SECOND EDITION

Introduction to ROCKET SCIENCE and ENGINEERING

TRAVIS S. TAYLOR

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Introduction to Rocket Science and Engineering

Second Edition

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To all those rocket scientists and engineers throughout history who have successfully designed the vehicles and technologies for spacecraft missions that fuel mankind's sense of wonder and dreams of space and to all those who will in the future, I dedicate this work.

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Preface

This book was written with the hope of being both accessible and practical for instructors teaching rocket science and engineering, for students taking classes in the subject, and for the enthusiast or professional needing further training in the topic. That being said, there are some recommended steps for reading and using this book.

Recommendation #1: Download OpenRocket onto your computer and download the latest technical document that goes along with it. This is free open-source code that is very good at designing and simulating rockets of various sizes. We will show how to use it to design an orbital launch vehicle in a later chapter of this book. Along with OpenRocket, you will also need to download the Thrust Curve Tool. The Thrust Curve Tool software will allow you to generate thrust curves of measured or even fictional engines that can then be ported into OpenRocket. OpenRocket can be found at

http://openrocket.sourceforge.net/

and Thrust Curve Tool can be found at

http://www.thrustgear.com/software.html.

We will use OpenRocket and Thrust Curve Tool to design a rocket later in the book. For the instructors, I have found it useful when teaching rocket science and engineering to announce on day one a design project that would require OpenRocket and other software such as Mathcad® (my preference) and MATLAB®. I have found it useful to announce at the first class that, before the final project is due, I will require each student to develop a design for a rocket that can place a certain payload into a specific orbit or altitude.

Recommendation #2: You will need some type of math-modeling and simulation software to become a rocket scientist or engineer. I prefer Mathcad, though I also use MATLAB from time to time and even spreadsheets like Excel.

Recommendation #3: Build, test, and fly a rocket. Design a small rocket in the modeling software and then actually build and fly it. If possible, use an altimeter to determine how closely the actual flight correlates with the model. Seeing the actual rocket hardware being constructed and integrated for flight is absolutely priceless in details that the modeling and simulation cannot teach. The students will learn that things such as fasteners, glues, bolt-and-nut combinations, shear pins, batteries, tape, and materials all come together in a way that simply cannot be fully understood without actually doing it. If class time allows or there is a corresponding lab, put a team together and build a bigger and more complex rocket.

Recommendation #4: If actually building a rocket is not in the cards, then purchase a rocket kit online and have the piece parts in the classroom for students to look at. Or, better yet, have them design them in a computer-aided design software package and three-dimensionally print them. Having the actual component available to point to when discussing it often helps.

Author

Travis S. Taylor ("Doc" Taylor to his friends) has earned his soubriquet the hard way: he has a doctorate in optical science and engineering, a doctorate in aerospace systems engineering, a master's degree in physics, and a master's degree in aerospace engineering, all from the University of Alabama in Huntsville. Added to this is a master's degree in astronomy from the University of Western Sydney (Australia) and a bachelor's degree in electrical engineering from Auburn University (Alabama). Dr. Taylor has worked on various programs for the Department of Defense and NASA for the past two decades. He is currently working on several advanced propulsion concepts, very large space telescopes, space-based beamed energy systems, next-generation space launch concepts, directed-energy weapons, nanosatellites, and low-cost launch vehicle concepts for the U.S. Army Space and Missile Defense Command. Dr. Taylor was one of the principal investigators of the Ares I Flight Test Planning effort for NASA Marshall Space Flight Center and was the principal investigator for the NASA Marshall Space Flight Center and Teledyne Brown Engineering effort to study and develop a Pluto–Kuiper belt orbiting probe using nuclear electric rocket engines.

In his copious spare time, Doc Travis is also a black-belt martial artist, a private pilot, and a SCUBA diver and races mountain bikes. He has also competed in triathlons, is a marathon runner and a CrossFitter, and has been the lead singer and rhythm guitarist of several hard rock bands. He has written 16 science fiction novels, two textbooks (including this one), and over a dozen technical papers. Dr. Taylor has appeared and starred in several television programs including the History Channel's *The Universe* and *Life After People*, National Geographic Channel's hit shows *Rocket City Rednecks* and *When Aliens Attack*, and The Weather Channel's *3 Scientists Walk Into a Bar*. He currently lives with his wife and two children in north Alabama just outside of Huntsville in view of the Saturn V rocket that is erected at the U.S. Space and Rocket Center.

Introduction

This book was written to be an introduction to the history and basics of rocket theory, design, experimentation, test, and applications. The book was written as an introductory overall view of the vast spectrum of knowledge the practicing rocket scientist or engineer must have to be successful. The knowledge covers areas from advanced mathematics, chemistry, and physics to logistics, systems engineering, and, yes, even politics. The great successful rocket scientists of history like Wernher von Braun, Robert Goddard, and Sergei Korolev understood what it truly meant to be rocket scientists from all aspects of the term. When most people think of rocket scientists, they think of the stereotypical nerd with horn-rimmed glasses and the pocket protector. Sure, there are rocket scientists who fit that description, but the new generation of rocket scientists are probably too young to recall von Braun and Walt Disney presenting concepts to the world through motion picture and television media with the polish that only a Disney production can produce. In those films, von Braun was far from stereotypical. The rocket scientist must be versatile indeed.

The material herein was compiled and written with the undergraduate student in mind. However, it is applicable and essential for any military or civilian space operator, manager, or designer who wants to achieve a better understanding of how rockets are designed and how they operate. The book was also written to be a good introduction and hopefully to spark excitement about the field and encourage those wishing to develop a more detailed and advanced understanding and study of the topic. By all means, go on to graduate school or further and become a true "rocket scientist."

1

What Are Rockets?

The 20th and 21st centuries brought forth the development of rockets that have enabled mankind to escape the bonds of planet Earth and go into the great "final frontier" that has mystified mankind since the first human looked up at the sky. Rockets have become commonplace in our everyday vernacular and culture to the extent that they are accepted technologies. We see them on television, in movies, in video games, and in books. Children have toys that apply general principles of rocket science and engineering and, for decades, have built rockets from kits that soar into the sky, generating excitement and enthusiasm. Rockets are exciting; it is as plain and simple as that!

That said, there is absolutely nothing "plain" or "simple" about a rocket. What the general "nonscientists" or "nonengineers" tend to misunderstand is how complicated and technically involved rockets actually are. The basic principles of rocket science might be easily explained to primary school-age students, but the devil is indeed in the details.

It is often stated as a major achievement of mankind that the Space Shuttle had something on the order of 2 million parts. The workings and functions of each of these parts are beyond the scope of this book, of course, but the understanding for the need of so many parts is something that will try to be emphasized herein. Rocket science and engineering are not a simple subject by any means; otherwise, the old joke about "it ain't rocket science" wouldn't be as funny as it is.

Therefore, this chapter will discuss a bit about how rockets were discovered and developed over mankind's history. The basic principles governing rockets and rocket science will also be discussed and will hopefully lead the readers to a point where the old joke has a much deeper and profound meaning.

1.1 The History of Rockets

1.1.1 400 BCE

One of the earliest mentions of anything rocket-like in history appears to be from the writings of Aulus Gellius, a Roman. Gellius writes about a Greek individual named Archytas who is from the city of Tarentum, a part of what is now known as Southern Italy. In this story by Gellius, the character Archytas uses a wooden pigeon suspended by wires and propelled by steam to amaze and mystify the Tarentum locals. This is related to the history of rockets simply because it is the earliest known mention of man using Newton's Third Law of action and reaction for a means of propulsion. It is especially interesting in that Newton's laws would not be developed for about 20 more centuries to come.

1.1.2 100 to 0 BCE

Sometime in the 1st century BCE, the Greek inventor Hero of Alexandria (70 to 10 BCE) is noted to have invented the device known as the aeolipile. The aeolipile was a steam-driven device that, like Archytas's pigeon, also implemented Newton's Third Law of action and reaction. Figure 1.1 shows an artist's rendition of the aeolipile. It should also be noted here that the device is sometimes described as Hero's engine.

The engine consisted of a fire to heat a reservoir of water, which was converted to steam. The steam then rose through tubes to a sphere, which collected the steam and became a pressure vessel as more and more steam became compressed into it. The sphere was suspended such that it could freely spin about a horizontal axis. On opposite sides of the sphere, orthogonal to the spin axis, were two small outlets for the steam. As the pressurized steam forced its way out of the pressure vessel and through the outlet "nozzles," the force of the steam caused the sphere to rotate about the spin axis. In actuality, Hero's engine contains most parts of a simple thermal rocket engine.

It is thought that the Chinese were also developing rockets in the form of fireworks sometime during this 1st century BCE. It is somewhat unclear as to the actual date when the first true rockets appeared, but it is certain that stories of rocket-like devices appear sporadically throughout this period in time. Some references suggest the Chinese had fireworks as early as the 2nd century BCE; however, others debate the claims.

1.1.3 0 to 100 AD

The Chinese most certainly began experimenting with compounds made from saltpeter (potassium nitrate, KNO_3), realgar (arsenic sulfide, As_4S_4), sulfur (S), and charcoal (carbon, C) by this time period, three of which are the basic ingredients to gunpowder. (Realgar

FIGURE 1.1
Hero of Alexandria's aeolipile demonstrates Newton's Third Law of action and reaction, which is the driving principle behind modern rocketry.

is not really required for gunpowder.) And, perhaps, the Chinese were experimenting with fireworks fashioned from these compounds.

1.1.4 850 AD

The earliest certain record of gunpowder is likely from the book written in 850 AD, translated as *Classified Essentials of the Mysterious Tao of the True Origin of Things.* In this book, it is indicated that Taoist alchemists derived gunpowder in their efforts to develop an "elixir of immortality." The book describes the alchemists being burned and even the building they were in burning down. It is believed that many Chinese alchemists were searching for this "elixir of immortality." It is likely that some of them found what they were searching for by accidentally blowing themselves up. There are historical reports of large repositories of gunpowder ingredients causing major fatal accidents in ancient China during this era.

1.1.5 904 AD

The Chinese began using gunpowder in warfare as incendiary projectiles by this time. These projectiles were known as "flying fires." They were fired as arrows, grenades, and catapults.

1.1.6 1132 to 1279 AD

Chinese military began to expand on the flying fires and began to use gunpowder as a propellant. The earliest recorded experiments were mortars being fired from bamboo tubes.

Around 1232 AD, the Chinese reportedly used the first true rocket in their fight with the Mongols (Figure 1.2). It was reported that, at the battle of Kai-Keng, the Chinese used a tube, which was capped at one end and contained gunpowder, that was lit from the open end. The ignition of the gunpowder within the capped tube created heat, smoke, and other exhaust gases that were forced out of the open end of the tube, creating thrust. The tube was controlled by placing a stick along its side that stabilized the solid rocket's flight path in the same way that a stick on a bottle rocket is used. Also, during this time frame, the

FIGURE 1.2
The Chinese fought the Mongols using arrows and bombs as depicted in this painting, circa 1293 AD.

English monk and alchemist Roger Bacon improved the formula for gunpowder for rockets. His work notably improved the range of rockets of the period.

1.1.7 1300 to 1600 AD

Frenchman Jean Froissart discovered a means of improving the accuracy of rockets. He realized that rockets were more accurate when launched from a tube. This was the birth of the bazooka and actually the launch tube. It is often published in textbooks that T. Przypkowski became the first European to study rocketry in detail in 1380 AD; however, Froissart and an Italian, Joanes de Fontana, were also studying the topic. de Fontana actually developed a rocket-powered torpedo. Joan of Arc, as well, was known to have used rockets at the battle of Orleans in 1449.

There is a tale of a Chinese official named Wan-Hu building a rocket chair and launching himself. The story says that 47 rockets and 2 kites were attached to a chair, and then Wan-Hu had himself launched by assistants. Once the fuses were lit, there was, according to the story, smoke and a loud roar, and then, when the smoke cleared, Wan-Hu was nowhere to be found. Gunpowder those days was quite unstable and was just as likely to explode as to burn in a rocket. It is more likely that Wan-Hu launched himself to oblivion in millions of tiny pieces rather than into the sky. Of course, it is not certain if this story is true or just that, a story. But, it is a good one, nonetheless.

In 1591, Johann von Schmidlap wrote a book about the nonmilitary uses of rockets. He described the use of sticks for stabilization and the possibility of mounting rockets on top of rockets—staging.

1.1.8 1600 to 1800 AD

This period offered a great deal of development in the understanding of rockets and the principles that drive them. In 1650, the Polish artillery expert, Kasimiez Siemienowicz, published designs for a staged rocket that would offer more destructive capabilities and potentially a farther range.

In 1696, the Englishman Robert Anderson published a document on how to build solid rockets. He described how to mix the propellants and then to pour them into molds. He also described how to prepare the molds. This is sometimes suggested as the first step in the mass production of rockets.

And, of course, in 1643, Sir Isaac Newton was born. In his publication, *De Motu Corporum* in 1684, he had the precursor to his laws of motion that would be later completed and published in *Principia* in 1687. It was through these laws of motion that other scientists and engineers could understand the whys and hows of rockets and rocket science.

Standing on Newton's foundations and with the development of calculus by him and Gottfried Leibniz (independently), the 1700s brought forth even greater understanding of rocketry. Leonhard Euler and Daniel Bernoulli both developed detailed understandings of the fluid dynamics of gas flow inside the rocket engine and of the aerodynamics of flowing air about the exterior.

In 1720, the Dutch professor, Willem Gravesande, was known for constructing model cars that were propelled by steam rockets. And, about this same time, German and Russian scientists were experimenting with heavy rockets that could lift as much as 45 kg. It is reported that these rockets were so powerful that they burned deep holes in the ground where they were launched. Also, during this time frame, rockets were seeing more use in military operations across Europe and India, with the latter using rockets in their fight with the British.

1.1.9 1800 to 1900 AD

British Admiral Sir William Congreve had apparently seen ample rocket use in the conflicts with India and had put his observations to task. He refined what he had seen to improve the understanding and application of rocketry for the British military. He carried on rocket experiments for this purpose.

In 1806, Frenchman Claude Ruggieri launched small animals in rockets equipped with parachutes. Perhaps this is the first mention of actual rocket passengers or occupants that were returned with some, at least potentially, safe method.

In 1807, Congreve's rockets were used against Napoleon and, in 1809, Congreve opened two rocket companies, and his rockets were later used against the United States in the War of 1812. It is reported that Congreve's rockets were fired against Fort McHenry and are the rockets mentioned in the American national anthem, "The Star-Spangled Banner." In the late 1820s, the Russians used Alexander Zasyadko's rockets against the Turks in the Russo–Turkish War.

In 1841, a patent was granted in England for the first-ever "rocket airplane." The patent was granted to one Charles Golightly. The idea apparently employed a steam-driven rocket. No prototype was ever constructed.

In 1844, the Congreve rockets were replaced by ones designed by the English inventor William Hale. Hale had been developing the "stickless rocket" for nearly two decades. The Congreve rockets used the stick concept for stabilization much in the same way seen on modern bottle rockets. Hale used three fins mounted on the rocket for stabilization. The rocket was also known to spin when launched and was often referred to as the "rotary rocket." This was actually the development of spin stabilization, which is used by many modern rockets today.

In the late 1800s, the era of modern rocketry was about to begin. In 1857 and in 1882, the two true founders of modern rocket science and engineering were born. One of them was Russian and the other an American.

The Russian schoolteacher Konstantin Tsiolkovsky (Figure 1.3) was born in Izhevskoye, Russia, to a middle-class family in 1857. As a child, Tsiolkovsky contracted scarlet fever and, as a result, developed a hearing impairment, which led to him being homeschooled until the age of 16.

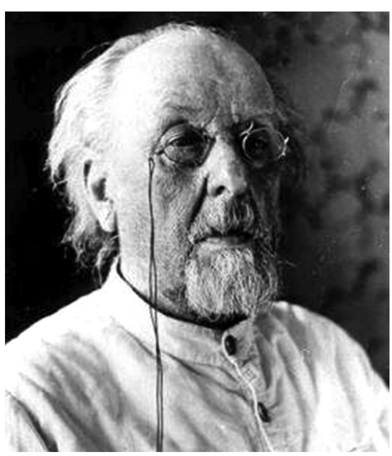


FIGURE 1.3
Russian high school math teacher Konstantin Tsiolkovsky is arguably *the* father of modern astronautics.

1.1.10 1900 to 1930 AD

Tsiolkovsky reportedly worked as a high school mathematics teacher until he retired in 1920. In 1903, he published what has become known to the rocket science community as the first true book or treatise on the subject. *The Exploration of Cosmic Space by Means of Reaction Devices* describes most of the aspects and intricacies of modern rocket science. Over the years, he would publish hundreds of papers on the topic. Mainly, he is noted for the idea of multistage rockets for cosmic rocket trains. For the details and theories in this treatise, Tsiolkovsky is often referred to as the father of modern astronautics. (The Russians sometimes call it cosmonautics.) In the book *Modeling Ships and Spacecraft: The Science and Art of Mastering the Oceans and Sky* (2013), Tsiolkovsky was quoted as being inspired by the science-fiction author Jules Verne:

> I do not remember how it got into my head to make the first calculations related to rockets. It seems to me the first seeds were planted by famous fantaseour, J. Verne. (p. 200)

Robert Goddard was born in 1882 in Worcester, Massachusetts. Goddard suffered from stomach problems as a child and, as a result, fell two years behind in his schoolwork. As he matured, he became deeply interested in reading and reportedly made regular visits to his local library. He received a bachelor's degree in physics from Worcester Polytechnic Institute in 1908, a master's degree from Clark University in 1910, and a PhD from Clark University in 1911. In 1912, he moved to Princeton University on a research fellowship.

Goddard's earliest experiments were with solid rockets, and, after various experiments in 1915, he became more and more convinced that liquid rocket fuel would enable the rocket to carry more payload to higher altitudes. Undoubtedly, World War I helped fund and fuel the need for Goddard's and others' research. In 1919, he published a book titled *A Method of Reaching Extreme Altitudes*, which is one of the reasons he is known as one of the founders of modern rocketry. He set about experimenting with liquid engines and launched his first successful flight on March 26, 1926. Figure 1.4 shows Goddard and his rocket. The flight lasted 2.5 sec and traveled a ground distance of about 56 m. The rocket peaked at only 12.5 m.

Another quite notable scientist of the era was Hermann Oberth who was born in Romania in 1894. In 1922, Oberth wrote to Goddard asking for a copy of his book on liquid rockets. Just one year later, Oberth published *The Rocket into Planetary Space*, which discussed the possibilities of manned flight and the effects that rocket flight would have on the human body. He also showed through Newtonian mechanics that a rocket could travel faster than its exhaust gas and how a rocket could place a satellite into space. As a result of his writings, many space clubs and organizations formed around the world. One of the most notable was the Verein fur Raumshiffahrt (VfR), which translates as the Society for Space Travel.

Some other notable events also occurred. In 1928, the first of nine volumes on interplanetary travel was published by the Russian professor Nikolai Rynin. The first manned rocket-powered car was tested by Opel, Valier, and others in Germany that same year. In the summer of 1928, a manned rocket glider was flown by Friedrich Staemer. The vehicle traveled nearly 2 km. In 1929, Goddard launched a rocket carrying a camera, a barometer, and a thermometer into space, which were all recovered after the flight. This was probably the first reconnaissance payload ever launched.

During this period, an event occurred that would not be important until the 1950s. In 1907, the Russian–Ukrainian Sergei Korolev (sometimes transliterated as Sergey Korolyov) was born. Korolev would become the catalyst and spark of the Russian space program that would eventually spawn the Cold War space race. He was to the Soviet space program what Wernher von Braun was to the American space program.

FIGURE 1.4
Robert Goddard and his first successful liquid fuel rocket engine that made him one of the founders of modern rocket science. (Courtesy of NASA.)

1.1.10.1 A Perspective

An interesting perspective on this era in time was the publicly perceived lack of understanding of the physics involved in rocketry and the opposition met by the founders of the field. It is often described in history books and television programs that scientists around the country and world opposed the idea that a rocket could travel to extremely high altitudes because there would be no air for the exhaust gases to push against. From the above historical discussion in Sections 1.1.1 through 1.1.10, it can be seen quite clearly that the knowledge and theoretical development were in place for such notions to be simply dismissed as physically incorrect.

In other words, Newton's laws had been published for centuries. The understanding of fluid dynamics about the vehicle and within it had been developed. And, at least Tsiolkovsky and Oberth, in two separate countries, understood that such a concept of high altitude and even space flight was possible. Looking back at history makes one wonder just how prevalent was this notion that rockets to space was impossible, or is this just a few particular statements made by a few scientists who failed to think through their statements that caused history to be described as it has? Were the statements of a few misguided scientists overplayed historically, and thus were the rocket scientists of the era all so incorrect? This topic would make a great discussion or historical study to determine which situation really was the case.

1.1.11 1930 to 1957 AD

The American Rocket Society was founded by David Lasser, G. Edward Pendray, and 10 others in April 1930. The purpose of the society was to promote public interest in the notion of space travel. During this time, Goddard had moved his rocket tests to New

Mexico near Roswell. He launched a rocket that reached over 800 km per hour and over 600 m in altitude.

A year later in 1931, Lasser published his book *The Conquest of Space* in the United States, and the Austrians launched a mail-carrying rocket. In Germany, the VfR launched a liquid fuel rocket. Another of the noted rocket scientists of history, Wernher von Braun, was involved with the experiments of the VfR where he assisted Oberth with his liquid fuel rocket tests.

In 1933, the Soviets fired off a rocket that consisted of both solid and liquid engines. The launch was near Moscow and reached over 400 m. The same year, the American Interplanetary Society also launched a rocket that reached over 75 m. In 1933, Wernher von Braun was awarded a research grant from the German Army Ordnance Department to study rocketry and was subsequently awarded his doctorate of physics from the University of Berlin in July of 1934. The complete contents of his work were kept classified until 1960 or so, but we now know that the thesis was titled *Construction, Theoretical, and Experimental Solution to the Problem of the Liquid Propellant Rocket*. In 1934, as an outcome of his work, von Braun and his team launched two rockets that both achieved altitudes of 2.2 and 3.5 km. The same year, one of Goddard's rockets broke the speed of sound. In 1936, the California Institute of Technology began testing rockets near Pasadena, California. This eventually became the Jet Propulsion Laboratory.

The rocket development community continued along at a steady pace through the early 1930s. But, in 1937, when the Nazis disbanded the VfR, or rather conscripted them to Peenemünde on the shore of the Baltic Sea, the pace quickened. The German rocket scientists were concentrated in this facility to develop weapons for Hitler. The Germans, including Oberth, were led by von Braun to develop the most advanced rockets ever known to man. It should be noted here that, up until about 1939, von Braun had some level of technical correspondence with Goddard and in fact had used the American rocket scientist's publications and plans to design the Aggregat line of rockets. The most famous of this line were the V1 and V2.

Throughout the early 1940s, the German rockets, V1 and V2, saw continued development and use for military applications during World War II. The V2 could travel nearly 200 km and had a payload that could destroy entire city blocks. Once the war was over, some of the scientists were captured by Russia and the United States; von Braun was among those who went to the United States.

The war saw other aspects of rocketry as well. In 1941, the United States launched a rocket-assisted airplane and then launched its first air-to-air and air-to-surface rockets in 1942. And, in 1945, at the end of the war, the Secretary of War ordered over 50 of the captured German scientists delivered to White Sands, New Mexico, to work on further rocket development. On August 10 of that year, Goddard died of cancer.

Following the war through the late 1940s, the United States made some progress with the development of liquid hydrogen– and liquid oxygen–based rockets at the Lewis Research Center in Cleveland, Ohio. Walter T. Olson directed this work. Also, the Soviets and the United States began launching rockets on a fairly steady basis, trying to reach space or perhaps merely to develop what would one day become intercontinental ballistic missiles (ICBMs). In 1949, President Harry Truman signed a bill making Cape Canaveral (later renamed Cape Kennedy), Florida, a rocket test range, and the Secretary of the Army relocated the German scientists from New Mexico to Huntsville, Alabama. The small north Alabama farm town soon boomed into the "Rocket City," which, nowadays, is still home to one of the largest concentrations of advance-degreed rocket scientists, both military and civil, than any other place in the United States.

1.1.12 1957 to 1961 AD

The space race between the Soviets and the Americans exploded into public purview when the Soviet Sputnik 1 was launched into orbit on October 4, 1957. Sputnik (Figure 1.5) was not only the first vehicle launched into space by mankind, but it was also the first orbital vehicle. The small satellite and launch program was led by Sergei Korolev (Figure 1.6) who was the driving force of the Soviet space program. His identity was kept completely secret, and, even to the workers of the project, he was known only as the Chief Designer. He

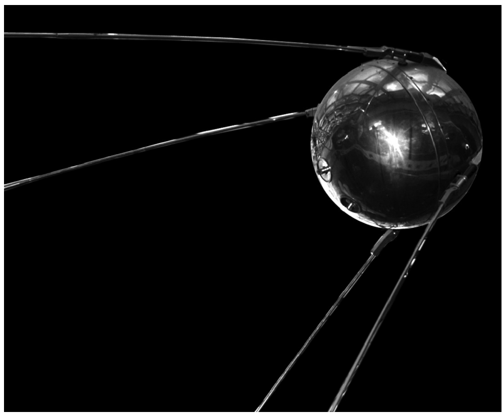


FIGURE 1.5
Sputnik 1 was the first orbital spacecraft launched by man. It was launched by the Soviets in 1957.

FIGURE 1.6
Sergei Korolev was the Chief Designer and the spark of the Soviet space program.

would continue his efforts in spacecraft development up until his untimely death in 1966. Korolev was, by trade, an aircraft designer but is described as being very good at what today would be called systems engineering. He used talents of design integration, program planning, and large effort organization to successfully build the first orbital rocket capabilities.

Sputnik triggered a great fervor, as well as fear, in the Americans. There was significant concern that, if the Soviets could launch a spacecraft into orbit, then they could just as easily launch nuclear weapons from space platforms. The driving force behind the Soviet program economically was the need to develop better ICBMs that could deliver nuclear payloads to the United States. It was Korolev who seized the opportunity to demonstrate spacecraft capabilities using the ICBM development programs.

In November of the same year, the Soviets launched Sputnik 2. This time, the spacecraft carried a dog, Laika. Laika was the first animal in space.

At the same time, the U.S. program was struggling to get off the launch pad and, in fact, had an unsuccessful firing of a Vanguard rocket. But, the next year, Satellite 1958 Alpha, dubbed Explorer 1, was launched on top of a modified Jupiter-C rocket developed by the Army Ballistic Missile Agency (ABMA). The satellite was the first to be launched by the United States, and its mission was to study the radiation enveloping the Earth. The project originators were William Pickering from the Jet Propulsion Laboratory, James Van Allen from the University of Iowa, and Wernher von Braun at the ABMA in Huntsville, Alabama. The Van Allen radiation belts around Earth were discovered with this mission. Figure 1.7 shows Pickering, Van Allen, and von Braun at the news conference after the successful mission holding up a model of the Explorer 1 spacecraft. Also, in 1958, the United States launched the first successful Vanguard rocket. Figure 1.8 shows the little spacecraft, which is still orbiting the Earth. It is the oldest artificial satellite orbiting the Earth, although it has lost power and is quiet. It was launched by a navy program to test three-stage launch vehicles. The Vanguard 1 telemetry data enabled scientists to discover that the Earth has

FIGURE 1.7
William Pickering, James Van Allen, and Wernher von Braun holding up a model of the Explorer 1 satellite. The rocket in the background is a model of the Jupiter-C launch vehicle. (Courtesy of NASA.)

FIGURE 1.8
The Vanguard 1 is the oldest artificial satellite orbiting the Earth. (Courtesy of NASA.)

an asymmetry and is shaped something like a pear, with the small end of the pear at the North Pole.

One of the most important actions for space exploration was made in 1958 when Congress approved the Space Act creating the National Aeronautics and Space Administration (NASA). NASA became the spearhead organization for all civil space research, development, and testing in the United States.

In 1959, the Soviets launched the Luna 1 spacecraft. Luna 1 was the first spacecraft to reach escape velocity of the Earth and travel onward into space to within 5,995 km of the Moon and then to travel onward into a heliocentric orbit (about the Sun) between Earth and Mars.

Luna 2 actually made it to the Moon and crashed there. The spacecraft impacted the surface of the Moon east of the Mare Serenitatis. Luna 2 is most famous for discovering the solar wind by using sensors that had been designed to detect ions in space. Luna 2 also confirmed that the Moon was lacking a magnetic field of any significance.

Luna 3 actually orbited the Moon and sent back images of the far side. The spacecraft used a camera to take photographs, which were developed onboard. The photographs were then scanned and sent back to Earth using technology very similar to a facsimile machine.

The year 1960 brought forth a new era for the world in weather prediction and warning. The first television weather satellite Tiros 1 was launched into orbit. Also, the first communications satellite, Echo 1, was launched. The Soviets launched two dogs on Sputnik 5 and successfully returned them to Earth. Strelka and Belka were the first cosmonauts to safely return home from space.

In 1961, the manned era of space exploration began. On April 12, 1961, the Soviet cosmonaut Yuri Gagarin became the first human to travel in space. He was launched atop a Vostok 1 with the call sign “Cedar.” His flight lasted 60 minutes, and the mission profile is shown in Figure 1.9. The Vostok 1 rocket was a direct derivative of the Soviet R-7 ICBM. The rocket burned kerosene and liquid oxygen as propellant. The rocket carrying Gagarin into space is shown in Figure 1.10.

On May 3, 1961, Alan Shepard became the first American in space with a suborbital flight. Shepard flew aboard the Freedom 7 spacecraft atop a U.S. Army–derived Redstone

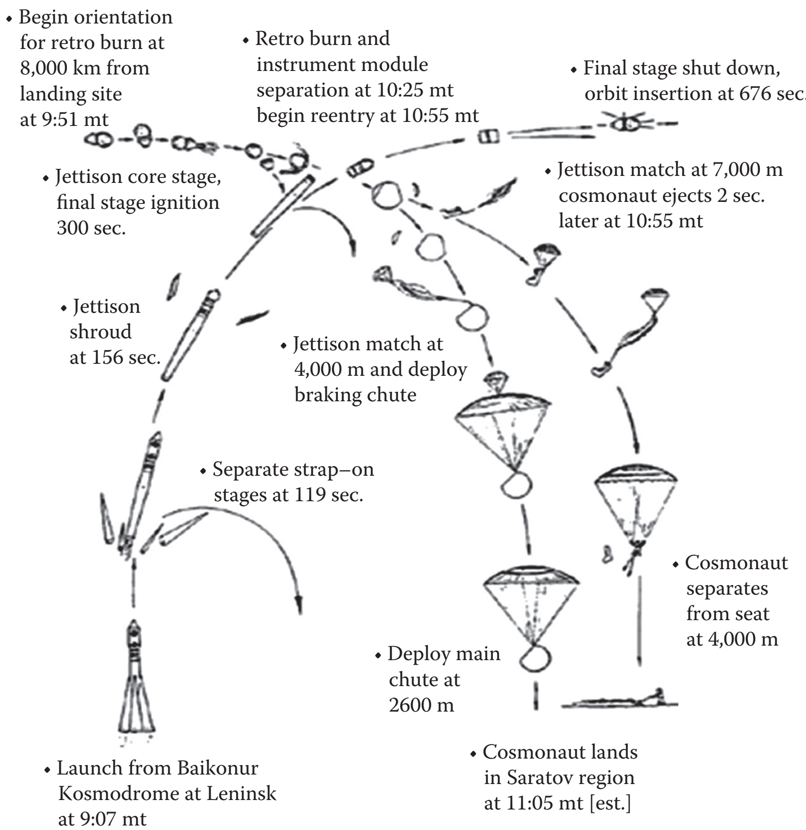


FIGURE 1.9
The flight profile of Yuri Gagarin's historic first manned flight into space aboard Vostok 1. (Courtesy of NASA.)

rocket (Figure 1.11). The rocket was developed by the ABMA in Huntsville, Alabama, and was derived from the German V-2 under the leadership of the German rocket scientist Wernher von Braun. The rocket burned alcohol and liquid oxygen.

On July 4, the United States flew a second suborbital flight with astronaut Virgil I. "Gus" Grissom. Then, on August 6, the Soviets orbited Gherman Titov for more than 25 h around the Earth, making him the first human to orbit for longer than a day.

1.1.13 1961 to Present

It was at this point (in 1961) that the manned space program started with extreme vigor fueled by the Cold War desire to be first to reach the Moon. The Soviets and the Americans launched flight after flight into space with new technologies and experiments to show that man could indeed perform in space. The improvements in rocketry were on levels of how much payload could be lifted into orbit and even to escape Earth and enter orbits about the Moon. Docking in space, living in space for prolonged periods of time, spacecraft control and guidance, and a sundry of other experiments eventually led to Americans walking on the Moon and Russians living for very long periods orbiting in space stations around the Earth.

FIGURE 1.10
The Vostok rocket launching Yuri Gagarin into space. (Courtesy of NASA.)

On July 16, 1969, a Saturn V spacecraft (Figure 1.12) launched Neil Armstrong, Edwin "Buzz" Aldrin, and Michael Collins into space and to the Moon on the NASA Apollo 11 mission. Once reaching the Moon, Armstrong and Aldrin descended in the lunar excursion module (LEM) (Figure 1.13a) to the lunar surface, where (on July 20, 1969) they were the first humans to set foot on an extraterrestrial body. The two men spent about 2.5 h on the lunar surface in extravehicular activity suits. They then launched the lunar module (LM) ascent stage and rendezvoused with Collins aboard the command module in lunar orbit. The three men returned to Earth and splashed down in the North Pacific Ocean on July 24, 1969. (*Author note:* That was my first birthday.)

The intricacies of rocket science and engineering were indeed refined during this Apollo era and long before there were handheld calculators, laptops, and smartphones. The modern rocket scientists often fail to comprehend the level of dependence on modern computational tools they have. Imagine how difficult the millions of calculations, designs, blueprints, meeting notes, and data logging that were done with pencil, paper, and slide rules. The Apollo era effort was indeed Herculean!

FIGURE 1.11
The Redstone rocket launching Alan Shepard and Freedom 7 into space. (Courtesy of NASA.)

At the end of the initial space race in the mid-1970s, the Cold War, Vietnam, and stressed economies of the United States and Russia could no longer afford to maintain the pace of spacecraft development. At this point in history, mankind did what has been described by some as the "great retreat from the Moon" in which humanity would not return at least until the second decade of the 21st century.

From the 1970s to the present, the experimentation with reusable launch vehicles was a particularly exciting development effort. The United States constructed a fleet of Space Shuttles (Figure 1.13b) that were partially reusable and implemented solid and liquid fuel systems. The Soviets attempted to copy the American Shuttle but had little success. The American program actually lost two of its Space Shuttles over the years of the program. The Space Shuttle Challenger exploded shortly after takeoff on January 28, 1986, killing all of its crew. On February 1, 2003, the Space Shuttle Columbia fell apart and disintegrated upon reentry, killing all of its crew.

The accidents and the cost of the Space Shuttle program, as well as the cost of the International Space Station construction for the most part, debilitated NASA and the American space program. From a manned-flight perspective, very little exciting events

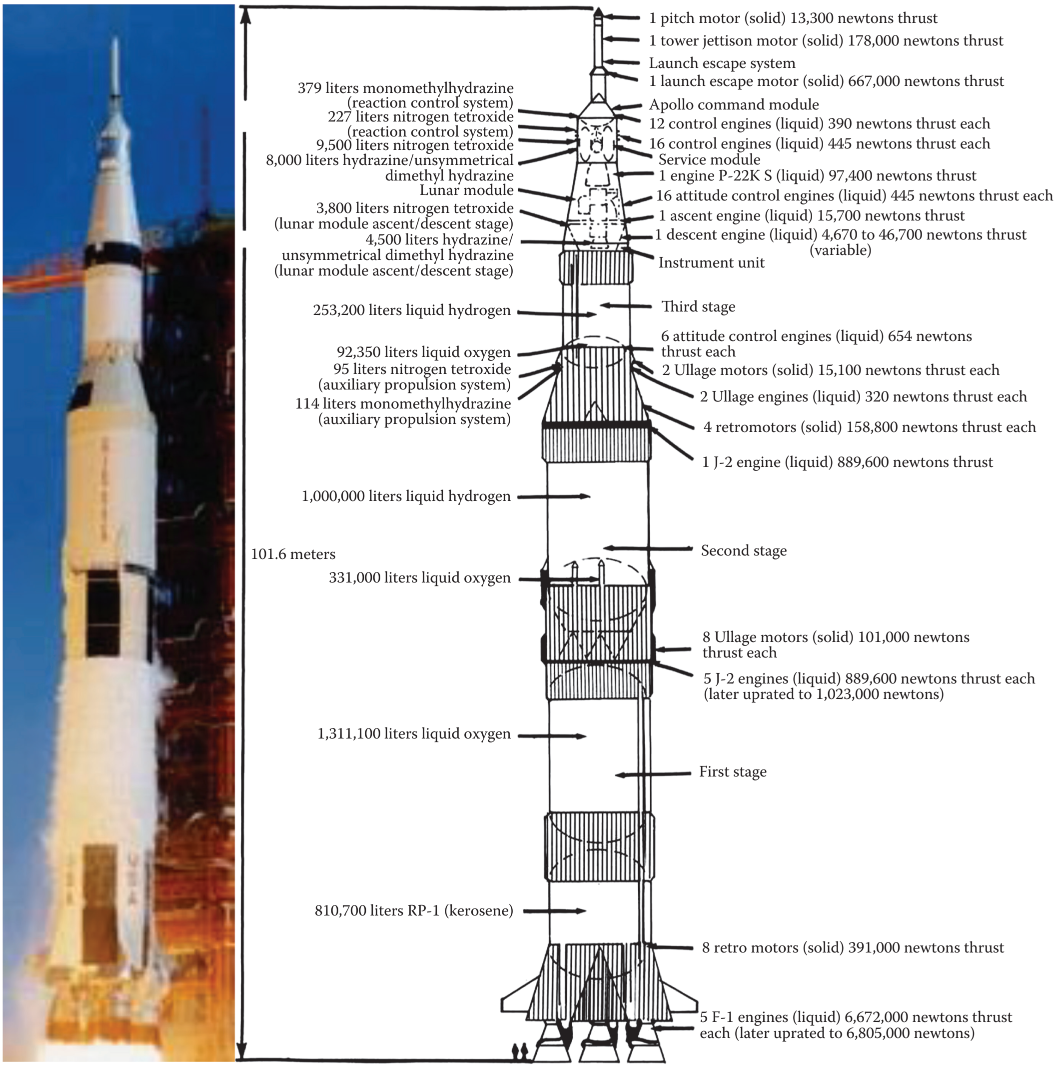


FIGURE 1.12
The Saturn V rocket launched the first manned flight to the Moon on July 16, 1969. (Courtesy of NASA.)

happened between the mid-1970s and 2004. There were program successes, and indeed the Space Shuttle was the most complex workhorse spacecraft ever designed by man. The rocket science and engineering involved in keeping the Shuttle flying required quite an effort and is why a major portion of NASA's budget was absorbed by it.

There were a few bright spots that occurred during this time frame. The Chinese developed a manned space program and joined the Russians and the Americans as having their own manned space vehicles. The Chinese based their launch systems on the ICBM technology of the Dong Feng Missile systems. They created a family of rockets called the Long March rockets that propelled the Shenzhou 5 spacecraft (Figure 1.14) carrying Yang Liwei into orbit on October 15, 2003. The Chinese have had other manned missions and began an unmanned lunar exploration program with hopes of landing a Chinese Taikonaut on the

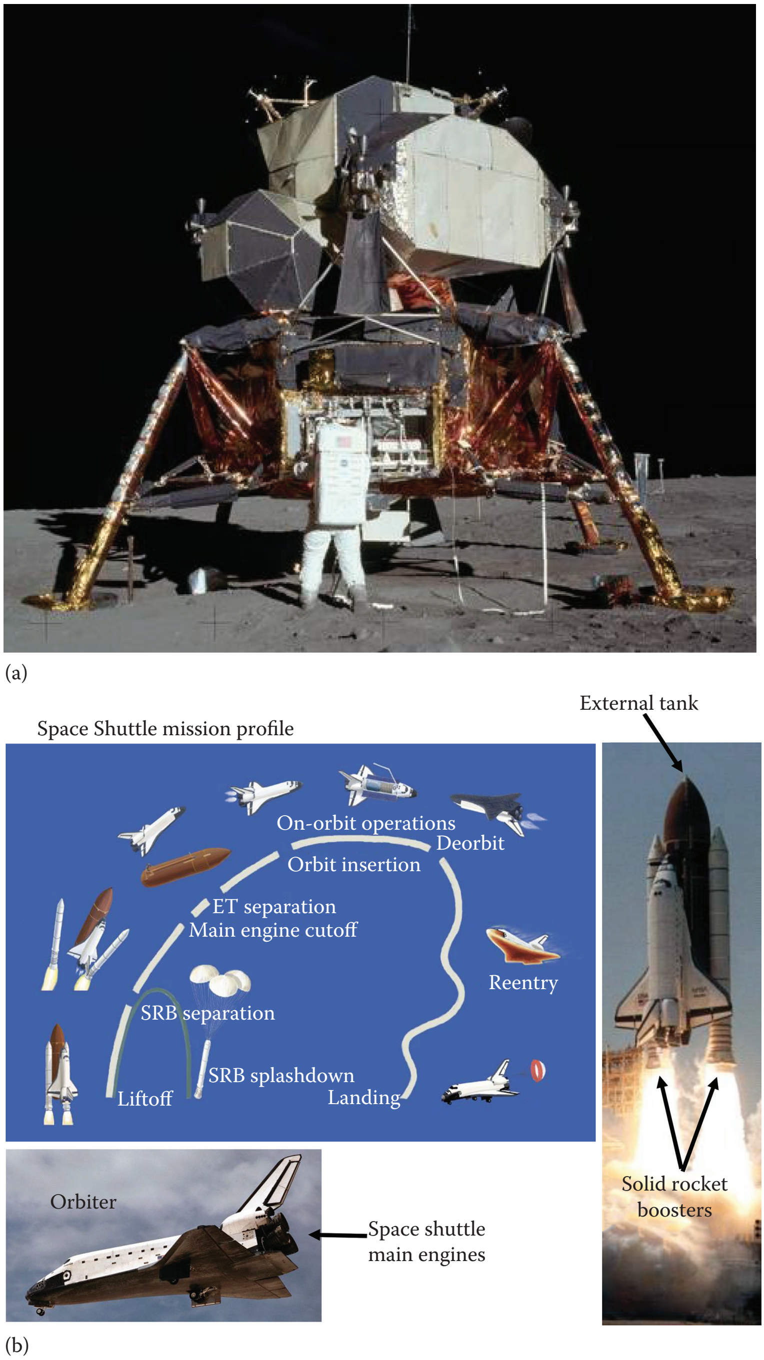


FIGURE 1.13
(a) The LEM used rockets to descend to and ascend from the Moon. (b) The Space Shuttle uses reusable SRBs, three SSMEs, and an expendable ET. The Orbiter carries the crew and payload and reenters using thermal protection tiles. The vehicle lands like an unpowered airplane. (Courtesy of NASA.)

