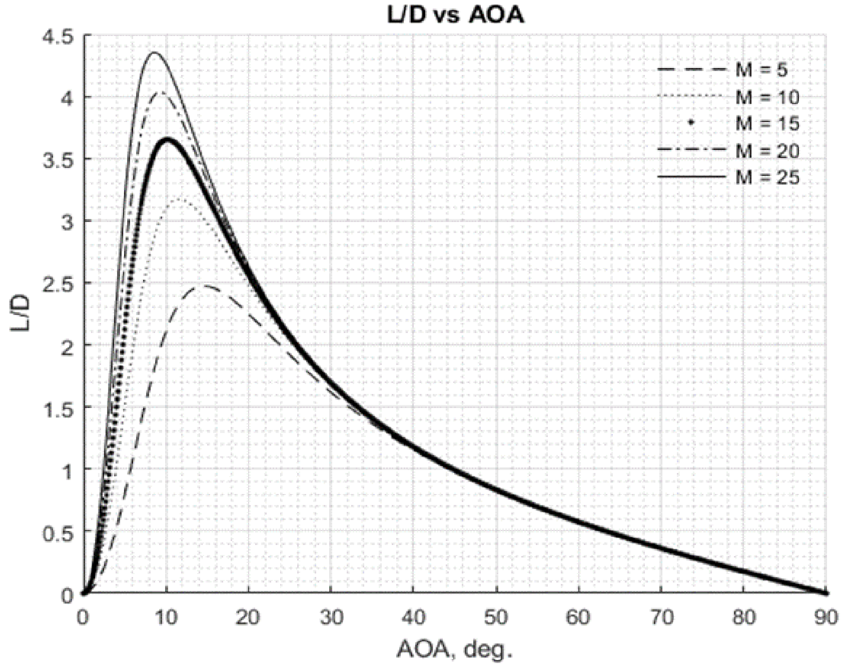


Figure 72. L/D vs AOA for various Mach Numbers *[16]*

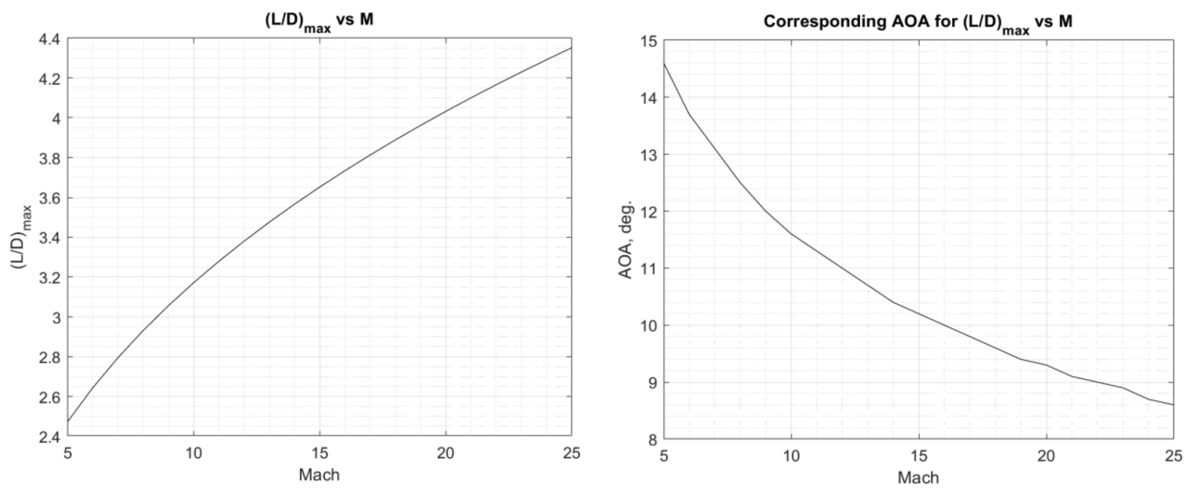


Figure 73. Maximum L/D and Associated Angle of Attack as Function of Mach Number *[16]*

The organization of the weights is recorded by the author to ensure that there is consistency between the weight calculations and no double-counting.

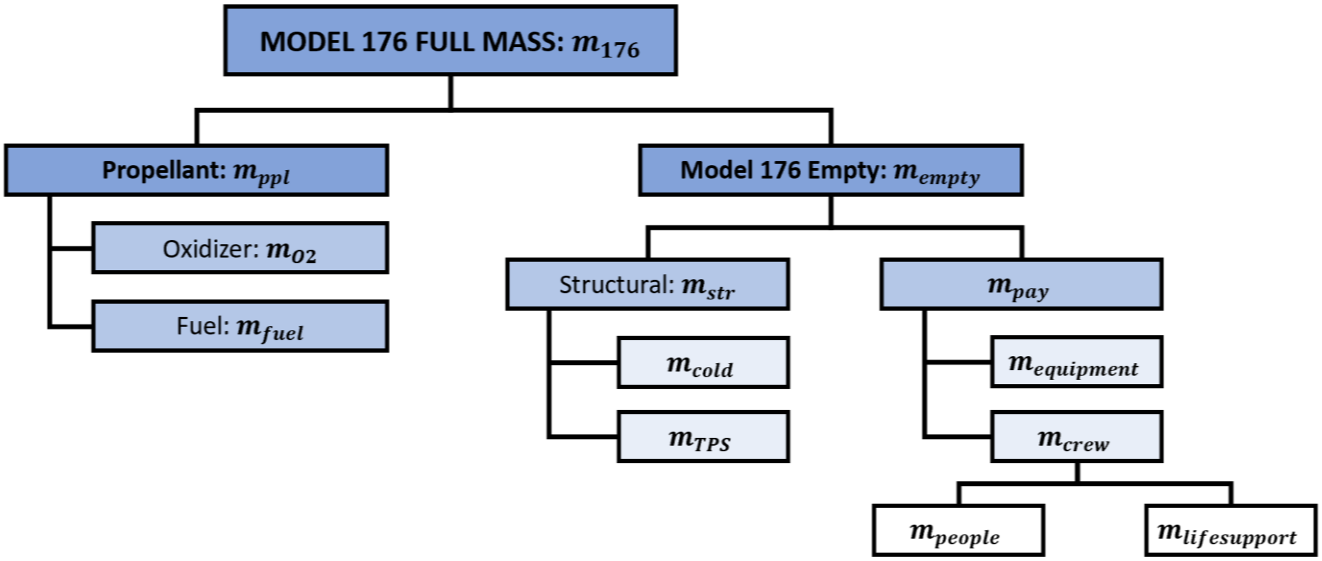


Figure 74. Upper Stage Mass Breakdown with Associated Variable Names

The weights shown above represent different methods for obtaining them. The payload weight is straightforwardly provided by the mission, where the crew needed and their associated life support system is easily computed in the mission chapter.

The CAD script is currently under development but will not be ready for the midterm as there is a huge difference between the methods used and the historical values, particularly when computing tau.

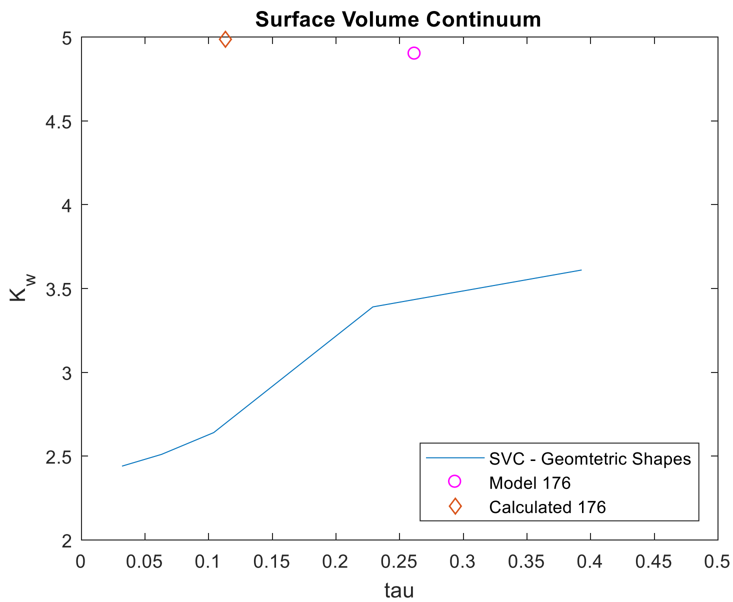


Figure 75. Initial Results from CAD Script *[20]*

D. Synthesis Script: Product Description

The Nassi-Schneiderman diagrams in the methodology chapter describe the process of analysis along with parallel processes, but the author believes it is also necessary to show code structure as it pertains to the various MATLAB subroutines developed by the individual disciplines. This more clearly displays the code hierarchy and the order they appear in the script. This is similarly shown in the table of contents within the script.

- Data Import
- Input Planet Geography
- Mission Trades
- Flight Domain Determination
- Hypersonic Analysis
    - Determine Requirements
    - Flight Profile and Disciplinary Analysis
    - Finalize Layout
    - Cost Evaluation
- Launch Analysis
- Vehicle Capability Determination
- Visualize
- Final Cost Evaluation
- Data Export

The flight profile and Disciplinary Analysis under "Hypersonic Analysis" is further broken down into the respective sub-team's analysis by the following structure:

- Mission Requirements
- Flight Profile
    - Flight Phase Bounds
    - Select Critical Flight Phase
- Trajectory Analysis
    - Critical Flight Phase
- Geometry Definition
- Propulsion
- Aerothermal
    - Aerodynamics
    - Heating
- Sizing
    - Geometry Estimate
    - Empty Weight Estimate
    - Volume Required Estimate
- Weight Estimation
- Structures
- Stability and Control
    - Stengel Method
    - Clarke/Trimmer Method
    - Wei Method (RCS Sizing)
- Geometry Redefinition
- Trajectory Analysis (Iterated)
- Comparison (Sizing)
- Cost Estimation

Below is a collection of variables which keep track of and standardize some common variables found in many disciplinary scripts.