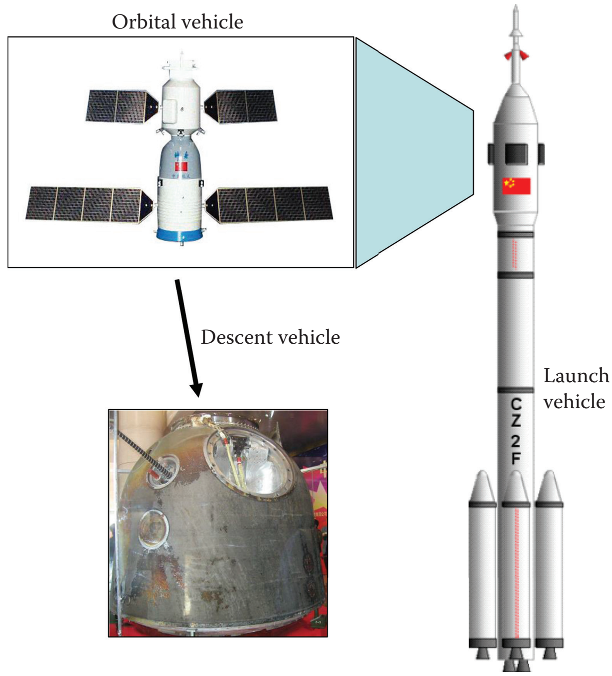


FIGURE 1.14
The Shenzhou 5 spacecraft carried the Chinese Taikonaut Yang Liwei into Space on October 15, 2003. The capsule was launched on a Long March rocket. (Modified GNU Free Documentation License images.)

Moon in the near future. The Chinese vigorous- manned space program plan sparked new interest in the American space community to return to the Moon.

1.1.14 X PRIZE

In 1995, Dr. Peter Diamandis addressed the National Space Society's (NSS's) International Space Development Conference and suggested a prize for nongovernment-funded rocket programs to demonstrate the first truly reusable manned spacecraft. The X PRIZE was modeled after many of the aviation prizes, such as the Orteig Prize that Charles Lindbergh won for his solo flight across the Atlantic Ocean. The X PRIZE was described as a $10-million award to the first team to launch a piloted spacecraft, carrying at least three crewmembers (or one human and mass equivalent of two others) to a 100-km altitude and return safely. Then, the mission would have to be repeated within two weeks from the first launch. The vehicle had to be the same with less than 10% of the vehicle replaced between missions. Several teams began developing launch vehicle concepts following the prize being announced.

On June 21, 2004, Scaled Composites, the small company owned by Burt Rutan, teamed with Tier One and Mojave Aerospace Ventures and launched the SpaceShipOne on its maiden suborbital flight. The venture was called the Tier One project. The spacecraft was completely developed by commercial or private funds and implemented several innovations. The spacecraft consisted of two stages. A larger mother ship called the White Knight carried the SpaceShipOne to an altitude of 14 km where it launched from there to over 100 km. The vehicle implemented a unique reentry design of moving wings that

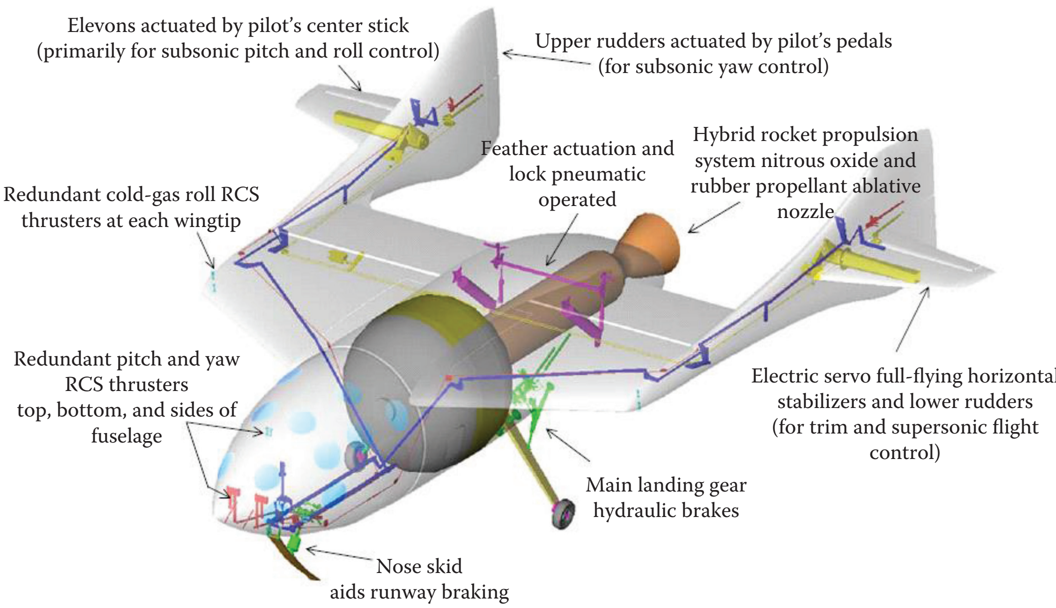


FIGURE 1.15
SpaceShipOne architecture overview shows the basic components of the spacecraft vehicle. (Courtesy of © 2004 Mojave Aerospace Ventures LLC, photograph by Scaled Composites. SpaceShipOne is a Paul G. Allen Project.)

adjusted to the drag as the atmosphere grows denser as the vehicle plummeted back to Earth. Once the vehicle was in thick-enough air to fly as an aircraft, it did so, gliding back to Earth safely. A second launch of the completely reusable spacecraft was successful on October 4, 2004. Mike Melvill piloted the first flight and Brian Binnie, the second. This effort was truly the first commercial manned spaceflight venture. Figure 1.15 shows an overview of the SpaceShipOne architecture. Figure 1.16 shows the White Knight carrying the SpaceShipOne vehicle to launch altitude and the SpaceShipOne on the in flight and on the runway. As can be seen in Figure 1.16, SpaceShipOne was designed with an inherently stable reentry system. The rotating wings of the vehicle adjust the angle of attack based on the atmospheric drag against them. The wings, therefore, remove the need for an exotic and much more expensive thermal protection system, such as using a capsule with a heat shield and then ballistic recovery parachutes or the thermal protection tiles that are used on the Space Shuttle. The design is nothing short of brilliant, as well as inexpensive. The actual cost of the Tier One project is often debated, but is approximated to be somewhere around $20 million in 2004. That is very inexpensive compared to other manned spacecraft designs.

1.1.15 Other Space Agencies

From the beginning of the space race to the present, several other countries have developed spacecraft and rocket technologies. Only the Americans, Russians, and Chinese have successful manned programs. However, the European Space Agency (ESA) has had astronauts fly on the Space Shuttle and on Russian Soyuz missions to the International

FIGURE 1.16
The White Knight carrying SpaceShipOne, SpaceShipOne spacecraft in flight, and on the runway. (Courtesy of © 2004 Mojave Aerospace Ventures LLC, photograph by Scaled Composites. SpaceShipOne is a Paul G. Allen Project.)

Space Station. Other countries, such as Canada and Israel, have flown astronauts on the American or Russian vehicles as well.

Many countries or coalitions of countries have their own space agency, but only a handful of them have their own launch vehicle core. Some of these include the following:

- ESA
- Centre National d'Etudes Spatiales (CNES—France)
- Indian Space Research Organization (ISRO—India)
- Iranian Space Agency (ISA—Iran)
- Israeli Space Agency
- Japan Aerospace eXploration Agency (JAXA—Japan)
- China National Space Administration (CNSA—People's Republic of China)
- Russian Federal Space Agency (FSA or RKA—Russia/Ukraine)
- NASA, U.S. Air Force (United States of America)

1.2 Rockets of the Modern Era

To begin understanding rocket science and engineering, it is a good supplement to have some knowledge of the rockets in use by many of the space agencies around the world. Therefore, in this section, a description of the launch vehicles of the space agencies listed in Section 1.1.15 is provided. New vehicles are continually being developed, so it is impossible to determine if this list is exhaustive or not. Also, many countries have missile programs or have purchased missiles with at least some suborbital capabilities; those will not be discussed here.

1.2.1 ESA and CNES

Currently, the ESA has the Ariane 5 and the Soyuz launch vehicles. The Ariane 5 is truly an ESA vehicle, whereas the Soyuz is purchased from the Russians. Figure 1.17 shows the details of the Ariane 5. More details of the Soyuz will be discussed in Section 1.2.7. It should also be noted that the ESA is developing a launch vehicle called Vega in cooperation with the Italian Space Agency. The Vega rocket is expected to be a single-body launcher with three solid stages and one upper liquid stage. It should be noted

Height	46–52 m
Diameter	Up to 5.4 m
Liftoff mass	746 Metric tons
Payload mass	6 Metric tons GTO 9.5 Metric tons LEO

FIGURE 1.17
The Ariane 5 Launch Vehicle is the primary rocket for ESA payloads. (Modified GNU Free Documentation License images.)

that CNES is also a partner with the Ariane 5 launch vehicle, and it is its primary vehicle as well.

1.2.2 ISRO (India)

Figure 1.18 shows the Polar Satellite Launch Vehicle (PSLV) and the Geosynchronous Satellite Launch Vehicle (GSLV) of the ISRO fleet. These two rockets have been used to launch many satellites into low Earth orbit (LEO) and to a geosynchronous Earth orbit (GEO). The PSLV has been flying since 1993 and the GSLV since 2001.

1.2.3 ISA (Iran)

The ISA has been developing satellite launch vehicles based on the North Korean Taepodong 2 missile system. The Shahab family of rockets' (Figure 1.19) main purpose is as that of the ICBMs, although the ISA is continuing research in space launch technology

Polar satellite launch vehicle (PSLV)

Height	44 m
Diameter	1.0–2.8 m
Mass	294,000 kg
Stages	4 with 6 solid strap-ons

Geosynchronous satellite launch vehicle (PSLV)

Height	49 m
Diameter	3.4 m
Mass	401,000 kg
Stages	3 with 4 liquid strap-ons

FIGURE 1.18
PSLV and the GSLVs of the ISRO.

FIGURE 1.19
The Shahab-3D suborbital rocket has a range of approximately 3,225 km.

improvements of these rockets. It is also known that the Iranian scientists have been present at many North Korean rocket tests.

1.2.4 Israeli Space Agency

Figure 1.20 illustrates the Shavit launch vehicle that has been launched four times. Two of the flights were unsuccessful. It is a three-staged system with solid rocket motors; however, there is a fourth-stage option that is a liquid engine. It should also be noted here that the Israeli Space Agency has opted to use the ISRO's PSLV for some of its spy satellites. To date, the PSLV has proved to be a much more reliable launch system than the Shavit.

1.2.5 JAXA (Japan)

The Japanese H-IIA and H-IIB launch vehicles are pictured in Figure 1.21. The rockets are liquid-fueled systems developed by Mitsubishi and ATK Thiokol. The H-IIA began flying in 2001, and the H-IIB first flew in 2006 and is still under development. The H-IIB rocket is to be the vehicle that JAXA will use to support the ISS in the future.

The M-V line of launch vehicles was designed by Nissan using the LG-118A Peacekeeper ICBM as a model. There were initial concerns that the M-V resembled an ICBM more than a launch vehicle. The vehicle is all solid propellant. The Japanese political leaders insist that the M-V is a space launch vehicle, even though there are some in the missile community that suggest the M-V could be converted into a weapon very easily and rapidly.

FIGURE 1.20
The Shavit launch vehicle of the Israeli Space Agency. (GNU Free Document License image.)

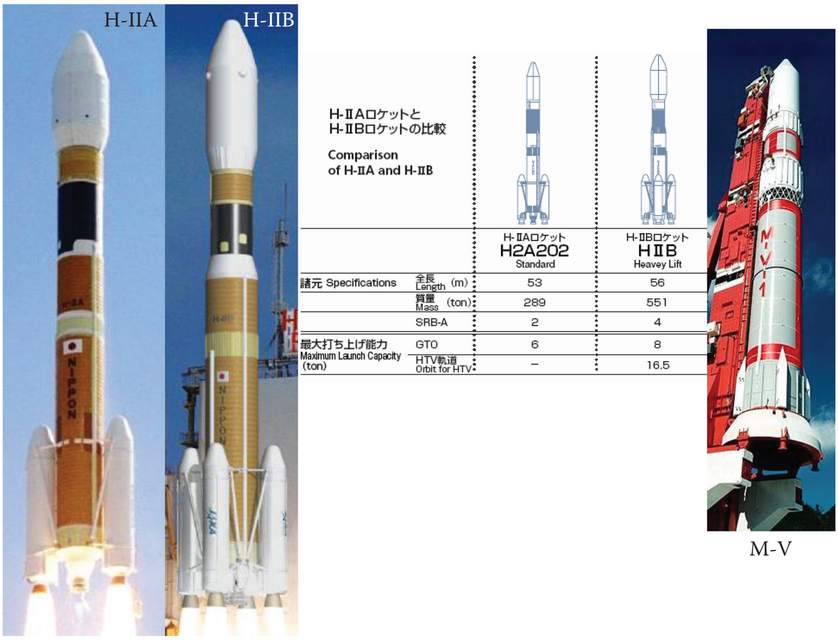


FIGURE 1.21
The Japanese H-II and M-V launch vehicles. (Courtesy of JAXA.)

1.2.6 CNSA (People's Republic of China)

The Long March family of rockets shown in Figure 1.22 has been in evolution since the 1970s and is mostly derivatives of the Dong Feng ICBMs. In English, the rocket nomenclature for the Long March rockets is sometimes an LM and sometimes a CZ. More commonly, the CZ seems to be used for some reason. The rockets typically use liquid propellants, such as unsymmetrical dimethylhydrazine (UDMH) and a tetroxide (dinitrogen tetroxide) oxidizer. The rockets range in capabilities from small payloads for LEO to heavy payloads for GEO.

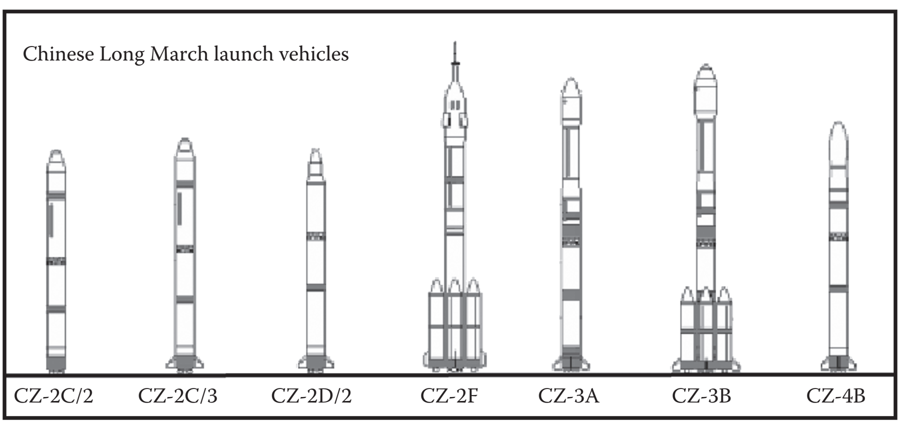


FIGURE 1.22
The Chinese Long March Family of launch vehicles.

The CZ-2F, which is also shown in Figure 1.14, was used to lift the first Chinese Taikonauts into orbit. The rocket can lift as much as 9,200 kg to LEO. The CZ-3B can lift as much as 5,100 kg to geosynchronous transfer orbit (GTO). The CNSA is developing future evolutions of the Long March rockets in order to increase payload to orbit capacity.

From a business standpoint, the Long March launch vehicles have launched many satellites. A significant number of the Iridium satellites were launched on CZ-2Cs. The Chinese space program continues to be very active and, in 2006, launched as many as 6 Long March rockets and as many as 94 since 1970.

1.2.7 Russian FSA (also known as RKA in Russian—Russia/Ukraine)

Actually, the National Space Agency of Ukraine (NSAU) and the RKA are two separate entities, although the NSAU launches are in cooperative programs with the RKA. Clearly the Russian and the American space race led to both countries having very mature launch vehicle programs. Figure 1.23 shows the Russian/Ukraine families of rockets currently in use. The launch vehicles range in capabilities from the Rockot's small 2 tons for LEO to the Proton's 5 tons to interplanetary orbit capabilities. The Rockot is a three-stage, liquid system based on the former Soviet ICBM designated UR-100N (SS-19 by the U.S. Department of Defense [DoD]). There was also an ESA version called the Eurockot.

The Kosmos rocket is a two-stage, liquid-fueled launch vehicle. It is capable of lifting 1.5 tons to LEO. The latest versions of the Kosmos rocket are designated as the Kosmos 3MU and implements digital control over the second stage burning of fuel for a more efficient rocket system.

The Dnepr is a converted ICBM three-stage, liquid-fueled rocket system. It is capable of launching 3 tons to LEO and 2 tons to a sun-synchronous orbit. Another claim to fame

FIGURE 1.23
The Russian and Ukraine launch vehicles demonstrate a heritage of space launch capabilities.

for the Dnepr is that it launches the so-called CubeSats on a secondary payload fairing. The CubeSats are small satellite busses that are about 10 cm on a side and often built by students and amateurs.

The Tsyklon was based on the R-36 ICBM and is a liquid-fueled, three-stage rocket manufactured in the Ukraine. The rocket's third stage is restartable.

The Soyuz rocket was originally based on the R-7 ICBM. The rocket is a liquid-based, three-stage launch vehicle and has become the most used launch vehicle in the world. The Soyuz vehicle is used to launch Progress supply spacecraft to the ISS. The Soyuz also is the launch vehicle for the Soyuz spacecraft, which are used to carry crew to the ISS, as well as other manned cosmonaut missions. The vehicle is also the first manned vehicle to carry a true "space tourist" Dennis Tito, who reportedly paid $20 million for a ride to the ISS and back.

The Zenit rocket is a three-stage liquid vehicle. It is capable of lifting about 5 tons to GTO and about 13 tons to LEO. It is manufactured in the Ukraine.

The Proton is *the* heavy lift vehicle for the Russians. It is comparable to the American Delta IV or Atlas 5 in capabilities. The liquid-fueled rocket is capable of launching over 22 tons to LEO and more than 5 tons to interplanetary destinations. It was originally designed to be a "super ICBM" capable of throwing very large nuclear warheads (more than 10 megatons) to ranges of more than 12,000 km. It was also considered as a launch vehicle to launch a two-manned spacecraft into a lunar injection orbit, but that was never attempted. The Proton rocket was used to launch the Salyut space stations, several Mir modules, and the Zarya and Zvezda modules for the ISS.

1.2.8 United States of America: NASA and the U.S. Air Force

Until about 2011, the United States had the largest fleet of launch vehicles in operation, and they consisted of both manned and unmanned systems. The previous manned program was NASA's Space Shuttle program, which is known officially as the Space Transportation System (STS). A picture of the Space Shuttle is seen in Figure 1.13. The STS fleet consisted of the *Discovery*, *Atlantis*, and the *Endeavor*. The fleet also included the *Enterprise,* which was a test vehicle that only flew in drop tests; the *Challenger,* which exploded 73 sec after launch on January 28, 1986; and the *Columbia,* which disintegrated on reentry on February 1, 2003.

Although the STS program had two major accidents in which both crews were lost, the shuttle had a long successful flight history of placing large payloads into orbit, including the Hubble Space Telescope and several of the components and modules for the ISS. The shuttle fleet was the workhorse of the American space program from the 1980s through the first decade of the 21st century. Between 1981 and 2011, the system flew 135 missions. The launch system could lift approximately 23,000 kg into LEO and at many different inclinations. The STS program was scheduled for decommission in 2010 and was officially put out to pasture in 2011. Unfortunately, there is no current replacement manned launch system in place to perform the shuttle's duties of carrying payloads and astronauts to LEO. However, NASA does have a development program under way for the shuttle's replacement (this will be discussed later in this section).

The United States also has quite a fleet of unmanned launch vehicles. Figure 1.24 shows the major workhorses of this fleet. These widely used expendable launch vehicles are the Atlas family built by Lockheed Martin and the Delta family built by Boeing. After some question of industrial espionage by Boeing, Lockheed Martin and Boeing formed the United Launch Alliance, which now acts as a one-stop shop for the U.S. government and

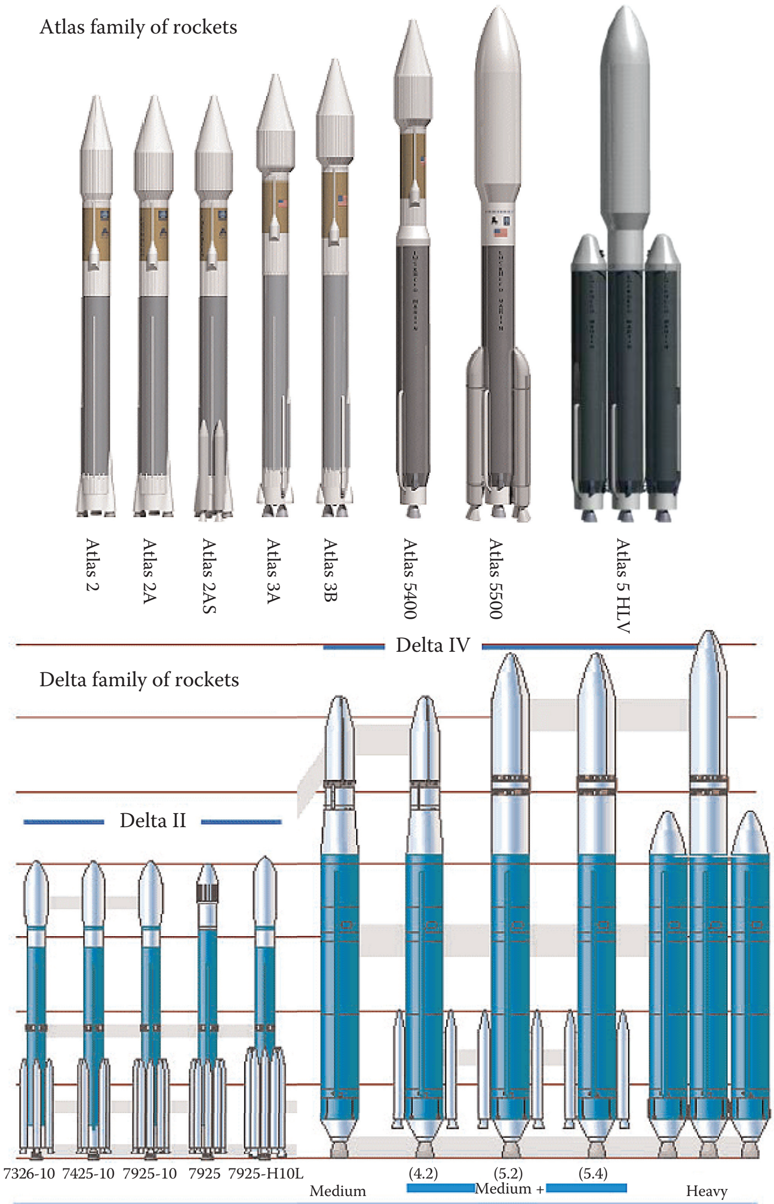


FIGURE 1.24
The U.S. EELVs demonstrate a heritage of space launch capabilities and are the current workhorses of the U.S. unmanned space fleet. The Atlas and Delta rockets were developed by Lockheed Martin and Boeing, respectively. (Courtesy of NASA.)

commercial customers to purchase a launch. Both Atlas and Delta rockets are constructed at the Boeing rocket plant in Decatur, Alabama, and the engineering operations are conducted at a Lockheed Martin complex in Littleton, Colorado. The two families of launch vehicles are similar in design and capabilities and were developed through the air force's evolved expendable launch vehicle (EELV) program.

The Delta IV rockets implement liquid hydrogen and liquid oxygen for propellant and oxidizer on the main stage and, in some cases, use added "strap-on" solid boosters made by Alliant Techsystems. The Delta I, Delta III, and the Atlas rockets use refined petroleum-1 (RP-1) and liquid oxygen for propellant and oxidizer, respectively, for their main-stage boosters.

Interestingly enough, the heavy-payload workhorses of the fleet, the Delta IV and the Atlas V, are similar in construction and as a requirement by the EELV program have the same payload fairing designs. The Atlas V uses a Russian-derived engine now manufactured by Pratt & Whitney known as the RD-180 for its main stage engine. The Delta IV uses a redesigned, modernized, and simplified version of the Space Shuttle Main Engines (SSMEs) built by Rocketdyne known as the RS-68.

The two families of vehicles can carry payloads into LEO and to GTO. The maximum payload to GTO is just under 11,000 kg for the Delta IV and just over 8,000 kg for the Atlas V. Several interplanetary probes have been launched on the Atlas V. While NASA seems to have historically preferred the Atlas V for its launches, the DoD, for some reason, has historically preferred the Delta IV. Lately, in the past few years since the formation of ULA the preferred launch vehicles for the United States has been the Atlas V.

1.2.9 Other Systems Are on the Way

There are other launch vehicles being tested and flown in the United States by commercial and government teams. Boeing created an international venture with Russia, Norway, and Ukraine to develop a rocket system based on the Zenit rocket that launches from the sea. The venture is known as Sea Launch and has had over 20 successful launches.

Lockheed Martin has successfully launched the Athena rocket many times and in different configurations. Lockheed Martin has also had several other successful launch vehicles, such as the Titan systems that have recently been decommissioned.

The company SpaceX founded by Elon Musk (cofounder of PayPal) has and is developing a family of launch vehicles to be competitive with the Delta and Atlas rockets known as the Falcon rockets. Figure 1.25 shows the Falcon 1, Falcon 5, and Falcon 9 rockets. The development of the vehicles has been supported by the Defense Research Projects Agency (DARPA), NASA, and the U.S. Air Force, as well as personal venture investments.

The Falcon 1 was launched in March 2006 but failed after 29 sec into flight. SpaceX believes they understand what caused the malfunction and have long since corrected it. Not long into the Falcon 1 effort the company to a change in direction and decided to focus mainly on the heavy launch vehicle Falcon 9 and moved forward with its development.

The Falcon rockets are still being developed and SpaceX has been contracted by the U.S. Air Force for several launches. In August of 2006, NASA awarded SpaceX a contract for $278 million to develop the Falcon 9 to deliver a manned capsule, called Dragon, to the ISS. In 2011 the contract was amended and increased to a value of about $396 million. This contract addition was for Falcon 9 development, testing, and Dragon capsule testing and demonstration. Also, in 2008, NASA gave SpaceX a contract for $1.6 billion for 12 resupply missions to ISS. The Dragon space vehicle and is supposed to be able to deliver seven crewmembers or cargo to LEO for ISS missions. The SpaceX Dragon is shown in Figure 1.26.

It should be noted here that the media, as well as SpaceX, NASA, and various politicians, claim that the Falcon 9 is a "commercial space" vehicle and was developed purely by private industry. From the previous paragraph, it is clear that this is not the case. The Falcon 9 development was supplemented by the U.S. government to the tune of almost $2 billion. Even Elon Musk admits that the vehicle would have matured much more slowly without

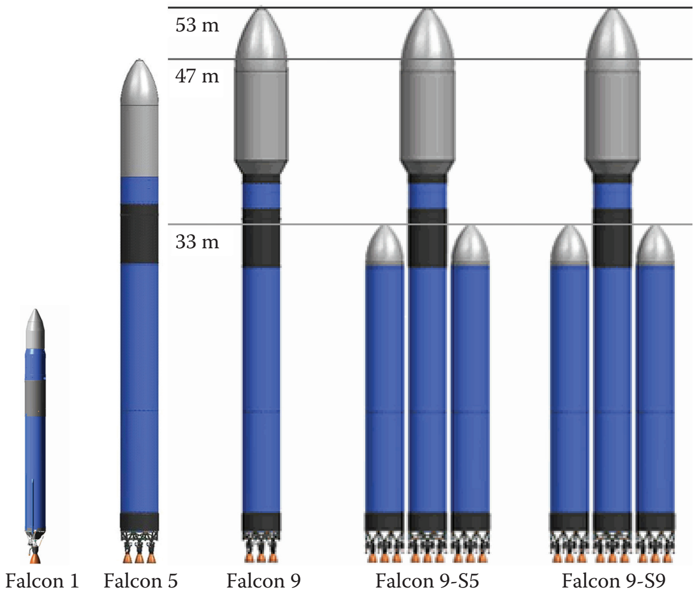


FIGURE 1.25
The SpaceX Falcon family of launch vehicles will compete with the Atlas and Delta rockets. (Modified GNU Free Documentation License images.)

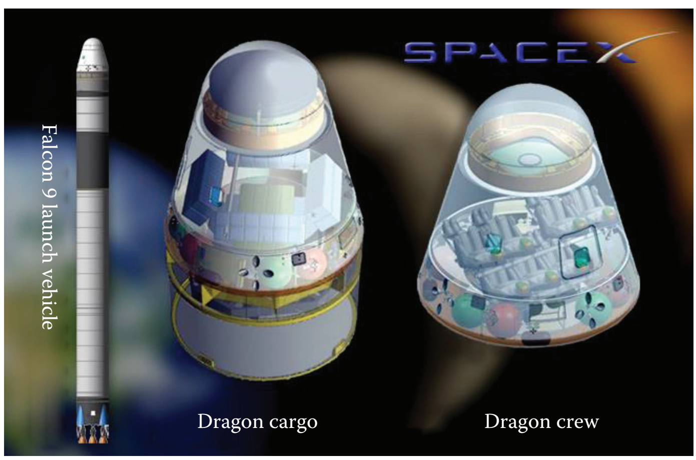


FIGURE 1.26
The SpaceX Dragon will launch on a Falcon 9 and carry a crew of seven to the ISS. (Courtesy of DARPA.)

the NASA contracts. It seriously does not take a rocket scientist to see that the Falcon 9 is mostly an extension of the EELV program but with a different contractor and spacecraft/rocket design concept.

The Falcon 9 has had fairly good success. It has been developed in three versions: The Falcon 9 v1.0 (retired), Falcon 9 v1.1 (retired), and the current version the Falcon 9 full thrust. Its current configuration is a two stage vehicle which implements nine Merlin 1D engines on stage 1 and one Merlin 1D Vacuum rated engine for stage 2. It can carry over 13,150 kg to LEO and as much as 5,300 kg to geosynchronous transfer orbit (GTO).

Currently SpaceX is testing a concept to reuse the stage 1 booster of the Falcon 9 by having it return to the landing site under rocket powered flight. Several tests of the Falcon 9R (reusable) have occurred to various degrees of success. On December 21, 2015, a first-stage booster was successfully brought back to Earth under rocket-powered flight and landed safely.

A company created by Richard Branson of Virgin Group and by the X PRIZE winning team Tier One (see Section 1.1.14) known as Virgin Galactic is currently developing a SpaceShipTwo/White Knight Two combination for launching paying customers into suborbital flights. A SpaceShipThree/White Knight Three combination for LEO flights would be the next phase. The launch vehicle system is based on the SpaceShipOne concept.

1.2.10 NASA Constellation Program

On January 14, 2004, President George W. Bush announced his "Vision for Space Exploration." This vision included the following steps:

- Complete ISS by 2010. (accomplished)
- Retire STS by 2010. (accomplished)
- Develop the Orion space vehicle formerly known as CEV (Crew Exploration Vehicle) by 2008 with its first manned mission by 2014. (never happened)
- Develop so-called "Shuttle-Derived Launch Vehicles." (stalled by 2009)
- Explore the Moon with unmanned missions by 2008. (never happened)
- Launch manned missions to the Moon by 2020. (unlikely at present pace)
- Explore Mars and other destinations with unmanned and manned missions (although no time frames given for these). (maybe someday)

Following President Bush's announcement, NASA went through several months of study efforts to determine the best approach to carry out this vision with the budget available. NASA decided on a move away from the aircraft-looking vehicle designs, such as the Space Shuttle and other X-vehicles developed through the 1980s and 1990s, to a more familiar looking rocket design. Figure 1.27 shows the Ares I and Ares V rockets that were down-selected to by the NASA launch vehicle community. The designs use "typical-looking" rockets with a space capsule atop them. The space capsule was to operate and reenter much in the same way as the capsules of the Apollo and Soyuz era doing away with the need for the exotic tiles that caused so many problems within the STS program.

The Ares I vehicle was to be developed first and was slated for flight testing in the 2009 time frame. The Ares I was to be used to launch the Orion space capsule to LEO carrying astronauts to space, as well as some cargo. This vehicle was to be used to transport astronauts to the ISS when the shuttle was retired. The Ares I main stage was a five-stage solid rocket booster (SRB) derived from the STS SRB design. The second stage engine was a liquid hydrogen and liquid oxygen (LH_2/LOX) engine derived from the Saturn IB and Saturn V J-2 engine, called the J-2X. Note that the "spike" on top of the Orion craft, which sits atop the Ares I, was an abort booster that could lift the spacecraft away from the launch vehicle in the event of an emergency.

The Ares V was to be the heavy lift–capable vehicle that would carry cargo or, for Moon missions, the Earth Departure Stage (EDS) and the Lunar Surface Access Module (LSAM). The EDS would dock with the Orion spacecraft in LEO and then travel onward to the Moon. The Ares V first stage was to consist of five RS-68 LH_2/LOX engines and two

FIGURE 1.27
The Ares V and Ares I launch vehicles of the NASA Constellation Program are the two vehicles to take America back to the Moon and beyond. (Courtesy of NASA.)

shuttle-derived SRBs. The EDS was the second stage and was to be propelled by a J-2X engine. The NASA plan was to have the first manned Moon mission by 2019. Figure 1.28 pictures the Saturn V of the Apollo program, the shuttle, and the Ares rockets for comparison. It should be noted here that although the rockets look like a "blast from the past," the systems were to be completely modern based on lessons learned through the Apollo and STS programs. The cockpit of the Orion spacecraft was planned to be a "glass cockpit," meaning that the control systems are computer based rather than older avionics displays and instruments. It was also planned to reenter like the Apollo capsules did, then parachute to Earth, and land on ground like the Soyuz spacecraft.

Illustrated in Figure 1.29 is the planned configurations for the Orion crew vehicle and the LSAM (which later became known as Altair). The Orion vehicle was be similar to the Apollo Command and Service Module (CSM) and the LSAM was to be similar to the Apollo LEM. However, as with the Ares I and Ares IV, the Orion and LSAM would be modern spacecraft with new technologies and updated systems (according to NASA). They were to also each be able to carry more payload and passengers.

Figure 1.30 is a slide from a NASA briefing on the expected mission profile for the Ares I, Ares V, Orion, and Altair vehicles. This diagram will help the readers in understanding what the major components of these rocket systems were to be used for in the Moon mission scenario. Even though the missions will never take place using these concepts, any future missions to the Moon using standard modern rocket technology and architectures would be similar in steps and activities to the Constellation mission concept.

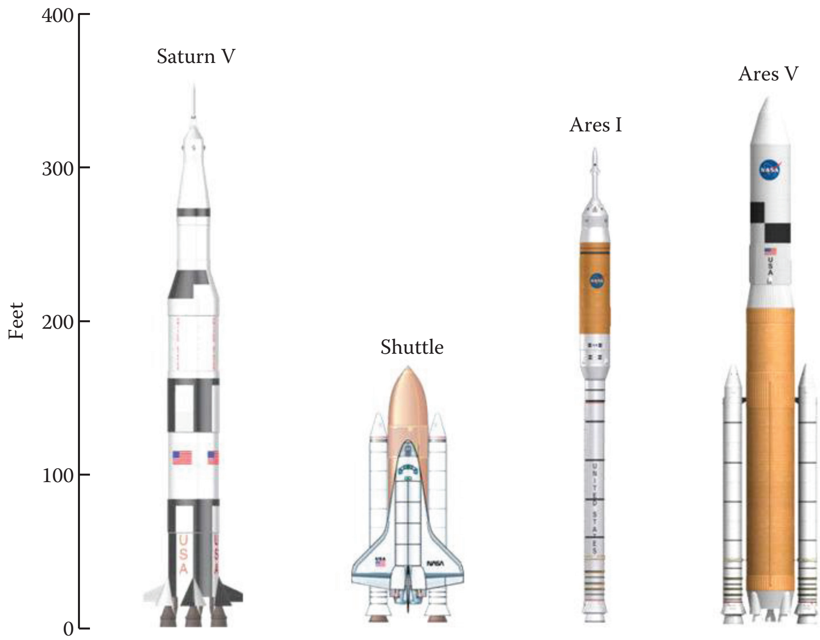


FIGURE 1.28
The major launch vehicles of NASA compared. The Ares I and Ares V vehicles will do the missions of both the Saturn V and the Space Shuttle. (Courtesy of NASA.)

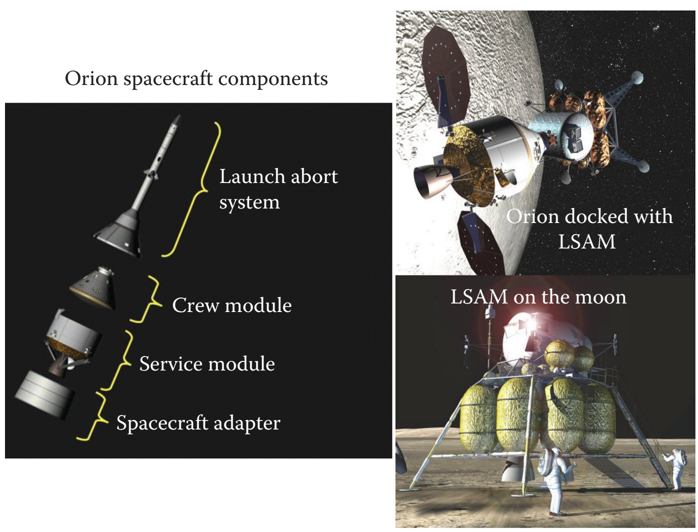


FIGURE 1.29
The Orion spacecraft and the LSAM. (Courtesy of NASA.)

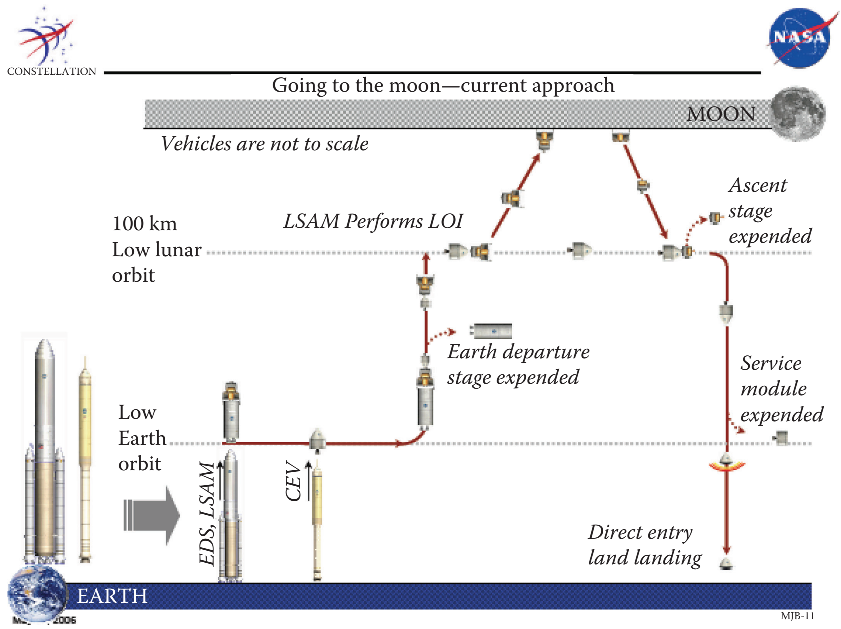


FIGURE 1.30
The Constellation Program mission profile for going to the Moon. (Courtesy of NASA.)

In February of 2010, President Barrack Obama proposed cancelling the Constellation program. After many activities from the White House and Capitol Hill, a new rocket program known as the Space Launch System (SLS) was started.

1.2.11 NASA SLS Program

The latest rocket development program at NASA is the SLS, which will take concepts of the Ares I and Ares V from the Constellation program and combine them into one rocket known as the Space Launch System. There is no particular plan such as going to the Moon for the SLS, but it is being designed to be a workhouse superheavy lift capability. Figure 1.31 shows an artist's rendering of the SLS Block 1 in flight. The Block 1 will employ both parallel staging through shuttle-derived SRBs on the side of stage 1 and serial staging using four shuttle-derived main engines on stage 1 and one RL10B-2 engine (EELV upper-stage engine) on stage 2. It is planned that the SLS Block 1 will be capable of putting between 70,000 and 130,000 kg into LEO. Future variants of the SLS are proposed to have similar capabilities. The SLS Block 1 engines all use liquid hydrogen and liquid oxygen for propellants.

It should be noted here that this section has been an overview of some of the rockets of the modern era. There are likely many candidates that were not discussed due to the fact that the field of rocketry is ever growing across the planet (such as the Orbital Sciences Corporation's Pegasus rocket that can carry small payloads to LEO). The commercialization of space has sparked several companies around the globe to begin rocket development efforts. Hopefully, this section has educated the student on the types of rocket systems that are available and some of the details about them.

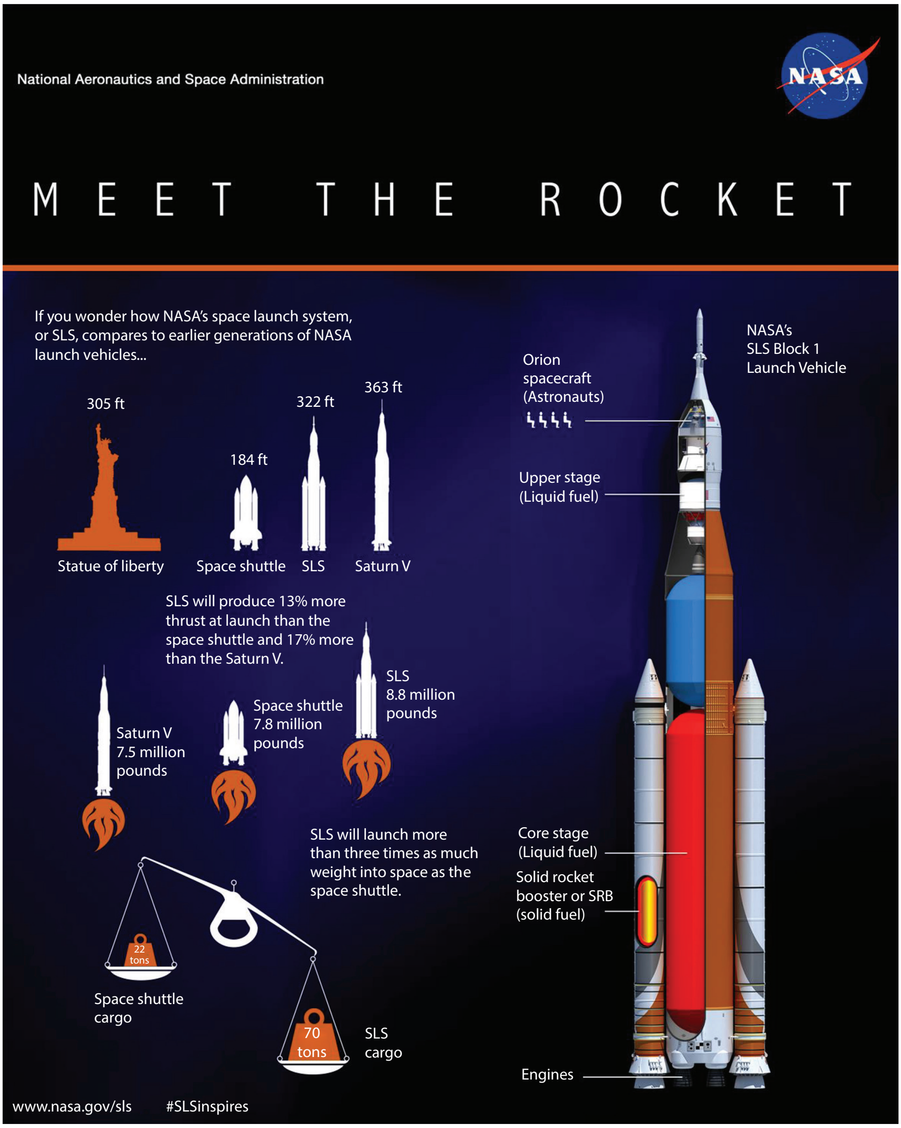


FIGURE 1.31
Space Launch System. (Courtesy of NASA.)

1.3 Rocket Anatomy and Nomenclature

So what are rockets made of? What are the pieces? Before discussing the physics and engineering principles behind rocketry, it is a good idea to learn some of the basic language of the discipline. It is likely throughout Sections 1.1 and 1.2 that there were some terms that were confusing and perhaps even foreign to a student first being exposed to rocket science. This is to be expected. Hopefully, Section 1.4 will clear up some of these questions before we get into the math, physics, chemistry, and other details of how rockets truly work.

Figure 1.32 is a block diagram of the major components of a rocket. Although the diagram is fairly detailed, it is only a very high-level description of the components of a rocket. There are other components on this level, such as the ground segment components, that are not given. Because the emphasis of this book is on rocket science and engineering, the diagram takes into account only the major components that are physically part of the rocket. These components and their functions include the following:

- *Structure:* Provides support structure for all the components, protects the inner workings of the vehicle, contains fairings and interfaces for subsystems and stages, and houses and/or supports moving components.
- *Propulsion:* Contains the fuel, oxidizer, flow systems, combustion chamber, nozzles, and other aspects needed for propelling the vehicle.
- *Power:* This subsystem contains power storage, conditions that power for use, and distributes it accordingly.
- *Guidance, navigation, and control (GN&C):* Contains attitude control system (ACS) and reaction control systems; these might include thrust vector controls (TVC) and control surfaces, such as fins or wings; navigational sensors, such as inertial navigation units and star trackers; and has computer subsystems to run GN&C functions.

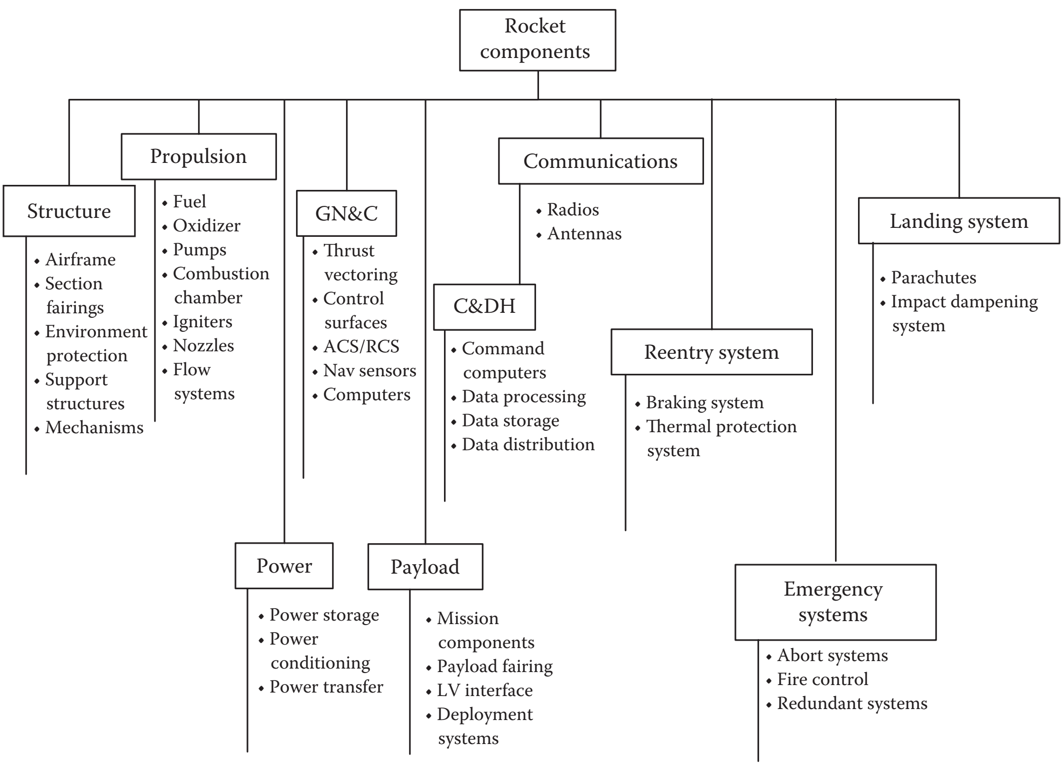


FIGURE 1.32
Block diagram showing the components of a rocket.

- *Payload:* This is the reason for the rocket and contains the science instruments, cargo, or crew.
- *Command and data handling (C&DH):* Has command computers, data processors, data storage systems, and the data distribution protocols and infrastructure.
- *Communications (Comm):* Contains radios, low- and high-gain antennas, and telemetry systems.
- *Reentry systems:* These are for rockets that must safely return a payload to Earth and contain braking systems, such as the orbital maneuvering system thrusters on the Space Shuttle, and reentry thermal protection, such as the shuttle tiles or the ceramic shields on the Apollo capsules.
- *Emergency systems:* These systems are for use in fault condition situations and include sensors for leak, fire, and damage detection, backup systems, and abort systems like the abort rockets on the Ares I, as discussed in Section 1.2.10.
- *Landing systems:* For return vehicles, there must be some means of landing the payload safely on Earth, which could include parachutes, wings, airbags, or even rockets.

Figure 1.33 is a diagram of the German V2 rocket showing the major components of that fairly basic rocket system. The rocket was designed to be a ballistic missile and, therefore,

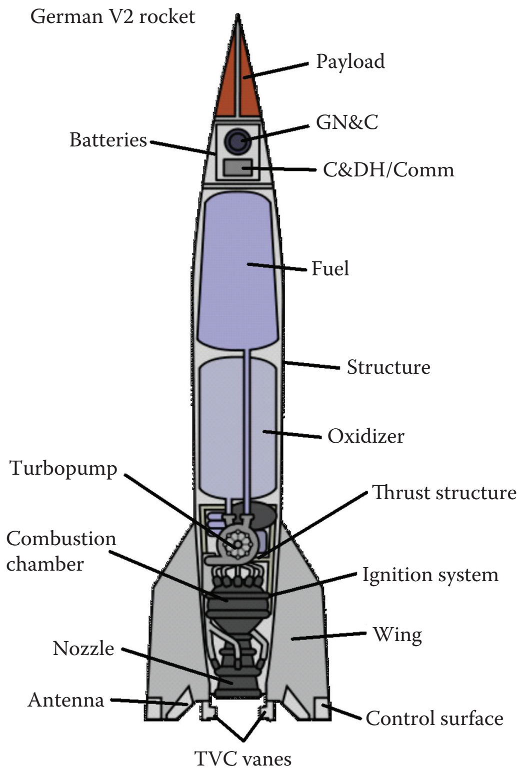


FIGURE 1.33
The German V2 liquid-fueled rocket and major components. This image is a modified "Wikimedia Commons" image.

did not contain landing system components. Also, in the case of the V2, the payload was a warhead. Interestingly enough, the V2 warhead was not high explosives because the payload container reached temperatures as high as 1,200 degrees Fahrenheit (1200°F), which would have detonated the higher-order explosive materials. The V2 could have used a better thermal protection system.

The V2 used a mixture of alcohol and water for fuel and LOX for the oxidizer and had a burn time of about 65 sec. The rocket implemented both control surfaces and TVC for ACS. The wings of the vehicle supplied aerodynamic stability.

For an even simpler description of a rocket, the best example is the hobbyist's model rocket. The types of model rockets that we all built as kids and launched with small cardboard cylinders filled with solid propellant are not really a "model rocket." In essence, the small cardboard, plastic, and balsa wood vehicles are truly rockets. Figure 1.34 shows the major components of the hobby rockets. It is useful to think about the small rocket systems with the components shown in Figures 1.32 and 1.33 in mind. A comparison of the figures also shows the difference in complexity between a liquid- and a solid-fueled rocket system.

For a more complex rocket design, Figure 1.35 gives a good view of the major subsystems of NASA's Ares I rocket. The first stage is a five-segment SRB that connects to the second stage via an interstaging cylinder. The final upper stage is the Orion spacecraft. This rocket combines the simplicity of a solid motor with the complexity of a liquid-fueled system. It also includes the added complexity of multiple stages.

Compare the vehicles in Figures 1.32 through 1.34 to gain a better understanding of the various components of a rocket. Realize that rockets can be as complex as the NASA vehicles or as simple as the hobby rocket. The key common ingredients become clear when comparing the three rockets. All the rockets use some sort of propellant, combustion

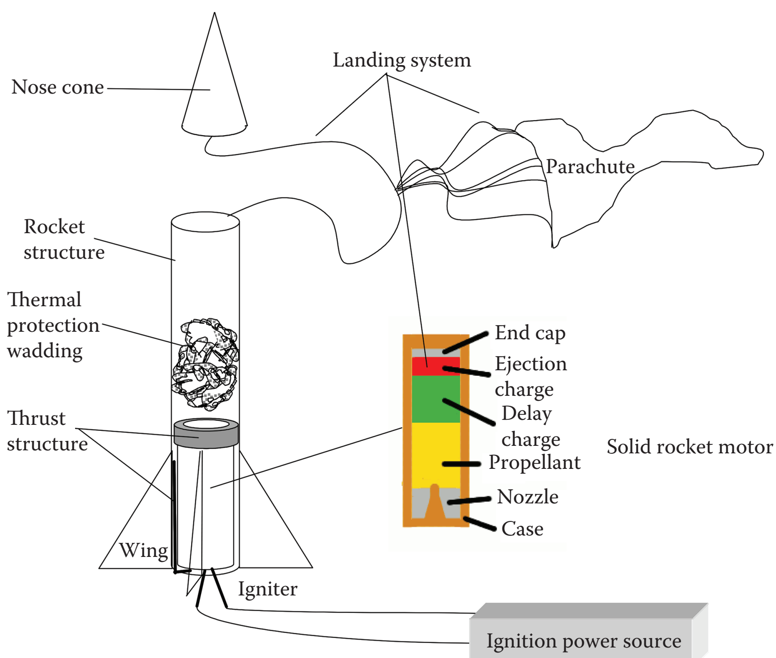


FIGURE 1.34
A hobby rocket is a simple solid-fueled rocket system.

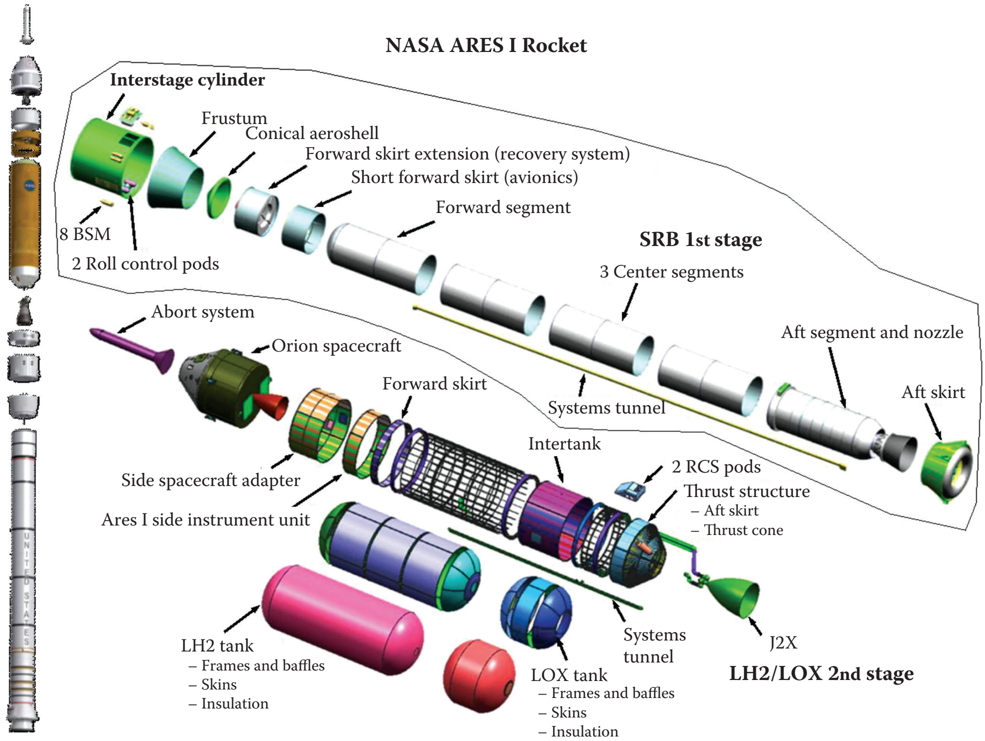


FIGURE 1.35
The Ares I rocket is a very complex system that combines solid and liquid propulsion, as well as multiple stages. (Courtesy of NASA.)

chamber, nozzle, structure, and flight control system. Note that the flight control system for the hobby rocket consists of passive control via the fixed wing surfaces.

1.4 Chapter Summary

In Chapter 1, we have discussed in some detail the history of rocketry and where and when and by whom some of the key discoveries and developments in the field were made. There are, of course, many details left out as the history of rocketry itself could fill at least one textbook, if not many.

We also discussed the launch vehicles of the modern era and what types of rockets are available in the first decade of the 21st century. From the list of rockets that were discussed, it is clear that rocketry is a global endeavor and will likely continue to be so. We also see that multinational and commercial efforts are ongoing to develop new types of launch vehicles with a broad range of flight capabilities.

Finally, we briefly touched on the anatomy of a rocket. A block diagram of the major components of rockets was given, and examples of various vehicles were used to demonstrate these components. It is clear that rockets can be as simple as the hobby solid rockets made of cardboard, wood, and plastic or as complex as the aluminum–lithium composite structures of NASA vehicles.

Exercises

1.1 Discuss the relevance of the *aeolipile* to rocket science and why it was considered the first demonstration of the principles of rocketry.

1.2 What are the main components of gunpowder?

1.3 What was *Principia*, and why is it relevant to rocket science?

1.4 Why were William Hale's rockets "better" than William Congreve's?

1.5 Compare and contrast the contributions to the development of rocketry by Konstantin Tsiolkovsky and Robert Goddard. Which one could be considered the "father of rocket science" and which one the "father of rocket engineering?"

1.6 Who was known as the Chief Designer and why?

1.7 Who was the Chief Designer's counterpart in the American space program?

1.8 What is the oldest spacecraft still in orbit?

1.9 What is UDMH? What is it used for? What is nitrogen tetroxide?

1.10 Draw a simple liquid fuel rocket and label all the major subcomponents.

2

Why Are Rockets Needed?

As discussed in Chapter 1, it is clear that rockets have been around for thousands of years. Only for the last 200 years or so have they really had viable uses other than for entertainment purposes or psychological warfare. So, why are rockets needed?

The cost of a launch vehicle can reach as high as several hundred million dollars and require a small army of people to build, prepare, and fly. There must be a good reason to expend such resources on such things; otherwise, people simply would not go to the trouble.

In this chapter, we will discuss why rockets are needed from a "top-level" answer of the economic, philosophical, and strategic point of view. Likewise, we will also discuss the "bottom-level" answer in detail of why the physics of the universe forces us to use rockets to complete particular activities. Once a good understanding of why rockets are needed is achieved, then, in Chapter 3, we will begin to discuss the details of how rockets work.

2.1 Missions and Payloads

After reading the history of rockets in Chapter 1, it might be a common perception to ascertain that from the beginning rockets were developed as missiles to deliver an explosive payload to the enemy at a distance. This perception is mostly true; however, the need for the rocket has a dichotomy that should not be overlooked. In fact, modern rocketry had two starts. The first major one can be traced to 1919 and the Treaty of Versailles that officially ended World War I. The treaty was between the Allied and Central powers and the German Empire. Among many things, this treaty would prevent Germany from being able to develop long-range artillery technology. From that point on, the Germans became very interested in developing rocket technology to take the place of the long-range artillery. It was the impact of the Treaty of Versailles that sparked the V2 missile development and successful launches. World War II saw over 3,000 V2 missile launches by the Germans. The success of the V2 led the Americans and the Soviets to long-range missile development efforts of their own that continued throughout the Cold War.

The second part of the dichotomy of modern rocketry development was sparked by the launch of Sputnik and the advent of the space race between the Americans and the Soviets. While the missile development efforts improved the rapid launch technologies, guidance and control, and throw-weight versus range capabilities, the space race led to the development of rocketry that would place payloads into orbit and even safely return them. The space race added an element from a scientific curiosity standpoint, in that science teams began seeing rockets as a means for sending payloads into orbit, deep space, and even to extraterrestrial bodies, such as the Moon, Venus, and Mars. The combination of these closely coupled, yet parallel, efforts is what led to the modern era of rocketry.

So, from the modern era history of rockets, we see that the need for rockets is to place a payload at some distance as rapidly as possible to locations where it is the only viable technological solution.

2.1.1 Missions

The mission for which a rocket system is used is driven by many factors. Military missions might need to deliver a payload to a target or place a craft, such as a spy satellite, into a particular orbit in space. There might be a need for telecommunications satellites to be placed in orbit. A new and interesting idea is the U.S. Marine mission known as Hot Eagle, which would be a rapid response vehicle that could deliver a small contingent of marines to any point on the globe within 2 h. The only present technology that can do this is rocket technology. Of course, this is just a concept mission, and no vehicle has been developed for such a task.

Commercial missions might include the need for telecommunications or delivering satellite television broadcasts globally or even space tourism. The great thing about commercial missions is that there are endless possibilities for potential missions. The key is to find a way to make money from the mission. For example, Deep Space Expeditions (a subset of Space Adventures) is planning to sell seats on a Soyuz spaceflight, which will orbit around the Moon for the sum of $100 million each. There are already potential buyers.

And, of course, there are science missions. The science missions to space, from lower Earth orbit (LEO) to the Kuiper belt and beyond, drive the rocket technologies to new capabilities. An example of this type of science mission would be to study the deep space or to determine if there is water or ice on the Moon or liquid water on Mars. The mission is typically designed to improve our understanding of the universe, as well as satisfy a scientific curiosity.

Whatever the mission might be, groups of experts in these fields define the requirements for a mission to as great a level of detail as possible. Then, once the needs are understood as well as can be with the knowledge of the problem and resources available, a payload is designed to accomplish the mission.

2.1.2 Payloads

The payload is the reason for building the rocket. Whether the payload is a warhead, a science instrument, or a communications device, the only known technology for delivering that payload is with a rocket. The absolute standard textbook in the space mission preparation community is called *Space Mission Analysis and Design*, 3rd ed. (2005; there are later editions now available, and the book is known as the SMAD, pronounced "smad") and is edited by James R. Wertz and Wiley J. Larson. It gives the following description for a payload:

> … the term payload includes all hardware above the launch-vehicle-to-spacecraft interface, excluding the payload's protective fairing, which is usually part of the launch system … payload consists of the entire spacecraft above the booster adapter interface.

> For the Shuttle, it is customary to speak of the payload as the spacecraft to be deployed or the sortie mission payload to be operated from the payload bay … (p. 719)

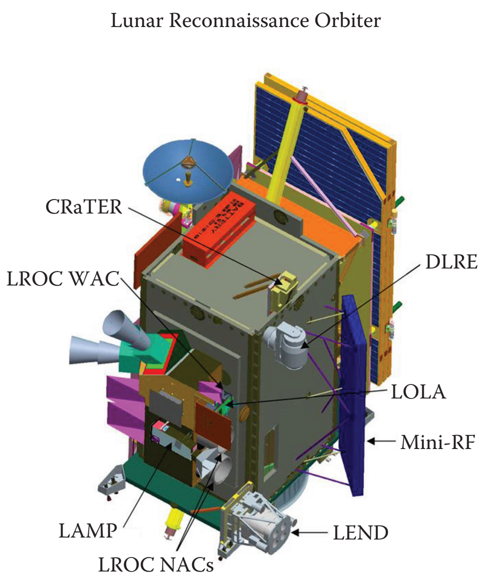
The description only tells us how to physically discern the payload from the launch vehicle. The payload is really the means for which a mission can be accomplished. In other words, the payload truly is the "means to an end" for the mission.

A very recent example of how mission requirements lead to the definition of a payload is the NASA Lunar Reconnaissance Orbiter (LRO) Measurement Announcement of Opportunity (AO) that was declared on June 18, 2004. The mission statement of the AO was as follows:

> NASA established an external group entitled the LRO Objectives/Requirements Definition Team (ORDT) that met in March 2004 to assist in defining specific LRO mission goals and measurement objectives needed for the initial steps in lunar robotic exploration. From the results of this external group, NASA has established the following high priority objectives for the initial robotic elements in the Lunar Exploration Program:
>
> - Characterization of the global lunar radiation environment and its biological impacts and potential mitigation, as well as investigation of shielding capabilities and validation of other deep space radiation mitigation strategies involving materials.
> - Determination of a high spatial resolution global geodetic grid for the Moon in three dimensions:
>   a. Global geodetic knowledge by means of spatially resolved topography, and
>   b. Detailed topographic characterization at landing site scales.
> - Assessment of the resources in the Moon's polar regions (and associated landing site safety evaluation), including characterization of permanently shadowed regions and evaluation of any water ice deposits.
> - High spatial resolution global resources assessment including elemental composition, mineralogy and regolith characteristics.

Of course, the mission statement had much greater detail than the summary of the objectives above (73 more pages), but this illustrates the general idea of how a science mission statement might read. The payloads derived from the LRO Measurements AO will meet these requirements as best as the state-of-the-art instruments can provide. Figure 2.1 shows the current conceptual configuration for the LRO and a brief description of the onboard science instruments. The LRO Measurements mission was defined, and the LRO payload was designed from them. The payload will be launched by an Atlas V rocket in the fall of 2008.

Whatever the mission and whatever the payload, there is always the need for the understanding of the rocket science and engineering principles governing how to put the payload where it needs to be and how to control it once it arrives at its mission destination. Once the LRO is launched on a launch vehicle, it will be hurled toward the Moon, and the larger vehicle will be left behind and expended. Along the route to the Moon, the LRO will have to make orbit corrections with small rockets, and it will have to stabilize itself in order to make measurements as it orbits the Moon. The aspects of rocket science and engineering are embedded into every aspect of such a mission.



CRaTER
The primary goal of CRaTER is to characterize the global lunar radiation environment and its biological impacts.

DIVINER
DIVINER will be measuring lunar surface temperatures at scales that provide essential information for future surface operations and exploration.

LAMP
Reflected Lyman sky-glow and starlight produce sufficient signal for even a small UV instrument like LAMP to see in the Moon's permanently shadowed regions.

LEND
LEND will provide measurements, create maps, detecting possible near-surface water ice deposits.

LOLA
The Lunar Orbiter Laser Altimeter (LOLA) investigation will provide a precise global lunar topographic model and geodetic grid that will serve as the foundation of this essential understanding.

LROC
The Lunar Reconnaissance Orbiter Camera (LROC) has been designed to address the measurement requirements of landing site certification and polar illumination.

Mini-RF
The mini-RF is a technology demonstration of an advanced single aperture radar (SAR) capable of measurements in X-band and S-band. Mini-RF will demonstrate new lightweight SAR, communication technologies and locate potential water-ice.

FIGURE 2.1
The Lunar Reconnaissance Orbiter and its instruments will carry out science missions from lunar orbit and will be the payload on an Atlas V rocket. (Courtesy of NASA.)

2.2 Trajectories

The LRO spacecraft mentioned in Section 2.1.2 will be a very complicated mission, a very complicated payload, and a very complicated spacecraft and will require a complicated set of calculations to determine the proper launch trajectories, lunar injection, and lunar orbiting maneuvers. From the characteristics of the orbits planned for the mission, the rocket scientists and engineers determined the appropriate launch vehicle, upper-stage rockets, and onboard thrusters to complete the mission successfully. Understanding how to put a spacecraft where it needs to be is the first step in understanding rocketry. As with all things, it is best to understand the basics before getting into the more complex problems. Therefore, we will start with some basics of simple trajectories and how to calculate them.

Once a rocket is launched, it burns its fuel until it is gone (or the engines are shut off), and, at that point, the vehicle has reached the so-called burnout velocity. At that point on the rocket's path, it becomes a freely flying projectile, unpowered, and forced to succumb to the laws of physics of projectile motion.

The basics of a projectile in motion are actually quite easy to understand if the following are accounted for properly and the right assumptions are made:

- Acceleration due to gravity is assumed constant.
- Neglect air resistance.
- Assume the Earth is flat.
- Assume the Earth's rotation has no impact on the motion of the projectile.

On a small scale of a few kilometers of payload and even a range of a few hundred kilometers, these assumptions work well (except for assumption 2, as wind often shows up in the real world). For now, we will accept these assumptions.

2.2.1 Example 2.1: Hobby Rocket

Let's start with the simple analysis of a hobby rocket's trajectory. Assume that the rocket starts from rest on the ground at $x = 0$ and $y = 0$ where x is horizontal and y is vertical. The rocket will be launched at a 75° angle with the x-axis. The rocket engine burns for 3 sec until main engine cutoff (MECO) when the solid propellant has been used up and has reached an altitude, y_{bo}, of 300 m, a downrange distance, x_{bo}, of 100 m, and a burnout velocity, v_{bo}, of 50 m/sec. Again, assuming no air friction or wind, how high will the rocket reach, how far will it travel, and what will the trajectory look like? Figure 2.2 shows the rocket flight scenario.

In order to determine the trajectory of this rocket, we first need to know the laws of projectile motion. Projectile motion is described by velocity, position, and time and is as follows:

$$v_x = v_o \cos\theta \tag{2.1}$$

$$v_y = v_o \sin\theta - gt \tag{2.2}$$

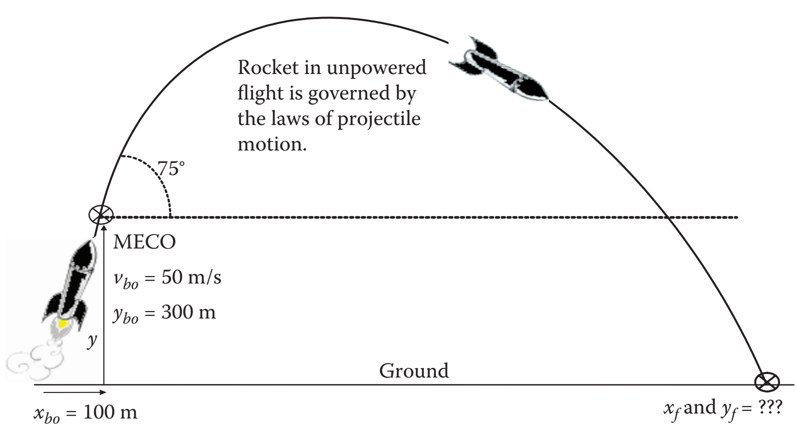


FIGURE 2.2
The hobby rocket trajectory, as described in Example 2.1.

$$x(t) = (v_o \cos\theta)t \tag{2.3}$$

$$y(t) = (v_o \sin\theta)t - \frac{1}{2}gt^2. \tag{2.4}$$

Here, v_o is the velocity of the rocket at MECO and, therefore, equal to the burnout velocity v_{bo}; g is the acceleration due to gravity equal to 9.8 m/s^2. The angle θ is the angle that the flight path velocity vector makes with the horizontal axis or

$$\tan\theta = \frac{v_y}{v_x}. \tag{2.5}$$

Solving for t in Equation 2.3 yields

$$t = \frac{x}{v_o \cos\theta}. \tag{2.6}$$

Substitute Equation 2.6 into Equation 2.4, and the result is

$$y(x) = (v_o \sin\theta)\frac{x}{v_o \cos\theta} - \frac{1}{2}g\left(\frac{x}{v_o \cos\theta}\right)^2. \tag{2.7}$$

Simplifying Equation 2.7 and substituting v_{bo} for v_o yields

$$y(x) = (\tan\theta)x - \left(\frac{g}{2v_{bo}^2 \cos^2\theta}\right)x^2. \tag{2.8}$$

Equation 2.7 is the expression for the position of the rocket as it travels along its trajectory after MECO. In order to account for the altitude the rocket has already reached at MECO, we must add $y(0) = y_{bo}$ to Equation 2.8:

$$y(x) = y_{bo} + (\tan\theta)x - \left(\frac{g}{2v_{bo}^2 \cos^2\theta}\right)x^2. \tag{2.9}$$

It should be noted here that Equations 2.8 and 2.9 are equations for a parabola. This tells us that the trajectory of the rocket follows a parabolic flight path. Also note that, if necessary, solving for an equation for $x(y)$ could be done easily enough using the quadratic formula. (This is left as an exercise for the students). Figure 2.3 shows a graph of the hobby rocket's trajectory as calculated from Equation 2.9. Note that the x-axis of the graph is $x + x_{bo}$ in order to account for the distance the rocket has traveled downrange at MECO.

Figure 2.3 describes the rocket's flight path from MECO to impact with the ground. There are a couple of ways to determine the flight's maximum height and range. The first way to determine the maximum range or impact point is simply to input $y = 0$ into Equation 2.9 and solve for x. Or, we could look at where the plot crosses the x-axis and see

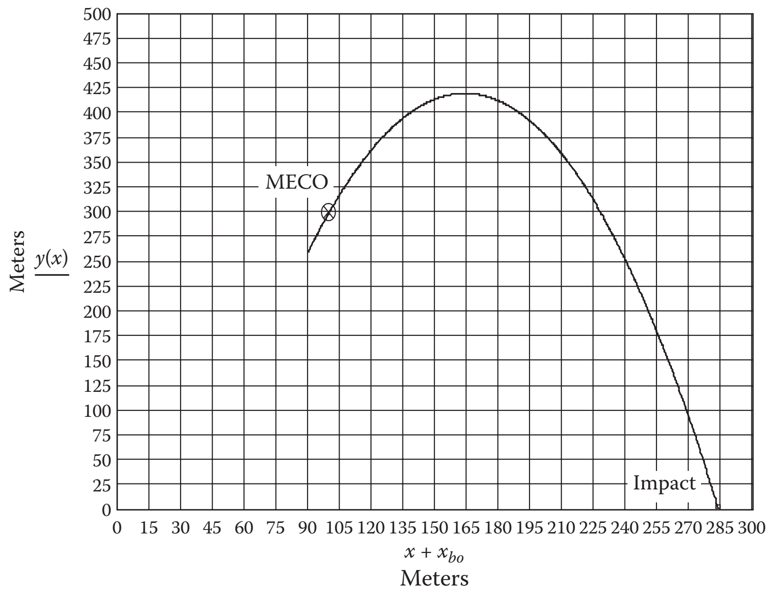


FIGURE 2.3
The trajectory of the hobby rocket as described in Example 2.1 and Equation 2.9.

that it is around 285 m. For the maximum height, again, looking at the graph tells us that the peak height is around 420 m.

In order to exactly determine the maximum height and the impact values, we need to do some more math. For maximum height, we need to realize that, once the rocket reaches the peak of the trajectory, for that brief instant, the velocity vector in the y direction is zero. In other words, Equation 2.2 is zero. Solving Equation 2.2 when $v_y = 0$ for t gives

$$t_{y\max} = \frac{v_o \sin\theta}{g}. \tag{2.10}$$

Substituting Equation 2.10 into Equation 2.4, recalling that $v_o = v_{bo}$, and then adding y_{bo} to account for the height at MECO result in

$$y_{\max} = y_{bo} + \frac{v_{bo}^2 \sin^2\theta}{2g}. \tag{2.11}$$

Equation 2.11 shows the maximum height the rocket reaches. Substituting the numbers from Example 2.1 gives $y_{\max} = 419$ m, which agrees well with the value we chose from the graph in Figure 2.3.

In order to determine the maximum range of the rocket, we have to modify Equations 2.3 and 2.4 to account for x_{bo} and y_{bo}, so they become

$$x(t) = x_{bo} + (v_{bo}\cos\theta)t \tag{2.12}$$

$$y(t) = y_{bo} + (v_{bo}\sin\theta)t - \frac{1}{2}gt^2. \tag{2.13}$$

Solve Equation 2.13 for $y(t_{xmax}) = 0$ and a quadratic equation for t_{xmax} is the result. After some calibration in algebra, an equation for x_{max} is found to be

$$x_{max} = x_{bo} + (v_{bo}\cos\theta)\left[\frac{v_{bo}\sin\theta + \sqrt{v_{bo}^2\sin^2\theta + 2gy_{bo}}}{g}\right]. \quad (2.14)$$

Substituting the numbers from the Example 2.1 above gives $x_{max} = 283.4$ m, which agrees well with the graph in Figure 2.3.

2.2.2 Fundamental Equations for Trajectory Analysis

We have now developed five equations that describe the flight path in position and time for our hobby rocket system (realizing that we made certain assumptions to begin with) that will work just as well for any other rocket following a similar path. In other words, if a rocket is a downrange missile, and we want to know what it does after MECO and where it will impact the ground, Equations 2.9 and 2.11 through 2.14 are the main equations for describing that rocket's flight path trajectory and are recaptured in Figure 2.4.

An important thing to notice in Equations 2.9, 2.11, and 2.14 is what happens to the equations if there is no gravity, or $g = 0$. In Equation 2.9, if there is no gravity, then the equation becomes linear with x, and the rocket will travel upward at the original angle forever. This is really emphasized in Equations 2.11 and 2.14 where the zero is in the denominator and, therefore, causing infinities. In other words, the values for y_{max} and x_{max} become infinity if there is no gravity, and the rocket would travel along at v_{bo} forever in a straight path at the original angle.

Another very interesting aspect of Equation 2.9 is illustrated in Figure 2.5. The figure shows several trajectories of an ICBM, as calculated by Equation 2.9. For a burnout altitude of 300 km, downrange distance at burnout of 100 km, and a burnout velocity of 5 km/sec, several trajectories were calculated using different flight path angles. There are two very important points to take from the figure. The first is that the initial flight path angle of 45° at MECO is the optimum angle for achieving maximum range. The second point is that there are two trajectory solutions to each downrange point at the MECO altitude (other than the maximum downrange point, which has only one trajectory solution at 45°). The two solutions for each point have complementary angles, the sum of which is 90°. It should

The fundamental equations for a rocket's flight path

$$y(x) = y_{bo} + (\tan\theta)x - \left(\frac{g}{2v_{bo}^2\cos^2\theta}\right)x^2 \quad (2.9)$$

$$y_{max} = y_{bo} + \frac{v_{bo}^2\sin^2\theta}{2g} \quad (2.11)$$

$$x(t) = x_{bo} + (v_{bo}\cos\theta)t \quad (2.12)$$

$$y(t) = y_{bo} + (v_{bo}\sin\theta)t - \frac{1}{2}gt^2 \quad (2.13)$$

$$x_{max} = x_{bo} + (v_{bo}\cos\theta)\left[\frac{v_{bo}\sin\theta + \sqrt{v_{bo}^2\sin^2\theta + 2gy_{bo}}}{g}\right] \quad (2.14)$$

FIGURE 2.4
Shown above are the five equations that describe the ballistic trajectory of a rocket from MECO to target impact.

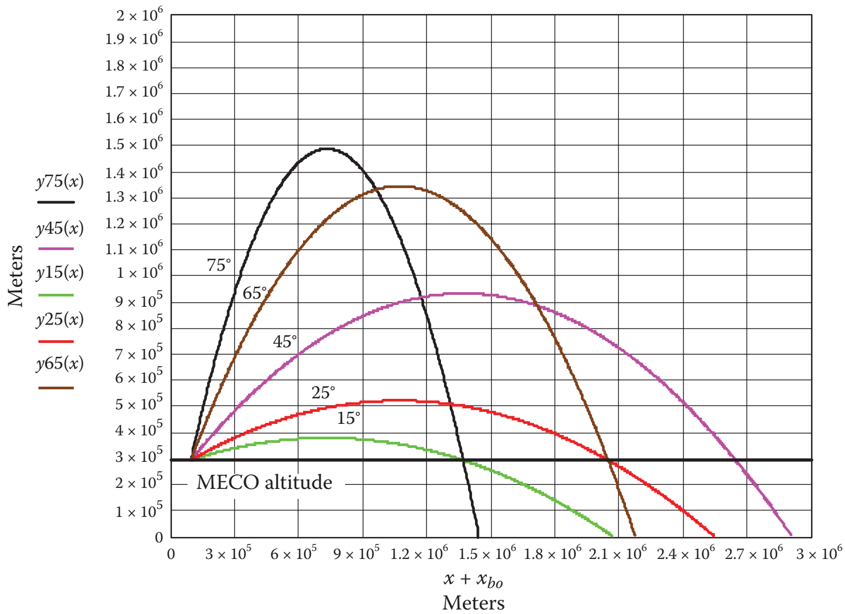


FIGURE 2.5
Multiple ICBM traces for different initial flight path angles show that 45° is optimum for maximum downrange and that a point downrange other than the maximum range can be reached by two complementary values for the initial flight path angle.

also be noted here that, although the two different angles allow the rocket to pass through the same point downrange, the transit times and maximum altitudes are different.