Weights:		
Variable	**Name**	**Units**
Empty	m_empty	kg
Max Take-off	m_TO	kg
Structural Weight	m_str	kg
Mission Equipment	m_equipment	kg
Payload	m_pay	kg
Crew/lifesupport	m_crew	kg
Propellant	m_ppl	kg

Trajectory:		
Variable	**Name**	**Units**
Delta V	dVtotal	m/s
Delta V's Vector	dVs	m/s
Critical Burn delta V	dV_crit	m/s
Field Length	Sg	m
Approach Velocity	V_approach	m/s
Critical Burn Time	time_burn_crit	sec

Geometry:		
Variable	**Name**	**Units**
GEOM:		
Planform Area	S_plan	m^2
Wetted Area	S_wet	m^2
Vehicle Length	l	m
Vehicle Width	w	m
Trapezoid Base Area	A_base	m^2
Body Aspect Ratio	AR	
Leading Edge Sweep	sweepLE	deg
Trailing Edge Sweep	sweepTE	deg
Side profile area	S_side	m^2
GEOMsub:		
Planform Area	S_plansub	m^2
Wetted Area	S_wetsub	m^2
Vehicle Length	l	m
Vehicle Width	w	m
Trapezoid Base Area	A_base	m^2
Subsonic Wing Width	b_sub	m
Subsonic Wing Chord	c_rootsub	m
GEOMfins:		
Upper Fin Area	A_finupp	m^2
Lower Fin Area	A_finlow	m^2
MAC Upper Fin	MAC_finupp	m
MAC Lower Fin	MAC_finlow	m
Dihedral of Upper Fin	Dihed_finupp	deg
Dihedral of Lower Fin	Dihed_finlow	deg
GEOMextended:		

Propulsion:		
Variable	**Name**	**Units**
Thrust requirement	T_req	N
Engine Length	L_eng	m
Engine Diameter	D_eng	m

Aerodynamics:		
Variable	**Name**	**Units**
Lift coefficient subsonic	CL_sub	
Lift-to-Drag Ratio	L2D	
Zero-lift Drag	CD0	

Figure 76. Common Variable Names in Synthesis Script

IX. Cost Analysis

The procedure for identifying costs is outlined in the literature review, and the steps for actual analysis and results are explained section by section in this chapter. These methods were developed by NASA in 2015 and is relatively straightforward. The simplicity of this cost analysis does not negate its importance, as the chief's responsibility to make a project economically viable is complemented with the disciplinary deliverable of producing a physically realizable conceptual design.

A. Understanding the Project

Understanding the project can be described by the elements described in the cost analysis section of the literature review. The author attempts to answer these questions to begin the cost estimate analysis below.

Data:

- Type of data needed: historical trends, cost estimate methodologies, professor guidance, market news
- Data availability: most of NASA data is public domain, recently unclassified 176 documents
- Outside Organization Co-operation: NASA documents are freely available, faculty is helpful and disposed to consult with team, SpaceX data is mostly proprietary and limited
- Non-disclosure agreements: none

Resources:

- People required: mainly the lead chief, though the support chief should analyze respective sub-team
- People available: both chiefs
- Budget required: 3 – 5 days (estimated in days to account for lack of a budget in an academic setting)
- Budget Available: 11 weeks

Expectations:

- Expected Outcome: inform the business case for conceptual design, allocate proper funds for further design work and program support
- Customer Expectation: a proper assessment of cost versus expected revenue from market analysis
- Team Expectation: a framework and constraint for economic feasibility with the same stringency as physical and technical feasibility
- Agency-wide Expectations: a successful program from the above two expectations (where market demands meet feasibility for a viable business model)

Schedule:

- Time to collect required data: 11 weeks
- Resources to meet time constraint: disciplinary input on costing, such as engine specifications, R & D required for TPS, Stability & Control system, ICI required, etc.

B. Work Breakdown Structure

Below is the preliminary work break down structure (WBS) for engineering requirements based on the team activities experienced to date and anticipated activities. This is a Level 3 WBS, where at this step the budget is undefined. The WBS will be expanded upwardly to a Level 4 to include the entire project as a whole after the engineering work. This will include operations, launch, construction, and miscellaneous expenses.

The difference between the engineering work and the other activities is that the engineering will represent the initial program cost, and the other activities are ongoing expenses as a function of usage, market size, and time in use. These will cut into the yearly profit or budget, while engineering is the large upfront cost. Additionally, the activities involved with the engineering work category are being conducted in this project at the conceptual level. There is a direct relationship between the technical challenges, solutions, and hardware specifications and the upfront cost of this program.

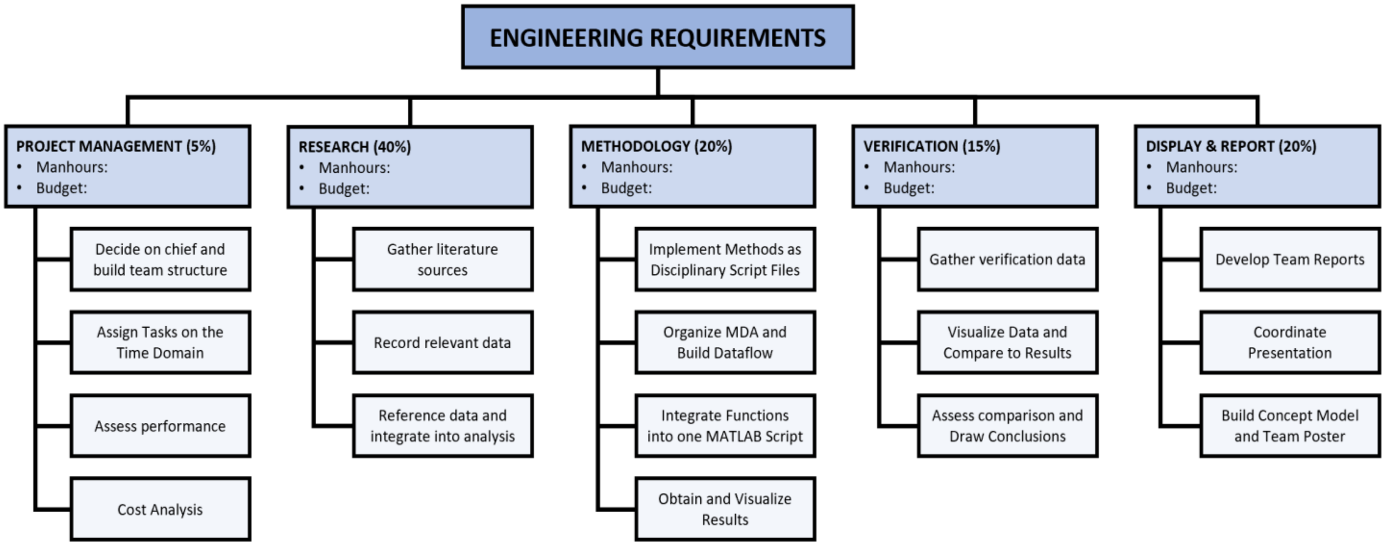


Figure 77. Level 3 Work Breakdown Structure for Engineering

Next to each Level 2 WBS element, there is an estimated percentage of productive time invested. This is subject to change but should provide a guideline for future budget and manhour estimations.

C. Define Project Technical Description

This section will define the aspects of this project which will be analyzed. The starting costs are the engineering costs to develop a craft in the following disciplines:

- Adapt two SpaceX launch platforms
- Design three mission layouts of the same concept upper stage design
  - Interior systems and equipment integration
  - Switchblade wing system
- Upper Stage Engine development
- Stability and Control systems
- Hypersonic Wind-tunnel Testing

By adapting an existing launch system, the costs will be drastically reduced. By integrating with existing vehicles and programs, a project manager can avoid falling into the problem of piecemeal decision making and keep his or her agency/organization within a reasonable budget [37].

D. Develop Ground Rules

One of the expected customer deliverables is a fully developed engine, which certainly adds to costs, as is the case with conventional aircraft. Most organizations elect to use off-the-shelf propulsion systems given its complexity. However, a fully designed propulsion system at the conceptual level will provide insight into what an ideal propulsion system would look like, where such an activity will aid in engine selection should the program require reductions in development expenditure or meeting schedule in later stages of design.

E. Development Costs

The engineering development costs of a program can be estimated by the dry weight of the spacecraft. This is from the combination of the various systems and subsystems associated with a craft proportionate to its scale. This is one of the reasons to choose the smallest planform area possible for a spaceplane. The author has elected to only add the *new* dry weight to avoid double counting the cost of identical systems across different variants.

The costs are first approximated in engineering man-years, which cost the program approximately 330,000 USD per man-year (2018 dollars) [38], assumed to be covering engineering salary and its associated overhead. The minimum number of man-years required (at 30% of historical business-as-usual) for developing dry weight minus engine development is given by the trend line equation [38]:

$$MY = 1.314 * W_{dry}^{0.628}$$

The associated dry weight is taken from reference [1].