2.2.3 Missing the Earth

Consider an ICBM launching from one side of the planet Earth and maintain the assumptions mentioned on Section 2.2.2 for simple trajectory calculations. We will use a flight angle of 45°. Also, for simplicity, we will assume that the MECO height and the downrange distance are zero. This means that the missile would be at v_{bo} from the surface rather than at some point above the surface of the Earth. This is obviously not a real-world situation, but it will be useful to illustrate an interesting point. Figure 2.6 shows the trajectory for the missile at several burnout velocities ranging from 5 km/sec to 12.5 km/sec. The figure also has a photo of the Earth scaled to the same scale as the graph and overlaid upon it. (The black dotted line simulates the circumference of the Earth; also, realize that the radius of the Earth is about 6,370 km.) Note that, once the burnout velocity reaches 11.2 km/sec, the missile trajectory downrange distance becomes equal to the diameter of the Earth. This simulation shows us that, for a parabolic trajectory, a rocket with a burnout velocity of 11.2 km/sec or greater will escape from the Earth. This is the so-called escape velocity and will be discussed in greater detail later in the chapter.

2.2.4 Example 2.2: Dong Feng 31 ICBM

The Chinese ICBM DF-31 is a three-stage solid-fueled rocket that can carry a 1-megaton nuclear warhead payload, which is approximately 700 kg. Assume MECO at 100-km

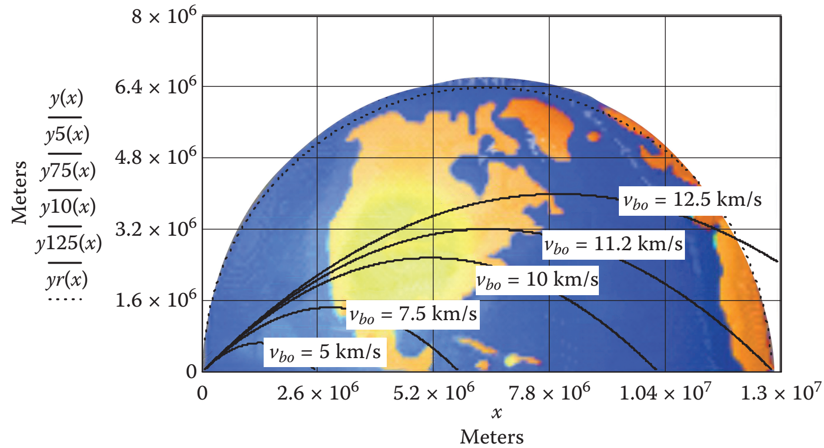


FIGURE 2.6
Trajectories for an ICBM with different initial velocities (burnout velocity) show that the missile will miss the Earth with velocity of 11.2 km/s or greater.

downrange and in altitude and a burnout velocity of 8.75 km/sec. Why should this missile be of concern to the American public (as well as many other parts of the world)? Assume an initial flight path angle, θ, of 45°.

We have all the data necessary to analyze this problem. Using Equation 2.9 and substituting in the values given, the trajectory for the DF-31 can be calculated, as shown in Figure 2.7. The figure shows that the DF-31 missile has a maximum range of about 8,000 km.

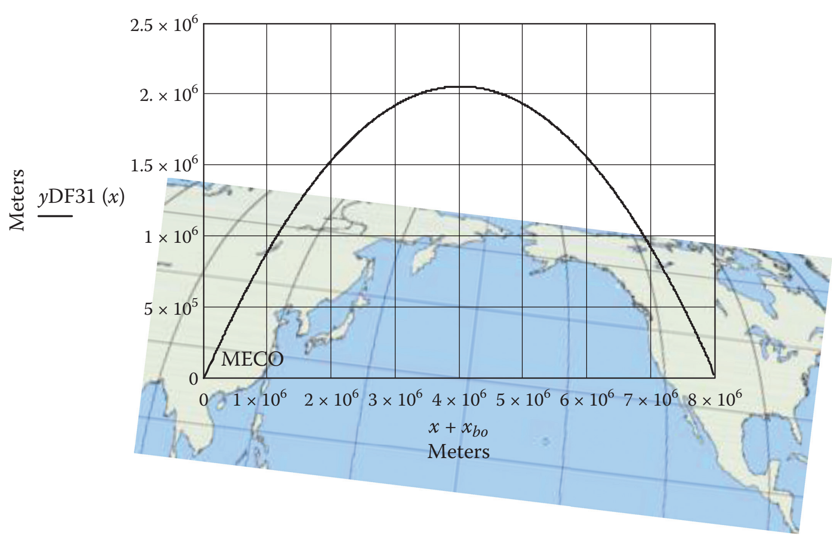


FIGURE 2.7
The Chinese DF-31 ICBM can deliver a 1-megaton nuclear warhead at a range of about 8,000 km.

2.3 Orbits

2.3.1 Newton's Universal Law of Gravitation

It has been documented that, one day in 1666, Sir Isaac Newton was visiting his mother in Cambridge, and, while he was in "a contemplative mood," he was "occasioned by the fall of an apple." The fall of that apple (whether it hit him on the head or not, as the folks at *Schoolhouse Rock!* convinced my generation, is unlikely) sparked Newton to develop the law of gravity and publish in 1667 his *Mathematical Principles of Natural Philosophy*. Newton's **Universal Law of Gravitation** tells us that every particle in the universe attracts every other particle by a gravitational force in a way that can be described as

$$F = \frac{Gm_1m_2}{r^2}. \tag{2.15}$$

The attractive force, F, between two masses, m_1 and m_2, is inversely proportional to the square of the distance, r, between the two masses. G is a proportionality constant and is known as the *gravitational constant*, which has been experimentally verified and measured to be

$$G = 6.672 \times 10^{-11}\,\frac{Nm^2}{kg^2}. \tag{2.16}$$

Newton considered many different aspects of falling bodies and how they interact through his law of gravitation given in Equation 2.15. A simple calculation to explain the acceleration of a falling object (such as an apple plummeting to the ground) due to Earth's gravity can be made by realizing that the force on a falling object, which is determined by

$$F = -mg \tag{2.17}$$

is due to gravitation and should be equal to Equation 2.15 or

$$F = -m_{apple}g = \frac{GM_{Earth}m_{apple}}{R_{Earth}^2}. \tag{2.18}$$

Where R_{Earth} is the radius of the Earth, solving for g and simplifying results in

$$g = -\frac{GM_{Earth}}{R_{Earth}^2}. \tag{2.19}$$

This simple relationship given in Equation 2.19 is very powerful when it comes to understanding the acceleration due to gravity on any planet. For example, in order to calculate the acceleration due to gravity on the Moon, we would simply substitute the mass and

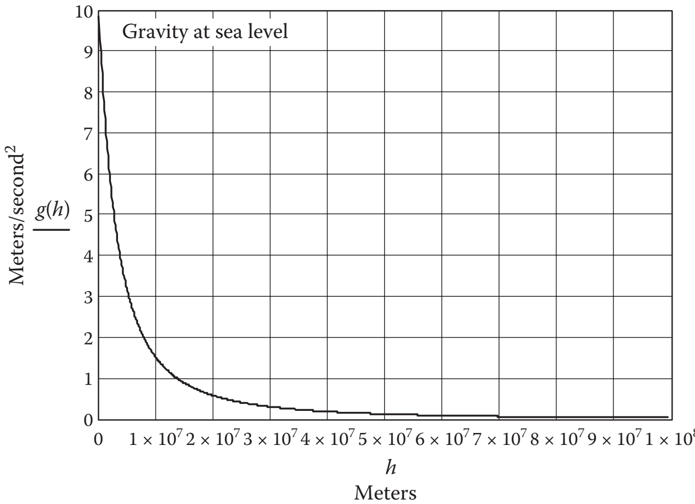


FIGURE 2.8
The acceleration due to gravity drops off as an inverse square of altitude.

radius for the Moon instead of the Earth. A slight modification to Equation 2.19 tells us another interesting thing. The modification is

$$g = -\frac{GM_{Earth}}{(R_{Earth} + h)^2}. \tag{2.20}$$

Equation 2.20 shows that the acceleration due to gravity drops off with altitude, h, above the Earth's surface, which will become important as we study rocketry in more detail. Figure 2.8 shows a graph of the acceleration due to gravity on Earth as a function of distance from the Earth. Note that the acceleration becomes fairly negligible at altitudes over 20,000 km.

2.3.2 Example 2.3: Acceleration due to Gravity on a Telecommunications Satellite

Consider a telecommunications satellite in a geostationary orbit at approximately 35,000-km altitude. What is the formula for determining the force on the satellite due to the Earth's gravity?

Starting with Equation 2.20a, we can determine the formula for g, which, with the numbers, is

$$g = -\frac{GM_{Earth}}{(R_{Earth} + 35,000,000m)^2}. \tag{2.20a}$$

Substituting Equation 2.20a into Equation 2.17 gives the answer:

$$F = -m_{satellite}g = -m_{satellite}\frac{GM_{Earth}}{(R_{Earth} + 35,000,000m)^2}. \tag{2.21}$$

Thus far, we have considered the force between two objects due to gravity. Now, we will investigate how this force of gravity influences the energy of the two object systems. In order to do this, we first have to determine the potential energy due to a gravitational force. Because gravity is a conservative force, we can calculate the potential energy between two points within a gravitational field as

$$U_f - U_i = \int_{r_i}^{r_f} F(r)\,dr = -\int_{r_i}^{r_f} \frac{GM_{Earth}m}{r^2}\,dr = -GM_{Earth}m\left(\frac{1}{r_f} - \frac{1}{r_i}\right). \tag{2.22}$$

If we assume the initial reference point to be at infinity, then that means the initial potential energy is zero, and, therefore, we can write Equation 2.22 as

$$U_f - 0 = -GM_{Earth}m\left(\frac{1}{r_f} - \frac{1}{\infty}\right) = -\frac{GM_{Earth}m}{r_f}. \tag{2.23}$$

Or simply,

$$U = -\frac{GM_{Earth}m}{r}. \tag{2.24}$$

Equations 2.15 and 2.24 are important in describing how two objects interact with each other in a gravitational field, and they tell us that force due to gravity drops off as $1/r^2$, while energy due to gravity falls off as $1/r$. Also, Equation 2.24 tells us that the energy potential is *negative*. What does that mean? Negative potential energy?

Actually, what Equation 2.24 is telling us is that the objects are inside a "potential well." In this case, the objects are in a "gravitational potential well." More importantly, in order to get one of the objects out of the potential well of another one would require that, in some manner, an amount of work would have to be expended to move that object out of the well. This gravitational potential well concept can be readily understood by considering the planet Earth as our major body and a rocket as the small body. The rocket sitting on the launch pad is at the bottom of the gravitational potential well (actually, the center of the Earth would be the center, but we will assume the rocket can only sit on the surface), and gravity is holding it to the ground. By expending the rocket's fuel in a very energetic combustion process, the rocket generates a force that pushes it up the well. From Equation 2.24, it is clear that as the distance, r, between the rocket and Earth gets larger and larger, the potential energy between them decreases. At a certain distance, the effect of the gravitational potential well of the Earth becomes negligible compared to other forces acting on it (such as the gravitational pull of the Moon or the Sun), and the rocket is said to be out of the Earth's "sphere of influence." The term is only applicable for three or more body systems and can be described as

$$r_{SOI} = a_p\left(\frac{m_{smallerbody}}{M_{largerbody}}\right)^{\frac{2}{5}}. \tag{2.25}$$

TABLE 2.1

The Sphere of Influence for the Planets of Our Solar System

Planet	r_{SOI} in km	r_{SOI} in Body Radii
Mercury	1.12×10^5	45
Venus	6.16×10^5	100
Earth	9.25×10^5	145
Moon	6.61×10^4	38
Mars	5.77×10^5	170
Jupiter	4.82×10^7	677
Saturn	5.48×10^7	901
Uranus	5.17×10^7	2025
Neptune	8.67×10^7	3866
Pluto	3.31×10^6	2753

Here, a_p is the length of the semimajor axis of the smaller body's orbit relative to the larger body. Semimajor axis is a term used to define the radius of an elliptical orbit along the long dimension of the ellipse. (This will be discussed more later in this chapter.) Table 2.1 shows a list of the planets of our solar system and the value for the radius of the sphere of influence, r_{SOI}.

In our discussion of a rocket within the Earth's sphere of influence, we talk about the rocket trying to leave the Earth's gravitational potential well. But, what if we simply wanted the rocket to circle the Earth at some altitude above it without ever falling back to the surface? Could that be done? In other words, what if we wanted the rocket to *orbit* the Earth?

2.3.3 A Circular Orbit

In the third book of Isaac Newton's *Principia* titled *De mundi systemate* or *On the System of the World* (1726), Newton drew a figure to describe what he thought would happen to a projectile being thrown from the top of a high mountain. The figure is shown in Figure 2.9. Newton described the figure and idea as follows:

> … the greater the velocity … with which (the projectile) is projected, the farther it goes before it falls to the Earth. We, therefore, may suppose the velocity to be so increased, that it would describe an arc of 1, 2, 5, 10, 100, 1000 miles before it arrived at the Earth, till at last, exceeding the limits of the Earth, it should pass into space without touching. (p. 517)

From the passage above, it is clear that Newton realized that the Earth was a sphere and that objects fell due to the Law of Universal Gravitation. What he suspected was that, if the projectile moved fast enough horizontally, as it fell, it would only fall the distance of the drop-off due to the curvature of the spherical Earth. Figure 2.9 illustrates his point quite well.

Let us consider Newton's concept in a little more detail and apply it to a real-world situation. In order for the projectile not to be slowed by atmospheric drag, it would need to be at an altitude just above where significant atmospheric drag occurs. We will assume that an altitude, y, of 100 km above the Earth will suffice.

Now, we need to determine what the drop-off due to the curvature of the Earth is for a given distance along the surface. Figure 2.10 shows a circle representing the Earth and the orbit we wish to maintain and how to go about calculating the rate of drop-off of the

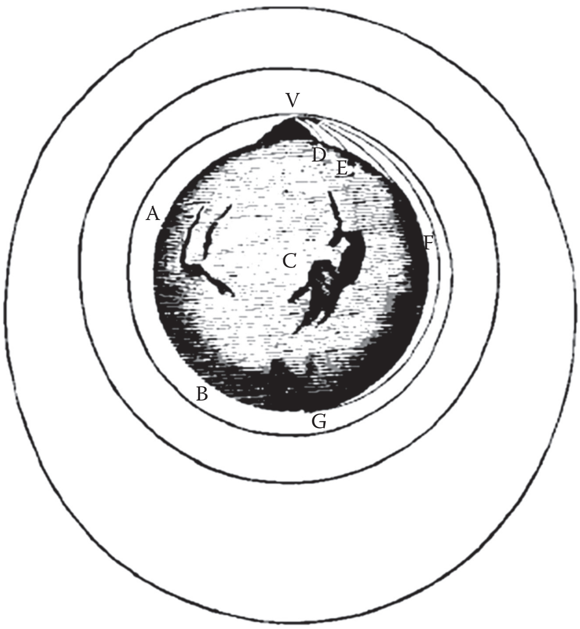


FIGURE 2.9
A projectile thrown from a mountaintop might not fall all the way to Earth if it is thrown fast enough. According to Newton, "Let AFB represent the surface of the Earth, C its center, VD, VE, VF, the curved lines which a body would describe if projected in a horizontal direction from the top of a mountain sufficiently with more and more velocity." (From Newton's *On the System of the World*.)

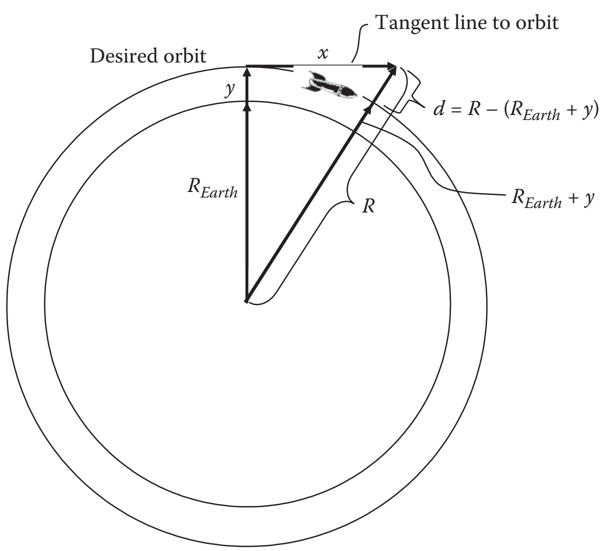


FIGURE 2.10
A rocket, if traveling at the right speed, can maintain a circular orbit about the Earth.

curvature of the orbit in order to keep our rocket at that orbit height. We should note here that, once the rocket reaches our orbit altitude and velocity, it would no longer use any engines to continue propulsion. With no drag from the atmosphere, it should continue along at the same horizontal velocity while falling vertically due to Earth's gravity.

If the rocket were to travel a distance x in a straight path above the surface of the Earth at a distance y in a tangent line with the orbit circle, the distance to the rocket is found by

$$R = \sqrt{(R_{Earth} + y)^2 + x^2}. \tag{2.26}$$

The distance, *d*, between the tangent line and the desired orbit height is

$$d = R - (R_{Earth} + y). \tag{2.27}$$

The drop-off of the Earth as a function of horizontal distance traveled is then

$$\frac{\Delta x}{\Delta y} = \frac{d}{x}. \tag{2.28}$$

By substituting values into Equations 2.26 through 2.28, the drop-off of the Earth can be found. Assuming $x = 1$ km, $y = 100$ km, and the radius of the Earth is 6,370 km allows for the calculation to be completed, giving a value for the drop-off of the Earth as

$$\frac{\Delta x}{\Delta y} = \frac{d}{x} = \frac{.00007728 \text{ km drop}}{1 \text{ km horizontal travel}}. \tag{2.29}$$

Now, consider that an object at rest at a height, *d*, when dropped, will reach a velocity in the y direction, v_y, after falling through that distance. The velocity in the *y* direction is determined by setting the potential energy equal to the kinetic energy and solving for it. Mathematically, this means

$$mgd = \frac{1}{2} m v_y^2. \tag{2.30}$$

Solving for v_y yields

$$v_y = \sqrt{2gd}. \tag{2.31}$$

Equation 2.31 describes the final velocity the rocket will have reached as it fell through the distance *y*. Using Equation 2.4 and realizing that the initial velocity in the *y* direction is zero, the time, *t*, required for the rocket to fall a distance d is found by

$$t = \sqrt{\frac{2d}{g}}. \tag{2.32}$$

When we substitute values into Equation 2.32, we see that a time of 0.1256 sec is required for the rocket to fall a distance from the tangent line to the orbit circle, as shown in Figure 2.10. More simply put, the rocket must travel 1 km horizontal to the surface of the orbit with the 0.1256 sec required for it to drop vertically by the distance d, as found in Equation 2.27. Rewriting this mathematically,

$$v_x = \frac{x}{t} = \frac{1\,\text{km}}{0.1256\,\text{s}} = 7.96\,\text{km/s}. \tag{2.33}$$

Equation 2.33 tells us that Newton was right. If a projectile (in our case, a rocket) has a tangential velocity to the surface of the Earth of 7.96 km/sec (and is sufficiently above the atmosphere), it will remain in a circular orbit about the Earth without the need of further propulsion or acceleration from engines. At this point, it would benefit the reader to work back through this section and try different orbit altitudes to calculate the orbital velocity needed for different circular orbits.

2.3.4 The Circle Is a Special Case of an Ellipse

Considering the rocket discussed in Section 2.3.3, we see that it can indeed travel in a circular orbit above the Earth if it has the right tangential velocity to match the vertical drop due to gravitational acceleration to the horizontal drop due to the curvature of the planet. The spacecraft traveling along this circular orbit can be described by

$$r^2 = (x - x_o)^2 + (y - y_o)^2. \tag{2.34}$$

Here, r is the radius of the circular orbit from the center of the Earth, and x_o and y_o are the coordinates for the center of the circle. Assuming x_o and y_o are at the origin, then Equation 2.34 becomes

$$r^2 = x^2 + y^2. \tag{2.35}$$

The above equation is the simplest expression for a circle in Cartesian coordinates.

Figure 2.11 shows an ellipse and its various labeled aspects. As our discussion of orbits progresses, we will see that the terms that describe the ellipse will be the language for orbital mechanics as well. The following is a list of the geometrical aspects of the ellipse and their definitions:

Apoapsis = the point on the ellipse farthest from the major focus

a = semimajor axis length

b = semiminor axis length

c = linear eccentricity of the ellipse

Foci = two focus points from which the ellipse is drawn by attaching a string to each focus and stretching the string into a triangle and drawing the ellipse at the vertex of the triangle (the same as the one center focus of a circle)

Major focus = a convention used in this book for the focus nearest the periapsis

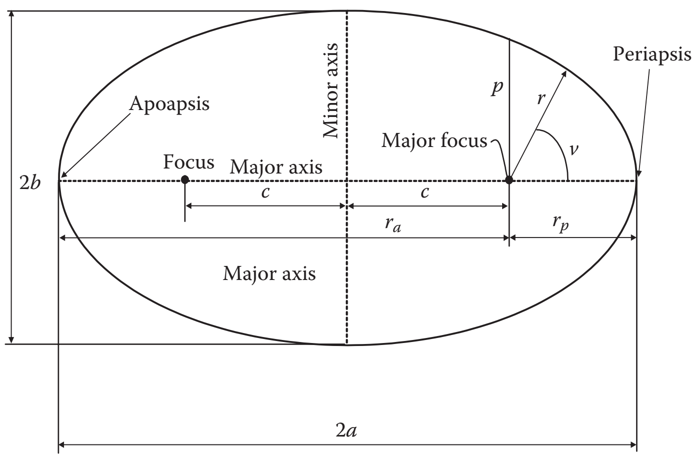


FIGURE 2.11
The terms defining the ellipse are important to orbital mechanics and will be referenced throughout this text.

p = semilatus rectum, which is the distance from the focus of the ellipse to the ellipse itself

Periapsis = the point on the ellipse closest to the major focus

r = radius in polar coordinates from the major focus to the ellipse. (For convenience, the origin is usually placed at the major focus.)

r_a = the distance from the major focus to the apoapsis

r_p = the distance from the major focus to the periapsis

ν = the angle between the major axis and the radius vector known as the "true anomaly"

Similar to Equations 2.34 and 2.35 that describe the circle, the ellipse is described, thus

$$1 = \frac{(x - x_o)^2}{a^2} + \frac{(y - y_o)^2}{b^2} \tag{2.36}$$

where the origin is not the center of the ellipse and

$$1 = \frac{x^2}{a^2} + \frac{y^2}{b^2} \tag{2.37}$$

where it is. Note that if the semimajor axis length, a, is equal to the semiminor axis length, b, or $a = b$, then Equation 2.37 can be rewritten as

$$1 = \frac{x^2}{a^2} + \frac{y^2}{a^2}. \tag{2.38}$$

Solving Equation 2.38 for a,

$$a^2 = x^2 + y^2, \tag{2.39}$$

which is the equation for a circle with $r = a$. Therefore, we see that the circle is the special case of an ellipse when the semimajor axis length and the semiminor axis length are equal. So, the circular orbit described in Section 2.3.4 is actually an elliptical one.

2.3.5 The Ellipse Is Actually a Conic Section

We have discussed circles as ellipses, and now we need to consider another important aspect of the ellipse. Actually, the ellipse is a particular case of something called a conic section. A conic section is a curve that is created by the intersection of the surface of a right circular cone and a plane. Figure 2.12 shows the possible configurations of these conic sections.

More precisely than the equations for the circle and ellipse, in Cartesian coordinates, the equation for the conic section is

$$Ax^2 + Bxy + Cy^2 + Dx + Ey + F = 0 \tag{2.40}$$

where A, B, C, D, E, and F are constant coefficients. The mathematics for solving Equation 2.40 for all of the conic section equations is a "hairy" algebraic undertaking and is beyond the scope of this text. The solutions for the equation are given in Figure 2.13.

The equations in Section 2.3.5 are useful in describing the conic sections in Cartesian coordinates, but are not as useful when describing orbits for the more convenient polar coordinate system. Equation 2.40 can be converted to polar coordinates and solved for the radius as a function of the true anomaly angle, resulting in one equation that describes all the conic sections rather than the four shown in Figure 2.13. In polar coordinates, the equation is

$$r = \frac{a(1 - e^2)}{1 + e\cos\nu}. \tag{2.43}$$

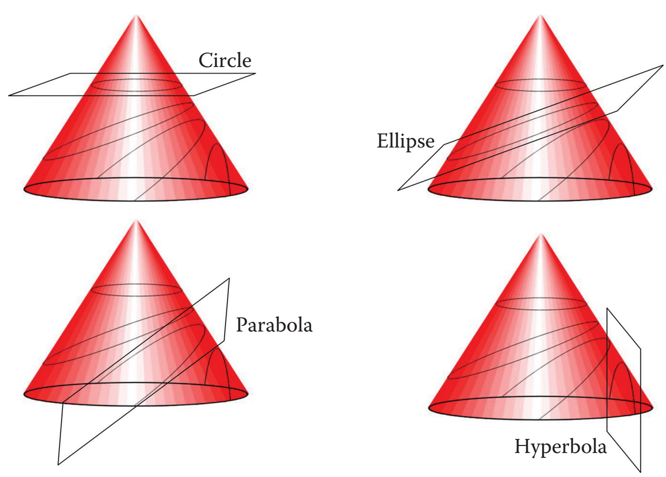


FIGURE 2.12
The circle, ellipse, parabola, and hyperbola are all conic sections.

Circle

$$r^2 = x^2 + y^2 \tag{2.35}$$

Ellipse

$$\frac{x^2}{a^2} + \frac{y^2}{b^2} = 1 \tag{2.37}$$

Parabola

$$y^2 = 4r_p x \tag{2.41}$$

Hyperbola

$$\frac{x^2}{a^2} - \frac{y^2}{b^2} = 1 \tag{2.42}$$

FIGURE 2.13
The Cartesian coordinate equations for the conic sections.

In Equation 2.43, the lowercase e is not to be confused with the base value of the natural logarithm ($e = 2.71828\ldots$) and is called the *eccentricity* of the conic section. The eccentricity is defined as

$$e = \sqrt{1 - \frac{b^2}{a^2}} = \frac{r_a - r_p}{r_a + r_p}. \tag{2.44}$$

The numerator of Equation 2.43 is equal to the semilatus rectum or

$$p = a(1 - e^2). \tag{2.45}$$

Also, it is important to realize that the different values for the eccentricity define the different conic sections as

Circle: $e = 0$
Ellipse: $0 < e < 1$
Parabola: $e = 1$
Hyperbola: $e > 1$

2.3.6 Kepler's Laws

So what does all this detailed discussion about ellipses and conic sections have to do with rocket science? The information about the conic sections in Section 2.3.5 will prove quite important in the development of orbits and trajectories, but we need a little more information about the physics involved with rockets and spacecraft and how they interact with planets and gravitational fields.

We have already discussed Newton's Law of Universal Gravitation, and now we will add to that discussion the physics governing uniform circular motion. Consider a spacecraft in a circular orbit about the Earth. Gravity pulls the spacecraft to the Earth, and the spacecraft pulls the Earth toward it. The magnitude of the force is given by

$$F = \frac{GM_{Earth} m_{spacecraft}}{(R + r)^2}. \tag{2.46}$$

In the above equation, R is the distance from the center of the Earth (larger mass) to the center of mass of the two-body system, and r is the distance from the center of mass to the spacecraft (smaller mass). Because the mass of the Earth is much larger than the mass of the spacecraft, the distance between the center of the Earth and the center of mass of the two-body systems is very small compared to the distance to the spacecraft, and Equation 2.46 can be rewritten as

$$F = \frac{GM_{Earth} m_{spacecraft}}{r^2}. \tag{2.47}$$

The spacecraft travels around the Earth in the circular orbit and is traveling with an angular velocity, ω, and the force opposing the gravitational force is called the *centrifugal force*. The centrifugal force is

$$F = m_{spacecraft} \omega^2 r. \tag{2.48}$$

Because the orbit is closed and the spacecraft stays locked in a path around the Earth, the gravitational force in Equation 2.47 must be equal to the centrifugal force in Equation 2.48. Therefore,

$$F = \frac{GM_{Earth} m_{spacecraft}}{r^2} = m_{spacecraft} \omega^2 r. \tag{2.49}$$

Simplifying Equation 2.49 results in

$$GM_{Earth} = \omega^2 r^2. \tag{2.50}$$

Realizing that the angular velocity is $2\pi/T$ where T is the period of the orbit in seconds, we can now solve for the period of the orbit (or time for the spacecraft to travel once around the planet) in Equation 2.50. Substituting $\omega = 2\pi/T$ into Equation 2.50 yields

$$T^2 = \frac{4\pi^2 r^3}{GM_{Earth}}. \tag{2.51}$$

Equation 2.51 tells us that the square of the period of the orbit is proportional to the cube of the radius of the orbit. This holds true for elliptical orbits as well where r is replaced with a in Equation 2.51. We have just derived Kepler's third law. Johannes Kepler discovered this law after more than 16 years of analysis of his mentor's (Tycho Brahe) lifelong data collection of the motion of the planets and stars visible to the naked eye. Also, from these data, he derived that all the planets map out elliptical orbits about the sun, which is Kepler's first law. He discovered this without the benefit of Newton's Law of Universal Gravitation.

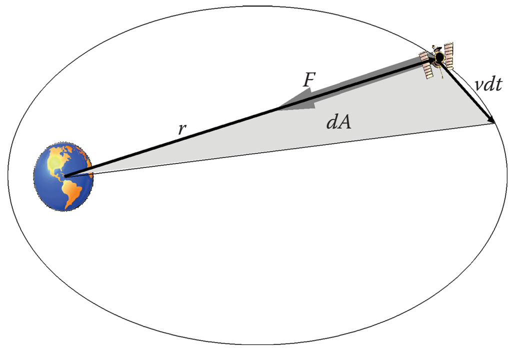


FIGURE 2.14
The force acting on a spacecraft due to Earth's gravity holds it in an elliptical orbit that follows Kepler's laws.

Figure 2.14 shows a spacecraft in an elliptical orbit about the Earth. The force due to gravity acting on the spacecraft is radially inward toward the center of the Earth. Any force that acts toward or away from a fixed point radially is called a *central force*. The equation for describing the torque on the spacecraft due to a central force is

$$\tau = r \times F \tag{2.52}$$

where τ is the torque, r is the radius vector from the Earth to the spacecraft, and F is the force vector due to gravity. The two vectors are both parallel to each other, and the cross product of parallel vectors is zero, so

$$\tau = r \times F = r\hat{r} \times F\hat{r} = rF \sin\theta = rF \sin(0) = 0 \tag{2.53}$$

where the angle θ is the angle between the vectors r and F. Torque is defined as the time rate of change of angular momentum L, which is the first derivative of the angular momentum or

$$\tau = \frac{dL}{dt}. \tag{2.54}$$

The angular momentum is defined as

$$L = r \times mv. \tag{2.55}$$

Here, v is the velocity vector of the spacecraft. Because in Equation 2.53, we can see that the torque is zero; then, we can set Equation 2.54 equal to zero and integrate it once. The integration shows us that the angular momentum is a constant. Equation 2.55 tells us that the angular momentum is also the spacecraft mass multiplied by the cross product of the radius vector and the velocity vector of the spacecraft. In other words,

$$L = r \times mv = \text{constant}. \tag{2.56}$$

So, L is a constant. Rewriting Equation 2.56 might give us some perspective as to what it truly means. Placing the constants on one side of the equation gives

$$\frac{L}{m} = r \times v. \tag{2.57}$$

Equation 2.57 still does not tell us much useful information about the spacecraft–Earth system, but, with a little more manipulation, it might. Consider Figure 2.14 where the shaded triangle represents the area, dA, that the spacecraft's radius vector sweeps out in an incremental time dt. The area of the triangle is one-half the base, vdt, multiplied by the height, r. If we multiply both sides of Equation 2.57 by dt/2, we find

$$\frac{L}{2m} dt = \frac{1}{2} r \times vdt. \tag{2.58}$$

Taking the magnitude of each side yields

$$\frac{|L|}{2m} dt = \frac{1}{2} |r \times vdt| = \frac{L}{2m} dt. \tag{2.59}$$

The middle part of Equation 2.59 is actually the area of the triangle in Figure 2.14, so

$$dA = \frac{L}{2m} dt. \tag{2.60}$$

Solve for the time derivative of dA (effectively dividing through by dt) and we get

$$\frac{dA}{dt} = \frac{L}{2m}. \tag{2.61}$$

Equation 2.61 is Kepler's second law. What it states is that the radius vector to the central body of an orbit to the orbiting body sweeps out equal areas over equal times. In other words, any 1-sec interval along the path of the spacecraft's orbit will sweep out a triangle with an area equal to any other 1-sec interval along any other part of the spacecraft's orbit.

We have thus far derived Kepler's third law and second law and stated Kepler's first law. Succinctly put, Kepler's laws are shown in Figure 2.15. Kepler developed these laws to describe the motion of planets orbiting the Sun, but we should be able to realize at this point that these laws will hold for any smaller body orbiting any larger body in any elliptical orbit. Orbits that follow these three laws are said to be following "Keplerian orbits." We also know now that these Keplerian orbits are particular cases of conic sections, as discussed in Section 2.3.5. In the next two sections, we will see where the other two cases of conic sections are important.

2.3.7 Newton's *Vis Viva* Equation

We have already discussed in detail Isaac Newton's Law of Universal Gravitation. We just spent a great deal of effort discussing conic sections and then Kepler's Laws. But our basic

Kepler's First Law
All planets move in elliptical orbits with the Sun at one of the focal points.

Kepler's Second Law
The radius vector joining any planet to the Sun sweeps out equal areas in equal times.

Kepler's Third Law
The square of the period of an orbit is proportional to the cube of the radius of that orbit.

FIGURE 2.15
Kepler's Laws for planetary motion.

toolbox needed to grasp rocket science is not quite complete. There are four more basic tools that are critical to our understanding of the subject.

The first three of these tools are the so-called Newton's laws of motion. We have actually been using the premise of these three laws throughout this chapter and will continue to do so for the rest of this book. Therefore, we will formally state them here in Figure 2.16.

As Newton developed these laws, along with the Law of Universal Gravitation, he was aware of Kepler's earlier work. In fact, until Newton, Kepler's Laws were practically a curve fit to experimental data. It was Newton who, through his laws of motion and of universal gravitation, proved that Kepler's Laws were correct.

Interestingly enough, Newton worked these proofs out years before he published them. It was not until Edmund Halley, who discovered Halley's Comet, asked Newton what shape an orbit of a planet would be if it were coasting along in a gravitational field. Newton immediately told Halley that the planet would follow an elliptical orbit and that he had already worked out the math on it years before.

Newton actually could not find his previous work and set about solving the problem once again. When he finished, he not only had developed a very detailed understanding of planetary orbits as Kepler's Laws describe, but he also formalized them to describe the orbital motion of any smaller body orbiting a larger one. It was through his brilliant combination of the laws of motion and gravity that he developed the so-called *vis viva* equation.

Newton's First Law—"the law of inertia"
An object in a state of uniform motion tends to remain in that state of motion unless an external force is applied to it. Likewise, an object at rest will remain at rest unless an outside force is acting on it.

Newton's Second Law—"$F = ma$"
The rate of change of momentum is proportional to the force impressed upon an object and is in the same direction of the force. The force, F, and acceleration, a, are both vectors and are expressed by the relationship with the object's mass, m, as $F = ma$.

Newton's Third Law—"the law of action and reaction"
For every action, there is an equal and opposite reaction.

FIGURE 2.16
Newton's Laws of Motion are necessary tools for understanding the basics of rocketry.

Vis viva is Latin for the phrase "living force" in which, in the case of orbits, the living force is gravity. Consider the potential energy of a satellite body orbiting a larger body, such as the Earth, as described by the potential energy form of Newton's Law of Universal Gravitation in Equation 2.24. Also, realize that the kinetic energy of the satellite is 1/2 mv^2. Therefore, the total energy of the satellite in its orbit is the sum of the kinetic and potential energies and is given as

$$E = \frac{1}{2}mv^2 - \frac{GM_{Earth}m}{r}. \tag{2.62}$$

The above equation is sometimes referred to as the *specific mechanical energy equation*. In the above, equation E is a constant and will remain the same at any point along the orbit. So, we can rewrite Equation 2.62 for two different points on the orbit as

$$E = \frac{1}{2}mv_1^2 - \frac{GM_{Earth}m}{r_1} = \frac{1}{2}mv_2^2 - \frac{GM_{Earth}m}{r_2}. \tag{2.63}$$

Reducing some common factors from Equation 2.63 yields

$$\frac{1}{2}v_1^2 - \frac{GM_{Earth}}{r_1} = \frac{1}{2}v_2^2 - \frac{GM_{Earth}}{r_2}. \tag{2.64}$$

From our discussion in Section 2.3.5 about ellipses, we know that the radius at the periapsis of an elliptical orbit is given by Equation 2.43 when the true anomaly angle is zero degrees and is, therefore,

$$r_{periapsis} = a(1-e). \tag{2.65}$$

Also, we need to know here that the velocity of the satellite at the periapsis is

$$v_{periapsis} = \sqrt{\frac{GM_{Earth}}{a}}\sqrt{\frac{1+e}{1-e}}. \tag{2.66}$$

Substituting Equations 2.65 and 2.66 into Equation 2.64 results in

$$\frac{1}{2}v_1^2 - \frac{GM_{Earth}}{r_1} = \frac{1}{2}\frac{GM_{Earth}}{a}\left(\frac{1+e}{1-e}\right) - \frac{GM_{Earth}}{a(1-e)}. \tag{2.67}$$

Reducing some common factors in this equation gives

$$\frac{1}{2}v_1^2 - \frac{GM_{Earth}}{r_1} = \frac{GM_{Earth}}{a(1-e)}\left(\frac{1+e}{2} - 1\right) = \frac{GM_{Earth}}{a(1-e)}\left(\frac{1}{2} + \frac{e}{2} - 1\right) \tag{2.68}$$

$$\frac{1}{2}v_1^2 - \frac{GM_{Earth}}{r_1} = \frac{GM_{Earth}}{a(1-e)}\left(\frac{-(1-e)}{2}\right) = -\frac{GM_{Earth}}{2a}. \tag{2.69}$$

Solving for v^2 results in

$$v_1^2 = GM_{Earth}\left(\frac{2}{r} - \frac{1}{a}\right). \tag{2.70}$$

Equation 2.70 is the *vis viva* equation. In most cases, the term GM_{Earth} is written as μ, and so the equation is usually written as

$$v^2(r) = \mu\left(\frac{2}{r} - \frac{1}{a}\right), \text{ or} \tag{2.71a}$$

$$v(r) = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}}. \tag{2.71b}$$

It should be mentioned here that Equation 2.71a is often referred to as the *orbital energy conservation equation*.

Equation 2.71a and b are equations for the conic sections that describe orbits, as discussed in Section 2.3.5. If the conic section is for a circular orbit, then $a = r$, and the equation becomes

$$v^2(r) = \frac{\mu}{r}. \tag{2.72}$$

or,

$$v_{circ}(r) = \sqrt{\frac{\mu}{r}}. \tag{2.73}$$

Now, consider an elliptical orbit where the apoapsis is at infinity. In other words, the satellite would never reach the apoapsis and would continue traveling away from Earth. This is the situation when the spacecraft has reached the *escape velocity* and is

$$v_{esc}(r) = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}} = \sqrt{\frac{2\mu}{r} - \frac{\mu}{\infty}} = \sqrt{\frac{2\mu}{r}}. \tag{2.74}$$

We have now developed the major equations needed in order to understand the basics of conic section orbits. With the polar coordinate equation for the conic sections in Equation 2.43, the shape of any section can be defined depending on the eccentricity and the true anomaly. A subset of Equation 2.43 when the eccentricity is in the range for elliptical orbits is a statement of Kepler's first law. Equations 2.61 and 2.51, respectively, describe Kepler's second law and Kepler's third law. Equation 2.71 allows us to calculate the velocity of a given orbit including the special cases given in Equations 2.73 and 2.74. Figure 2.17 is a summary of these equations, which are the basic mathematical tools for understanding orbital mechanics.

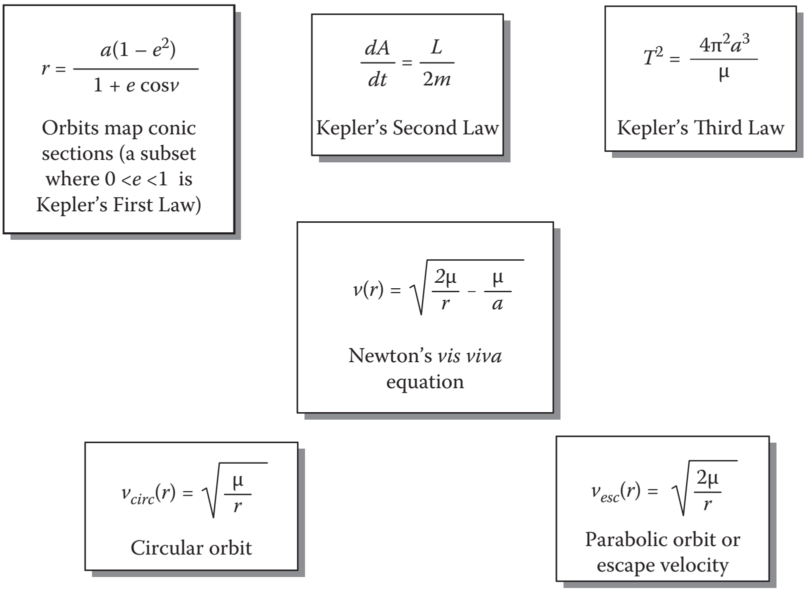


FIGURE 2.17
The basic mathematical tools for understanding orbits are listed here. Orbits map out conic sections, follow Kepler's laws, and can be described by Newton's *vis viva* equation.