SYSTEM WEIGHTS (by variant) [kg]			
System	**Reconnaissance**	**Command & Control**	**Point-to-Point**
Mission Payload	1,156	5,332	2,126
Crew and Life Support	266	266 (duplicate)	266 (duplicate)
Equipment	1,469	1,528	1,678
Structural System	6,098	6,098 (duplicate)	6,098 (duplicate)
Contingency	138	241	174
Totals (skip duplicates):	**9,717**	**7,573**	**4,426**

Figure 78. System Weights by Variant (Engines not Included) *[38]*

The development costs minus the engine are then determined by these weights, where each man-year costs 330,000 USD in 2018.

$$(Cost)_{dev} - ((Cost)_{dev})_{eng} = 330{,}000 * 1.314 * [9717^{0.628} + 7573^{0.628} + 4426^{0.628}] = \mathbf{341\ million\ USD}$$

The development costs for a single engine designed for orbital maneuvering and re-use capability are approximated with a similar historical engine, the LR-91 [17]. For a program of 5 years at the flight rate given in the business case, the amortization of the development costs are approximately 8,000 USD per flight.

X. ABET Outcomes

A. Outcome C: Design System or Process to Meet Needs

The main task for the chief was to derive and define a set of mission requirements from the design intent of the Model 176 to build a mission profile and request deliverables from team members. This required a literature review (see associated chapter) to find which military and market needs were to be addressed, and which vehicle parameters were required to achieve the missions required to address those needs.

To successfully reverse-engineer the Model 176, the author managed the team along a timeline, both near-term (weekly goals) and long-term (entire semester) to complete given tasks. This was determined by both the author and the respective disciplines from their literature search on what the important deliverables were. Towards the beginning, the weekly goals guided the starting activities, but as the semester and disciplines progressed in their literature research, activities began to be dominated by goals determined by disciplinary literature research.

The processes developed by synthesis and the author is outlined in the methodology chapter. It includes a parallel split between the two sub-teams: launch vehicle and hypersonic vehicles. These sub-teams are further split into two parallel processes per sub-team: sizing and disciplinary work. These processes, guided by mission and disciplinary requirements, are compared against the volume and weight budgets tabulated by CAD.

B. Outcome D: Ability to Function on Multidisciplinary Teams

The author has taken the responsibility of chief engineer, whose main task is to create and guide the disciplinary activities into a cohesive multi-disciplinary design process. Again, this is demonstrated in the methodology chapter. Following this methodology requires that all team members pull their weight, though that cannot be expected to be the case in the real world. This section as well as the team management chapter describe the lessons learned in dealing with these issues.

Dealing with lapses in performance was first done by example. As should be expected at the very minimum, the chief was never late to a meeting and never skipped a day of class (this was also demonstrated by the supporting chief). In addition to this, the chiefs could not have a lapse in performance, especially since their deliverables were always under the watchful eye of the class faculty head and graduate teaching assistant. The threat of being deposed ensured that the chief position could work effectively and not hold the team back. The unofficial demands imposed on the author has certainly provided the pressure needed to produce a quality product and manage time effectively. This is as much of a learned skill as was the becoming acquainted with the technical aspects.

Secondly, few disciplinary teams worked as a team of one, so that if one member was absent or lagging behind, there was usually a member to pick up the slack. The expectations of a timeline could keep teams on track with their deliverables as well. Attendance was also found to be a determinant of performance, so the author kept track of this per sub-team.

As a last resort, disciplines which failed to provide deliverables were given a grace period, where the author would notify the graduate teaching assistant of a lapse in performance. Fortunately, non-performance did not go past this stage, though time-delays were usually dealt with by the dependent disciplines by using substitute data or methods. There really was no excuse to point fingers at a discipline for not providing deliverables, since all teams proved to be competent in making due with a slow-down.

C. Outcome F: Understand Professional and Ethical Responsibility

The number one ethical responsibility of an aerospace engineer is to provide a safe design. The two disciplines which are the critical here are structures and stability/control. Their unique analysis is situated at the end of the multi-disciplinary analysis so that they can ensure that the human crew can be safe and stable throughout the flight regime. If there are any dangers posed by the trajectory and aerothermal loads, these disciplines will document their analysis methodology and report results honestly.

Grading and managerial incentives are aligned such that reporting a failure in results is not punished. It is important not to ignore this aspect of the capstone project, as the environment in which the disciplines are expected to perform does have an influence in the daily decisions and reporting of analysis and results. Assumptions are recorded and presented, where historical validation is a requirement for both the reports and the presentation.

Referencing the work of source documents and figures is a base requirement for producing this report. This includes referencing the author’s own works and describing why it is used. The more which is written on the rationale of the author, the more transparent the report is. This transparency is what allows for future preliminary design engineers to go along a path which allows for a working design, as well as to expand on the conceptual analysis under the correct assumptions and methodology.

D. Outcome G: Ability to Communicate Effectively

The overall format of this report serves as a communication tool by its hand-made figures, tables, and explanations. The most useful of these, along with those made by other team members, were then put together in the midterm and final presentations. Any figure which is not referenced was constructed by the author to visually communicate key concepts in this project.

One of the goals of the presentation was to provide ample analysis for the specialist, as the flight phase story was told from the relevant disciplines' perspective. Additionally, by creating the overall presentation according to the order of flight phase, a layman audience and decision maker can follow the mission profile along all three variants. One of the presentation roles of the author was to interject between each flight phase to address the main idea, as well as explain some concepts to the layman. This was heavily aided by presentation visuals, many of which can be found in this report.

E. Outcome H: Understand Impact of Engineering Solutions in Global and Societal Context

The entire business case rests on the ability to provide a solution to address global market needs in addition to United States military objectives. Aspects of the business case such as competition analysis acknowledge the wider aerospace industry and their capabilities. Additionally, the market analysis conducted provides a numerical basis for the supposed global need.

The mission profiles and historical context also look into the wider world to guide the team methodology.

F. Outcome I: Recognize the Need and Ability to Engage in Lifelong Learning

Much of the literature search completed was done throughout the semester, and as represented by the semester timeline, was continued throughout the summer project. If this is representative of industry, there is certainly a need for lifelong learning.

Hypersonic flight and space vehicle design is not formally taught in the aerospace degree plan, and because of this, team members needed to catch up in their literature search to successfully fulfill deliverables accurately. If the project's subject had been about helicopters, students would be in the same situation. An engineer's performance is not only assessed by their knowledge coming into a position, but by their ability to quickly become situated in a specialized set of knowledge they could not possibly have known before entering that position.

XI. Conclusion

A. Closing Remarks

This report summarizes the activities and results of the lead chief engineer for the first half of the semester dedicated to the MAE 4351 Senior Capstone Project. The project assignment was to develop the multi-disciplinary methodology to reverse engineer the Douglas Model-176 and an associated SpaceX launch system. The team was split into two functional sub-teams between the upper-stage Model-176 and its launch system.

After an introductory section, literature review was conducted for all the disciplines, particularly in the author-specific roles regarding management, the business case, synthesis, and costing. This was brought together from a wide variety of sources: aerospace conceptual design literature, the professor's recommended text, teammates' input, and reports found online.

In accordance with the presentation format, the business case for the combined vehicle was developed starting with market research. This market research provided a basis to run an operation supporting space tourism and low-Earth orbit access at a high flight frequency at a rate of once an hour, sustained by a 48-hour turn-around time for a civilian fleet size of fifty. This flight frequency allows for the amortization of infrastructure and development costs to allow for an affordable ticket price of 73k USD per seat. This civilian market will support a military capability which will not be as frequent.

The three missions are designed around the operations, where the civilian version is simply designed to carry passengers to a 200km orbit. The military variant will be equipped to enter a 600km orbit and stay there for up to a week. Both of these variants are carried by the Falcon Block 5 launch vehicle. The reconnaissance vehicle was designed to conduct its mission at a 400km orbit and perform a synergetic maneuver to increase its inclination-change capability. Various trades were examined to determine the best mission profile for meeting geopolitical goals and achieving a realizable fuel requirement by the use of aerodynamic performance.

The methodology was inspired by disciplinary literature review and underwent several iterations. Starting with a flow diagram, the author attempted to illustrate how the vehicle may be reverse engineered per flight phase. After revision and iteration with the synthesis team and constructive critique from Dr. Chudoba, the multi-disciplinary analysis underwent several drastic changes. The flow chart was thrown out in favor of the Nassi-Schneiderman format, which was designed as a two-tiered MDA to improve robustness at the higher level. Lower levels would iterate or could change with respect to each sub-team's implementation into actual MATLAB code.

The history of program development was added around week 4 to document the activities of the lead chief engineer that do not necessarily produce tangible results directly. This also doubled as a more in-depth look at the weekly issues and lessons learned, which may prove valuable in the future.

B. Future Work

Future work entails exploring additional launch systems to reduce flight costs. The currently existing launch vehicles are oversized for the most frequent variant: the civilian point-to-point. However, the development of a specialized launch system may not be the right move to minimize program costs.

The author recommends the exploration of an additional civilian variant redesigned to carry cargo. High-value goods which are time-critical may have a significant market to further sustain a large flight rate by slightly modifying the existing civilian variant. The two variants will have the same trajectory capability, but one will not need the rigorous certification associated with transporting human passengers.

Appendix

A. Databases

The following tables show the databases for the disciplines concerned. These are imported into the synthesis script to make judgements on the given data and plot trends. The propulsion database is shown below, which is a compendium of historical engine data.