2.4 Orbit Changes and Maneuvers

2.4.1 In-Plane Orbit Changes

Consider a spacecraft in a circular orbit at radius, r. If, for some reason, our mission required that the spacecraft needed to change its present circular orbit to an elliptical one, this can be done by adding velocity to the spacecraft. Actually, any conic section orbit can be transferred into any other conic section orbit simply by adjusting the velocity and providing that both orbits are in the same plane or *coplanar*.

So, let's assume the spacecraft is to be placed into an elliptical orbit of periapsis $r_p = r$ and apoapsis r_a. From Figure 2.11, in our ellipse discussion, we can see that the semimajor axis, a, of the orbit can be found as

$$a = \frac{r_a + r_p}{2}. \tag{2.75}$$

Equation 2.73 is the velocity of a circular orbit, and we can use Equation 2.71b to determine the velocity for the elliptical orbit. Substituting $r_p = r$ into Equation 2.73 gives us

$$v_{circ}(r) = \sqrt{\frac{\mu}{r_p}} \tag{2.76}$$

and Equation 2.75 into Equation 2.71b results in

$$v(r) = \sqrt{\frac{2\mu}{r_p} - \frac{2\mu}{r_a + r_p}}. \tag{2.77}$$

The velocity change or Δv (pronounced "delta-vee") needed to go from the circular orbit to the elliptical orbit is then

$$\Delta v = v - v_{circ} = \sqrt{\frac{2\mu}{r_p} - \frac{2\mu}{r_a + r_p}} - \sqrt{\frac{\mu}{r_p}}. \tag{2.78}$$

Simplifying Equation 2.78 gives us an equation for simple coplanar orbit changes from circular to elliptical orbits:

$$\Delta v = \sqrt{\frac{\mu}{r_p} - \frac{2\mu}{r_a + r_p}}. \tag{2.79}$$

Figure 2.18 illustrates this maneuver. It should also be mentioned here that the amount of Δv required to change from a circular orbit to an elliptical one is the same as the amount of Δv required to go from the elliptical orbit back to a circular one. All coplanar orbital maneuvers are reversible.

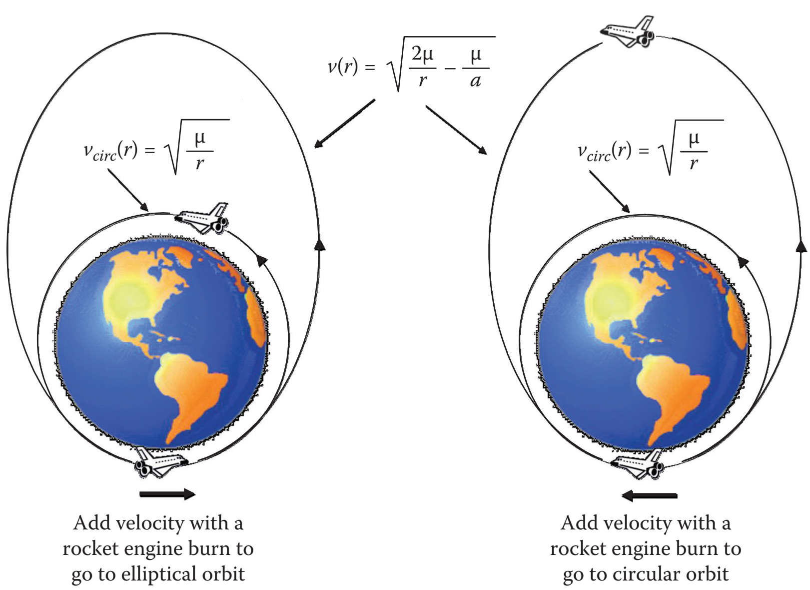


FIGURE 2.18
Coplanar orbit changes are reversible and can be done simply by adding velocity.

2.4.2 Example 2.4: Hohmann Transfer Orbit

When the Space Shuttle is in orbit, it is typically at LEO in a circular orbit at approximately 325-km altitude. On occasion, the Shuttle has been used to service or deploy satellites at higher altitudes. One particular set of examples are the Hubble Space Telescope (HST) service missions. The HST orbit altitude is approximately 500-km high. Using the coplanar orbit change process described in Figure 2.18 and Equation 2.79, the Δv requirements to push the Shuttle into an elliptical orbit that will reach the HST altitude and then to circularize the orbit can be determined. Figure 2.19 shows the rocket burns needed for the Shuttle to reach the HST orbit.

The initial burn is calculated just as in the discussion in Section 2.4.1. Substituting the values given into Equation 2.79,

$$\begin{aligned} \Delta v &= \sqrt{\frac{\mu}{r_p} - \frac{2\mu}{r_a + r_p}} \\ &= \sqrt{\frac{398,600\,\text{km}^3/\text{s}^2}{(325+6,370)\,\text{km}} - \frac{2(398,600\,\text{km}^3/\text{s}^2)}{(500+6,370)\,\text{km} + (325+6,370)\,\text{km}}} \\ &= 0.876\,\text{km/s}. \end{aligned} \tag{2.80}$$

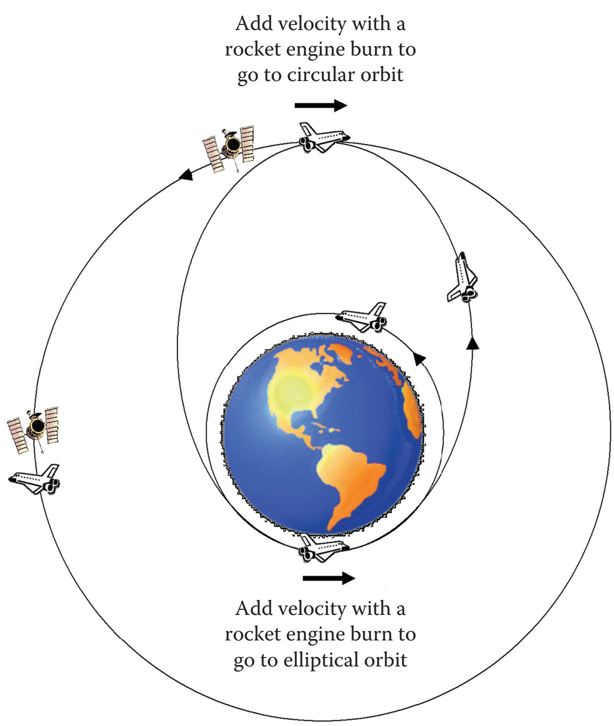


FIGURE 2.19
Rocket engine burns are needed for the Shuttle to reach the HST. This circular-to-elliptical-to-circular maneuver is called a Hohmann transfer.

So, a $\Delta v = 0.876$ km/sec is required to enter into the elliptical orbit with a periapsis at the lower orbit and an apoapsis at the HST orbit.

Once the Shuttle reaches the apoapsis of the elliptical transfer orbit, it must then conduct a burn to speed up into the circular orbit of the HST. In order to calculate the Δv needed for this circularization, we must realize that the circular orbit radius is ra and that we are subtracting the elliptical orbit velocity from the circular orbit velocity, so there is a sign change, and Equation 2.79 must be rewritten as

$$\Delta v = \sqrt{\frac{2\mu}{r_a + r_p} - \frac{\mu}{r_a}}. \tag{2.81}$$

Substituting values into Equation 2.81,

$$\begin{aligned}\Delta v &= \sqrt{\frac{2\mu}{r_a + r_p} - \frac{\mu}{r_a}} \\ &= \sqrt{\frac{2(398{,}600\ \text{km}^3/\text{s}^2)}{(500+6{,}370)\,\text{km} + (325+6{,}370)\,\text{km}} - \frac{398{,}600\,\text{km}^3/\text{s}^2}{(500+6{,}370)\,\text{km}}} \\ &= 2.076\,\text{km/s}.\end{aligned} \tag{2.82}$$

We see from this calculation that the burn required to circularize the Shuttle into the HST orbit must supply a $\Delta v = 2.076$ km/sec. For the Shuttle to return to its lower orbit, it simply has to do the same burns, but pointed in the opposite directions. Also note here that the timing of the Shuttle reaching the apoapsis of the transfer ellipse and the HST approaching the same point in space is critical for them to rendezvous.

The orbit maneuver discussed in this example is known as the Hohmann transfer. It was discovered by the German engineer Walter Hohmann in 1925. Applying Kepler's third law enables us to determine the time it would take for the Hohmann transfer. First, we must calculate the semimajor axis, a,

$$a = \frac{r_a + r_p}{2} = \frac{(500+6{,}370)\,\text{km} + (325+6{,}370)\,\text{km}}{2} = 6{,}782.5\,\text{km}. \tag{2.83}$$

The period of the transfer ellipse is then

$$T = \sqrt{\frac{4\pi^2 a^3}{\mu}} = \sqrt{\frac{4\pi^2(6{,}782.5\ \text{km})^3}{398{,}600\,\text{km}^3/\text{s}^2}} = 5{,}556.2\,\text{s}. \tag{2.84}$$

The orbit transfer is actually half the period, as can be seen from Figure 2.18. Therefore, the transfer time is 1,567.8 sec or about 26 min.

2.4.3 Bielliptical Transfer

Figure 2.20 shows the bielliptical transfer maneuver. This maneuver is a three-burn method of transferring from one circular orbit to another. In cases where the larger orbit radius is

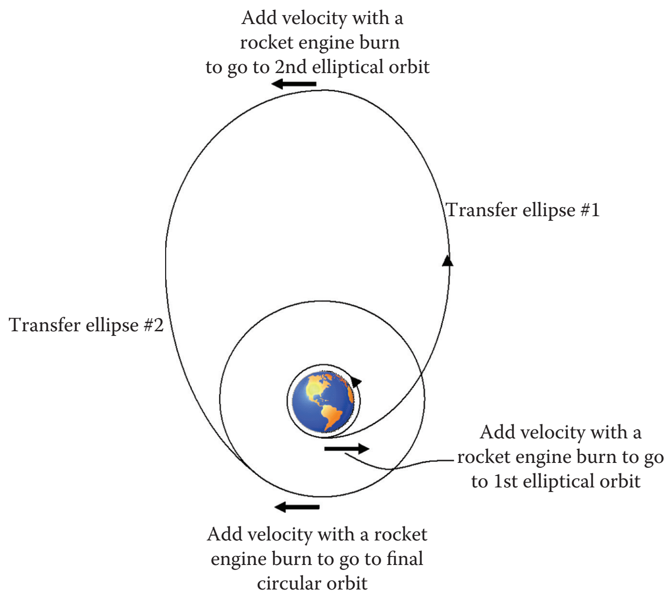


FIGURE 2.20
Three rocket burns are required for the more efficient bielliptical transfer maneuver. The maneuver is complicated and seldom used.

15.58 times larger than that of the smaller orbit radius, it is actually more energy efficient by as much as 8%. However, the added burn makes the maneuver more complicated and, therefore, more risky that a failure could occur. So, the maneuver is seldom used in spacecraft missions, but it is worth mentioning here.

2.4.4 Plane Changes

To this point, we have only discussed coplanar orbital maneuvers. Now, we will consider changing an orbit's inclination or the plane within which the orbit lies. The Space Shuttle actually has to do this on occasion because the International Space Station (ISS) is at an inclination of 51.6°, but it typically does this on ascent. However, if the shuttle were already in an orbit at 28° where the HST is located and needed to change planes to the ISS inclination, it could, though it would require a lot of fuel. A simple plane change is shown in Figure 2.21, and the $\Delta v_{planechange}$ required at the point where the two orbits intersect is

$$\Delta v_{planechange} = 2v \sin\left(\frac{\Delta i}{2}\right) \tag{2.85}$$

where Δi is the change in inclination angle desired.

2.4.5 Interplanetary Trajectories

We have yet to discuss in detail the hyperbolic conic section, which represents orbits with eccentricities greater than 1. The *vis viva* equation is a little different for such orbits because

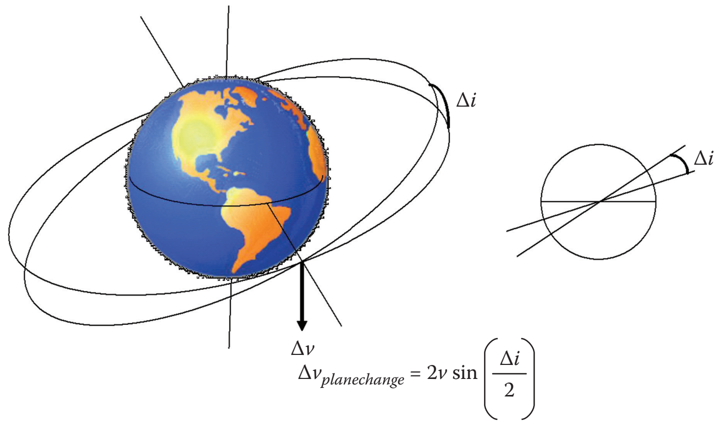


FIGURE 2.21
In order to conduct a plane change or change in inclination, Δv must be added where the orbits intersect.

they have excess energy at infinity and, therefore, have a slightly different solution. In our discussion of the elliptical orbits in Section 2.3.5, we showed from the energy in Equation 2.62 that the energy of the system is a constant and is balanced between kinetic and potential energy. Also, when the apoapsis is at infinity, the ellipse is a parabola, and both the kinetic and potential energy at that point are zero. For a hyperbolic orbit, this is not the case. There is excess kinetic energy at infinity. This results in the *vis viva* equation for a hyperbolic trajectory to be

$$v(r) = \sqrt{\frac{2\mu}{r} + \frac{\mu}{a}} = \sqrt{\frac{2\mu}{r} + C_3} = \sqrt{\frac{2\mu}{r} + v_\infty^2}\ . \tag{2.86}$$

In Equation 2.85, the C_3 is called the *characteristic energy* and sometimes the *launch energy*. It is equal to the square of the *hyperbolic excess velocity*. Quite often, orbital mechanics are heard discussing the "see three" of a mission profile, and it is this characteristic energy to which they are referring. C_3 is a measure of the amount of speed a spacecraft needs to lose before it can achieve an orbit around a particular planet. It is also a measure of the amount of speed a spacecraft must gain in order to leave a circular orbit and achieve escape velocity. Adding C_3 to the circular orbit velocity in Equation 2.73 gives the escape velocity in Equation 2.74.

If a spacecraft is launched such that it has a C_3 greater than zero, then it will escape from Earth on a hyperbolic trajectory. This is how interplanetary spacecraft missions are planned. Figure 2.22 shows a typical mission profile for an interplanetary mission. For example purposes, we will assume a mission from Earth to Mars. The spacecraft is launched with excess velocity, and, once it has escaped Earth's sphere of influence, as described in Equation 2.25 and Table 2.1, it is then in the coasting phase. Occasionally, there are trajectory correction burns conducted in order to optimize the trajectory, but these are for navigation and control, not for speed. The spacecraft coasts until it reaches the Martian sphere of influence. (Again, see Equation 2.25 and Table 2.1.) At this point, a braking burn must be done in order to shed the C_3 (relative to Mars) so that it can enter a stable Mars orbit. Subsequent rocket burns can be conducted to circularize to the desired orbit. Also note that Table 2.2 gives gravitational parameters for many bodies in the solar system that are useful for calculating orbits around other planets.

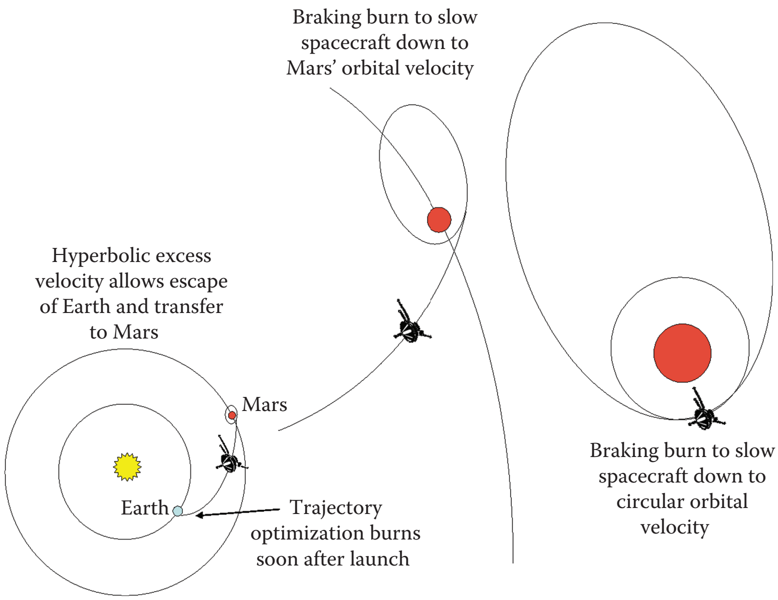


FIGURE 2.22
Typical mission profile for a planetary mission.

TABLE 2.2
The Gravitational Parameters for the Bodies of Our Solar System

Body	μ in km³/s²
Sun	132,712,440,018
Mercury	22,032
Venus	324,859
Earth	398,600
Moon	4,903
Mars	42,828
Jupiter	126,686,534
Saturn	37,931,187
Uranus	5,793,947
Neptune	6,836,529
Pluto	1,001

2.4.6 Gravitational Assist

Another example of interplanetary mission is the flyby where the spacecraft only comes close to a planet or body. This is done sometimes for scientific reasons and others for what is sometimes called a "gravitational assist" or "slingshot." Figure 2.23 shows the Pioneer 10 spacecraft trajectory profile for the Jupiter flyby. The spacecraft conducted various course corrections and optimizations soon after launch and then coasted with hyperbolic excess velocity to Jupiter's sphere of influence. At this point, the spacecraft swung by the planet, conducted various experiments, then picked up velocity due to the gravitational assist of Jupiter, and then traveled onward on a trajectory that would lead it out of the solar system.

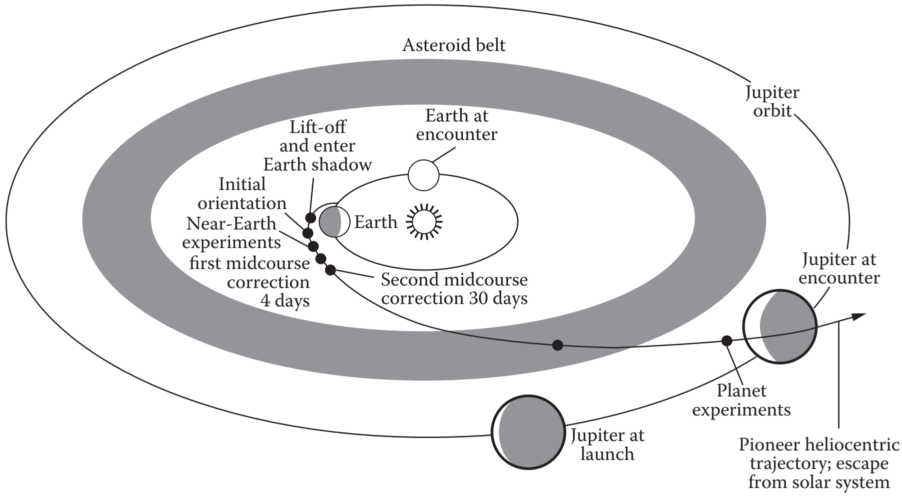


FIGURE 2.23
The Pioneer 10 spacecraft conducted a flyby of Jupiter and used the planet's gravity for an extra velocity boost. (Courtesy of NASA.)

There are a lot of misconceptions about gravitational velocity assists. The gravity of the flyby planet does not add velocity to the vehicle because it is falling inward and picks up speed. In fact, the spacecraft will lose the same amount of speed as it leaves the planet's gravity as it gained by falling inward to it. This is easily demonstrated by tossing a ball upwards in the air. As the ball leaves your hand, it will have an initial velocity; as it reaches its peak, it has zero velocity and begins to fall back to Earth. When the ball lands back in your hand, neglecting air resistance, it will have the same velocity as when it was thrown upward. The same can be said for a rocket entering into a planet's gravitational field and then leaving it. Energy must be conserved so that no excess velocity is given to the spacecraft due to the gravitational field. So where does it come from?

The mechanism of the velocity boost from a gravitational assist can be cleared up by a simple discussion of velocity and momentum. Consider a man sitting in a boat afloat in a lake moving away from shore at a constant velocity. His buddy is standing on the bank of the lake and tosses the boater a baseball. (Assume the ball is thrown in the same direction of travel as the boat.) The man in the boat catches the baseball, and the boat has imparted to it the momentum of the baseball. In actuality, the ball has much less mass than the boat and is not traveling exceedingly fast so the momentum imparted to the boat is too small for the boater to notice, but there is a momentum transfer nonetheless. The velocity change is calculated from the law of conservation of momentum:

$$m_{ball+\mathrm{boat}} v_{ball+boat} = m_{ball} v_{ball} + m_{boat} v_{boat}. \tag{2.87}$$

Solving for the velocity of the boat after the ball is caught gives

$$v_{ball+boat} = \frac{m_{ball} v_{ball} + m_{boat} v_{boat}}{m_{ball+boat}}. \tag{2.88}$$

Now, if the boater, in return, throws the baseball back to his buddy on the bank, the same amount of momentum is again imparted to the boat due to Newton's third law. After some algebra, the final velocity of the boat is found to be

$$v_{boatfinal} = \frac{2m_{ball}v_{ball}}{m_{boat}} + v_{boat}. \tag{2.89}$$

This momentum transfer is what happens during a gravitational assist to a spacecraft. The ball becomes the planet, and the boat becomes the spacecraft. The velocities are sun-centered trajectory (heliocentric or sun-relative) velocities, and Equation 2.88 becomes

$$v_{spacecraftfinal} = \frac{2m_{planet}v_{planet}}{m_{spacecraft}} + v_{spacecraft}. \tag{2.90}$$

Note that the velocities are written as vectors because the spacecraft and the planet have planar dimensions. From Equation 2.89, it can be determined that a significant boost in velocity vector can be achieved by approaching a planet. The planet is slowed down slightly, but even far less so than the boater is sped up by the baseball. The loss of velocity of the planet is determined by

$$v_{planetfinal} = \frac{2m_{spacecraft}v_{spacecraft}}{m_{planet}} + v_{planet}. \tag{2.91}$$

From Equation 2.90, it is clear that, because the mass of the planet is much larger than the mass of the spacecraft, the velocity of the planet after the gravity assist is practically the same as it was before the assist. Figure 2.24 illustrates the gravity assist maneuver.

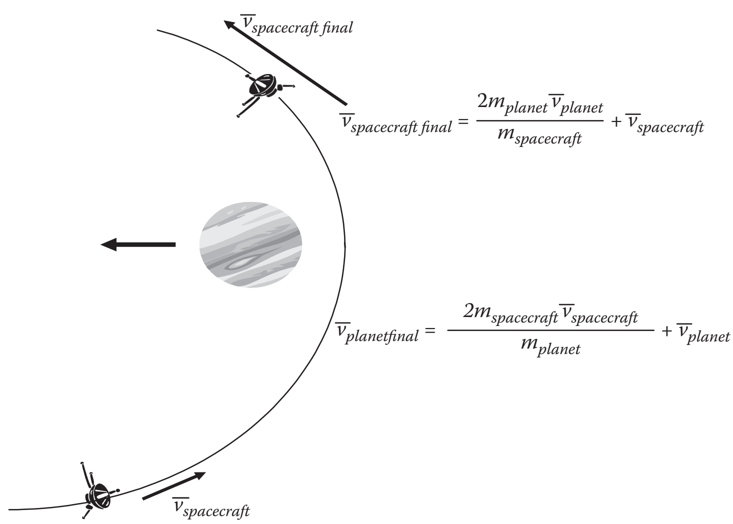


FIGURE 2.24
Gravity assist maneuver.

2.5 Ballistic Missile Trajectories

2.5.1 Ballistic Missile Trajectories Are Conic Sections

We have discussed trajectories and orbits in enough detail at this point to look deeper at the flight path of ballistic missiles. In actuality, the flight path of a ballistic missile consists of three parts: (1) powered flight, (2) free-flight, and (3) reentry. Figure 2.25 illustrates these flight path components, as well as the fact that the free-flight trajectory of the missile is actually an elliptical conic section. The free-flight trajectory can be assumed to be a symmetrical ellipse that begins at MECO or at missile propellant burnout and continues until reentry. Reentry height, h_{re}, and burnout height, h_{bo}, are equal to each other. Also note that the periapsis of the ellipse is inside the radius of the Earth; otherwise, the missile would be in an elliptical orbit about the planet. There are solutions where the periapsis is outside the radius of the Earth, but the path of the ellipse is not. Such cases are not clear orbits about the planet and are instead a trajectory.

If the burnout height and some of the elliptical parameters are known, the free-flight trajectory ellipse can be written as

$$r_{bo} = \frac{a(1-e^2)}{1+e\cos\nu_{bo}}. \tag{2.92}$$

The total range angle, Λ, that the missile traverses is given by

$$\Lambda = \Gamma + \Psi + \Omega. \tag{2.93}$$

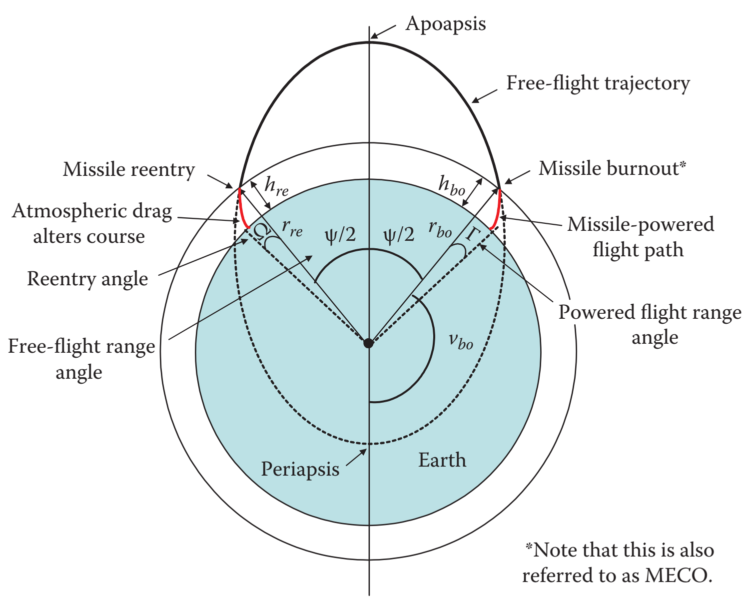


FIGURE 2.25
The ballistic missile trajectory is actually an elliptical conic section.

Figure 2.25 and Equations 2.90 and 2.91 give an overview of the flight path characteristics of a ballistic missile. There are many other factors, such as the rotation of the Earth, and the oblateness of the Earth, accounting for launch inclination, and many others that complicate the missile trajectory model. The devil is indeed in the details, and, very quickly, it becomes clear why any antiballistic missile becomes a very complex beast. The ranges and angles are very large and accuracies out to many decimal places are required in order to allow for a ballistic missile intercept. The slightest error over such long ranges and large angles can cause missile intercepts to miss by many kilometers.

As complex as the problem of understanding missile trajectories is, this section gives a good general overview of the topic. The ballistic missile problem is truly beyond the scope of this text and, in fact, can fill complete volumes all by itself.

2.6 Chapter Summary

In this chapter, we have actually given the reason for rocket propulsion. In Section 2.1, we discussed missions and payloads, which are the things we desire to accomplish and how we plan to accomplish them. Realizing that many missions require placing payloads at long ranges very rapidly (such as missiles delivering explosive payloads to a target) or placing payloads into orbit, the only technological solution known to date for doing this is the rocket. Hence, we understand the need for rockets and rocket science.

In Section 2.2, we began discussing in more detail some of these mission parameters. Missiles typically deliver payloads on a ballistic trajectory. In order to truly understand what a rocket must do in order to deliver a payload, we looked into the details of trajectories and how to mathematically model them.

Likewise, in Section 2.3, we discussed the details of orbits and how to calculate them. We learned of Newton's laws of motion and universal gravitation and of Kepler's Laws. We also derived the so-called *vis viva* equation, which is the most powerful equation in understanding orbits. We also learned that orbits are mathematically described as the conic section, and we discussed in detail some aspects of conic sections.

In Section 2.4, we discussed orbital maneuvers and changes and how to calculate them. Also discussed in this section was how to convert an elliptical orbit to a circular one and how to transfer from one orbit to another via some different approaches. We also investigated interplanetary trajectories and how to fly-by a distant planet, how to enter into an orbit about a distant planet, and how to use a planet's gravity well for a heliocentric velocity boost in a gravity assist maneuver.

Finally, in Section 2.5, we revisited the ballistic missile trajectory with the tools that we learned in the earlier sections in this chapter. Now, we realize that a missile trajectory is actually a conic section and that it can be described in much the same way that an elliptical orbit is modeled.

Chapter 2 has given us an understanding of why rockets are needed and some mathematical tools that are the basics of trajectory and orbital mechanics. The tools used by any good trajectory or orbital mechanic to "fix" any flight path are rockets. This is why we have rocket science and engineering. Without the rocket scientists and engineers, there would be no such tools. From a more romantic aspect, as well as practical (not that anyone has ever confused romantics and practicality), we can note that scientists and mathematicians were imagining and calculating interplanetary orbits and trajectories centuries before they had any idea how to achieve them. It is the rocket scientists and engineers who now have offered us the means to attain these flights of fancy.

Exercises

2.1 Discuss the dichotomy of rocket science in the modern era.

2.2 In your own words, give a definition for a rocket mission.

2.3 What is a payload?

2.4 What is the so-called "SMAD"?

2.5 Give the four basic assumptions required for understanding the basics of projectile motion.

2.6 Define MECO.

2.7 Equation 2.9 gives the parabolic flight path of a rocket trajectory as height, y, as a function of range, x, or $y(x)$. Use the quadratic equation to solve for x as a function of y to give a range equation as a function of height.

2.8 A rocket is launched with a burnout velocity of 75 m/sec, a burnout altitude of 300 m, and a burnout range of 100 m. Assuming a flight path angle of 75°, calculate the final range of the rocket when it impacts the ground.

2.9 Calculate the maximum altitude reached by the rocket in Exercise 2.8.

2.10 Redo Exercise 2.8 to determine the range at MECO altitude. What is the range at MECO if the initial flight path angle is 15°?

2.11 What is the force due to gravitational attraction between the Earth and the Moon? Assume the Moon is 400,000 km from Earth and the mass of the Earth is 5.99×10^{24} kg, and the mass of the Moon is 7.36×10^{22} kg.

2.12 A satellite is in a circular orbit at 100 km above the Earth. What is the orbital velocity of the satellite? How long does it take for the satellite to make one complete orbit around the Earth?

2.13 What is the semilatus rectum?

2.14 Give the equation for a conic section.

2.15 A spacecraft is traveling in an orbit with a periapsis at 100 km and an apoapsis at 1,000 km. What is the eccentricity of the orbit? This orbit is what type of conic section?

2.16 Calculate the semilatus rectum of the spacecraft orbit in Exercise 2.15.

2.17 What is the period of the orbit described in Exercise 2.15?

2.18 What is the velocity of the spacecraft in Exercise 2.15?

2.19 Calculate the Δv needed to circularize an elliptical orbit with an apoapsis at 500 km above the Earth and a periapsis at 325 km above the Earth. (Hint: See Example 2.3.)

2.20 Calculate the Δv burns needed to conduct a Hohmann transfer from a 300-km circular orbit around Earth to a 35,000-km circular orbit around Earth.

2.21 Calculate the transfer time for the Hohmann transfer given in Exercise 2.20.

2.22 A Space Shuttle is in a 325-km circular orbit in a 28° inclination. How much Δv is needed to move the Shuttle to a 51° inclination?

2.23 What is C_3?

2.24 A Mars probe leaves Earth's sphere of influence with a C_3 of 16 km^2/sec^2. How much Δv is required for the probe to enter a Mars orbit with a periapsis at 100 km and an apoapsis at 1,000 km?

2.25 In order to go from Equations 2.87 to 2.88 (as well as Equations 2.89 and 2.90), some algebra was needed. Do the algebra calculation showing all the steps.

2.26 A ballistic missile has a powered flight range angle of 4° and a reentry range angle of 5°. If the missile has a total ground range of 8,000 km, what is its free-flight range angle? (Hint: Assume the 8,000-km range is the distance the missile travels around the circumference of the Earth. The radius of the Earth is 6,370 km.)

3

How Do Rockets Work?

In Chapter 1, we briefly answered the question, what are rockets? It began by talking about the history of rocket science and engineering and the key incidents in history that led to the development of the modern rocket era. We also discussed some of the types of modern rockets, as well as a very basic description of the components and subsystems required to build a rocket. We also talked a little about the types of rocket development programs that are being initiated to assure the future of rocketry.

We then, in Chapter 2, discussed why rockets are needed. We explored the ideas of mission requirements and payloads. We discussed in some detail trajectories and orbits, which are the places and velocities that we would like to place payloads and conduct missions. In order to conduct these missions along these trajectories or from these orbits, we must be able to reach the trajectories and orbits that we desire. That is why rockets are needed.

In this chapter, we will begin to explain how rockets can actually accomplish the goals set forth for them in Chapter 2. How do rockets work? That is a question that has taken centuries of development of the laws of nature and engineering practices to answer. In order to understand the answer to this question, we have to begin our discussion with a dialogue about thrust, momentum, and impulse. From there, we will derive the so-called "rocket equation" and then understand the basics of how rockets work.

3.1 Thrust

First and foremost, thrust is a *force* and is measured in *newtons* (*N*). Older rocket scientists might often revert to using pounds of thrust, but we will stick to the International System of Units here. It is a force generated by some propulsive element in order to overcome other forces acting on a body in order to manipulate that body's position and velocity vector. It is the force that is used to propel a rocket or a spacecraft to the destination trajectory or orbit or the landing site desired. Airplanes use propellers or jet engines to generate thrust. Rockets use rocket engines to generate thrust.

Mathematically speaking, thrust is the net external force acting on an object that can be calculated as the rate of change of the momentum of the body. In other words,

$$p = mv \tag{3.1}$$

where p is the momentum of a mass, m, moving with scalar velocity, v. The average force on the mass, therefore, is

$$Thrust = F = ma = \frac{dp}{dt} = m\frac{dv}{dt} + v\frac{dm}{dt}. \tag{3.2}$$

Equation 3.2 describes the conservation of momentum of a mass that is moving with a varying velocity plus the possibility that some of the system is at constant velocity, but changing in mass. What does this really mean?

Figure 3.1 shows a rocket with both fuel and oxidizer tanks. As the fuel is burned with the oxidizer in the combustion chamber, high-pressure gases are created from the chemical combustion process. As the gases push against the combustion chamber walls with higher pressure than the outside ambient pressure, they are forced out the opening at the bottom of the chamber called the *throat*, through a nozzle where the velocity of the exhaust is accelerated, and then out into the space behind the rocket. It is the escaping of these highly accelerated exhaust gases that propel the rocket through Newton's Third Law.

It should be noted here that Newton's Third Law is a restatement of the law of conservation of momentum for this system. The simplest rocket would have one particle for propellant (like the baseball and boat example given in Chapter 2). As the propellant mass is thrown from the main mass of the rocket, the rocket is accelerated in the other direction, and momentum is conserved. As the calculation of momentum conservation and velocity change for a ball being thrown from a boat was already discussed in Chapter 2, we will not redo it here. If the concept is confusing to the readers at this point, then it can be reexamined.

With Figure 3.1 in mind, we will reconsider Equation 3.2. The thrust equation can be rewritten as

$$F_{Thrust} = \frac{dm}{dt} v + m\frac{dv}{dt} = \dot{m}v_e + m\dot{v}_e. \tag{3.3}$$

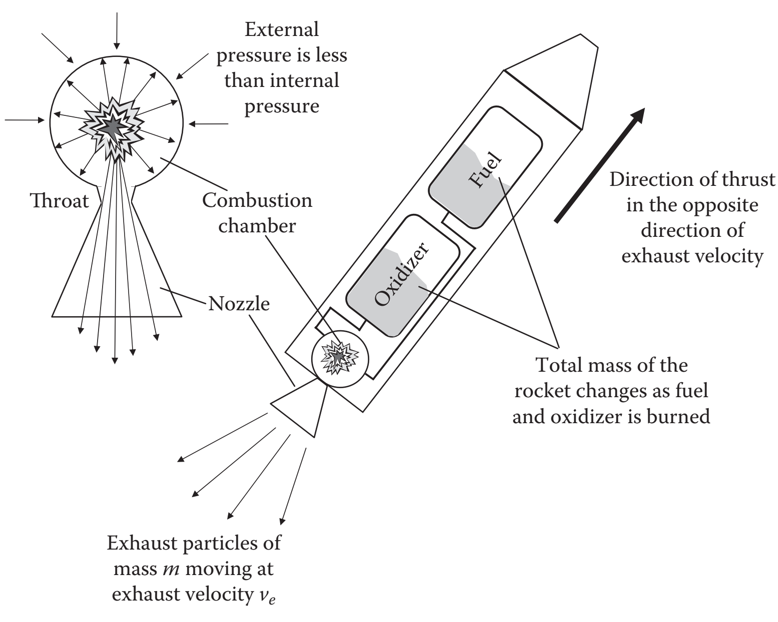


FIGURE 3.1
The liquid rocket engine's system mass varies with time as the fuel and oxidizer are burned in the combustion chamber, generating exhaust gases that are accelerated through the nozzle-generating thrust.

For convenience, the rate of change convention of placing a dot over the variable is used, and, for example, the "m-dot" is used to describe the time rate of change of mass from the rocket. The escaping gases have an exhaust velocity v_e. The left part of the right-hand side of Equation 3.3 is a useful and understandable quantity. The m-dot tells us the rate at which the rocket engine is burning fuel and oxidizer and is controlled by the throttle of the vehicle. This is the same type of throttling that occurs when you step on the accelerator pedal of an automobile. When you hear the expression, "Roger, Shuttle, go at throttle up"—this is exactly what is taking place. The astronauts onboard the Space Shuttle are increasing the throttle, or, in other words, they are increasing the m-dot. The exhaust velocity is a constant and is determined by the chemicals used in the combustion process, as well as the geometrical design of the chamber, throat, and nozzle. Once the rocket engine is designed and built, the exhaust velocity does not change unless a different fuel or oxidizer is used (which is typically not good for the rocket as they are not usually designed but for one fuel/oxidizer combination).

The right part of the right-hand side of Equation 3.3 is a bit more erroneous and does not immediately suggest to us design parameters. However, "v-dot" is acceleration by definition, and, thus, the right part can be described as the force component due to the exhaust mass escaping the rocket that is being accelerated by the diverging nozzle. Another way to look at this is that the exhaust gas is quickly flowing out of this nozzle and is being accelerated faster and faster as it approaches the exit. When the gas exits the nozzle, the flowing gas leaving the rocket in the opposite direction then pushes the rocket forward. To quantify this, we must first realize that force can also be written as pressure, P, multiplied by the area, A, that the pressure is incident upon or

$$F = PA. \tag{3.4}$$

Therefore, if the pressure inside the nozzle just before it exits is the exhaust pressure P_e, and the pressure outside the nozzle is P_o, then the force due to the pressure difference on the nozzle exit surface area A_e is

$$F_{nozzle} = (P_e - P_o)A_e. \tag{3.5}$$

Substituting the right-hand side of Equation 3.5 into Equation 3.2 gives us the rocket thrust equation

$$F_{Thrust} = \dot{m}v_e + (P_e - P_o)A_e. \tag{3.6}$$

Equation 3.6 is an important design equation for rocketry. From it, we can design a rocket engine in general terms. The exhaust velocity and pressure is determined by the fuel/oxidizer combination, throat, and nozzle design. The nozzle design drives the definition of the exit area of the nozzle. Therefore, from Equation 3.6, a rocket can be designed to generate a particular desired thrust. The equation is also powerful in allowing us to analyze a particular engine and determine the thrust that it can generate given its particular design.

3.2 Specific Impulse

As we discussed in Section 3.1, thrust is the force generated by the rocket engine that propels a rocket along its trajectory through the air and space. Specifically, the engine houses a combustion process where a gas is heated and expanded and then forced out the rear of the rocket in the opposite direction as that of the motion of the rocket itself. As we will discuss in Chapter 4, this gas is forced out the back of the rocket through a converging, diverging nozzle system that accelerates the gas flow. For now, just consider the fact that the gas flow is accelerated out of the rocket engine and that that is the main purpose of the rocket engine—to accelerate exhaust gas flow.