	Basic and Historical Data						Propellant Data			Configuration	
Engine	Predecessor	Origin	First Flight	Designer	Application	Vehicle	Fuel Type	Propellant	Cycle	Chamber	Nozzle Ratio
RD-180	RD-170	USSR	2000	NPO Energomash	Booster	US Atlas V launch vehicle	Liquid	LOX/RP-1	Staged Combustion	2	36.87
Rocketdyne F-1		US		Rocketdyne		Saturn V	Liquid	LOX/RP-1	Gas-generator		16
Merlin 1D		US		SpaceX	Booster/Upper Stage Engine	Falcon 9/Falcon Heavy	Liquid	LOX/RP-1	Gas-generator		
Space Shuttle main engine	HG-3	US	1981	Rocketdyne		Space Shuttle/Space Launch System	Liquid	LOX/LH2	Staged Combustion		69
Rocketdyne J-2	HG-3/J-2X	US	1966	MSFC/Rocketdyne	Upper Stage Engine	Saturn IB/Saturn V	Liquid	LOX/LH2	Gas-generator		27.5
RD-250		USSR	1965	OKB-456		R-36/Tsyklon-2/Tsyklon-3	Liquid	N_2O_4/UDMH	Oxidizer Rich Staged Combustion	2	
BE-4	BE-3	US		Blue Origin			Liquid	Oxygen/Methane			
NK-33	NK-15/NK-15V	USSR	1970	Kuznetsov Design Bureau	First/Second Stage Engine		Liquid	LOX/Kerosene	Staged Combustion		
RL10		US	1962	Pratt & Whitney/MSFC	Upper Stage Engine	Atlas/Titan/Delta IV/Saturn I	Liquid	LOX/LH2	Expander		84 or 280
RD-843	RD-869	Ukraine	2012	Yuzhnoye Design Bureau	Upper Stage Engine	AVUM	Liquid	N_2O_4/UDMH	Pressure Fed	1	
RD-270	RD-270M	USSR	1969	V. Glushko		UR-700/UR-900	Liquid	N_2O_4/UDMH	Full-flow Staged Combustion		
RD-855		USSR	1965	Yuzhnoye Design Bureau	Vernier Engine	R-36/Tsyklon-2/Tsyklon-3	Liquid	N_2O_4/UDMH	Gas-generator	4	
Vinci	HM-7B	France		Airbus Safran Launchers	Booster/Upper Stage Engine		Liquid	LOX/LH2	Expander	1	240
RD-856		USSR	1965	Yuzhnoye Design Bureau	Vernier Engine	R-36/Tsyklon-2/Tsyklon-3	Liquid	N_2O_4/UDMH	Gas-generator	4	
RS-68		US		Rocketdyne/Pratt & Whitney/Aerojet	First Stage	Delta IV	Liquid	LOX/LH2	Gas-generator		21.5
BE-3		US	2015	Blue Origin		New Shepard	Liquid	LOX/LH2	Combustion Tap-off		
RD-253		USSR	1965	Energomash/V. Glushko	First Stage Booster	Proton	Liquid	N_2O_4/UDMH	Staged Combustion	1	26.2
Vulcain		France	1996	Snecma	Main Engine	Ariane 5	Liquid	LOX/LH2	Gas-generator		45.1
RD-8		USSR	1985	Yuzhnoye Design Bureau	Second Stage Vernier	Zenit	Liquid	LOX/RG-1	Staged Combustion	4	
Aestus		Germany	1997	Ottobrunn Space Propulsion Centre	Upper Stage Engine	ESA	Liquid	N_2O_4/MMH	Pressure Fed		84
RD-0237	RD-0225	USSR	1973	OKB-154	Gimbaling Engine	UR-100N/Strela	Liquid	N_2O_4/UDMH	Pressure Fed	1	
TR-201	LMDE	US	1972		Upper Stage/Spacecraft	Delta-P	Liquid	N_2O_4/Aerozine 50	Pressure Fed	1	
RD-864		USSR	1977	Yuzhnoye Design Bureau	Upper Stage Engine	R-36M/Dnepr	Liquid	N_2O_4/UDMH	Gas-generator	4	
AJ10-118K		US	1957		Upper Stage/Spacecraft	Delta-K/Orion/Transtage	Liquid	N_2O_4/Aerozine 50	Pressure Fed	1	
RD-170		USSR	1985		Main Engine	Energia	Liquid	LOX/RG-1	Oxidizer Rich Staged Combustion	4	36.87
LR-87-3		US			Main Engine		Liquid	LOX/RP-1	Gas-generator		
RD-0146		USSR	2001	KBKhA Design Bureau	Upper Stage Engine		Liquid	LOX/LH2	Expander	1	
HM7B	HM4	France	1979	Snecma	Upper Stage Engine	Ariane 5	Liquid	LOX/LH2	Gas-generator	1	83.1
RD-801		Ukraine		Yuzhnoye Design Bureau	First Stage		Liquid	LOX/RG-1	Staged Combustion	1	
LR91	LR-91-9	US	1964	Aerojet	Upper Stage Engine	Titan III/Titan IV	Liquid	N_2O_4/Aerozine 50	Gas-generator	1	49.2
YF-20B		China	1995	Academy of Aerospace Liquid Propulsion Technology		Feng Bao 1/Long March 2/3/4	Liquid	N_2O_4/UDMH	Gas-generator	1	12.69
YF-40B		China	1995	Academy of Aerospace Liquid Propulsion Technology		Long March 1D/Long March 4	Liquid	N_2O_4/UDMH	Gas-generator	2	55
YF-1B	C2.1100	China	1958	Academy of Aerospace Liquid Propulsion Technology		DF-3A/DF-4/Long March 1	Liquid	N_2O_4/UDMH	Gas-generator	1	10
RD-810		Ukraine		Yuzhnoye Design Bureau	First Stage		Liquid	LOX/RG-1	Staged Combustion	1	
RD-119		USSR	1960	Energomash/V. Glushko		Kosmos-2	Liquid	LOX/UDMH	Gas-generator	1	1350
RD-191		USSR	2001	NPO Energomash	Main Engine		Liquid	LOX/RP-1	Oxidizer Rich Staged Combustion		37
RD-0255 = RD-0256 + RD-0257											
RD-0256		USSR	1981	OKB-154		R-36M/Dnepr	Liquid	LOX/UDMH	Oxidizer Rich Staged Combustion	1	
RD-0257	RD-0230	USSR	1981	OKB-154		R-36M/Dnepr	Liquid	N_2O_4/UDMH	Gas-generator	4	
RD-0233		USSR	1973	OKB-154		UR-100N/Rokot/Strela	Liquid	N_2O_4/UDMH	Oxidizer Rich Staged Combustion	1	
RD-0243 = RD-0244 + RD-0245											
RD-0244		USSR	1981	OKB-154		RSM-54/Shtil	Liquid	N_2O_4/UDMH	Oxidizer Rich Staged Combustion	1	
RD-0245		USSR	1981	OKB-154		RSM-54/Shtil	Liquid	N_2O_4/UDMH	Gas-generator	4	14
RD-0216		USSR	1965	OKB-154	ICBM Propulsion	UR-100	Liquid	N_2O_4/UDMH	Oxidizer Rich Staged Combustion	1	
RD-0236		USSR	1973	OKB-154	Vernier Engine	UR-100N/Rokot/Strela	Liquid	N_2O_4/UDMH	Gas-generator	4	
XLR81	Bell 8081	US	1963		Upper Stage Engine	Thor/Thorad/Atlas/Titan	Liquid	RFNA/UDMH	Gas-generator	1	45
S5.92		USSR	1988	KB KhIMMASH	Upper Stage Engine	Soyuz/Zenit	Liquid	N_2O_4/UDMH	Gas-generator	1	
S5.98M		USSR	1990	KB KhIMMASH	Upper Stage Engine	Proton-M/Rokot	Liquid	N_2O_4/UDMH	Gas-generator	1	
RS-27A	RS-27	US	1989	Rocketdyne	Booster	Delta 7000	Liquid	LOX/RP-1	Gas-generator	1	
KVD-1		USSR	2001	KB KhIMMASH	Upper Stage Engine	GSLV Mk 1	Liquid	LOX/LH2	Staged Combustion	1	
Rocketdyne H-1		US	1963		Booster	Saturn 1	Liquid	LOX/RP-1	Gas-generator	1	
RD-0120		USSR	1987	OKB-154	Sustainer Engine	Energia	Liquid	LOX/LH2	Staged Combustion	1	85.7
RD-193	RD-191	USSR	2011	NPO Energomash	Main Engine		Liquid	LOX/RP-1	Oxidizer Rich Staged Combustion	1	
RD-107		USSR		OKB-456	Booster	R-7 Family	Liquid	LOX/Kerosene	Gas-generator		
RD-0124 (14D23)	RD-0110	USSR	2006	KBKhA Design Bureau	Upper Stage Engine	Soyuz-2	Liquid	LOX/RG-1	Staged Combustion	4	

Figure 79. Propulsion Database (Part A)