To better understand the thrust generated by this accelerated gas flow, a few physics concepts need to be discussed. In classical physics, there is a phenomenon where momentum is imparted to a baseball by a bat (assuming the batter keeps his eye on the ball) called *impulse*. Impulse, I (sometimes called *total impulse*), in its purist sense, is defined as the total integrated force with respect to time and is written as

$$I = \int F\,dt. \tag{3.7}$$

Here, F is force as a constant or a function of time, and dt is the incremental time change variable. Recall from Newton's Second Law that force can be written as the derivative of momentum with respect to time or

$$I = \int F\,dt = \int \frac{dp}{dt}dt = \int dp = \Delta p. \tag{3.8}$$

Equation 3.8 is referred to as the *impulse–momentum theorem*. And, once again, the theorem basically tells us that a force applied to an object over a given amount of time produces an effect, and that effect is an *impulse*. Another way to think of impulse is that it is the change in momentum of an object due to an applied force. Recall the baseball's momentum being changed by hitting a bat. Also note from Equation 3.8 that impulse will have the same units as momentum of kg m/sec or a N · s. Integrating Equation 3.8 yields

$$I = Ft = \Delta p. \tag{3.9}$$

Considering a force generated by a changing mass with a constant velocity, the impulse can be written as

$$I = \int_{m_f}^{m_i} v\frac{dm}{dt}dt = \Delta p = (m_i - m_f)v. \tag{3.10}$$

As our discussion continues, it will become clear why Equation 3.10 might be useful. Clearly, if we have a rocket that is in need of a course change or even a liftoff, we must change its momentum, and Equation 3.10 shows us how. If we start off with an initial mass for the rocket and apply an impulse by ejecting propellant mass out of the back of it at a constant exhaust velocity, we can rewrite the equation as

$$I = (m_i - m_f)v = \Delta m_{propellant} v_e. \tag{3.11}$$

Solving for the exhaust velocity results in

$$\frac{I}{\Delta m_{propellant}} = v_e. \tag{3.12}$$

Before we go further with this line of reasoning, we need to identify something. We said in Equation 3.6 that we defined thrust as something more than simply the m-dot times the exhaust velocity. Reexamining that equation and defining a new parameter called the *equivalent velocity* or sometimes the *effective exhaust velocity*, *C*, we rewrite Equation 3.6 as

$$F_{Thrust} = \dot{m}v_e + (P_e - P_o)A_e = \dot{m}C \tag{3.13}$$

and we see that Equations 3.11 and 3.12 should really be the *equivalent velocity* rather than just the exhaust velocity, thus,

$$\frac{I}{\Delta m_{propellant}} = C. \tag{3.14}$$

This equation tells us that the total impulse imparted to a rocket divided by the propellant mass ejected is equal to the equivalent velocity. The relation is useful in describing the total rocket thrust, but it doesn't really tell us anything about the rocket itself. By defining a new parameter, we can make some very powerful assessments with Equation 3.14.

That parameter is known as the specific impulse, I_{sp}. This is a more useful parameter (given in seconds only) and is written as

$$I_{sp} = \frac{I}{\Delta m_{propellant} g} = \frac{C}{g}. \tag{3.15}$$

What Equation 3.15 tells us is that the I_{sp} of a rocket engine is the total number of seconds that the rocket can deliver thrust equal to the weight of the total propellant mass under acceleration due to one standard Earth gravity, *g*. This is an efficiency number that we use to describe rocket engines. The higher the I_{sp}, the more efficiently the engine can apply Δ-*v* to the spacecraft. Table 3.1 shows some specific impulse values for various rocket systems. From that data, it becomes clear that specific impulse is not the only important parameter when it comes to discussing rockets. For example, a launch vehicle engine will have I_{sp} typically around 200–500 sec. An ion engine for a deep space mission will have much higher I_{sp} values, upwards of 3,000 sec. Why are they so different, and why would we use one over the other?

TABLE 3.1

Specific Impulse for Various Rockets

Rocket	I_{sp} in s
SSME	363
RS-68	365
SRB	269
NSTAR	3,100
NERVA	800

Launch vehicles are typically employed to lift very heavy payloads into orbit or on an interplanetary trajectory as quickly as possible. The key component needed for such missions is thrust—as much thrust as usually can be applied within engineering capabilities. Applying this much thrust in such a short time requires a large amount of propellant mass. This is why launch vehicle rockets are mostly propellant and oxidizer with small areas on top for payloads. An example of this is the Delta IV heavy launch vehicle that employs three common booster core liquid hydrogen fuel– and liquid oxygen oxidizer–driven Boeing/Rocketdyne RS-68 engines, as discussed in Chapter 1. Each of the three engines can supply just under 3.4 MN (meganewtons) of thrust with an I_{sp} of 362 sec at sea level.

On the other hand, interplanetary missions (after launch) typically need to apply thrust continuously for a long period of time. These are usually the small payloads that are atop the launch vehicles; thus, they have very little mass budget for fuel. This means that they cannot apply large thrusts for long periods of time, or they will run out of fuel. Hence, a more propellant-efficient engine, such as an ion thruster, is needed. The ion thrusters use small amounts of propellant mass at a time, but accelerate that mass to very high equivalent velocities. Figure 3.2 illustrates images of NASA's Deep Space Probe 1 that used an ion engine that only generated 0.09 N of thrust but had an I_{sp} of over 3,100 sec.

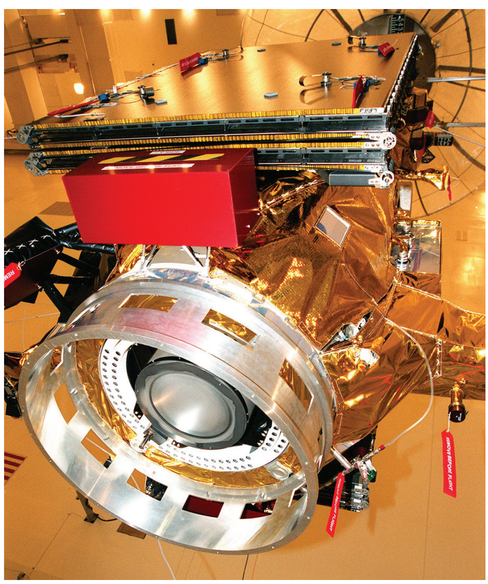


FIGURE 3.2
The Deep Space 1 probe before it was launched in 1998. This view gives a good vantage point of the ion engine that fired successfully for 678 days. (Courtesy of NASA.)

3.2.1 Example 3.1: I_{sp} of the Space Shuttle Main Engines

The three Space Shuttle Main Engines (SSMEs) of the Orbiter each provide about 1.8 MN of thrust with an I_{sp} of 363 sec at sea level. What is the mass flow rate of an SSME?

The first step is to determine the equivalent velocity, C, of the engine. From Equation 3.15, we see that $C = g\, I_{sp}$ = 3,557.4 m/sec. Then, from Equation 3.13, we can solve for the mass flow rate, $\dot{m}$, which is

$$\dot{m} = \frac{F_{thrust}}{C} = \frac{1.8\times 10^{6}}{3{,}557.4} = 505.99 \text{ kg/sec}.$$

In other words, one SSME uses about a half ton of propellant each second. Do not forget that there are three of them.

3.3 Weight Flow Rate

What we have learned from Sections 3.1 and 3.2 is that the two parameters, thrust and specific impulse, are key in defining rocket engines for particular applications. Knowing these parameters tells us a lot on how to size the rocket engine for a particular mission. There is another parameter that we should also mention here. Let us reconsider Equation 3.15 by rewriting it as

$$I_{sp} = \frac{I}{\Delta m_{propellant}\, g} = \frac{C}{g} = \frac{F_{thrust}\Delta t}{\Delta m_{propellant}\, g} = \frac{F_{thrust}}{\dfrac{\Delta m_{propellant}}{\Delta t}\, g}. \tag{3.16}$$

Realizing that the change in propellant mass over the period of time thrusting occurs multiplied by the gravitational acceleration of one standard Earth gravity is the parameter known as *weight flow rate*, $\dot{W}$, then it becomes

$$I_{sp} = \frac{F_{thrust}}{\dot{W}}. \tag{3.17}$$

Another way to look at Equation 3.17 is that the *weight flow rate* is the ratio of rocket thrust to specific impulse,

$$\dot{W} = \frac{F_{thrust}}{I_{sp}}. \tag{3.18}$$

Two different rocket engines will likely have different values for I_{sp}, thrust, and weight flow rate, but it is these three parameters that enable rocket engineers to begin the initial sizing of the engine. The thermodynamic properties of the engine (combustion, gas dynamics, and nozzle design) will determine the I_{sp}. The overall weight of the spacecraft and rocket

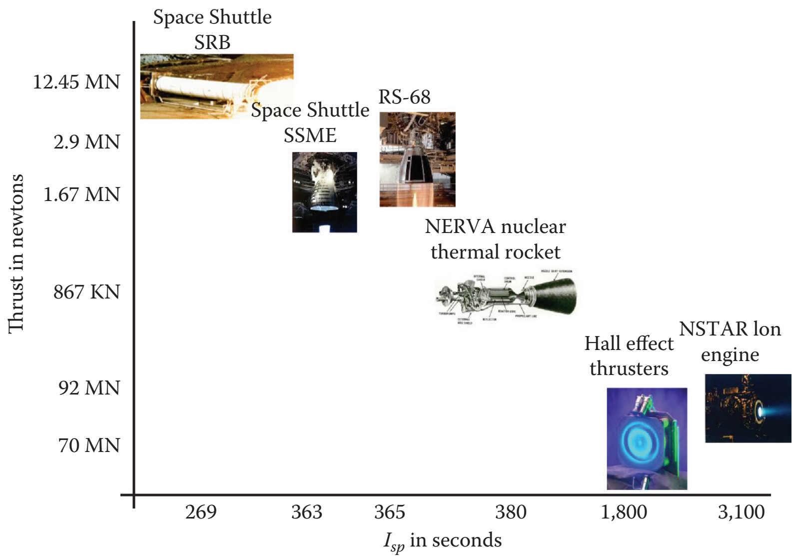


FIGURE 3.3
Various rocket engines shown as thrust versus I_{sp}. (Courtesy of NASA.)

combination will drive the thrust requirement. And, having these two values allows us, through Equation 3.18, to determine the weight flow rate of propellant mass needed, which will lead to design knowledge about how big the nozzle throat of the rocket must be. Figure 3.3 shows a graphic of the thrust versus the specific impulse for a few engines. From Equation 3.18, we see that the weight flow rate is the slope of this graph. This is a useful design tool telling us that, for a given thrust and specific impulse, we will need to construct an engine that can handle the flow rates shown on the graph and determined by Equation 3.18.

3.4 Tsiolkovsky's Rocket Equation

As discussed in Chapter 1, the father of rocket science was the Russian mathematics teacher Konstantin Eduardovich Tsiolkovsky. In 1903, he published a paper in which he derived the so-called rocket equation. It is this derivation that shows us the basis for rocket propulsion. He even went so far as to describe using multistaged rockets, which we will discuss in Section 3.5.

It turns out that the derivation for the rocket equation is not an extremely complicated one. Most certainly, the tools were available to make the discovery as far back as Newton's time. In fact, that is where the derivation starts, with Newton's Second Law of Motion.

Consider the rocket system shown in Figure 3.4 where there is a rocket vehicle and propellant with total mass, M, an equivalent exhaust velocity, C, and a mass flow rate, $\dot{m}$. The thrust then follows Newton's Second Law and is written as

$$F_{Thrust} = \dot{m}C = \frac{dM}{dt}C. \tag{3.19}$$

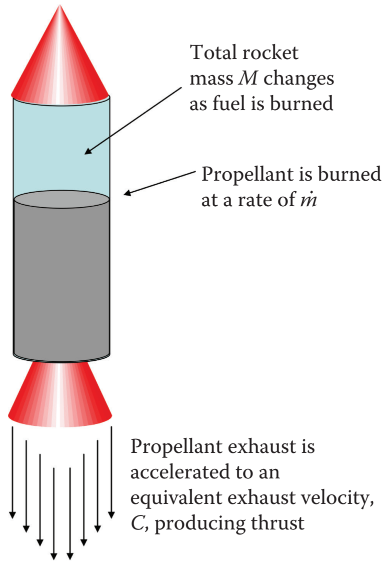


FIGURE 3.4
As the rocket engine fires, the total mass of the rocket decreases, propellant is burned, and the exhaust is accelerated.

Also applying Newton's Second Law in that the total force of the rocket can be defined by the total mass of the rocket vehicle and propellant times the total acceleration, then

$$F = Ma = M\frac{dv}{dt} \tag{3.20}$$

where v is the velocity of the rocket. Setting the right-hand sides of Equations 3.19 and 3.20 equal to each other and realizing that they are forces in the opposite direction (Newton's Third Law) give

$$M\frac{dv}{dt} = -\frac{dM}{dt}C. \tag{3.21}$$

Simplifying Equations 3.19 and 3.20 leads to the following differential equation:

$$dv = -C\frac{dM}{M}. \tag{3.22}$$

Assuming that the rocket starts out with a velocity of v_o and ends with velocity v_f and the initial mass is M_o and the final mass is M_f, then we can solve Equation 3.22 by integrating it through these limits:

$$\int_{v_o}^{v_f} dv = -C\int_{M_o}^{M_f}\frac{dM}{M}. \tag{3.23}$$

Integrating and applying the limits result in the rocket equation

$$v_f - v_o = -C(\ln(M_f) - \ln(M_o)) = C\ln\left(\frac{M_o}{M_f}\right). \tag{3.24}$$

Realizing that the left-hand side of Equation 3.24 is the change in velocity, Δv, then it can be rewritten as

$$\Delta v = C\ln\left(\frac{M_o}{M_f}\right). \tag{3.25}$$

In many textbooks or when talking to rocket scientists and engineers, it is likely to hear the ratio of the initial mass of the rocket to the final mass in the argument of the natural logarithm of Equation 3.25 as the *mass ratio*, or sometimes the *propellant mass ratio,* and sometimes as the *mass fraction*. In some cases, the mass ratio is given as *MR,* making Equation 3.25 look like

$$\Delta v = C\ln(MR). \tag{3.26}$$

For the purposes of clarity in this book, we will stick to Equation 3.25.

Now, simplifying Equation 3.25 by dividing both sides by C and raising each side to the exponent yield

$$M_o = M_f e^{\frac{\Delta v}{C}}. \tag{3.27}$$

From Equation 3.15, we see that $C = g{\cdot}I_{sp}$. Substituting this value into Equation 3.27 gives

$$M_o = M_f e^{\frac{\Delta v}{gI_{sp}}}. \tag{3.28}$$

This is yet another formulation of the rocket equation and is expressed in useful rocket science and engineering terms. As we saw in Chapter 2, the Δ-v is an all-important parameter, and we are realizing within this chapter the importance of the other terms in Equation 3.28.

Now, let's reconsider this derivation taking into account the gravitational force on a launching rocket. We will start by writing the total force on the rocket as

$$F = Ma = M\frac{dv}{dt} = -F_{thrust} - Mg. \tag{3.29}$$

Again, force due to thrust is the right-hand side of Equation 3.19; thus, Equation 3.29 becomes

$$M\frac{dv}{dt} = -C\frac{dM}{dt} - Mg. \tag{3.30}$$

Simplifying and integrating

$$\int_{v_o}^{v_f} dv = -C\int_{M_o}^{M_f} \frac{dM}{M} - g\int_{t_o}^{t_f} dt\,, \tag{3.31}$$

$$(v_f - v_o) = C\ln\left(\frac{M_o}{M_f}\right) - g(t_f - t_o). \tag{3.32}$$

Again, realizing the left-hand argument is Δv and the time argument on the right-hand side is the Δt or time of rocket burn, t_b, then Equation 3.32 becomes

$$\Delta v = C\ln\left(\frac{M_o}{M_f}\right) - gt_b. \tag{3.33}$$

Finally, moving the time of burn segment to the right-hand side, taking the exponent of each side, and substituting in for C result in the adjusted-for gravity rocket equation

$$M_o = M_f e^{\frac{\Delta v + gt_b}{gI_{sp}}}. \tag{3.34}$$

The initial and final mass notation in Equations 3.28 and 3.34 might become confusing, especially because the mass diminishes with time. It is common to see these masses described as the "full-of-fuel" or just "full" mass of the rocket and the "empty" mass of the rocket. Thus, the equations are sometimes written as

$$M_{full} = M_{empty} e^{\frac{\Delta v}{gI_{sp}}} \tag{3.35}$$

and

$$M_{full} = M_{empty} e^{\frac{\Delta v + gt_b}{gI_{sp}}}. \tag{3.36}$$

Figures 3.5 and 3.6 show graphs of the mass ratio as a function of the Δ-v and the specific impulse, respectively. It is clear that the Δ-v goes up as the *MR* goes up, but the I_{sp} goes down as *MR* goes down.

Equation 3.36 is a fairly useful tool in that it can be used to design parameters around a mission. Assuming that we use the procedures found in Chapter 2 to develop the Δv needed for a space mission for a particular craft, then we can use Equation 3.36 to determine the *MR* required to do this, at what I_{sp}, and for how long of a burn time. Taking the natural log of both sides gives us the burn time equation

$$t_b = \ln\left(\frac{M_{full}}{M_{empty}}\right) I_{sp} - \frac{\Delta v}{g}. \tag{3.37}$$

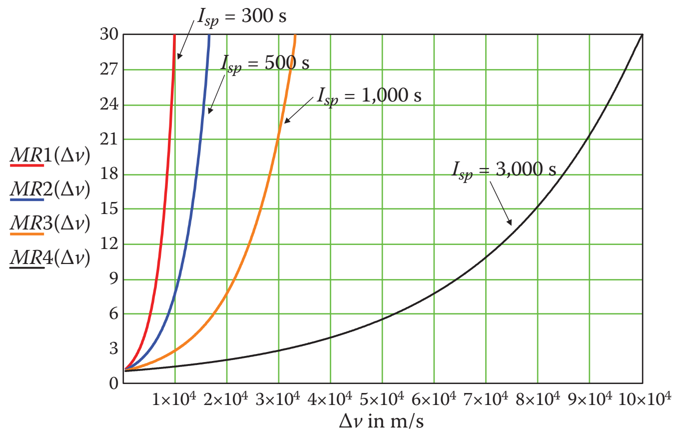


FIGURE 3.5
Mass ratio, *MR*, versus Δv for various I_{sp} values.

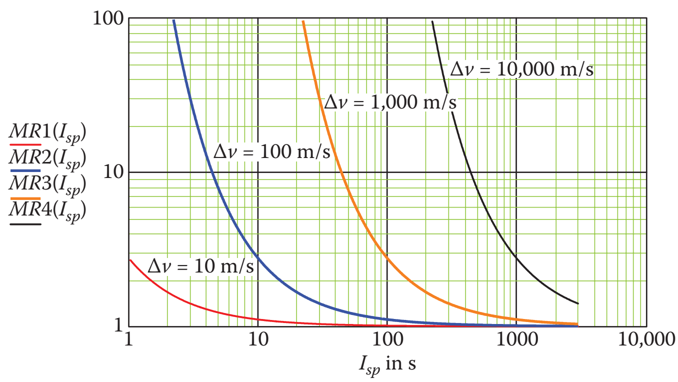


FIGURE 3.6
Mass Ratio, *MR*, versus I_{sp} for various Δv values.

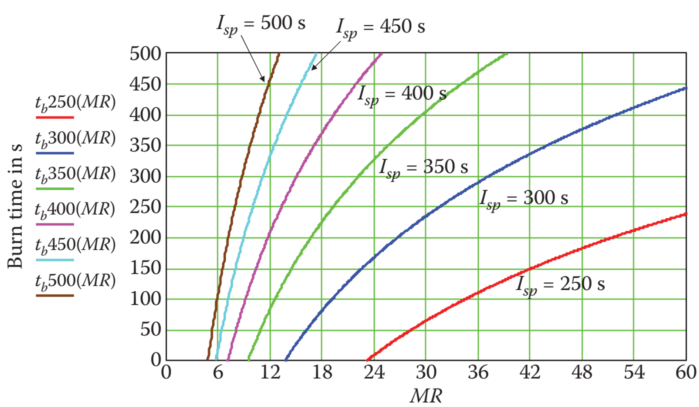


FIGURE 3.7
Burn time in seconds versus the mass ratio.

Figure 3.7 shows a graph of the burn time as a function of the mass ratio for a required Δ-v of 7,700 m/sec and for several values of I_{sp}. Clearly, as the specific impulse increases, the burn time and the mass ratio reduce.

3.5 Staging

We now see that as a launch vehicle rocket lifts itself up, it discards a lot of fuel and oxidizer mass, as can be seen by the *MR* of the system. From some of the figures of launch vehicle rockets shown in Chapter 1, Section 1.2, it is quite clear that these systems are very large in size, and, even though they are made of lightweight space-age materials, they are still quite heavy. Therefore, it makes sense that throwing away the empty fuel and oxidizer tanks, and the structure supporting them, and the engines that used them would enable the use of a secondary rocket system that has a smaller initial or full mass and, therefore, change the performance of the rocket, as described by Equations 3.35 through 3.37.

This concept is called the *staging* of the launch vehicle. Figure 3.8 illustrates the two basic types of staging that are typical with rocket systems. The first type of staging is *serial staging*, and the second is *parallel staging*. Serial rocket staging is a system that stacks stages one atop the other, whereas parallel staging is a system that straps boosters beside each other.

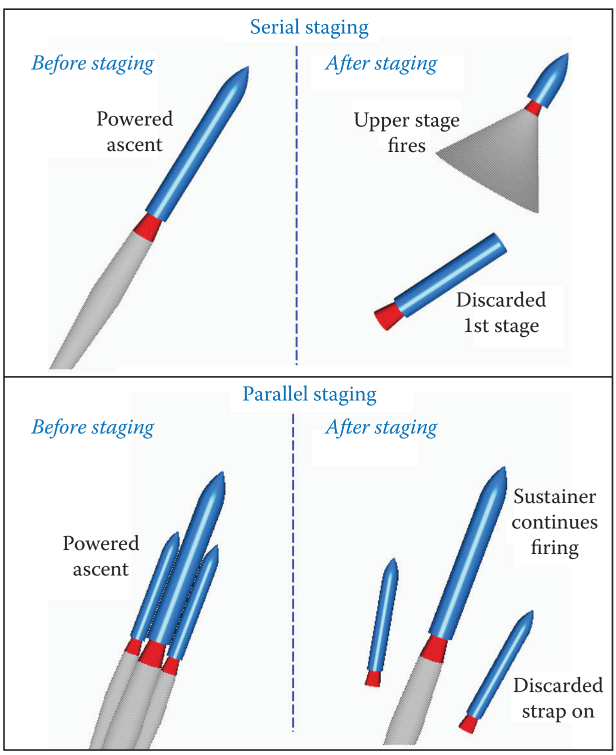


FIGURE 3.8
Two standard types of booster staging. (Courtesy of NASA.)

The Space Shuttle is an example of parallel staging, whereas the Saturn V of the Apollo program was serial-staged rockets.

There is another type of staging, which we will call *hybrid staging*. Hybrid staging is a combination of the two basic types. An example of a hybrid-staged rocket system is the Delta IV family of launch vehicles, as discussed in Chapter 1 and shown in detail in Figure 3.9. This vehicle has parallel strap-on boosters (solid Gem 60s for the normal and medium class and liquid RS-68s for the heavy class), as well as an upper-stage liquid booster (RL-10B-2 engine).

Before we can develop a model for staging rockets, we need to discuss the impact staging has on some of the rocket's parameters. Specifically, we should take a closer look at describing the rocket's mass. The basic components of the rocket vehicle are the structure, which houses the engines and tanks for each stage; the fuel and oxidizer, which we will just call propellant for now; and the payload, which consists of instruments and, sometimes, astronauts. Each of these contribute to the rocket system's mass, and, thus, the total full mass of the rocket system can be written as

$$M_{total} = M_{structure} + M_{propellant} + M_{payload}. \tag{3.38}$$

For a two-staged rocket, Equation 3.38 becomes

$$M_{total} = M_{stage1structure} + M_{stage1propellant} + M_{stage2structure} + M_{stage2propellant} + M_{payload}. \tag{3.39}$$

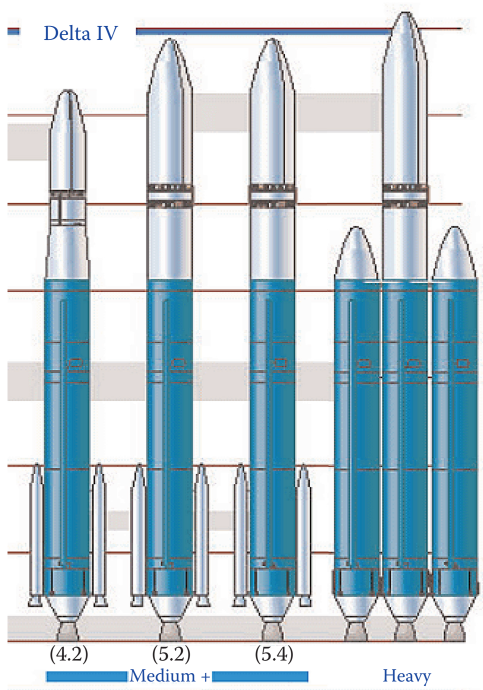


FIGURE 3.9
The Delta IV Medium + to Heavy rockets use hybrid staging including parallel strap-on boosters and serial upper-stage boosters. (Courtesy of NASA.)

For convenience from this point onward, we will use subscripts of *s* for structure and *p* for propellant with a number to denote the stage it represents. Hence, Equation 3.39 is more simply written as

$$M_{total} = M_{s1} + M_{p1} + M_{s2} + M_{p2} + M_{payload}. \tag{3.40}$$

For a multistaged rocket, Equation 3.40 can be written as

$$M_{total} = M_{s1} + M_{p1} + M_{s2} + M_{p2} + \ldots + M_{sN} + M_{pN} + M_{payload}. \tag{3.41}$$

Now, consider how a system like the one described by Equation 3.41 impacts the calculation of the mass ratio and, therefore, the rocket equation. The approach to account for this is to develop a mass ratio calculation for each stage. The *MR* for the initial state of the multistaged rocket for the first-stage burn is

$$\begin{aligned} MR_1 &= \frac{M_{total}}{M_{total} - M_{p1}} \\ &= \frac{M_{s1} + M_{p1} + M_{s2} + M_{p2} + \ldots + M_{sN} + M_{pN} + M_{payload}}{M_{s1} + M_{s2} + M_{p2} + \ldots + M_{sN} + M_{pN} + M_{payload}}. \end{aligned} \tag{3.42}$$

After the first stage is expended and jettisoned, the rocket then becomes smaller, and the new mass ratio becomes

$$MR_2 = \frac{M_{s2} + M_{p2} + \ldots + M_{sN} + M_{pN} + M_{payload}}{M_{s2} + \ldots + M_{sN} + M_{pN} + M_{payload}}. \tag{3.43}$$

Likewise, the mass ratio for the *N*th stage is

$$MR_N = \frac{M_{sN} + M_{pN} + M_{payload}}{M_{sN} + M_{payload}}. \tag{3.44}$$

Using the above mass ratio formulas for the multiple stages, along with Equation 3.36, the performance of the rocket at each stage can be determined. This approach allows rocket scientists and engineers to determine how many stages are needed to achieve the final desired Δv for the mission payload. We should note that there is an optimum mass ratio for each of the stages. It is outside the scope of this text to prove this here, but it turns out that the optimum for the rocket system is to design it such that the mass ratios for each stage are equal to one another if each stage uses the same type of propellant or engines with similar performance parameters. Using this knowledge, rocket engineers can determine what mass for structure and propellant can be allowed for each stage.

3.5.1 Example 3.2: Two-Stage Rocket

Given a two-stage rocket with each stage of equal mass and same engines, find the equation that describes the total Δ-v achieved by the rocket after second-stage burnout.

Start with Equation 3.35 and solve for the total Δ-*v*, which is

$$\Delta v_{total} = gI_{sp} \ln\left(\frac{M_{full}}{M_{empty}}\right). \tag{3.45}$$

The mass ratio for stage 1 is

$$MR_1 = \frac{M_{total}}{M_{total} - M_{p1}} = \frac{M_{s1} + M_{p1} + M_{s2} + M_{p2} + M_{payload}}{M_{s1} + M_{s2} + M_{p2} + M_{payload}}. \tag{3.46}$$

So, the Δv for the first stage is

$$\Delta v_1 = gI_{sp} \ln\left(\frac{M_{s1} + M_{p1} + M_{s2} + M_{p2} + M_{payload}}{M_{s1} + M_{s2} + M_{p2} + M_{payload}}\right). \tag{3.47}$$

The mass ratio for stage 2 is

$$MR_2 = \frac{M_{s2} + M_{p2} + M_{payload}}{M_{s2} + M_{payload}}. \tag{3.48}$$

The Δv for stage 2 is then

$$\Delta v_2 = gI_{sp} \ln\left(\frac{M_{s2} + M_{p2} + M_{payload}}{M_{s2} + M_{payload}}\right). \tag{3.49}$$

The total Δv for the two-stage rocket, therefore, is written as

$$\begin{aligned} \Delta v_{total} &= \Delta v_1 + \Delta v_2 \\ &= gI_{sp} ln\left(\frac{M_{s1} + M_{p1} + M_{s2} + M_{p2} + M_{payload}}{M_{s1} + M_{s2} + M_{p2} + M_{payload}}\right) \\ &\quad + gI_{sp} ln\left(\frac{M_{s2} + M_{p2} + M_{payload}}{M_{s2} + M_{payload}}\right). \end{aligned} \tag{3.50}$$

3.6 Rocket Dynamics, Guidance, and Control

To this point in our development of rocket science and engineering understanding, we have discussed details of the anatomy of a rocket (see Section 1.3) and, in this chapter, we

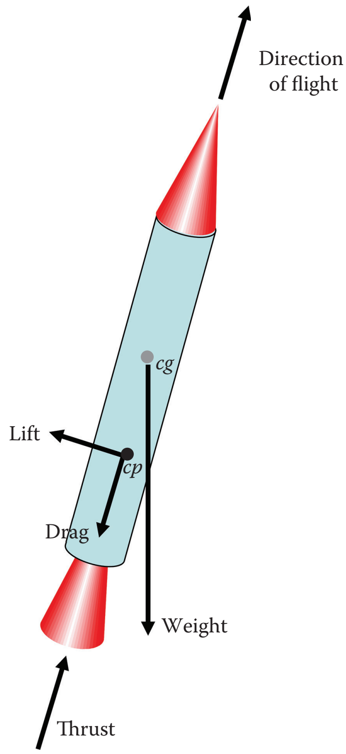


FIGURE 3.10
The forces acting on a rocket.

have discussed the basic components of propulsion and staging. At the most basic level, a rocket can be described as four major subsystems:

1. Payload system that carries instruments and astronauts
2. Propulsion system that contains the engines, pumps, and propellants
3. Structural system that houses all components and is also called the "frame"
4. Guidance and control system

All of the components interact with each other and are integral subsystems to the overall rocket vehicle system.

The guidance and control system is impacted by the rocket design because tanks, structure, propellant mass, engine mass, and the location within or on the rocket of all these units change the rocket vehicle system's center of gravity. Figure 3.10 shows a graphic of a rocket with all the dynamic forces impinging on it. These forces include (1) aerodynamic lift, (2) aerodynamic drag, (3) the weight, and (4) the rocket engine thrust. These are the basic four forces on all rockets.

3.6.1 Aerodynamic Forces

The lift and drag on the rocket vehicle are only applicable if the rocket is in an atmosphere. Note that, sometimes, the atmosphere can be perceived to be a vacuum when it really isn't.

The International Space Station (ISS) is at an orbit of over 400 km and is in a vacuum that is deadly to astronauts without spacesuits. However, the drag on the ISS is appreciable enough even at that orbital altitude that it often has to be reboosted in order to maintain its orbit. Although the number of molecules of air (atomic oxygen at that altitude mostly) is extremely small, the surface area of the ISS is large enough, and the relative velocity between the atmosphere and the spacecraft is fast enough that a significant drag force is imparted to the space station.

Of course, the aerodynamic forces are particular to the size and shape of the vehicle, as well as the angle of attack of that shape with respect to the direction of motion. The lift force, L, is found by

$$L = C_L A \frac{\rho v^2}{2}. \tag{3.51}$$

In Equation 3.51, C_L is the lift coefficient, A is the effective area of the surface impacting the air, ρ is the atmospheric density at the altitude of flight, and v is velocity of the rocket.

Similar to the lift, the drag force is calculated as

$$D = C_D A \frac{\rho v^2}{2} \tag{3.52}$$

where C_D is the drag coefficient of the vehicle. We should note that the quantity $\rho v^2/2$ in both of the above equations is known as the dynamic pressure and is often referred to as Q. During a shuttle launch just before the announcement that the shuttle is "go at throttle up," it is usually announced that the shuttle has just passed through "max-Q." Max-Q is the point where the spacecraft has just pushed through the atmosphere at the highest dynamic pressure it will meet during the flight trajectory. Max-Q for the Space Shuttle is at an altitude of about 11 km and at a velocity of about 442 m/sec. Figure 3.11 shows a graph of the velocity of the airflow against the Space Shuttle versus the time of flight. It also shows the density of the atmosphere the Shuttle meets as it flies through its trajectory. Somewhere

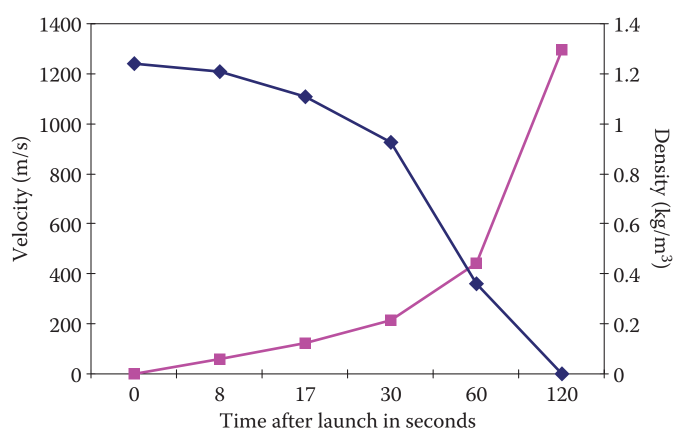


FIGURE 3.11
The airflow velocity and atmospheric density against the Space Shuttle versus time after launch.

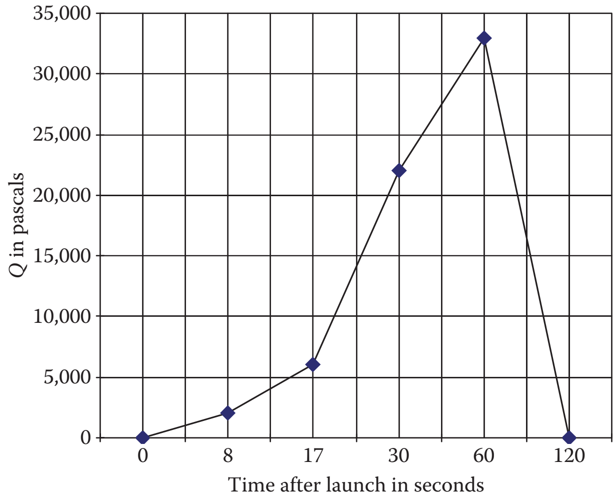


FIGURE 3.12
Dynamic pressure, *Q*, against the Space Shuttle versus time after launch.

around 60 sec, the vehicle goes through max-Q. Figure 3.12 shows the dynamic pressure on the Space Shuttle versus flight time. The peak pressure is around 35,000 Pa (pascals).

The dynamic pressure due to the aerodynamic forces on the Space Shuttle is the reason that we hear the "go at throttle up" call around 68 sec or so. This is because the rocket system is not designed to plow through the atmosphere at maximum velocity prior to this because the max-Q would be too large. It is not necessarily too large for the frame to handle, but throttling more fuel through the engines would be an inefficient use of fuel by pushing harder and harder against the pressure for only slight changes in Δ-v with huge fuel penalties. Therefore, waiting just a few seconds more at a lower mass flow rate until the dynamic pressure starts dropping off rapidly is a more fuel-efficient flight profile. After passing through the max-Q, the rocket then goes full throttle.

3.6.2 Example 3.3: Drag Force on the Space Shuttle

Given the information in this section, the affective area diameter of the shuttle is about 20 m, and that the drag coefficient, C_D, for the shuttle is about 0.2, calculate the drag force on the vehicle at max-Q.

Using Equation 3.52 and substituting in the numerical values, we see

$$D = C_D A \frac{\rho v^2}{2} = 0.2\pi(10\text{ m})^2(3{,}500\text{ Pa}) = 2.199 \times 10^6\text{ N} \approx 2.2\text{ MN}. \qquad (3.53)$$

3.6.3 Rocket Stability and the Restoring Force

Now that we understand weight, thrust, lift, and drag, we can discuss how they impact the flight dynamics of a rocket in a little more detail. Refer back to Figure 3.10 and the forces on a rocket. The weight is directed downward from the center of gravity of the rocket, as shown. The thrust is parallel with the direction of travel of the rocket. The drag is parallel, but in the opposite direction of the thrust. And, the lift forces are typically perpendicular to the flight path unless the rocket structure is of an unorthodox design. Another crucial aspect of the flight dynamics is that the four forces incident on the rocket are changing.

So far, we have treated these forces as scalar quantities, but, in actuality, they are vectors that are changing in direction and magnitude throughout the flight path. As standard practice for understanding forces acting on a body, we sum up the four force vectors to determine the total force, and it is written as

$$\sum F = F_{thrust} + mg + L + D. \tag{3.54}$$

Note that, in this case, the gravitational parameter **g** was used as a vector. The direction of **g** would be radially inward from the center of gravity, *cg*, (the gray dot in Figure 3.10) of the rocket to the center of the Earth and can be in whatever coordinate system deemed appropriate. Another important parameter mentioned in Figure 3.10 that is key in Equation 3.54, but is not shown, is the angle of displacement, α. The angle of displacement is the angle between the flight path line and the vertical. This is sometimes referred to as just the *displacement angle*.

Again, refer back to Figure 3.10. An important parameter shown as the black dot is the center of pressure, *cp*. The two parameters *cg* and *cp* are very important to stable rocket flight and, therefore, deserve some discussion here. We'll start with the *cg*.

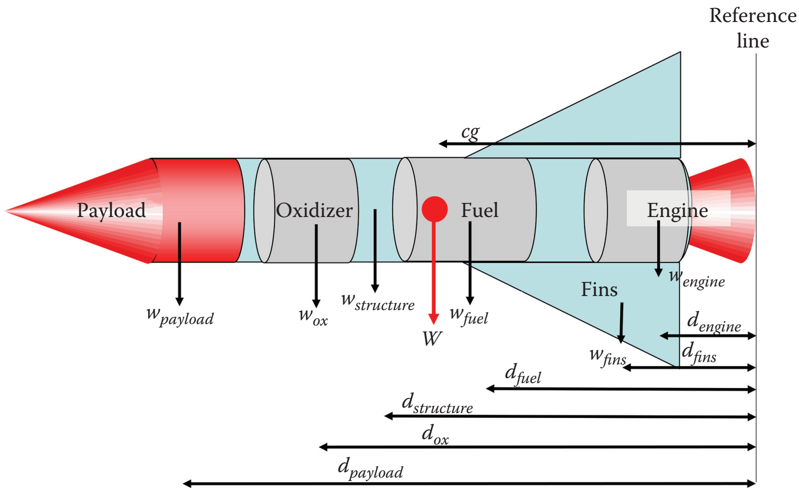
The center of gravity is sometimes referred to as the "center of mass" of a system when talking about two bodies. But, in rocketry and in reference to a rocket vehicle, the *cg* is the term used the most. The *cg* of a solid object (assuming uniform, or near-uniform density) is defined as average position of all particles making up that object weighted by the masses of the particles and is typically the geometric centroid of the object and is a distance measured in meters. Mathematically speaking, the *cg* is

$$cg = \frac{\sum_i w_i d_i}{\sum_i w_i} = \frac{\sum_i m_i g d_i}{\sum_i m_i g} = \frac{\sum_i m_i d_i}{\sum_i m_i}. \tag{3.55}$$

Note that Equation 3.55 shows that the *cg* is the same, whether it is measured as the center of weight or the center of mass, and, hence, the usage center of gravity makes the most sense. We should also note that, without the aid of detailed engineering drawings of all components with known densities, it is easier just to weigh the components as opposed to calculating their masses. In some cases, it is not.

Figure 3.13 shows a typical liquid-fueled rocket lying horizontally with the *cg* marked with a large dot. The components (or particles, as from our definition in Equation 3.55) making up the rocket are the payload (in the nose cone), the oxidizer, fuel, structure, engine, and the guidance fins. The total weight, *W*, of the rocket times the location of the *cg* measured from a reference line at the rear of the rocket is equal to the sum of the weight of each of the components listed in the previous sentence multiplied by their respective distances from the reference line. Mathematically speaking, the *cg* times the weight of the rocket is found from the following equation:

$$\begin{aligned} cgW = {} & w_{payload} d_{payload} + w_{ox} d_{ox} + w_{fuel} d_{fuel} \\ & + w_{structure} d_{structure} + w_{engine} d_{engine} + w_{fins} d_{fins}. \end{aligned} \tag{3.56}$$



$$cg = \frac{w_{payload}\,d_{payload} + w_{ox}\,d_{ox} + w_{fuel}\,d_{fuel} + w_{structure}\,d_{structure} + w_{engine}\,d_{engine} + w_{fins}\,d_{fins}}{W}$$

FIGURE 3.13
Center of gravity of a rocket is calculated as shown above.

Or, from Equation 3.55, we see that

$$\frac{w_{payload}d_{payload} + w_{ox}d_{ox} + w_{fuel}d_{fuel} + w_{structure}d_{structure} + w_{engine}d_{engine} + w_{fins}d_{fins}}{W}. \tag{3.57}$$

Also,

$$cg = \frac{m_{payload}d_{payload} + m_{ox}d_{ox} + m_{fuel}d_{fuel} + m_{structure}d_{structure} + m_{engine}d_{engine} + m_{fins}d_{fins}}{M_{total}}. \tag{3.58}$$

Again, both Equations 3.57 and 3.58 are useful in that sometimes, it is easier to weigh the components of the vehicle, and, sometimes, it is not. Think of the Space Shuttle that has over 2 million moving parts. It would take quite some time to weigh all precisely with propellants and lubricants flowing through them. It would also be difficult to raise a vehicle like the Shuttle on a pendulum or balance (like a teeter-totter) to determine the *cg*. Various methods for calculating the *cg* for such vehicles are used that include some measurements, as well as detailed engineering solid modeling. Yet, another tool needed for modern rocket experts is that of three-dimensional solid modeling on high-performance computers.

Let us now talk about the center of pressure. The *cp* is the location on a solid body where the sum of all the forces from pressure fields (in our case, mostly aerodynamic pressure fields) act while creating no moment about that location. In other words, the *cp* of a solid object with a total surface area, *A*, is defined as the average position of all particles having an exterior surface areal component making up that object weighted by the area of the

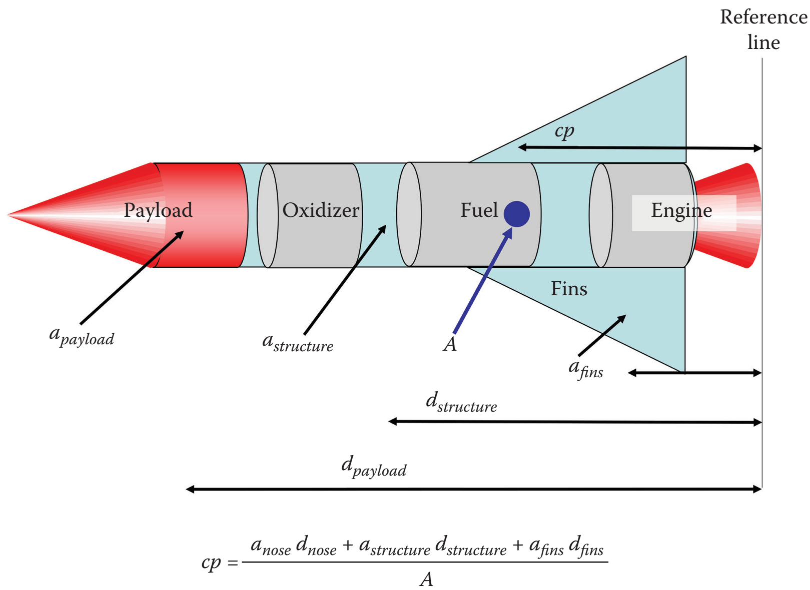


FIGURE 3.14
Center of pressure of a rocket is calculated as shown above.

individual particles and is a distance measured in meters. The *cp* is shown as the large dot in Figure 3.14. Mathematically speaking, the *cp* is

$$cp = \frac{\sum_i a_i d_i}{\sum_i a_i} = \frac{\sum_i a_i d_i}{A}. \quad (3.59)$$

Like Equations 3.57 and 3.58, it can be mathematically calculated via the following equation:

$$cp = \frac{a_{nose}d_{nose} + a_{structure}d_{structure} + a_{fins}d_{fins}}{A}. \quad (3.60)$$

One other note here is that, in some reference, the *cp* is measured from the nose of the vehicle as opposed to the same reference line as the *cg*. In this text, we will always use the same reference line (the rear- or bottommost point on the rocket) as the reference line for both parameters.

Now that we have defined *cg* and *cp*, we can continue with our development of rocket flight stability. Figure 3.15 shows three flight conditions of a rocket in flight. These conditions are (1) *powered*, (2) *stable*, and (3) *coasting*.

In the real world, there are things that cannot always be accounted for, such as wind gusts, shear forces, pings from flying foam insulation debris, geese, lightning, and any myriad things that can perturb the rocket's flight path. These external forcing functions add what is sometimes referred to as "wobble" to the vehicle. This wobble is simply a perturbation to the normal attitude vector of the rocket. If such a perturbation occurs during *powered* flight (while the engines are thrusting), the rocket will be displaced by some

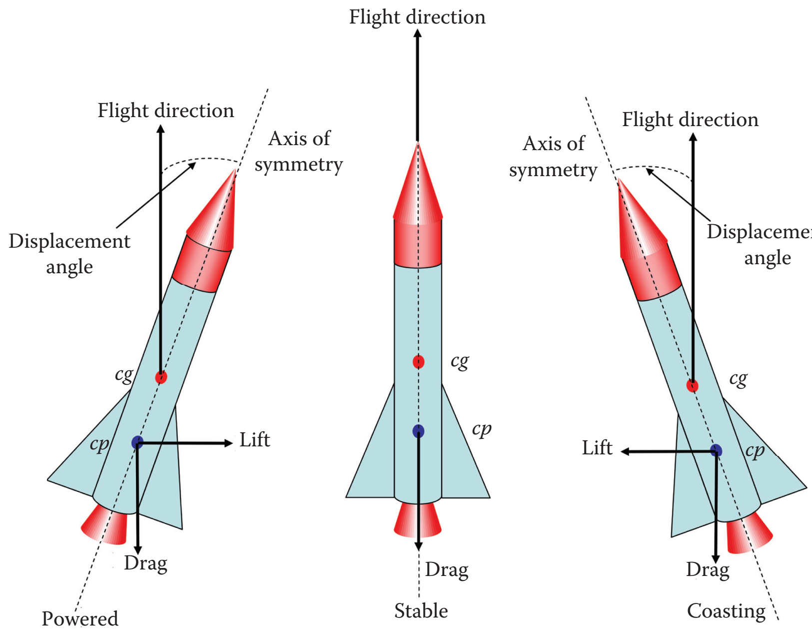


FIGURE 3.15
Three modes of rocket flight stability.

displacement angle, and the vehicle, while still traveling along the same flight path, will not be aligned in the optimum aerodynamic configuration. The lift and drag forces of the rocket will increase, creating a torque about the *cg*. The same happens if the rocket is in the *coasting* mode (the engines are off), except the lift and drag forces create a torque in the opposite direction.

A very important aspect of rocket design is that the *cp* MUST be below the *cg*, and here is why. If the *cp* is below the *cg*, then the torques created by the perturbing forces during both *powered* and *coasting* flight will be in directions that self-correct the perturbation and place the rocket back into a properly optimized aerodynamic attitude. This is known as the *restoring force*, and this force puts the rocket back into the *stable* flight mode.

If, for some reason, the *cp* is above the *cg*, then any perturbing forces will create torques that have a destabilizing effect and will send the rocket spinning madly out of control. Any true model rocket enthusiast has seen this at least once during his or her life. A nose cone that is too light or a motor that is too heavy will push the cg down the rocket body. If the *cg* is too close to the *cp* or below it, then the rocket will be unstable. A rocket designed like this is susceptible to very slight perturbations that can cause the rocket to fall head over heels, so to speak. At that point, the *cg* is above the *cp* and the model rocket, which is still perturbed and under a torque that is spinning the rocket body about the center of mass, will spin back over to the point with the top-heavy tail on top. Still, under the perturbations of spin, it will fall over once again placing the *cp* on top and so on until the rocket goes careening into oblivion or atop the neighbor's roof. Either case is not an optimum *stable* flight attitude.

In the ideal situation, the cg is above the cp, and, therefore, if the cg starts to cause the rocket to fall over, the dynamic pressure or *restoring force* against the lower body self-corrects the rocket's attitude. It is possible to make the nose too heavy or the counteracting lower body *restoring force* too low (fins too small, for example) such that the rocket cannot handle much in the way of perturbing forces. In this case, the rocket is likely to fall head over heels once again.

Some missile systems in use today still use simple solid motors with aerodynamic stability systems (fins). It is important in the design phase of these systems to heed the above section closely. However, most modern larger rocket systems do not rely on aerodynamic-driven control systems. In fact, most modern rockets adjust the attitude by tilting the thrust of the main engine. This technique is known as *thrust vector control* (TVC) and is what is happening when we see the nozzles of launch vehicles moving around at the base of the rocket. This is also the reason why most modern launch vehicles do not need fins.

3.6.4 Rocket Attitude Control Systems

Figure 3.16 shows the four basic types of systems for *attitude control* for rockets. They are moveable aerodynamic structures, such as fins, gimbaled thrust, vernier thruster rockets, and thrust vanes. Moveable aerodynamic structures, such as fins, function in the same way as ailerons and rudders on aircraft and require the rocket to be in an atmosphere that is dense enough for the control surface to be of any use. The other three methods, gimbaled thrust, vernier thruster rockets, and thrust vanes, are all variations on the same technique known as TVC.

TVC works by actually redirecting the thrust vector by swiveling the main engine nozzles, using smaller vernier rockets to thrust with a desired vector, or moving a vane in front of the main engine thrust to redirect it.

Gimbaled thrust is the technique employed on the Space Shuttle Orbiter. The SSMEs of the system, as well as the nozzles of the solid rocket boosters, are gimbaled and are used throughout the Earth-to-orbit phase to correct and optimize the flight trajectory. The Soyuz rocket uses combinations of aerodynamic fins and vernier thrusters. The Shuttle Orbiter actually has vernier rockets in the nose in order to adjust attitude while in orbit.

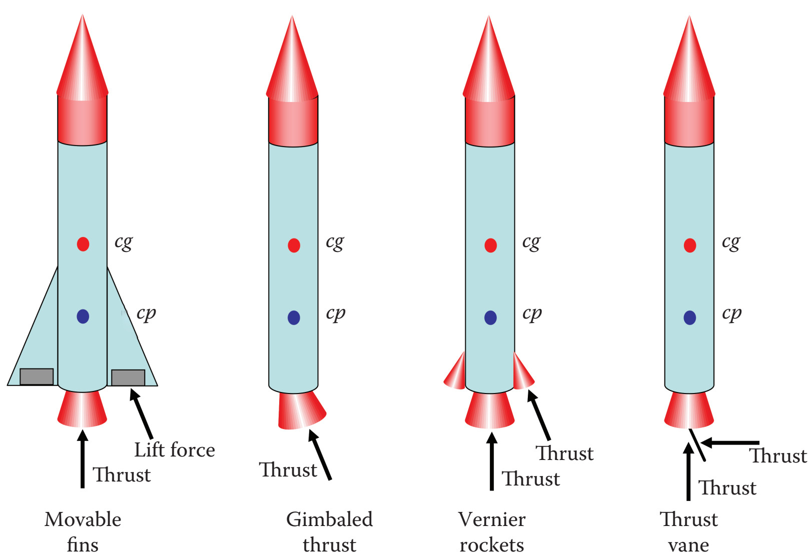


FIGURE 3.16
Four types of rocket attitude control.

The Mercury, Gemini, and Apollo spacecraft also employed vernier thrusters for in-space maneuvering and control. The German V2 rockets of the World War II era used thrust vanes for control. Thrust vanes are typically seen on modern fighter jets and have also been implemented on some experimental rocket vehicles.

3.6.5 Eight Degrees of Freedom

Figure 3.17 shows a rocket in flight along the x-axis. If the rocket spins about the x-axis, this is called a *roll* or a *rolling maneuver*. If the rocket rotates its nose up or down and, therefore, spins about the y-axis, this is called a *pitch* or a *pitching maneuver*. If the rocket rotates the nose toward the y-axis or the negative y-axis about the z-axis, this is called a *yaw* or *yaw maneuver*. The rocket can also have motion in the forward or backward directions along the x-axis due to drag or thrust. We have just described eight degrees of freedom of motion for the rocket. Rocket scientists and engineers will often refer to this as 8-DOF of motion. Sometimes, we will hear the term 6-DOF, but this is when drag and thrust are neglected. So, the 8-DOF dynamics can be described as follows:

Positive roll, θ_x
Negative roll, $-\theta_x$
Positive pitch, θ_y
Negative pitch, $-\theta_y$
Positive yaw, θ_z
Negative yaw, $-\theta_z$
Forward thrust, Δx
Drag or negative thrust, $-\Delta x$

In order for the rocket to correct for perturbations and disturbances along its flight path, attitude corrections within these 8-DOFs must be continually made. Accomplishing this controlled flight is quite an endeavor.

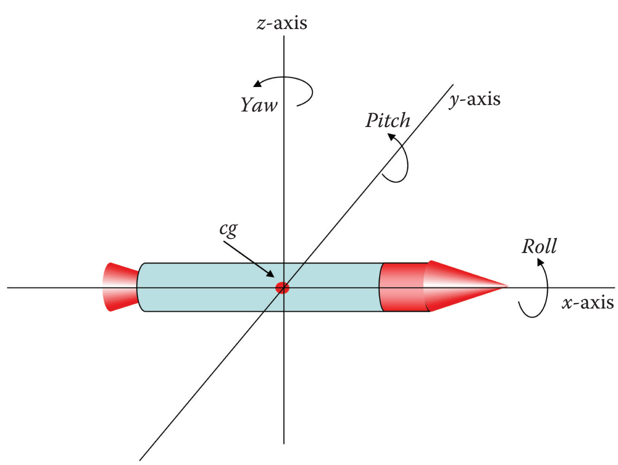


FIGURE 3.17
Rocket roll, pitch, and yaw maneuvers.

Figure 3.18 shows a typical attitude control system (ACS). Note that there is a different circuit for each axis. This is because the control for each axis can be separated from the others, simplifying the dynamics and complexity of the ACS itself. Do note, however, that there are inputs from each of the other two axes into the disturbance torque to account for any errors that an attitude correction for one axis might induce on another.

The initial state of the rocket is input into the circuit and is compared to the attitude from sensor data giving an attitude error value. Then, a control processor (computer) takes the difference data (error in attitude) and calculates if there are attitude correction thrusts that need to be generated. Also, at this point, the command unit can input other attitude maneuver commands into the system. Here is where a pilot's input from a joystick might come into play.

The commands for correction thrusts are then sent to the ACS thrusters, which fire for the calculated amount of time and with the appropriate force. Then, the control actions and external forcing disturbances move the rocket vehicle in space, as well as the bending, flexing, and sloshing components of the rocket. The rocket achieves a new state of attitude, which is then fed back into the initial-state input side of the circuit for the process to start all over. The controller circuit will determine through the same process if the external forces and the correction thrusts placed the rocket in the optimum attitude and will decide if too great or too little correction thrusts were made. This is a continuous process as long as the rocket is in flight.

Though it is beyond the scope of this text to develop the control algorithm in detail for a rocket vehicle, we can discuss it in general. The ACS control circuit shown in Figure 3.18 implements what is known as a *proportional, integral, derivative* (PID) controller. The open-loop PID control circuit is described mathematically as

$$\theta_{out}(t) = K_p\theta_{in}(t) + K_i\int_0^t \theta_{in}(t)\,dt + K_d\frac{d\theta_{in}(t)}{dt}. \tag{3.61}$$

Here, K_p is the proportional gain, K_i is the integral gain, and K_d is the derivative gain. The proportional gain is equivalent to a thrust in one direction, which would lead the rocket attitude to vibrate like an undamped mechanical spring. It is this proportional gain that determines the largest corrections. For example, if the attitude is incorrect by 7°, then the proportional controller sends a signal to the thrusters to correct for 7°. The derivative control component adds a damping thrust. It determines the rate at which a thrust should decrease or increase to damp out the disturbance. The integral controller looks at a longer period of time of the changing attitude and looks for longer-acting attitude errors. These

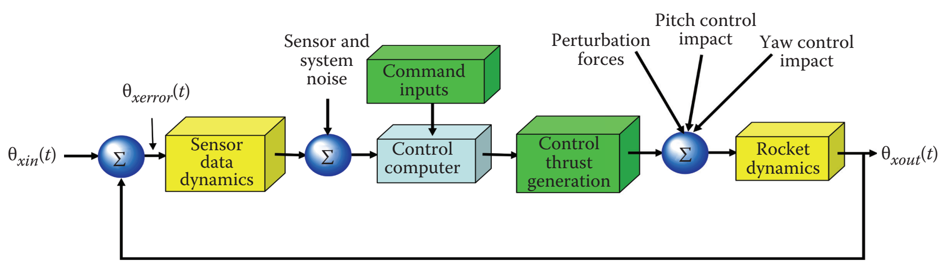


FIGURE 3.18
Attitude control circuit for a rocket system. Note that the diagram is only for the roll control of the vehicle.

errors are usually instantaneously small, but can cause large course errors over time. The integral component is needed as a check and balance to each of the other more abrupt control components to maintain flight path accuracy.

It should also be noted that, sometimes, in the literature, the PID controller is also called a *position*, *integral*, *derivative* controller. Due to the large mechanical forces involved with rocket systems, the PID controller is quite ideally suited for the task of rocket ACS.

It is beyond the scope of this text to develop in detail a complete closed feedback loop model for a rocket's ACS. However, a typical system follows the math of a damped oscillator similar to a damped mechanical spring. A general solution for such a system is

$$\theta_{out}(t) = \theta_{in} - \theta_{in} e^{-\zeta \varpi t} \cos(\omega t + \beta). \qquad (3.62)$$

Here, θ_{in} is the input state of the controller or the initial condition of the attitude of the rocket, ζ is the damping coefficient, ω is the frequency of the system, and β is the center frequency of the bandwidth of the system. With a much more detailed analysis of a rocket system, these coefficients can be solved in terms of the PID coefficients, and the equation might vary in complexity from system to system, but developing that solution is unnecessary here as we are merely trying to get an idea of how the controller for the ACS works. Complete textbooks and doctoral dissertations are written on the complex issues involving detailed ACS.

Figure 3.19 shows a graph of Equation 3.62 for several combinations of values for the control equation coefficients. Notice that it takes a certain amount of interplay between all

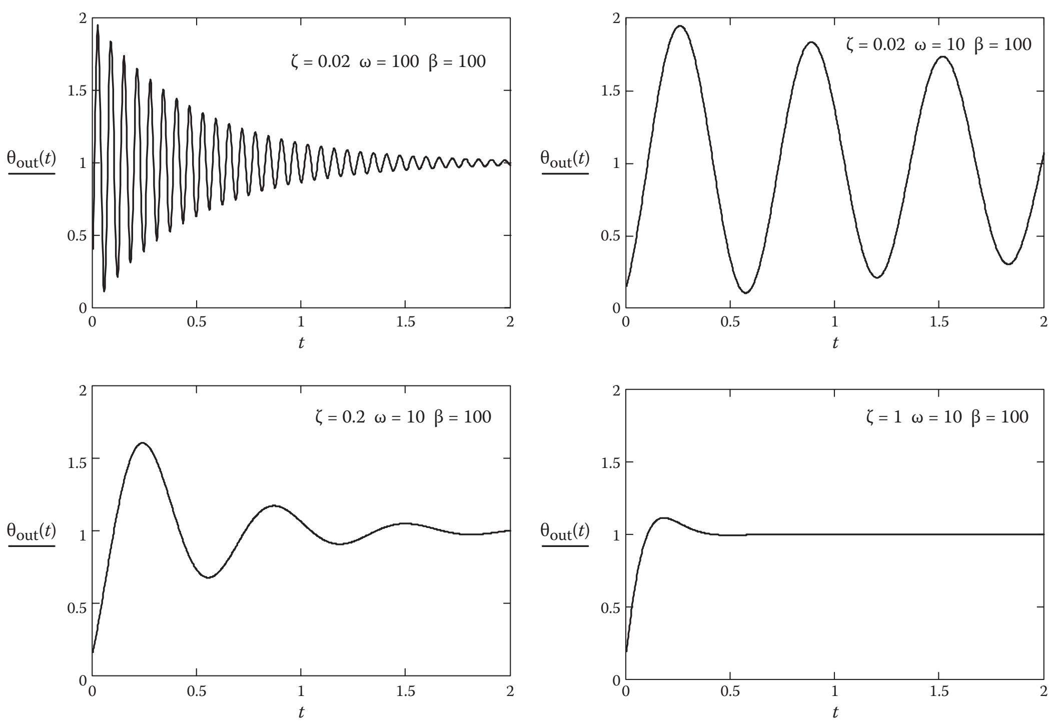


FIGURE 3.19
Attitude control circuit output as a function of time for various values of the control parameter coefficients. Ideal response would be immediately settling with no ringing.

three of them for the system to be stable where the oscillations are completely damped out. As an exercise for the readers, model Equation 3.62 in a math-modeling software package like Mathcad® and compare outputs due to different values for the three constants. When the right gains are used, the rocket's many degrees of attitude freedom can be controlled.

3.6.6 Inverted Pendulum

To this point, we have discussed the control of the rocket while in flight. What we have learned, but have not emphasized the design importance yet, is one of three rocket design rules that will be discussed in a later chapter. That rule is shown in Figure 3.15 and can be stated simply as follows:

Rocket design rule #1: For a rocket to have stable flight, the cg MUST be higher up the rocket than the cp.

A caveat to this rule is that, with a complicated-enough control algorithm, the rocket can be forced into proper flight attitude. However, it is not efficient to create such a complex control system when we could just design the rocket right in the first place, which is, actually, a statement of the second of the three rules. But we are getting ahead of ourselves.

The thing to take away from the discussion of stable flight is that a well-designed rocket will be top heavy. While in flight, this is not a problem because the *restoring force* keeps it stable. But what causes the *restoring force*?

Once the rocket is moving fast enough, a laminar flow of air is moving over the rocket body. It is this aerodynamic drag between the flow and the rocket's body and aerodynamic surfaces that causes the *restoring force* and hence the stable flight attitude. But what about before the rocket reaches a flight velocity that is within the stable flight regime? What about just as it comes off the pad or the launch rail?

Consider a well-designed rocket with the *cg* above the *cp*. In other words, it is top heavy. Just as soon as it comes off the pad, the rocket is unstable. Another situation when it would be unstable would be if it were performing a vertical landing like the SpaceX Grasshopper or the Blue Origin's New Shepard. While landing, these rockets are moving too slowly for the restoring force to be of any use.

A perfect demonstration for this situation is to take a broom and attempt to balance the broom upside down in the palm of your hand, as shown in Figure 3.20. In order to keep the broom end upright, we see that we must continuously adjust the position of our hand and therefore apply a force to "restore" the broom's upright attitude. It is a difficult task, but the human brain can master the task with practice. It actually turns out that a PID controller, as discussed in Section 3.6.4, has difficulty controlling this type of system. There are other controllers, such as the linear quadratic regulator and the state space controller, that do a better job but are outside the scope of this text.

We must realize that the upside-down broom is what is commonly referred to as the "inverted spherical pendulum" problem. Allowing the system to move in all directions is very complicated to model and is outside the scope of this text. However, the two-dimensional version, or simply, "the inverted pendulum on a cart" problem, is very apropos to study herein.

Consider the inverted pendulum on a cart shown in Figure 3.21. The pendulum is made of a massless stiff rod of length, l, with a mass, m, at the top. It is pinned to the cart at the bottom such that it can only swing in the x–z plane. The cart has a mass, M, and is also constrained to motion in the x–z plane. The rod, when off the vertical axis, will make an angle, θ. We will apply a force to the cart on either side as our controlling force, which would be analogous to the thrust required of retro rockets, or from a gimbaled main engine, or thrust vane, or aerodynamic surface.

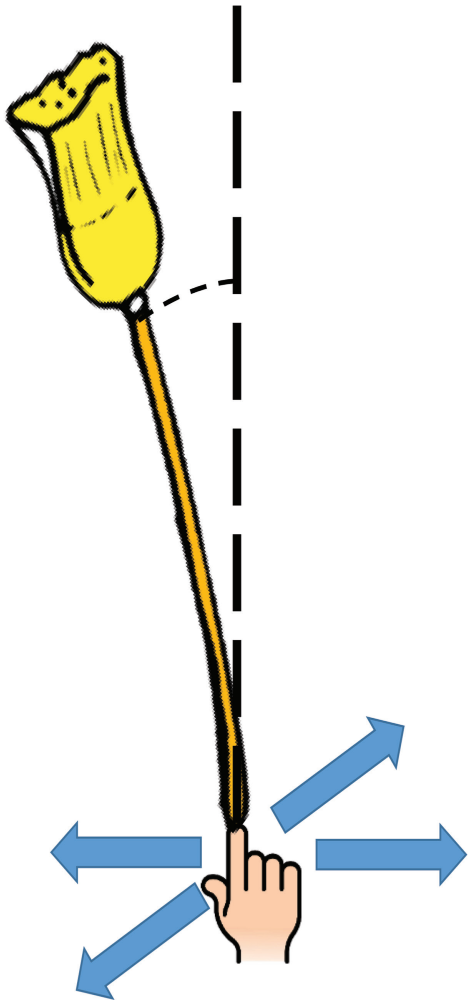


FIGURE 3.20
Balancing an upside-down broom is an inverted pendulum.

The equations of motion are found from the Lagrangian of the system, whereas

$$L = T - V. \tag{3.63}$$

T is the kinetic energy, and V is the potential energy of the system. The kinetic energy of the pendulum on the cart system is

$$T = \frac{1}{2}Mv_1^2 + \frac{1}{2}mv_2^2 \tag{3.64}$$

and the potential energy system is

$$V = mglcos(\theta). \tag{3.65}$$

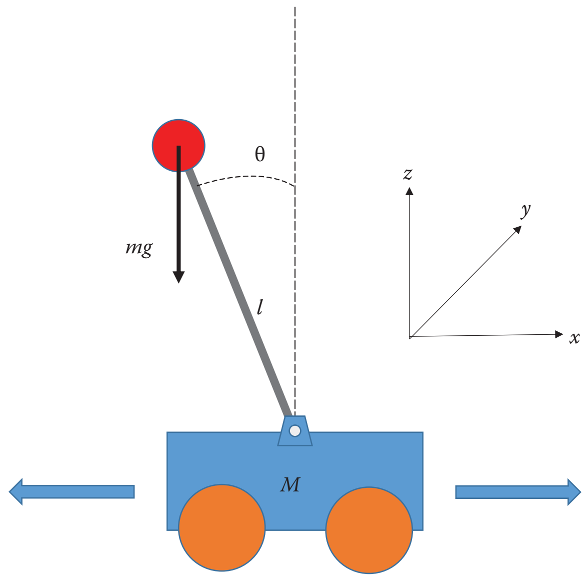


FIGURE 3.21
Diagram of inverted pendulum on a cart.

The Lagrangian becomes

$$L = \frac{1}{2}Mv_1^2 + \frac{1}{2}mv_2^2 - mglcos(\theta). \tag{3.66}$$

The velocity of the cart and the velocity of the pendulum mass are found by taking the derivative of their positions on the x-axis respectively. So,

$$v_1^2 = \dot{x}^2 \tag{3.67}$$

and

$$v_2^2 = \left[\frac{d}{dt}(x - lsin(\theta))\right]^2 + \left[\frac{d}{dt}(lcos(\theta))\right]^2. \tag{3.68}$$

Performing the derivative operations and simplifying Equation 3.68 yield

$$v_2^2 = \dot{x}^2 - 2l\dot{x}\dot{\theta}\cos(\theta) + l^2\dot{\theta}^2. \tag{3.69}$$

Finally, the Lagrangian becomes

$$L = \frac{1}{2}(M + m)\dot{x}^2 - ml\dot{x}\dot{\theta}\cos(\theta) + \frac{1}{2}ml^2\dot{\theta}^2 - mglcos(\theta). \tag{3.70}$$

The equations of motion are found from the Lagrangian by the following:

$$\frac{d}{dt}\frac{\partial L}{\partial \dot{x}} - \frac{\partial L}{\partial x} = F \tag{3.71}$$

and

$$\frac{d}{dt}\frac{\partial L}{\partial \dot{\theta}} - \frac{\partial L}{\partial \theta} = 0. \tag{3.72}$$

Substituting Equation 3.70 into Equations 3.71 and 3.72, we get the two equations of motion to be

$$(M + m)\ddot{x} - ml\ddot{\theta}\cos(\theta) + ml\dot{\theta}^2 \sin(\theta) = F \tag{3.73}$$

and

$$l\ddot{\theta} - g\sin(\theta) = \ddot{x}\cos(\theta). \tag{3.74}$$

Now, we must realize that we want to keep the pendulum (a.k.a our rocket) in the upright position and as close to the vertical flight path axis as possible. In other words, we want θ to be as close to the vertical as possible, and therefore it remains a small angle. We can linearize the equations of motion and simplify them if we use the small angle approximations

$$\cos(\theta) \sim 1 \tag{3.75}$$

and

$$\sin(\theta) \sim \theta. \tag{3.76}$$

Also, note that

$$\dot{\theta}^2 \sim 0. \tag{3.77}$$

The equations of motion become

$$(M + m)\ddot{x} - ml\ddot{\theta} = F \tag{3.78}$$

and

$$l\ddot{\theta} - g\theta = \ddot{x}. \tag{3.79}$$

Equations 3.77 and 3.78 are somewhat easier to deal with since they are linearized. They can be solved in an ordinary differential equation solver in MATLAB® or Mathcad.

Now, in order to determine how to control this spacecraft, we need a single controller transfer function. In other words, we need to equate the thrust force with the angle of the pendulum. First, we will take the Laplace transform of Equations 3.78 and 3.79 to get

$$(M+m)X(s)s^2 - ml\Theta(s)s^2 = F(s) \tag{3.80}$$

and

$$l\Theta(s)s^2 - g\Theta(s) = X(s)s^2. \tag{3.81}$$

We can now solve Equation 3.81 for $X(s)$ and substitute it back into Equation 3.80 to get

$$(M+m)\left(l\Theta(s) - \frac{g\Theta(s)}{s^2}\right)s^2 - ml\Theta(s)s^2 = F(s). \tag{3.82}$$

Simplifying Equation 3.82 and solving for the transfer function of $H(s) = \Theta(s)/F(s)$ result in the following:

$$H(s) = \frac{\Theta(s)}{F(s)} = \frac{1}{\frac{Ml}{(M+m)g}\left[s^2 - \frac{1}{Ml}\right]}. \tag{3.83}$$

Let $(M + m)g = b$ and $\frac{1}{Ml} = a^2$ and rewrite Equation 3.83 as

$$H(s) = ba\frac{a}{[s^2 - a^2]}. \tag{3.84}$$

Equation 3.84 is the open-loop transfer function for the inverted pendulum system. Taking the inverse Laplace transform gives us the time-varying transfer function and gives the time domain transfer function as

$$H(t) = \frac{\theta(t)}{F(t)} = basinh(at). \tag{3.85}$$

Figure 3.22 shows a plot of the time domain transfer function with arbitrary values for the constants. The plot shows that the system is unbounded and quickly becomes uncontrollable. This means we must put a controller like the PID controller discussed in Section 3.6.4 in loop with the system in order to control it. This is actually common sense if we

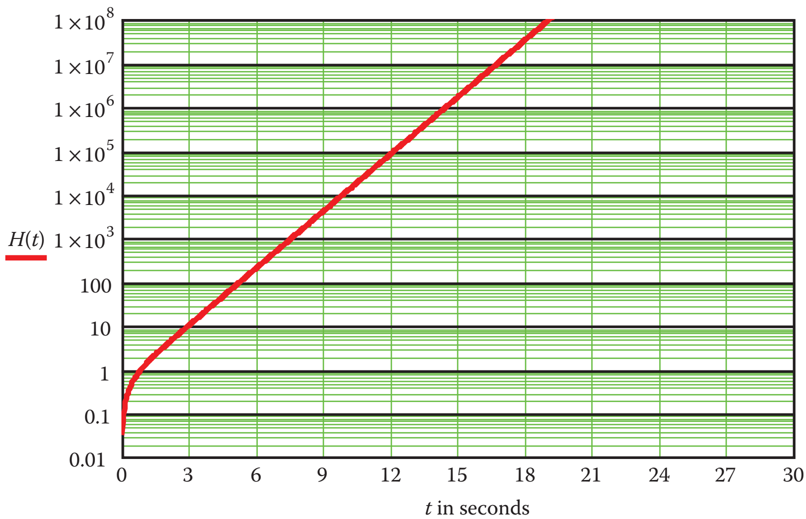


FIGURE 3.22
Time domain transfer function of inverted pendulum system.

think about it. If the top-heavy pendulum is perturbed off vertical, then we will need a control force to push the bottom of the rod back underneath the top. This is how rockets are controlled before they reach the stable flight regime. While the devil is in the details of designing a good controller for a rocket to achieve stable flight, or for an inverted pendulum to sit upright for that matter, we can look at the analysis in this section and get some understanding about rocket design choices.

A key parameter to investigate is that of the maximum control authority required by our retro rockets to keep the rocket body upright and flying straight. This force can be approximated from Equation 3.78 by realizing a couple of things. First, let $\alpha = \ddot{\theta}$, which is the angular acceleration of the mass at the top of the pendulum, or our rocket, which is top heavy. Second, we must realize that $\ddot{x}$ is the acceleration of the system in the x direction, which is the acceleration due to gravity, g, times the sine of the angle, θ. Rewriting Equation 3.78 gives us

$$F_{max}(\alpha) = (M + m)gsin(\theta) - ml\alpha. \tag{3.86}$$

From the inspection of Equation 3.86, it is clear that the worst case of control force needed to upright the rocket would be in the instant before a control thrust in the wrong direction had been given and the left part of the equation on the right-hand side is negative. This occurs and is maximum when $\theta = -\frac{\pi}{2}$, and Equation 3.86 becomes

$$F_{max}(\alpha) = -(M + m)g - ml\alpha\,. \tag{3.87}$$

Figure 3.23 shows a graph of Equation 3.87. The graph shows us the thrust requirement to overcome perturbations in angular accelerations of the rocket or pendulum.

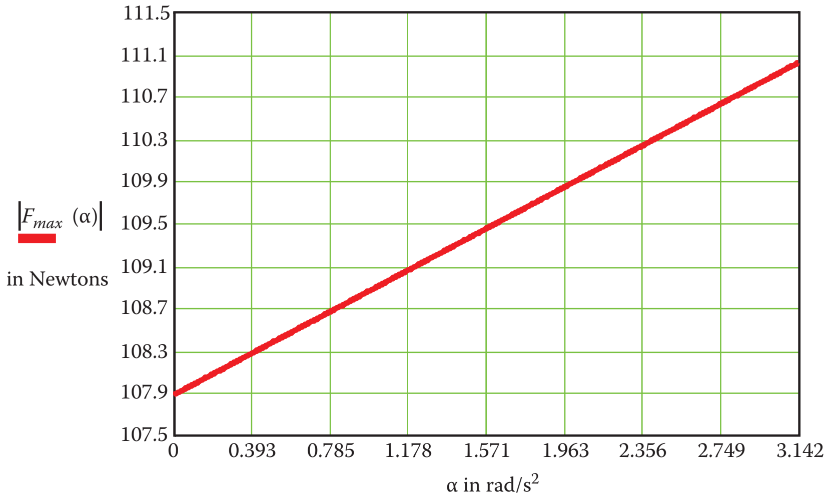


FIGURE 3.23
Magnitude of the maximum force, F_{max}, required to compensate inverted pendulum as a function of angular acceleration.

3.7 Chapter Summary

In this chapter, we have learned a great deal about how rockets work. We started in Section 3.1 and developed the concept of thrust from Newton's laws of motion. From the laws of motion, we developed definitions for mass flow rate and how this impacts rockets through throttling. And we found the rocket thrust equation, which is a good tool that is useful for designing a rocket engine for a particular mission.

The derivation of the thrust led us to understanding another key parameter for rocket engine design, and that is the specific impulse. In Section 3.2, we developed the calculation for specific impulse, and we discussed how this parameter is important when describing the efficiency of a rocket engine.

Likewise, the final design parameter for rocket engines discussed in this chapter was developed in Section 3.3. The weight flow rate was discussed, and it is clear now that with the thrust, specific impulse, and the weight flow rate, a rocket scientist or an engineer has most of the information needed to understand the capabilities of a given rocket engine. These are also important tools in the first steps of mission design. Knowing the type of mission tells the engineer if a high thrust is needed, such as for a launch vehicle, or if a high specific impulse is needed, as in interplanetary missions. The weight flow rate then tells the designer something about the actual physical size needed for the rocket engine to achieve the desired thrust and/or specific impulse.

In Section 3.4, we found the famous Tsiolkovsky rocket equation. This is the bread and butter for rocket scientists and engineers. The rocket equation allows us to understand how a rocket functions over a complete flight trajectory, from the beginning of the flight, when its fuel tanks are full, to the end of it, when all the propellant has been burned up and the tanks are empty. The equation tells us that there is a ratio of full-to-empty mass of the rocket that is the key parameter in determining how much Δv the rocket can supply to a payload. And, finally, the equation leads us to realize that, in some cases, it is better to have multiple stages of rocket engines for a more efficient system design. Staging was discussed in detail in Section 3.5.

To complete our discussion of how rockets work, we discussed in Section 3.6 the flight dynamics and how to control the attitude of a rocket vehicle during flight. We developed important concepts of rocket makeup including the center of gravity and the center of pressure. We showed how to calculate these parameters and why they are important to rocket scientists and engineers. Then, we developed the actual process for controlling the rocket's attitude during flight and discussed the PID controller. We also discussed the inverted pendulum and how it pertains to rockets in the stage of flight before enough dynamic pressure is available for stable flight. This occurs right at liftoff and for vertical landing scenarios. We learned that the inverted pendulum is directly applicable in describing rockets and how to determine the control forces required to maintain upright stability.

From the basics of thrust to the complexity of ACS, we now have a basic understanding of rocket vehicles and rocket flight. From Chapter 1, Chapter 2, and now Chapter 3, we are beginning to see how intricately detailed and massively complex the field of rocket science and engineering has become. And we have yet to discuss combustion chambers and nozzles in any detail. That will come in the next chapter.

Exercises

3.1 What do you call the opening at the bottom of the combustion chamber?

3.2 Highly accelerated exhaust gases leaving the rocket engine nozzle propel the spacecraft through which of Newton's laws of motion?

3.3 Why is m-dot important to the astronaut phrase "throttle up"?

3.4 What does "throttle up" mean?

3.5 What is the *impulse momentum theorem*, and what does it tell us?

3.6 What is the difference between the *effective exhaust velocity* and the *equivalent velocity*?

3.7 Define *equivalent velocity*.

3.8 Define *specific impulse*.

3.9 What is the importance of the *weight flow rate*?

3.10 What are three key parameters for rocket engine design?

3.11 What is the *propellant mass ratio*? What else is it sometimes called?

3.12 What is *hybrid staging*?

3.13 What are three types of staging?

3.14 Describe the four major subsystems of a rocket.

3.15 What is max-Q?

3.16 Why do most launch vehicles wait until after max-Q to "go at throttle up"?

3.17 What are three rocket flight conditions?

3.18 What is the *restoring force*?

3.19 Discuss the four types of attitude correction systems.

3.20 What is a PID controller?

3.21 Given a rocket nozzle with an exit area of 1 m^2 and an exit pressure of 101,325 Pa, what is the force on the nozzle due to the pressure difference inside and outside the rocket if the rocket is at sea level?

3.22 In Exercise 3.21, calculate the force on the nozzle if the rocket is in space and the pressure outside the rocket is zero.

3.23 In Exercises 3.21 and 3.22, determine the force on the rocket if the m-dot of the engine is 1 kg/sec and the exhaust velocity is 400 m/sec.

3.24 A rocket engine has an I_{sp} of 363 sec and can produce a thrust of 2 MN. Calculate the equivalent velocity for the engine.

3.25 In Exercise 3.24, determine the m-dot of the engine.

3.26 In Exercises 3.24 and 3.25, determine the *mass ratio* required to reach a Δ-v of 7,700 m/sec.

3.27 In Exercises 3.24 through 3.26, determine the burn time required to achieve the Δ-v of 7,700 m/sec assuming the *mass ratio* calculated in Exercise 3.26.

3.28 Given a two-stage launch vehicle with an engine that produces an I_{sp} = 400 sec, a payload mass of 10,000 kg, stage 1 structure mass of 10,000 kg, and stage 2 structure mass of 10,000 kg, determine the mass ratio and the total mass of propellant required to reach low Earth orbit. Assume the total Δ-v required is 7,700 m/sec. Determine the Δ-v after each stage and the propellant mass for each stage.

3.29 Assume that the drag coefficient for the ISS is 0.2 and its velocity is 27,744 km/h. The density of the atmosphere at ISS's orbit is about 1×10^{-11} kg/m^3. If the surface area of the ISS is about 3,000 m^2, what is the drag force?

3.30 Consider three blocks of density 1 kg/m^3. Block 1 is 1 m per side in dimension. Block 2 is 2 m per side in dimension. Block 3 is 3 m per side in dimension. The blocks are oriented in such that the largest block, Block 3, is on the bottom. Block 2 is then stacked on Block 3, and then Block 1 is stacked on Block 2. The faces of the blocks are aligned, and the center of each the blocks make a straight line upward through them. Figure 3.24 shows the blocks and how they are stacked.