Performance							Dimensions					
Thrust$_{vac}$ (lbf	Thrust$_{S.L.}$ (lbf	T/W	Chamber Pressure (ps	Specific Impulse$_{vac}$ (	Specific Impulse$_{S.L.}$ (	Burn Time (s)	Length (ft)	Diameter (ft)	Dry Weight (lbf)	O/F	Flame Temp	Notes
930000	860000	78.44	3870	338	311	270	11.66666667	10.33333333	12080			
1746000	1522000	94.1	1015	304	263	165	18.5	12.2	18500	2.27		
205000	190000	179.8	1410	311	282			4.1	1030			
512300	418000	73.1	2994	452.3	366		14	8	7775			
232250	109302	73.18	763	421	200	500	11.1	6.8	3942	5.5		
198000	177000		1208	301	270				1737	2.6		
	550000		1950									
380000	340000	137	2151	331	297		12	6.583333333	2730			
25000				450 to 465.5		700	13.6	7	611			
551			296	315.5		667			35.1			
1510000	1410000	189.91	3790	322	301		16.16666667	11	7430			
74000	64200		952.9	292	254	127	5.75	12.5	710			
40470			882	465			13.8	7	1213			
12190			1038.5	280.5		163	2.916666667	11	248			
	705000	47.4	1488	412			17.1	8	14870			
110000												
370000	330000	156.2	2130	316	285		9.8	4.916666667	2380	2.67		
256300			1500	431			10	5.8	2900			
17640			1123.6	342		1100	4.916666667	13.33333333	840			
6654			160	324		1100	7.2	4.3	245			
1100				200								
9419		31.4	101.5	301			7.44	4.52	249			
4500			590	309		600	13.16666667	4.666666667	439	1.8		
9824			131	319				2.75	220			
1777000	1631000	75	3556	337	309	150	13.33333333	12.5	23700			
164785	145451		855.7	290	256		12.6	3.74	1850			
15400			860	470			11.67	6.416666667				
13980			536.6	444.6			6.59	3.25	364	5		
300000	269000		2600	336	300.7	200			3590	2.65		
105000	52300	80.85	860	316	160	247	9.25	5.333333333	1299	1.86		
183500	164400		1030	289	259	160		2.75	1571			
23000			670	303		412	3.916666667	2.083333333	183			
68300	61900		1030	267.4	242.5	140		1.833333333				
473000	422000		2600	335.5	299				6200			
24000	14700		1150	352	220	260	7.083333333	3.358333333	371	1.5		
470000	432000	89	3740	337	310.7	325	13.33333333	4.75	5050	2.6		
170000										2.6		
120000			2970	310	285	121						
153000			3990	310	280	74						
47000			2130	300	280	79						
49000			2520	313								
3540				293								
16000			506	393		365	83.2	35.5	296	2.55		
4410			1394	327		2000	3.375	2.75	165			
4410			1420	328.6		3200	3.75	3.108333333	209	2		
236993.6	200102	102.47	700	302	255	265	12.4	5.58	2528			
15600			810	462		800	7	5.166666667	622	6		
	205000	102.47	700	289	255	155	8.8	4.9	2200			
440900	343000	43.95	3180	455	353.2	500	14.91666667	7.916666667	7610	6		
469000	430000	103		337.5	311.2		9.9	6.89	4189			
220000	180000			313	256				2620			
66200		52.5	2280	359		300	5.166666667	7.833333333	1261			

Figure 80. Propulsion Database (Part B)

B. MATLAB Code

Below is the class script written by the author.

```
%%
%             ================================================
% ========================================================================
% ------------------------------------------------------------------------
% --------------   ASCENSION AEROSPACE DESIGN SCRIPT   -------------------
% ------------------------------------------------------------------------
% ========================================================================
%             ================================================

%{
WRITTEN BY:
Leonardo Pinero
%}

%{
CODE DESCRIPTION:
The following script will synthesize the two main sub-team scripts: one for
the Hypersonic Lifting Body Sub-team, and another for the Launch Sub-team.
The code takes mission and geographic parameters as inputs, and should
output a conceptual design of a combined vehicle reversed engineered from
the SpaceX launch vehicles and the McDonnel Model-176.
%}

%{
TABLE OF CONTENTS:
    Data Import
    Input Planet Geography
    Mission Trades
    Flight Domain Determination
    Hypersonic Analysis
        Determine L/D_req
        Flight Profile and Disciplinary Analysis
        Finalize Layout
        Cost Evaluation
    Launch Analysis
        ...
        ...
    Vehicle Capability Determination
    Visualize
    Final Cost Evaluation
%}



%%
% ================================================
%                  DATA IMPORT
% ================================================
%{
This section will preload all the outside variables. It can be commented
out to greatly save on computation time for further iterations.
%}
```

```
%

% User Interface:
Planet = xlsread('DesignMAE4351INPUT.xlsx', 'Planet', );
Mission = xlsread('DesignMAE4351INPUT.xlsx', 'Mission', );
Trade = xlsread('DesignMAE4351INPUT.xlsx', 'Trade', );

% Technical Database:
DATA = xlsread('DesignMAE4351DATA.xlsx', 'Eng', );


%

%%
% ================================================
%              INPUT PLANET GEOGRAPHY
% ================================================
% Written by: Leonardo Pinero
%{
inputs: Planet Data
outputs: Coordinates and flight path geometry
%}
Re = 6371000;  % planet radius [m]
g0 = 9.807;  % surface gravitational acceleration [m/s^2]

% ATMOSPHERIC PROPERTIES
% ------------------------------
T0 = 240; % average atmospheric temperature [K]

% composition:
N2  = 0.78;  % nitrogen
O2  = 0.21;  % oxygen
H2  = 0.00;  % hydrogen
CO2 = 0.00;  % carbon dioxide
Ar2 = 0.01;  % argon
MOL = (N2*28.04 + O2*32 + H2*2.02 + CO2*44.01 + Ar2*79.88)/1000; %
atmospheric molecular weight

ATMOS_beta = MOL*g0/(8.314*T0);  % atmospheric scale height^(-1) [1/m]
ATMOS_scale = 1/ATMOS_beta;  % atmospheric scale height [m]

% SURFACE LOCATIONS
% ------------------------------
% Launch:
LAT_launch = 30;



%%
% ================================================
%                 MISSION TRADES
% ================================================
% Written by: Leonardo Pinero
%{
```

```
inputs: mission requirements and user preferences
outputs: defined mission
%}


% TARGET FOR PARKING ORBITS:
% -------------------------------
% Target 1 (after launch):
LAT_park = 40;  % inclination of stable parking orbit [deg]
ALT_park = 200000;  % altitude of stable parking orbit [m]
VEL_park = sqrt(g0*Re^2/(Re + ALT_park));  % velocity of circular parking
orbit [m/s]


% CROSS RANGE REQUIREMENT:
% -------------------------------
REQCrossRange = 1;  % cross range mission requirement
switch REQCrossRange % should be based on planetary radius
    case 1  % percent global coverage
%         cover = 0.95;  % percent coverage
%         L2Dreq = L2Dreq_globe(cover, Re);
        L2Dreq = 3.0;
    case 2  % continental US landing (Earth only)
        L2Dreq = 2.7;
    case 3  % Russian ladning (Earth only)
        L2Dreq = 1.8;
end



%%
% ================================================
%           FLIGHT DOMAIN DETERMINATION
% ================================================
% Written by: Leonardo Pinero
% TARGET COMBINED VEHICLE:
% -------------------------------
REQy = ALT_park;  % apogee of combined vehicle set to meet parking orbit [m]
REQi = LAT_park;  % inclination of combined vehicle set to that of parking
orbit [deg]
REQvel = 7600;    % last velocity of combined vehicle [m/s]
if REQvel > VEL_park
    error('Launch overshoot. Consider decreasing REQvel requirements or
increasing ALT_park.')
end
REQboost = VEL_park - REQvel;

ALT_stage = 100000;  % altitude of staging (should be launch performance
output)

START = [ALT_stage, REQvel, ALT_park];  % [combined vehicle altitidue,
combined vehicle velocity, desired parking orbit]
```

```
%%
% ================================================
%             HYPERSONIC ANALYSIS
% ================================================

% DETERMINE L/D_REQ
% -------------------------------
%{
inputs: Down Range requirement (DR)
outputs: Lateral Range (LR), L/D_req
%}
cover = 0.30;  % 30 percent circumferential coverage upon landing
L2Dreq_globe(cover, Re) % computation based on historical data


% FLIGHT PROFILE & DISC. ANALYSIS
% -------------------------------
%{
inputs:
outputs:
%}
iter = 0;
[W_176, Output Areas] = Hypersonic(Mission, DATAhyper, W_carried, START,
iter);


% FINALIZE LAYOUT
% -------------------------------
%{
inputs:
outputs:
%}

% COST EVALUATION
% -------------------------------
%{
inputs:
outputs: cost estimation
%}



%%
% ================================================
%               LAUNCH ANALYSIS
% ================================================



%%
% ================================================
%        VEHICLE CAPABILITY DETERMINATION
% ================================================
```

```
%%
% ==============================================
%                   VISUALIZE
% ==============================================



%%
% ==============================================
%             FINAL COST ESTIMATION
% ==============================================
```

Below is hypersonic sub-team script written by the author, combining all of its disciplines and including their methods' functions. This code interfaces with the launch sub-team.

```
function [W_176, Areas, STARTiter] = Hypersonic(Mission, DATAhyper,
W_carried, START, iter)

% ======================================================================
% ----------------   HYPERSONIC STAGE DESIGN SCRIPT   --------------------
% ======================================================================

%{
WRITTEN BY:
Leonardo Pinero
%}

%{
CODE DESCRIPTION:
The following script will take two inputs: one database and a user
interface. The user interface contains the mission of the vehicle: a man-
rated hypersonic re-entry vehicle that can perform space operations, both
military and civilian. With user guidance, the script will converge on a
final conceptual design and give output plots and values.
%}

%{
TABLE OF CONTENTS:
    Mission Requirements
    Flight Profile
        Flight Phase Bounds
        Select Critical Flight Phase
    Trajectory Analysis
        Critical Flight Phase
    Geometry Definition
    Propulsion
    Aerothermal
        Aerodynamics
        Heating
```

```
    Sizing
        Geometry Estimate
        Empty Weight Estimate
        Volume Required Estimate
    Weight Estimation
    Structures
    Stability and Control
    Geometry Redefinition
    Trajectory Analysis (Iterated)
    Comparison (Sizing)
    Cost Estimation
%}



%%
% ==============================================
%             MISSION REQUIREMENTS
% ==============================================
%{
This section will process the Mission input and turn it into useful
variables for the disciplines.
%}

%{
Mission input breakdown:
[Inclination Change, Number of Crew Members, Time in Space, W_mission,
(L/D)_req]
%}

DeltLAT = Mission(1);
Ncrew = Mission(2);
Ndays = Mission(3);
L2Dreq = Mission(4);

VOL_pay = 1.247*Ncrew^0.136*Ndays^0.150;
m_crew = lifesupport(Ncrew, Ndays); % life support and crew estimation [kg]
m_pay = m_crew + m_equipment; % payload weight estimation



%%
% ==============================================
%               FLIGHT PROFILE
% ==============================================
%{
This section will actually develop a plotted flight profile which matches
the mission description. This is how the team will visualize the next
iteration.
%}



%%
```

```
% ================================================
%             TRAJECTORY ANALYSIS
% ================================================
%{
This section will take the flight profile and compute the velocity changes
needed to effect such a trajectory.
%}

% initial condition:
if iter == 0
    hi = START(1);
    Vi = START(2);
    h_des = START(3);
end

 % deltaV calculations:
dVs = Dvcalc(hi,Vi,h_des);

% Propulsion design-critical burn (thrust need)
dV_crit = 800;%max(abs(dVs));  % [m/s]
time_burn_crit = 240;  % [s]

% AERO INCLINATION TURN
% -------------------------------
[dV,V,hret] = AeroInc(W,S,CL,LD,h,sel);

%%
% ================================================
%             GEOMETRY DEFINITION
% ================================================
%{
This section will store information about the geometry and output it to
sizing and disciplines.
%}
[GEOM, GEOMsub, GEOMfins, GEOMextended GEOMall] = CADdefine(DATAhyper);

%{
GEOM: [Planform Area (S_plan), Wetted Area (S_wet), length of vehicle (l),
vehicle width (w), trapezoidal base (A_base), body aspect ratio (AR),
leading edge sweep (sweepLE), trailing edge sweep (sweepTE), Side profile
area (S_side) ]

GEOMsub: [Planform Area (S_plansub), Wetted Area (S_wetsub), length of
vehicle (l), vehicle width (w), trapezoidal base (A_base), subsonic wing
width (b_sub), switchblade wing chord (c_rootsub) ]

GEOMfins: [ upper fin area (A_finupp), lower fin area (A_finlow), MAC upper
fin (MAC_finupp), MAC lower fin (MAC_finlow), dihedral of the upper fin
(Dihed_finupp), dihedral of the lower fin (Dihed_finlow) ]

GEOMextended: [ Weight balance inputs ]

GEOMall: [GEOM; GEOMsub; GEOMfins; GEOMextended] matrix of geometry
(preallocate with a zero matrix of a large enough size to avoid matrix
dimensional issues)
```

```
%}


l = GEOM(1);
w = GEOM(2);
S_plan = GEOM(3);
S_wet =

% Output to rocket team:
Areas = [S_side, S_plan];



%%
% ================================================
%                  PROPULSION
% ================================================

% SIZING ENGINE FOR THRUST REQ
% -------------------------------
T_req = W_carried*dV_crit/time_burn_crit;  % critical burn requirement from
traj
[T2W_eng, D_eng, L_eng, m_eng] = PropulsionSizingFunction(T_req);
% _eng subscript relates to engine
VOL_eng = pi*(D_eng/2)^2*(0.5*L_eng);  % engine volume assumed to be
cylindrical, considers volume INSIDE ship

% SIZING TANKS FOR DELTA V REQ
% -------------------------------



%%
% ================================================
%                 AEROTHERMAL
% ================================================

% SUBSONIC AERODYNAMICS
% -------------------------------
[CLsub, CDsub, L2Dsub, L2D_maxsub] = subsonic(WL_sub, l, A_b, b_sub,
S_wetsub, S_refsub);

% HYPERSONIC AERODYNAMICS
% -------------------------------
[CL, CD, L2D, L2D_max] = hyper(l, A_b, S_wet, Sref, W_empty);

% NEUTRAL POINT
% -------------------------------
[NPsub, NP] = NPcalc(GEOM);

% THERMAL LOADS
% -------------------------------
Thermmap(W,S,CL,R0,Tmin,Tmax,Tint,points)  % Written by Chris Miller
```

```
%%
% ===============================================
%                   SIZING
% ===============================================



%%
% ===============================================
%                WEIGHT ESTIMATION
% ===============================================
% Build massINPUT
massINPUT = [ ];

[m_str, m_component] = massEstimate(massINPUT);
m_empty = m_str + m_ppl + m_pay


%%
% ===============================================
%                 STRUCTURES
% ===============================================
%{
Takes aerothermal loads and outputs required structural weight and
geometry.
%}



%%
% ===============================================
%             STABILITY AND CONTROL
% ===============================================

% STENGEL METHOD
% --------------------------------
StaCon_Input = [l, w, S_plan, A_base, VOL, croot_sub, b, CG_empty, yMAC, MAC,
NPsub, NP, CD0sub, sweepLE, sweepTE, AR, S_finupp, S_finlow, MAC_finupp,
MAC_finlow, Dihed_finupp, Dihed_finlow];
%{
StaCon_Input (25 GEOM components in index order):

body length, maximum body width, reference area, base area, volume,
wing root chord, wing span, CG location from tip, yMAC, MAC, NP subsonic
(from tip), NP supersonic (from tip), parasitic drag, LE sweep, TE sweep,
AR of the body, upper fin area, lower fin area, MAC upper fin, MAC lower
fin, dihedral of the upper fin, dihedral of the lower fin.
%}

% Stengel Method:
[SC1] = StaCon_Function(StaCon_Input);
% SC1 Guide 1-9: M, Cma, CYb, Cnb, Clb, Cmde, CYdr, Cndr, Clda
figure
hold on
plot(SC1(1,:),SC1(2,:), 'k', 'LineWidth', 2)
```

```
title('C_{M\alpha} vs. Mach')
xlabel('Mach Number')
ylabel('C_{M\alpha}')
grid on
grid minor
hold off

% CLARKE/TRIMMER METHOD
% -------------------------------


% WEI METHOD (RCS SIZING)
% -------------------------------



%%
% ==============================================
%             GEOMETRY REDEFINITION
% ==============================================

% GEOMETRY SIZING CHECK
% -------------------------------
%{
This function will act as the CAD check and is a MATLAB stand-in for the
discipline’s tools: SOLIDWORS and OpenVSP.
%}
GeomSizing(S_plan, S_wet, GEOM, VOL)  % void function



%%
% ==============================================
%             COMPARISON (SIZING)
% ==============================================



%%
% ==============================================
%          TRAJECTORY ANALYSIS (ITERATED)
% ==============================================

% SUBSONIC ANALYSIS
% -------------------------------
[range_sub, V_approach, Sg, gamma_approach] =
subsonic_perf(L2D_maxsub,CLsub,CL_maxsub,S_plansub,W_empty,hsub);

% Sg = field length



%%
% ==============================================
%             COMPARISON (BUDGET)
```

```
% ==============================================



%%
% ==============================================
%              COST ESTIMATION
% ==============================================



end
```

Below is the author's script to relate cross-range capability to required lift-to-drag.

```
function L2Dreq_globe(cover, Re)
% Written by: Leonardo Pinero

L2D = 0:0.025:3;
LR = 0.539957*(1.667 + 68.016.*L2D + 706.67.*L2D.^2 - 91.111.*L2D.^3);  %
[km]
DR = 4866.6 + 4.70417.*LR;  % [km]
circum = zeros(length(L2D), 1) + (Re/1000)*2*pi;
circum_half = circum./2;

reqDR = cover*circum;

hold on
plot(L2D, LR, 'k--')
plot(L2D, DR, 'k')
plot(L2D, circum_half, 'k', 'LineWidth', 2)
plot(L2D, reqDR, 'r', 'LineWidth', 3)
legend('Lateral Range', 'Down Range', 'Half of Earth''s Circumference',
'Required Range', 'Location', 'Best')
title('Range vs. L/D (30% Circumferential Coverage Requirement)')
xlabel('Lift to Drag Ratio')
ylabel('Range [km]')
grid on
grid minor
hold off

end
```

Below are the trajectory functions from performance, used to derive the vehicle's disciplinary requirements from the mission profile and Earth's geography [15].

Inclination change:

```
function  [dV,V,hret] = AeroInc(W,S,CL,LD,h,sel)
%AeroInc_a
%Christopher Miller
%7/2/18
```

```
%inputs
W = W * 2.2; %vehicle weight, lb
S = S* 10.763; %wing reference area, ft^2
%CL = .2; %lift coefficient
%LD = 3; %lift-drag ratio
h = h*3.28; %ft
WSCL = W/(S*CL);
WCDA = 1000;

%constants
R0 = 6371000 * 3.28; %ft
B = 1/24000; %1/ft
u0 = 25940; %ft/s
d0 = .07648; %sea level air density, lbm/ft^3

dVi = Dvcalc(h/3.28,0,34.4*6076/3.28);
dV = dVi(1);

 %insertion
ri = R0+h;
if sel == 0
    %hohmann
    uiu0 = sqrt((2*1.01)/((ri/R0)*(1.01+ri/R0)));
    gammar = 0;
    uru0 = sqrt(2)/1.01*sqrt((1.01*(ri/R0)/(1.01+ri/R0)));
    ui = uiu0*u0; %iniital velocity
    ur = uru0*u0; %reentry velocity
end
if sel == 1
    %ballistic
    uiu0 = sqrt(1.01)*R0/ri;
    gammar = atan(1.01*R0/ri - 1);
    uru0 = 1/sqrt(1.01)*cos(gammar);
    ui = uiu0*u0;
    ur = uru0*u0;
end

%pullout
if sel == 1
    u0u = ((ri/R0)*(ui/u0)*cos(0))/(((63708.8*3.28 + R0)/R0)*cos(gammar))^-1;
    u = ur/((exp(gammar*(1/LD)*(1 + (u0u^2-1)/(B*R0*(1-cos(gammar)))))));
    hp = -1/B*log((2*WSCL*B/d0)*(1-cos(gammar)+1/(B*R0)*(u0u^2 - 1)));
    uru = exp((R0*d0*(deg2rad(2))*exp(-B*hp))/(2*WCDA*cos(deg2rad(2))*(u0u^2-
1)));
end

%glide
lambda = pi/3; %60 deg. minor circle
pos = pi/2.88703; %position coordinate for 30 deg plane change
i = asin(sin(pos)*cos(lambda)*sqrt(1 + (1-
cos(pos))/(1+cos(pos))*sin(lambda)^2));
rad2deg(i)
xi = ur/u0;
```

```
Fxi = 1 - xi^2/sin(lambda)^2 - 1/sin(lambda)*sqrt(xi^4/sin(lambda)^2 - 2*xi^2
+ 1);
Fx = exp(log(abs(Fxi))-2*pos/LD);
x = sin(lambda)*sqrt(-Fx^2 + 2*Fx + csc(lambda)^2 - 1)/(sqrt(2)*sqrt(Fx));
dV(2) = (u0-x*u0)/3.28;
dVi = Dvcalc(34.4*6076/3.28,0,h/3.28);
dV(3) = dVi(1);
dV(4) = dVi(2);
V = [ur/3.28 x*u0/3.28];
hret = 34.4*6076/3.28;

end
```

Delta V calculations:

```
function dVs = Dvcalc(h,Vin,h_des)
% Written by: Christopher Miller

%input
%h = 200000;
%V = [7784 0];
Re = 6371000; %km
me = 5.972e24; %kg
G = 6.67408e-11;
mu = G*me;
a_des = h_des + Re;
if Vin == 0
    Vin = sqrt(mu/(h+Re));
    V = [Vin 0];
else
    V = Vin;
end

R = [0; Re+h];
r = norm(R);
v = norm(V);
e = ((v.^2-mu/r)*R - (dot(R,V)*V))/mu;
es = norm(e);
a = 1/(2/r-v^2/mu);
E_1 = -mu/(2*a);
apo = a*(1+es^2);
peri = a*(1-es^2);
va_1 = sqrt(2*(E_1+mu/apo));

E_2 = -mu/(2*((a_des+peri)/2));

vp_2 = sqrt(2*(E_2+mu/apo));
dVs(1) =  vp_2-va_1;

vcirc = sqrt(mu/a_des);
va_2 = sqrt(2*(E_2+mu/a_des));
dVs(2) = vcirc-va_2;

end
```

Subsonic performance:

```
function [range_sub,Vapp,Sgland,gammaa]  =
subsonic_perf(LDmax,CL,CLmax,S,W,hi)
% Written by: Christopher Miller

g = 9.81;

gammaa = atan(LDmax^-1);
range_sub = hi*cos(gammaa)/sin(gammaa);

rho = atmos(0:10:10000,'Units','SI');

V = sqrt(W./(.5.*rho.*S*CL*(cos(gammaa)+LDmax^-1*sin(gammaa))));

plot(V,0:10:10000,'k')
grid on
grid minor
xlabel('Velocity, m/s')
ylabel('Altitude, m')
title('Subsonic glide velocity')

fprintf('Subsonic glide range: %0.0f m\n',range_sub)
n = 1.5;
WSland = W/S;
Vsland = sqrt(WSland * 2/(rho(1)* CLmax));
Vapp = Vsland * 1.2;
Vtd = Vsland * 1.1;
gammaapp = gammaa;
Rapp = mean(Vapp+Vtd).^2/(g*(n-1));
hflare = Rapp*(1-cos(gammaapp));
Ktapp = -.3;
Kaapp = rho(1)/(2*(WSland))*(.3*.05-.0095-.0535*.05^2);
Sgland = 1/(2*g*Kaapp)*log((Ktapp)/(Ktapp+Kaapp*Vtd^2)) + 3*Vtd;
fprintf('Approach angle: %0.0f degrees\n',gammaapp*180/pi)
fprintf('Approach speed: %0.2f m/s, \n',Vsland)
fprintf('Landing distance assuming 3 second ground roll after touchdown:
%0.0f m\n', Sgland)

end
```

Thermal mapping:

```
function stuff = Thermmap(W,S,CL,R0,Tmin,Tmax,Tint,points)
% Written by: Christopher Miller

WSCL = W/(S*CL);%input('Specify W/(S*CL): ');
% points = 100;%input('Specify number of data points: ');
% R0 = .1;%input('Specify LE radius (m): ');
% Tmin = 300;%input('Specify desired min temp (K): ');
% Tmax = 1900;%input('Specify desired max temp (K): ');
% Tint = 200;%input('Specify desired temp interval: ');

T = [Tmin:Tint:Tmax];
Re = 6371000;
```

```
g = 9.81;

H = [0:100000/(points-1):100000];
rho = atmos(H,'units','SI');

V = (1./(g*(Re+H))+ rho./(2*WSCL)).^(-1/2);
V2 = (1./(g*(Re+H))+ rho./(2*600)).^(-1/2);

lim = size(T);
i = 1;
Vt = zeros(lim(2),points);
while i <= lim(2)
   Vt(i,:) = ((T(i).^4*5.67e-8*.8)./(1.83e-4*sqrt(rho./R0))).^(1/3);
    i = i + 1;
end
hold on
grid on
grid minor
for i = 1:lim(2)
    if mod(i,2) == 1
        plot(Vt(i,:),H,'k')
    end
    if mod(i,2) == 0
        plot(Vt(i,:),H,'--k')
    end
end
plot(V,H,'r')
plot(V2,H,'--r')
axis([0 10000 50000 100000])
xlabel('Velocity,m/s')
ylabel('Altitude,m')
entries = cell(length(T),1);
for i = 1:length(T)
    entries{i} = [num2str(T(i)) ' K'];
end
entries{length(T)+1} = ['Re-entry profile, W/SCL = ' num2str(WSCL)];
entries{length(T)+2} = ['Re-entry profile, W/SCL = 600'];
title('Thermal Map, temp increases moving right')
legend(entries,'Location','SE')
end
```

Trajectory code:

```
%Performance and Trajectory - code link
%Christopher Miller
%7/7/2018

clear
clc
close all

%Test Case Variables
LDmaxsub = 6;
CLsub = .2;
CLmaxsub = .5;
```

```
Ssub = 15;
Wsub = 10000;
hsub = 10000;
hi = 70000;
Vi = [6800 100];
h_des = 200000;
W = 12000;
S = 60;
LDmax = 3;
CL = .15;
h_circ = 200000;
sel = 0;
R0 = .1;
Tmin = 700;
Tmax = 1900;
Tstep = 200;
points = 100;

%Subsonic - range calculation, landing speed, field length
%Inputs: L/Dmax, CL for L/Dmax, foldout wing area, vehicle weight, initial
%height
%Outputs: range during subsonic flight,landing field length, approach
%velocity, approach flight path angle, plot of optimum glide speed vs
altitude

[Range, Vapp, Sg, gammaa] =
subsonic_perf(LDmaxsub,CLsub,CLmaxsub,Ssub,Wsub,hsub);

%Dvcalc - calculates dV's required for Hohmann transfers between orbits
%Inputs: vehicle altitude, current velocity, desired circular orbit
%altitude
%Outputs: a vector with 2 values for the impulses required (at
%initial/transfer orbit apogees) to insert the vehicle into the desired
%circular orbit

dVi = Dvcalc(hi,Vi,h_des);

%AeroInc - calculates the enviroment and delta-V requirement for the
%orbital plane change maneuver (assumes 30deg)
%Inputs: Vehicle weight, hypersonic reference area, Cl for l/dmax,
%l/dmax, initial orbit altitude (must be circular), selector for hohmann or
%ballistic reentry (suggest hohmann for now ballistic is wonky)
%Outputs: four required dV's, max/min atmosphere speed, altitude for those
%speeds (constant altitude glide phase)

[dVs, Vspec, h_t] = AeroInc(W,S,CL,LDmax,h_circ,sel);

%Thermmap - determines re-entry trajectory and thermal map for said
%trajectory
%Inputs: W,S,CL, nose radius, Temperature ranges as three variables: Tmin,
Tmax, Tstep; # data points (typically 100)
%Outputs: Plot of thermal map, currently outputs no variables

hold on
figure
```

```
Thermmap(W,S,CL,R0,Tmin,Tmax,Tstep,points);
```

Below is CAD's life support function which aids in defining payload parameters from the mission description. [36] This is based on the realities and human studies of living in space.

```
function [W_tot] = lifesupport(Ncrew, Ndays)
% Written by Jared Arizpe
% Recon
% This will calculate crew/passenger payload for 'x' days
W_pass = 92; %Weight of 6' male in kg
W_crew = (Ncrew*W_pass); %total weight of crew
O2 = 0.9; %O2 usage/man day in kg.
H2O = 9.1; %H2O usage/man day in kg. Includes drinking and wash
Food = 0.6; %Food usage/man day in kg.
W_lifesupport = (O2+H2O+Food)*Ncrew*Ndays;

W_tot = W_crew + W_lifesupport;
end
```

Descent script made to replicate literature search plots and use as a tool for testing out different design parameters.

```
%{
===========================================================================
========================   PLOTTING DESCENT   ==============================
===========================================================================
%}
close all



%%
% CONSTANT SETTINGS
% ----------------------------------
Re = 6371000; % earth's radius [m]
g0 = 9.807;  % gravitational acceleration at sea level [m/s^2]
K = g0*Re^2;  % gravitational constant



%%
% INITIALIZE
% ----------------------------------
% Kinematics:
% gamma =  % entry angle
% Vorb =  % orbital speed

% Design:
% CL =  % coefficient of lift
% S =  % planform area
% CD =  % coefficient of drag

% Altitudes examined:
```

```
% h1 = 1000*(0:.1:30);  % low altitudes high resolution [m]
% h2 = 1000*(30:2:150);  % high altitudes low resolution [m]
% h = [h1, h2(2:end)];  % combine into one vector
% clear h1 h2
h = 1000*(40:2:150);  % altitude observed: 40 km - 150 km

% Pre-load atmos properties
[~,~,~,rho] = atmoscoesa(h, 'None');  % 1976 extended atmosphere



%%
% TRAJECTORY PLOTTER
% -----------------------------------
figure
%{
WSCL1 = 1:20;
WSCL2 = 30:10:140;
WSCL3 = 150:50:700;
WSCL4 = 750:250:2000;
%}

% WSCL = [WSCL1 WSCL2 WSCL3 WSCL4];
WSCL = [30, 100, 1000];
Vc = sqrt(K./(Re + h));

hold on
title('Descent Trajectories for Different Values of $\frac{W}{SC_L}$',
'Interpreter', 'Latex')
xlabel('Velocity [km/s]', 'Interpreter', 'Latex')
ylabel('Altitude [km]', 'Interpreter', 'Latex')
grid on
grid minor
for i = 1:length(WSCL)
    V = (1./Vc.^2 + rho./(2.*WSCL(i))).^(-1/2);
    if mod(i,3) == 0
        plot(V./1000,h./1000, 'k')  % converting from [m] to [km]
    elseif mod(i,3) == 1
        plot(V./1000,h./1000, 'k:', 'LineWidth', 2)  % converting from [m] to
[km]
    else
        plot(V./1000,h./1000, 'k--')  % converting from [m] to [km]
    end
end

% legend entries:
WSCLentries = cell(length(WSCL),1);
for i = 1:length(WSCL)
    WSCLentries{i} = ['W/(SC_L) = ' num2str(WSCL(i))];
end
hold off
legend(WSCLentries, 'Location', 'Best')
```

Below is the analysis code used for mission trade studies and associated sub-routines (continuously modified to make different plots).

```
% NYLAND PLOTTER
% ---------------------------
% written by: Leonardo Pinero
% date: 31 July 2018
% close all

% Inputs:
PLANET = [1.2, 8500, 6371000, 9.807]; %Mars: [0.02, 11100,
3396000, 3.71];
DV = 123:200:20000;
L2D = 3;
K = PLANET(4)*PLANET(3)^2;
rh = PLANET(3) - PLANET(2)*log(0.000676/PLANET(1));  % density
to turn: 0.000676 kg/m^3


% PLOT DELTA I vs DELTA V
DI = zeros(length(L2D), length(DV));  % preallocate
hold on
for i = 1:length(L2D)
    for j = 1:length(DV)
    DI(i,j) = nyland(DV(j), L2D(i), PLANET);
    end
    %{
    % USE IF NEED TO SEE REQ'S FOR HIGHER DELTA I ANGLES
    DIbig = unwrap(3*DI, pi);
    DI = DIbig./3;
    %}
    if mod(i,4) == 0
        plot(DV, rad2deg(DI(i,:)), 'k', 'LineWidth', 1.5)
    elseif mod(i,4) == 1
        plot(DV, rad2deg(DI(i,:)), 'b--', 'LineWidth', 1.5)
    elseif mod(i,4) == 2
        plot(DV, rad2deg(DI(i,:)), 'b--')
    else
        plot(DV, rad2deg(DI(i,:)), 'k--', 'LineWidth', 1.5)
    end
end
DVprop = 0:200:20000;
DIprop = zeros(1, length(DVprop));  % preallocate
for i = 1:length(DVprop)
    DIprop(i) = PropIncline(DVprop(i), PLANET);
end
```

```
plot(DVprop, rad2deg(DIprop), 'b', 'LineWidth', 2)
ufinish = sqrt(K/rh);
xlim([0, ufinish])
DeltaIreq = zeros(length(DV),1) + 30;
plot(DV, DeltaIreq, 'k:', 'LineWidth', 3)
title('$\Delta$i per $\Delta$V with $\frac{L}{D}$ = 3 on Earth
and Mars', 'Interpreter', 'Latex')
xlabel('$\Delta V_{required}  [\frac{m}{s}]$', 'Interpreter',
'Latex')
ylabel('$\Delta$i [deg]', 'Interpreter', 'Latex')
grid on
grid minor

% legend entries:
L2Dentries = cell(length(L2D),1);
for i = 1:length(L2D)
    L2Dentries{i} = ['Earth L/D = ' num2str(L2D(i))];
end
L2Dentries{length(L2D) + 1} = 'Earth Propulsive Maneuver';
L2Dentries{length(L2D) + 2} = 'Mission';
hold off
legend(L2Dentries, 'Location', 'Best')
```

Nyland analysis sub-routine shown below.

```
function [DeltaI, phi, dv2] = nyland(DeltaV, L2Dturn, PLANET)

%{
% test case:
PLANET = [7317, 6371000, 9.81];  % scale, radius, surface
gravity
DeltaV = 2100;
L2Dturn = 2;
%}

% DATA/SETTINGS:
% -----------------------------------
% break down planet rack:
rho_sea = PLANET(1);
atmos_scale = PLANET(2);  % atmospheric scale height
Re = PLANET(3);  % planet radius
g0 = PLANET(4);  % surface gravity

% deal with negative L2D's
L2Dturn = abs(L2Dturn);
```

```
% environmental constants
K = g0*Re^2;
u0 = sqrt(K/Re);  % earth skimming velocity

gamma = deg2rad(45);  % minor circle cone angle



% METHODS:
% ref: Nyland, Rand Corporation
% ----------------------------------
% find delta V required to descend and re-circularize:
ALTturn = -atmos_scale*log(0.000676/rho_sea);  % density to
turn: 0.000676 kg/m^3

ri = Re + 200000;  % initial sattelite height
rh = Re + ALTturn;

% deltaV required to burn to and from this altitude
ui = sqrt(K/ri);  % initial circular velocity at ri
uapo = sqrt((2*K*rh)/(ri*(ri + rh)));  % descent ellipse
uperi = sqrt((2*K*ri)/(rh*(ri + rh)));  % descent ellipse
ufinish = sqrt(K/rh); % velocity at end of turn (descent ellipse
now circularized)

dv1 = ui - uapo;  % dv1 is used twice for mission (we need to
return home!)
dv4 = dv1;  % recircularize
dv3 = uperi - ufinish;  % dv req to get from unfinish back up to
orbit
dvatmos = DeltaV - (2*dv1 + dv4);  % deltaV available after
orbit ops
dv2 = dvatmos - dv3;

if dv2 < 0
    if dv3 < 0
        DeltaI = 0;
    else
        while dv2 < 0
            ufinish = (ufinish + uperi)/2; % cut the turn short,
not enough fuel!

            % run dv calcs again until we can get some dv2
            ui = sqrt(K/ri);
            uapo = sqrt((2*K*rh)/(ri*(ri + rh)));
            uperi = sqrt((2*K*ri)/(rh*(ri + rh)));
```

```
            dv1 = ui - uapo;
            dv4 = dv1;
            dv3 = uperi - ufinish;
            dvatmos = DeltaV - (2*dv1 + dv4);
            dv2 = dvatmos - dv3;

            u = ufinish - dv2;
            phi = 0.5*L2Dturn*(log(rangefunc(uperi, u0, gamma))
- log(rangefunc(u, u0, gamma)));
            DeltaI = asin(sin(phi)*sin(gamma)*sqrt(1 + (1 -
cos(phi))/(1 + cos(phi))*(sin(gamma))^2));

        end
    end

else
    u = ufinish - dv2;
    phi = 0.5*L2Dturn*(log(rangefunc(uperi, u0, gamma)) -
log(rangefunc(u, u0, gamma)));
    DeltaI = asin(sin(phi)*sin(gamma)*sqrt(1 + (1 - cos(phi))/(1
+ cos(phi))*(sin(gamma))^2));
end


% corrections for high performance
% NOT NEEDED (removed):
%{
if (deg2rad(90) - DeltaI) < -87.5
    DeltaI = real(DeltaI + deg2rad(180));  % maxed out
performance for gamma = 45 deg
elseif isreal(DeltaI) == 0
    DeltaI = real(deg2rad(90) - DeltaI);  % maxed out
performance for gamma = 45 deg
end
%}

DeltaI = abs(DeltaI);

% correct for high performance:
%{
[DeltaITEST1, ~] = nylandTEST(0.95*DeltaV, L2Dturn, PLANET);
if DeltaITEST1 > DeltaI % passed the north pole, increased dv
incorrectly lowers DeltaI
    DeltaI = 2*nylandTEST(ufinish, L2Dturn, PLANET) - DeltaI;  %
restate the true performance
end
```

```
%}
    function [F] = rangefunc(u, u0, gamma)
        x = u/u0;
        F = 1 - x^2/(sin(gamma))^2 -
1/sin(gamma)*sqrt(x^4/sin(gamma)^2 - 2*x^2 + 1);
    end


end
```

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