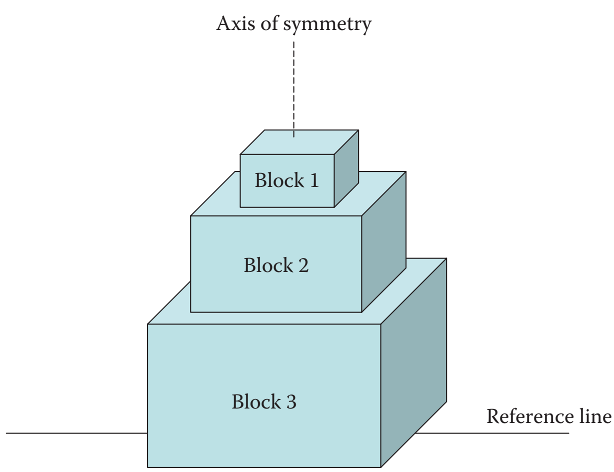


FIGURE 3.24
Diagram of stacked blocks for Exercise 3.30.

With the bottom of the stack as the reference line, calculate the center of gravity of the stack of blocks.

3.31 In Exercise 3.29, if Block 1 was twice as tall and the three blocks remain in the same stacked configuration, calculate the center of gravity.

3.32 In Exercise 3.29, calculate the center of pressure for the stacked blocks.

3.33 In Exercise 3.30, calculate the center of pressure for the stacked blocks.

3.34 Define 8-DOF and explain each component in detail.

4

How Do Rocket Engines Work?

In Chapter 1, we discussed how rocketry and rockets were developed over history. That gave us a detailed understanding of when breakthroughs in the science and engineering of rockets came about chronologically. In Chapter 2, we learned why the rockets are needed, which is to put something into orbit or to launch a payload on a trajectory. And, in Chapter 3, we developed the basics of rocketry and learned the concepts of thrust, specific impulse, weight flow rate, staging, and the rocket equation. So, we now have an understanding of rocketry from a historical, mission-need, and overall system perspective. Mostly, we talked about things that were outside of the rocket or acting upon the rocket. Though we did discuss thrust coming from the rocket, we didn't really talk about how the rocket generates that thrust. Now we shall.

The question we will answer in this chapter is what goes on inside the rocket system to generate the propulsion force. Mainly, this will be a discussion of the rocket engine, its components, and the physics involved in the generation of the propulsive force.

4.1 Basic Rocket Engine

Figure 4.1 shows a block diagram and a photo depicting the basic components of a rocket engine. To begin with, the engine needs some form of propellant. This includes both fuel and oxidizer. The main energy that will be converted to propulsion energy is stored in the propellant if it is a combustion-type engine. If the engine is simply a thermal engine, then the energy could be stored electrically or in nuclear fissile material. In the purely thermal engines, a heat source is used to heat an exhaust gas. The exhaust gas is practically inert and might be something as simple as water. In these cases, the propellant is simply a means to convert the heat energy into propulsive energy.

But, in most typical modern rocket engines, the heat is generated through a chemical reaction between the propellant chemicals. The fuel and the oxidizer are typically mixed together in an *exothermic reaction*. An exothermic reaction is defined as a reaction where chemical bonds are broken with less energy required than that needed to make the bonds. The excess energy is released as heat. A more simple definition is that an exothermic reaction is any reaction that releases heat. A very pertinent example of such a reaction is the mixing of liquid hydrogen, H_2, with liquid oxygen, O_2. The chemical reaction is as follows:

$$2H_2 + O_2 + \text{Ignition Heat} \rightarrow H_2O + \text{Excess Heat}. \tag{4.1}$$

A spark for ignition on the left side of the equation enables the burning of the liquids together to produce water and a large amount of heat as the by-products of the reaction. As can be seen in Equation 4.1, there is a proper mixing ratio of liquid hydrogen to liquid

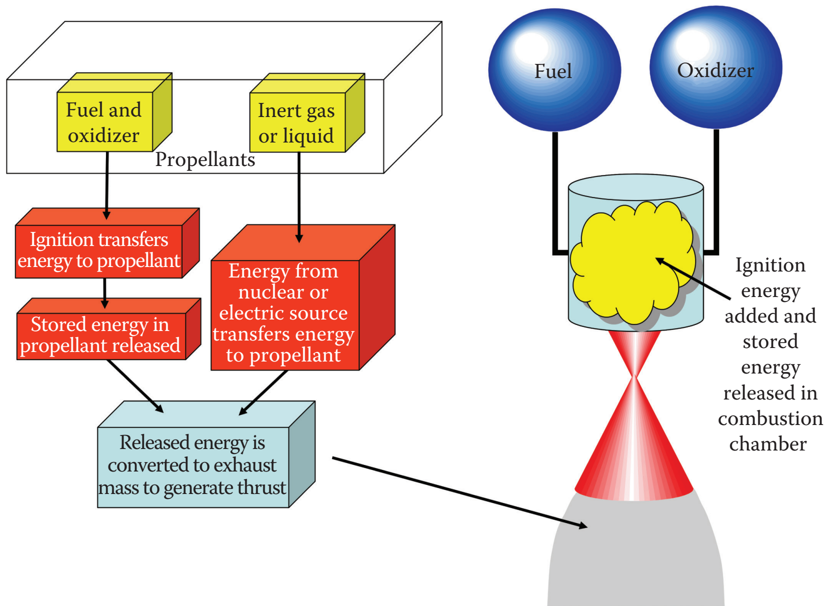


FIGURE 4.1
Shown are the basic components of a rocket engine.

oxygen. Two diatomic hydrogen molecules per one diatomic oxygen need to be in the chamber for an efficient use of the propellants. This ratio is known as the *stoichiometric ratio* and is what we all became familiar with in high school chemistry when the teachers had us balancing chemical equations. Don't be misled, though. Even though there are twice as many diatomic hydrogen molecules needed in the mix, it doesn't mean that there is twice as much by mass. Recall that the molecular weight of hydrogen is much less than that of oxygen.

The reaction in Equation 4.1 is that which occurs inside the engines of the Space Shuttle. The Space Shuttle Main Engines (SSMEs) react liquid hydrogen and liquid oxygen together to generate the thrust that drives the rocket into space. With each mole of liquid oxygen burned, 483.6 kJ of heat are produced. The SSMEs burn about 500,000 kg of O_2 during launch. Using the molecular weight of diatomic oxygen, we can find the number of moles burned as

$$\#\,\text{moles} = \frac{500,000\ \text{kg}}{0.032\ \text{kg/mole}} = 15,625,000\ \text{moles}. \tag{4.2}$$

Multiplying the number of moles by the heat generated per mole gives the total heat, ΔH, released to be

$$\Delta H = (15,625,000\ \text{moles})(483.6\ \text{kJ/mole}) = 7,556,250,000\ \text{kJ} \approx 7.6\ \text{TJ}. \tag{4.3}$$

That is quite a bit of heat, indeed.

The calculation in Equation 4.3 shows us that the exothermic reaction within a rocket engine releases a tremendous amount of heat energy, which, in turn, heats up the remaining gas products. In the case of the SSMEs, the combustion by-product is water, as shown in Equation 4.1. As these products (water vapor) get superheated inside the combustion chamber, they are forced out of the rear of the engine and are accelerated by a nozzle as they exit. Once they reach the exit of the nozzle at extremely high exhaust velocities, the result is a net reaction force against the rocket following the law of conservation of momentum and Newton's Laws (as discussed in Chapter 3).

In some instances, as with the SSMEs, the fuel and the oxidizer need an igniter to spark the reaction. Simply mixing the propellant fluids isn't enough to start the reaction; therefore, energy is added to the system. As the SSMEs prepare to fire, they use spark plugs to ignite an internal "blowtorch" of hydrogen and oxygen, which blows the flame through the rest of the combustion chamber. Once the reaction is started, it will continue to burn as long as there is propellant flow. Often, people confuse the sparks they see flying across the bottom of the SSMEs just before launch as the igniters. These sparks are used to keep any excess propellant gas from pooling in dangerous quantities underneath the engines. The spark shower keeps any propellant clouds ignited before they have time to pool.

In some engines, no igniter is needed, such as in the Space Shuttle orbital maneuvering system (OMS) thrusters. Those smaller rocket engines implement a single engine based on the Apollo Service Module's Service Propulsion System. The engine uses monomethylhydrazine (MMH) for fuel and nitrogen tetroxide (N_2O_4) for oxidizer. When the two propellants are mixed, they are volatile enough to spark the reaction without an external ignition source. A self-starting reaction like this is called *hypergolic*. The advantages of using hypergolic systems are fairly obvious. The mechanical systems are much less complex. The combustion rate of a hypergolic engine can be controlled by two flow control valves: one to control the fuel and one the oxidizer. Another advantage to hypergolic propellants is that large explosive quantities can't gather in one place. This is because the two compounds are volatile with each other, and, as they come into contact, they start to burn. A disadvantage of hypergolic systems is that they typically have a significantly lower I_{sp} than nonhypergolic ones.

RS-68 engine J-2X engine

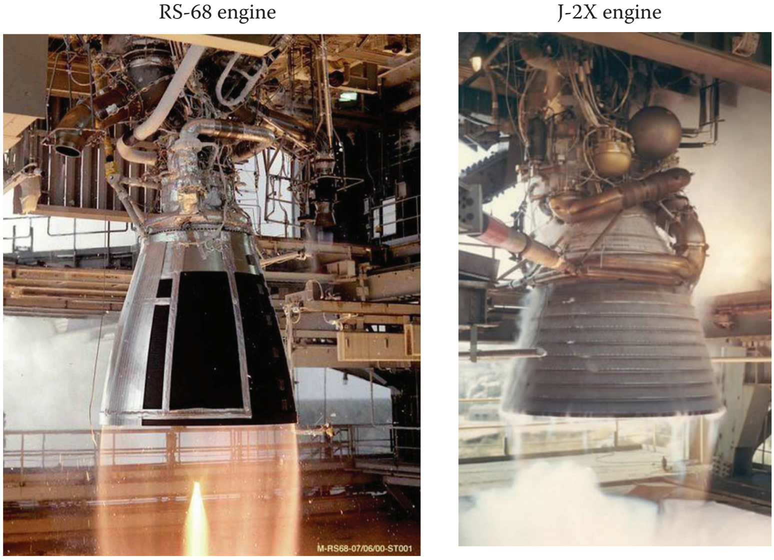


FIGURE 4.2
The RS-68 of the Delta IV and the J-2X of the Ares I are two modern liquid rocket engines.

Once the propellants are mixed and reacted within the combustion chamber of the rocket, they expand, and the force of the combustion is redirected out of the chamber through a nozzle. The simplest description of a rocket nozzle is that it is a component of a rocket (or an air-breathing engine like a jet) that produces thrust by the redirection and acceleration of exhaust gases. The nozzle converts the thermal energy of the chemical reactions in the combustion chamber (or the heated gas in a nuclear thermal engine) into kinetic energy through thermodynamic expansion by directing the kinetic energy vector along the axis of the rocket's flight path, which is in line with the nozzle axis. Figure 4.2 shows the J-2X engine that was evolved from the ones used on the Saturn IV upper stage and will be used on the Ares-I upper stage and the RS-68, which is used on the Delta IV and will be utilized as the main engines of the Ares V.

4.2 Thermodynamic Expansion and the Rocket Nozzle

Inside the combustion chamber of the rocket engine, for some reason, there is heat. Depending on the rocket engine type, there is a combustion process or, at the very least, a *thermal expansion* process taking place. What is *thermal expansion*? As the gas inside the combustion chamber is heated by the heating source (reaction, or heater), the gas follows the laws of physics and expands. If the walls of the chamber are rigid and will not expand, then the pressure inside the chamber increases. For a chemical rocket, there will be a fuel and an oxidizer burning (as discussed in Section 4.2) that generates superheated gases as by-products. Nuclear thermal rockets (NTRs) use a fission reactor to heat water or other fluids and gases into steam within the chamber. Solar thermal propulsion (STP) uses sunlight to convert the liquid fuel to pressurized heated gases. In each of these types of engines, this thermal energy is trapped within the combustion chamber. The pressure within the chamber can be determined by

$$P = \rho \frac{R_u}{M} T = \rho R T \tag{4.4}$$

where P is the pressure within the chamber, ρ is the density of the gas, R_u is the universal gas constant 8,314.41 J/kmol K, R is the specific gas constant, M is the molecular mass of the gas, and T is the temperature of the gas in K. From Equation 4.4, we can see that the hotter the combustion process in the chamber gets, the higher the pressure of the gas in the chamber will be. This equation is a statement of the *ideal gas law*, and this relationship between temperature and pressure is shown in Figure 4.3.

As mentioned earlier in this section, the combustion chamber is rigid, and the gas is not allowed to expand as it is heated there. So, the chamber is attached to a converging nozzle, as was shown in Figure 3.1 where the pressurized gas escapes and is forced down through it. As it flows through the converging nozzle, the gas is compressed and accelerated until it reaches the throat of the converging nozzle where a diverging nozzle is attached. If the nozzle is designed properly, the converging side will accelerate the flow of gas to the speed of sound at the *throat*. As the gas exits from the diverging nozzle, it is expanded and accelerated to supersonic velocities. *Note:* An important

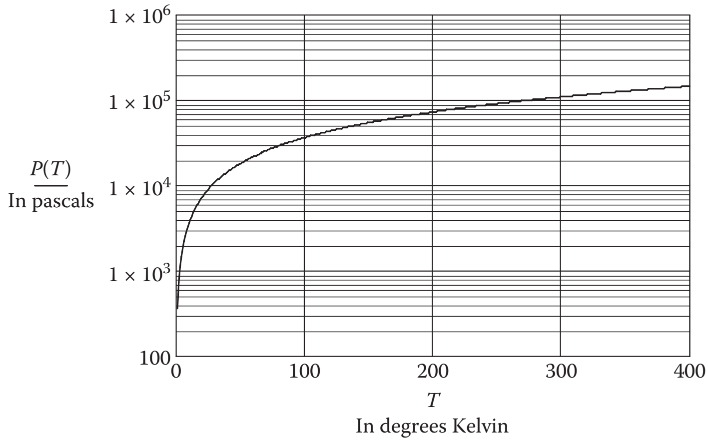


FIGURE 4.3
The relationship of pressure versus temperature of air using the ideal gas law.

phenomenon of fluid flow is that subsonic velocity–flowing fluids actually accelerate in converging nozzles, whereas supersonic flows accelerate in diverging nozzles. Some other properties to note about the flow of these heated gases through the rocket engine are the following:

- The flow is *adiabatic*, which means that, once it is heated in the combustion chamber and becomes *adiabatic*, no heat is transferred into or out of it.
- The flow is *reversible*, which means that, if the flow goes the other direction, the supersonic gases will slow down as they go backward through the diverging nozzle (now converging) and vice versa; energy is conserved in the system.
- The flow is *isentropic* by definition because an *isentropic flow* is both *adiabatic* and *reversible*.
- All chemical reactions and/or thermal energy additions take place within the combustion chamber before the flow becomes *isentropic*; this is called *frozen flow*.
- Because energy and momentum are conserved, then mass flow must remain constant (no throttling); this is known as *steady flow*.

4.2.1 Isentropic Flow

In the study of thermodynamics, we find that an *isentropic process* is defined as a process, whereas a system's entropy always remains constant. The *second law of thermodynamics* states that the temperature of a system multiplied by the change in entropy, ΔS, of that system is greater than or equal to the heat gained by the system, ΔQ. In other words,

$$T\Delta S \geq \Delta Q. \tag{4.5}$$

This is the law of thermodynamics that is often stated as "all things tend toward disorder." However, if the process is reversible, then the relationship becomes

$$T\Delta S = \Delta Q. \tag{4.6}$$

It is important to realize that Equation 4.6 tells us that the process is closed off from outside influence and that there is no transfer of heat energy outside the system. This also means that the process is *adiabatic*, and, as we mentioned in Section 4.2, a *reversible* and *adiabatic* system or process is *isentropic*. Therefore, we have come to a good general description of the isentropic flow, which is, "No energy enters or leaves the flow."

Isentropic flow typically only occurs when all flow variables change slowly and with small amplitude variations. This is the case with the converging–diverging nozzle. But, before we can go further in understanding the flow in the nozzle, we need to develop a few more mathematical tools.

The enthalpy, H, of the system is

$$H = U + PV \tag{4.7}$$

where U is the internal energy of the system, P is pressure, and V is volume. The total energy per mass in a fluid system or process is known as the *specific enthalpy*, h, and is written as

$$h = \frac{H}{m} = \frac{U}{m} + P\frac{V}{m} = u + Pv_{sp} \tag{4.8}$$

where h is the specific enthalpy of the system, m is the mass of the ideal gas flowing in the system, u is the specific internal energy, and v_{sp} is the specific volume (which is the inverse of density).

As the flow is forced down the converging nozzle and is accelerated, this suggests that the kinetic energy of the gas is increasing. But, we know that within the engine is an isentropic flow, so there must be a trade-off in the overall energy of the flow somewhere. This is governed by the Bernoulli Principle, which states

$$h + \frac{1}{2}v^2 = constant \tag{4.9}$$

where v is the velocity of the flow, and the velocity-squared segment of the equation is known as the specific kinetic energy. Equation 4.9 shows us that, if the specific kinetic energy of the flow increases due to an increase in velocity, then the specific enthalpy must go down because the sum of the specific enthalpy and the specific kinetic energy must remain constant. The opposite must be true as well.

This tells us how the gas flows from the combustion chamber, down the converging nozzle, and to the throat of the converging nozzle. At this point, the nozzle begins to expand becoming a diverging nozzle. As the area of the nozzle increases, the flow is accelerated. Why?

Before we can answer this question, we need to discuss some aspects of supersonic flow. A nozzle that is designed properly will accelerate the subsonic flow in the converging part of the nozzle until it reaches the throat. At the throat, the nozzle passes through the sound barrier hopefully without a shock wave being generated, and then it accelerates out the diverging side of the nozzle as supersonic flow or flow that is faster than the speed of sound. The speed of sound, a_o, in a material (in the case of the rocket engine flow, the material is made up of exhaust gases) is given by

$$a_o = \sqrt{\gamma RT} \tag{4.10}$$

where γ is the ratio of the heat capacity at constant pressure, C_p, to the heat capacity at constant volume, C_v, and is also called the *specific heat ratio* and sometimes the *isentropic expansion factor*. The velocity of the flow divided by the speed of sound in the flow is called the Mach number, $M_\#$, and is

$$M_\# = \frac{v}{a_o}. \tag{4.11}$$

The mass flow rate through the converging diverging nozzle of cross-sectional area A is

$$\dot{m} = \rho v A = constant. \tag{4.12}$$

Taking the derivative of Equation 4.12 results in

$$vAd\rho + \rho Adv + \rho vdA = 0. \tag{4.13}$$

Divide Equation 4.13 by ρvA:

$$\frac{d\rho}{\rho} + \frac{dv}{v} + \frac{dA}{A} = 0. \tag{4.14}$$

We must now introduce an isentropic flow equation, which is

$$\frac{dP}{P} = \gamma \frac{d\rho}{\rho}. \tag{4.15}$$

Rewriting Equation 4.15 and recalling that $P = \gamma RT$, we get

$$dP = \gamma P \frac{d\rho}{\rho} = \gamma RT d\rho. \tag{4.16}$$

Substituting Equation 4.10 into Equation 4.16 results in

$$dP = a_o^2 d\rho. \tag{4.17}$$

The conservation of momentum of the gas flow can be written as follows:

$$\rho v dv = -dP. \tag{4.18}$$

Substituting Equation 4.17 into this gives us

$$\rho v dv = -a_o^2 d\rho. \tag{4.19}$$

Rewriting Equation 4.19 and multiplying by v/v, we get

$$\frac{v}{va_o^2} v dv = -\frac{d\rho}{\rho}, \tag{4.20}$$

or

$$\frac{v^2}{a_o^2}\frac{dv}{v} = -\frac{d\rho}{\rho}, \tag{4.21}$$

and, finally,

$$-M_{\#}^2 \frac{dv}{v} = \frac{d\rho}{\rho}. \tag{4.22}$$

Now, substitute Equation 4.22 into Equation 4.14 to get

$$-M_{\#}^2 \frac{dv}{v} + \frac{dv}{v} + \frac{dA}{A} = 0. \tag{4.23}$$

Simplifying

$$\left(1 - M_{\#}^2\right)\frac{dv}{v} = -\frac{dA}{A}. \tag{4.24}$$

Equation 4.24 is a flow equation describing the flow through a nozzle system in relation to the velocity of the flow field, the Mach number, and the effective cross-sectional area. The equation shows us that if the Mach number is greater than 1, then a positive area change in the nozzle indicates a positive velocity change. For Mach numbers less than 1, if the area change is positive, then the velocity change is negative. This is the proof that we were looking for as to why the converging nozzle speeds up the subsonic flow and the diverging nozzle speeds up the supersonic flow.

Figure 4.4 shows a converging–diverging nozzle illustrating the different regimes of flow and how the velocity increases as the area changes, which was described by Equation 2.24. The convergent–divergent nozzle design is also known as the *de Laval* nozzle after the Swedish inventor Gustaf de Laval who developed it in the 19th century. As mentioned earlier in this section, the subsonic flow on the converging side breaks the sound barrier

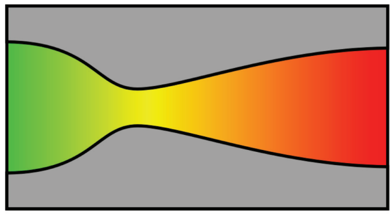


FIGURE 4.4
Velocity flow through a convergent–divergent or de Laval nozzle increases from left to right. (GNU free documentation license image.)

at the throat if the engine is designed properly. If $M_{\#} = 1$ at the throat, then the mass flow through the nozzle is said to be a *choked flow* or sometimes just *choked*.

4.3 Exit Velocity

We have discussed the exit velocity in previous sections, but we only defined it as a parameter of a rocket engine. We will now use the idea of the thermodynamic expansion and the isentropic flow used in Sections 4.2 and 4.2.1 to actually derive the exit velocity in terms of the combustion chamber and nozzle system. We will start by writing an equation for the heat in the combustion chamber and equating that to the kinetic energy of the exhaust particles as follows:

$$\Delta Q = K, \tag{4.25}$$

$$mC_p \Delta T = \frac{1}{2} m v_e^2. \tag{4.26}$$

Realizing that $\Delta T = T_c - T_e$, where T_c is the combustion chamber temperature and T_e is the exit temperature of the exhaust gases. Solving for the exit velocity yields

$$v_e^2 = 2C_p \Delta T = 2C_p (T_c - T_e). \tag{4.27}$$

At this point, we need the following definitions:

$$C_p = \frac{\gamma}{\gamma - 1} \frac{R_u}{M}, \tag{4.28}$$

$$T_e P_e^{\frac{\gamma}{\gamma-1}} = constant, \tag{4.29}$$

$$T_c P_c^{\frac{\gamma}{\gamma-1}} = constant. \tag{4.30}$$

Because the flow is isentropic, then the constants on the right-hand side of Equations 4.29 and 4.30 are equal, and, therefore, dividing Equation 4.29 by Equation 4.30 results in

$$\frac{T_e P_e^{\frac{\gamma}{\gamma-1}}}{T_c P_c^{\frac{\gamma}{\gamma-1}}} = \frac{constant}{constant} = 1. \tag{4.31}$$

Rearranging Equation 4.31 gives us a relationship between the temperature and pressure ratios as

$$\frac{T_e}{T_c} = \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}. \tag{4.32}$$

Now substitute Equation 4.32 into Equation 4.27, and we get

$$v_e^2 = 2C_p(T_c - T_e) = 2C_p T_c\left(1 - \frac{T_e}{T_c}\right) = 2C_p T_c\left(1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right). \tag{4.33}$$

Inserting Equation 4.28,

$$v_e^2 = \frac{2\gamma}{\gamma - 1}\frac{R_u}{M}T_c\left(1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right). \tag{4.34}$$

Equation 4.34 is a design equation of sorts for the nozzle of the rocket engine. If the nozzle is designed properly, then the pressure of the flow at the nozzle exit is equal to the ambient pressure. If the nozzle is designed for space where the ambient pressure is effectively zero, then Equation 4.34 can be written as

$$v_e^2 = \frac{2\gamma}{\gamma - 1}\frac{R_u}{M}T_c, \tag{4.35}$$

Though we will mostly use Equations 4.34 and 4.35 as they are written in their squared form, it is useful to see them as simply the exit velocity as well. So, taking the square root of the two equations gives the final equation for the exit velocity to be

$$v_e = \sqrt{\frac{2\gamma}{\gamma-1}\frac{R_u}{M}T_c\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)} \tag{4.36}$$

and

$$v_e = \sqrt{\frac{2\gamma}{\gamma-1}\frac{R_u}{M}T_c}\,. \tag{4.37}$$

Note that the two equations for the exit velocity are functions only of the chamber temperature and pressure, the exit pressure, the molecular mass of the gas, and the isentropic expansion factor. All of these parameters can be fixed through design choices.

Figure 4.5 shows the exit velocity as a function of flow pressure. There are four graphs, each with varying isentropic expansion factor. You will notice that the difference is small between each of the graphs. Figure 4.6 illustrates the exit velocity as a function of flow pressure with four different chamber temperatures, and Figure 4.7 has four different molecular weights. The most significant changes in all three figures are when the molecular weight of the exhaust gas is changed. Also note from each of the graphs that the maximum exit velocities occur when the pressure ratio is zero. In other words, when the exit pressure of the rocket is zero, a maximum thrust output is achieved. What this tells us is that rocket engines produce the most thrust in the vacuum of space.

Figure 4.7 also tells us something else, and that is the heavier the exhaust gas particles are, the lower the exit velocity. This might immediately lead us to believe that propellants with lower molecular weights are optimum. This is misleading. Recall that the key parameters of rocket engine performance include thrust, specific impulse, and mass flow rate through the engine. Each of these parameters is coupled to the exit velocity, so trade-offs must be made.

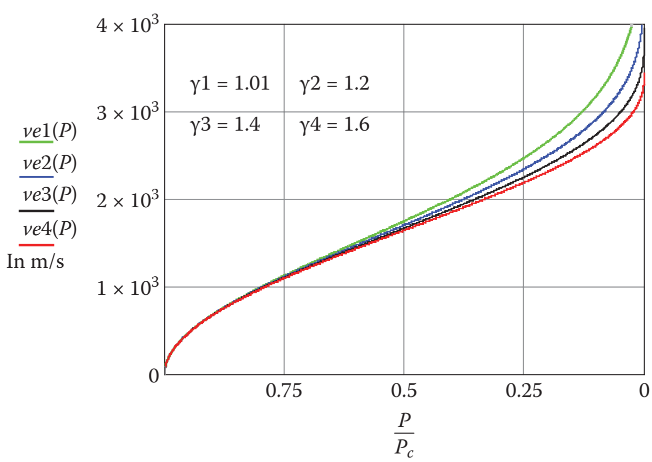


FIGURE 4.5
Exit velocity through a convergent–divergent nozzle as a function of pressure. $M = 12$, $T_c = 3{,}215\ K$, and $P_c = 5$ MPa.

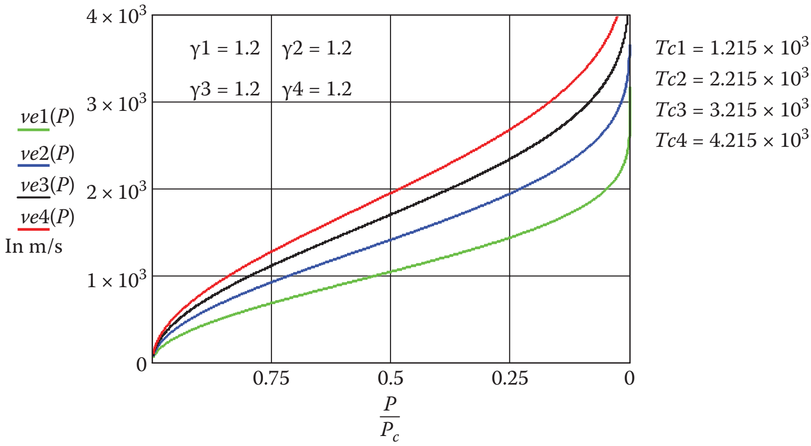


FIGURE 4.6
Exit velocity through a convergent–divergent nozzle as a function of pressure with varying chamber temperatures. $M = 12$ and $P_c = 5$ MPa.

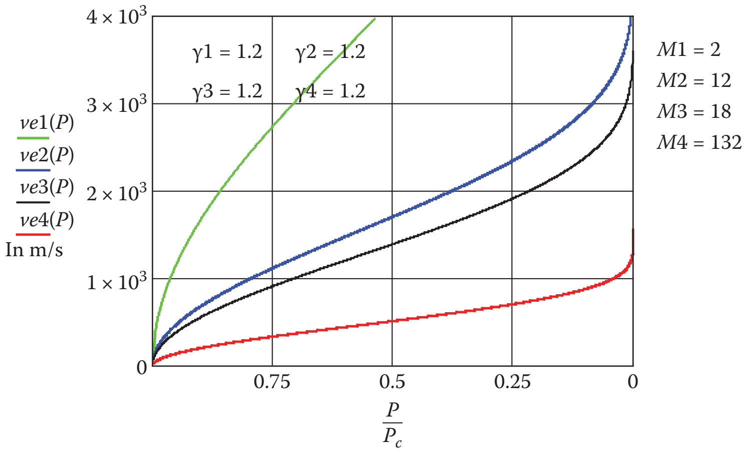


FIGURE 4.7
Exit velocity through a convergent–divergent nozzle as a function of pressure with varying molecular weights (deuterium, carbon, oxygen, xenon). $T_c = 3{,}215$ K and $P_c = 5$ MPa.

We will develop a formula for the m-dot through the engine in much the same way as we did the exit velocity to perhaps shed some light on these trade-offs. Before we begin, we will need to add a few more mathematical tools to our toolbox. They are

$$\rho_c = P_c \frac{M}{R_u T_c}, \tag{4.38}$$

$$\frac{\rho}{\rho_c} = \left(\frac{P}{P_c}\right)^{\frac{1}{\gamma}}, \tag{4.39}$$

and

$$\rho = P_c \frac{M}{R_u T_c}\left(\frac{P}{P_c}\right)^{\frac{1}{\gamma}}. \tag{4.40}$$

These three equations will come in handy at some point.

Now, consider the equation for m-dot and Equation 4.36:

$$\dot{m} = \rho v_e A = \rho A\sqrt{\frac{2\gamma}{\gamma-1}\frac{R_u}{M}T_c\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}. \tag{4.41}$$

Moving the density inside the square root and substituting Equation 4.40 for it give

$$\dot{m} = A\sqrt{\left(P_c \frac{M}{R_u T_c}\left(\frac{P}{P_c}\right)^{\frac{1}{\gamma}}\right)^2 \frac{2\gamma}{\gamma-1}\frac{R_u}{M}T_c\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}. \tag{4.42}$$

Simplifying Equation 4.42 gives us an equation for the m-dot to be

$$\dot{m} = P_c A\sqrt{\frac{2\gamma}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P_e}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}. \tag{4.43}$$

We should also realize at this point that this is an equation for pressure throughout the nozzle, not just at the exit. Therefore, Equation 4.43 can be written more generally as

$$\dot{m} = P_c A\sqrt{\frac{2\gamma}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}. \tag{4.44}$$

Equation 4.44 is powerful in that it tells us the mass flow rate through the nozzle, but, because the flow is isentropic, the mass flow rate is constant. What varies in the equation is the cross-sectional area of the nozzle at any given point along the nozzle's axis, as well as the pressure of the flow. Thus, by rewriting Equation 4.44 as

$$\frac{\dot{m}}{A} = P_c\sqrt{\frac{2\gamma}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)} \tag{4.45}$$

we obtain an equation for the so-called *flow density*. In actuality, the flow density is really the m-dot passing through a given surface area and is truly an areal density. But, suffice it to say that most rocket scientists and engineers call it the flow density, and it is given in units of kg/m^2. Figure 4.8 is a graph of the flow density as a function of the pressure through the nozzle. The graph actually tells us the correct shape of the nozzle. Where the flow pressure is equal to the chamber pressure at the nozzle's input, the mass flow per area is a minimum. Also, at the exit where the exit pressure is a minimum, the mass flow per area is a minimum. This tells us that the area of the inlet and outlet sides of the nozzle is a maximum. In the middle of the graph around $P/P_c = 0.6$, the flow is at a maximum. Therefore, at some point between the inlet (expansion chamber) and the outlet of the nozzle, there is a minimum area. This is the throat. And this equation and graph reveal that we need a converging–diverging nozzle, and the only thing we need to know about the rocket engine is the isentropic expansion factor, the molecular weight of the exhaust gas, the temperature inside the expansion chamber, and the pressure inside the expansion chamber. All are thermodynamic properties that lead us to design parameters and decide the actual physical shape of the rocket nozzle.

A quick comparison of Figures 4.7 and 4.8 is worthwhile. The rocket engine parameters are the same for each of the graphs, and, therefore, they are directly comparable to each other. Note that, as the exit velocity is decreased by an increase in molecular weight, the flow density increases. If we multiply Equation 4.36 by Equation 4.45, we get the thrust of the engine divided by the cross-sectional area, which we will call the *areal normalized thrust* or the *thrust density:*

$$\frac{F_{thrust}}{A} = \frac{\dot{m}}{A} v_e$$

$$= P_c\sqrt{\frac{2\gamma}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}\sqrt{\frac{2\gamma}{\gamma-1}\frac{R_u}{M}T_c\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}. \tag{4.46}$$

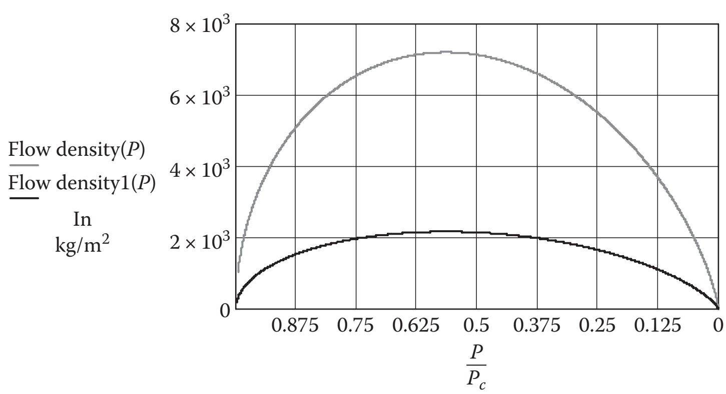


FIGURE 4.8
Flow density versus pressure for two different exhaust gases (carbon and xenon). T_c = 3,215 K and P_c = 5 MPa.

Figure 4.9 is a graph of the thrust density as a function of the normalized pressure through the system. The graph is of two of the scenarios given in Figures 4.7 and 4.8, but note that there is only one curve. This shows us that the thrust through the engine will be the same when any of the design variables are changed, except for the isentropic expansion factor or the chamber pressure. Figure 4.10 shows the thrust density of the engine with different isentropic expansion factors. This change does indeed cause the two thrust density curves to differ with otherwise the same engine design parameters. The lower isentropic expansion factor (approaching 1.0) allows for the higher thrust density. What we can learn from these variations is that the engine thrust performance is basically a function of the thermodynamics of the exhaust gas itself. Unfortunately, the isentropic expansion factor is typically a value of 1.2 for hydrocarbon fuel and oxidizer exhaust gases and only varies slightly. Also, the other potential means of increasing the thrust density would be to raise the chamber pressure. However, the chamber pressure is limited by the strengths of materials of the engine walls. If the walls are built too rigid, then they become impractically heavy to a point of diminishing returns. This is part of the reason that the performance of thermodynamic rocket engines is limited by the very physics of their design.

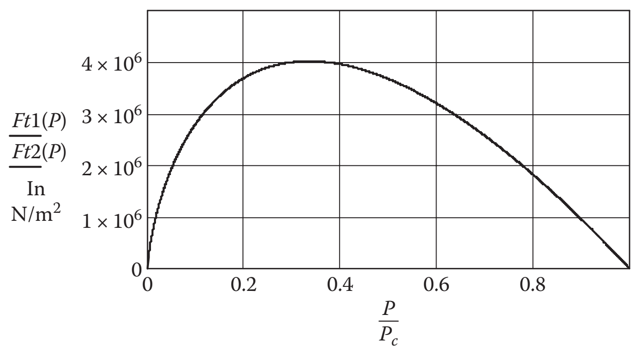


FIGURE 4.9
Thrust density versus pressure for two cases where only the isentropic expansion factor and chamber pressure are the same.

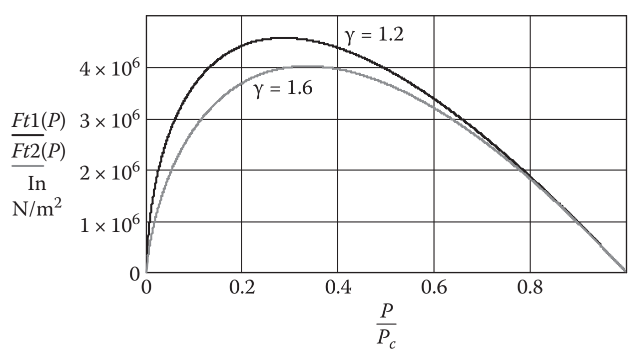


FIGURE 4.10
Thrust density versus pressure for two cases where the isentropic expansion factors are different.

4.4 Rocket Engine Area Ratio and Lengths

4.4.1 Nozzle Area Expansion Ratio

Although we talked in fairly great mathematical detail about the pressure and mass flow through the rocket engine and how its cross-sectional area must change along the engine's axis, we didn't really discuss how to actually design a rocket nozzle from all of that math. In this section, we will go a step closer to being able to physically design a rocket nozzle. First, we need to reexamine Equation 4.45. Solving the flow density equation for the cross-sectional area gives us

$$A = \frac{\dot{m}}{P_c\sqrt{\frac{2\gamma}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}}. \tag{4.47}$$

Now, we need to define the parameter known as the *expansion ratio*, ε, to be

$$\varepsilon = \frac{A_e}{A_t}, \tag{4.48}$$

where A_e is the cross-sectional area of the exit of the nozzle, and A_t is the cross-sectional area of the throat of the nozzle. Note that, in some textbooks, the throat area is also denoted as A^* and is pronounced "A-star." Substituting Equation 4.47 with appropriate subscripts for the exit and throat pressures, as well as doing some simplifying, yields

$$\varepsilon = \frac{A_e}{A_t} = \frac{\dfrac{\dot{m}}{P_c\sqrt{\frac{2\gamma}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P_e}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}}}{\dfrac{\dot{m}}{P_c\sqrt{\frac{2}{\gamma-1}\frac{M}{R_u T_c}\left(\frac{P_t}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_t}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}}} = \frac{\sqrt{\left(\frac{P_t}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_t}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}}{\sqrt{\left(\frac{P_e}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}} \tag{4.49}$$

Equation 4.49 gives us a means of calculating the *expansion ratio* of the rocket engine using only the design parameters of the chamber pressure and the exit pressure. However, it also involves the throat pressure, which is not straightforward to determine. It is non-trivial to show that, by differentiating Equation 4.47 and realizing that the minimum occurs at the throat, the area of the throat can be found by

$$A_t = \frac{\dot{m}}{P_c\sqrt{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\frac{M}{R_u T_c}}}. \tag{4.50}$$

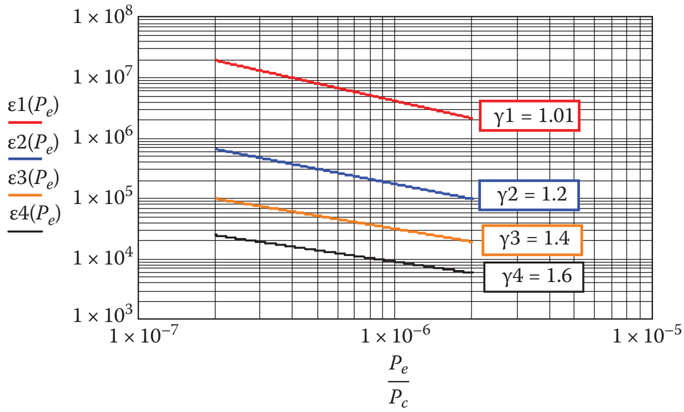


FIGURE 4.11
Area ratio versus pressure ratio for four different values of the isentropic expansion factor. As the isentropic expansion factor goes up, the ratio goes down.

Plugging Equation 4.50 into Equation 4.49 gives us

$$\varepsilon = \frac{A_e}{A_t} = \sqrt{\frac{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}{\left(\frac{P_e}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}}. \tag{4.51}$$

Now, we have an equation for the expansion ratio that is dependent only on the isentropic expansion factor, the exit pressure, and the chamber pressure of the rocket engine. Figure 4.11 shows the expansion ratio as a function of the pressure ratio and different isentropic expansion factors. Note that the higher the isentropic expansion factor is, the smaller the expansion ratio becomes. We should also note here, but we won't derive it, that Equation 4.51 can also be written in terms of the Mach number as

$$\varepsilon = \frac{A_e}{A_t} = \sqrt{\frac{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}{\left(\frac{P_e}{P_c}\right)^{\frac{2}{\gamma}}\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}} = \frac{1}{M_{\#}}\left(\frac{1+\frac{\gamma-1}{2}M_{\#}^2}{\frac{\gamma+1}{2}}\right)^{\frac{\gamma+1}{2(\gamma-1)}}. \tag{4.52}$$

4.4.2 Nozzle Design

Figure 4.12 shows a typical nozzle with an inlet area, an exit area, and a throat area as depicted. Standard rocket nozzles use a *nozzle divergence half-angle*, θ, of 15°. It is nontrivial to show that the optimum half-angle divergence is between 12° and 18° and is beyond the

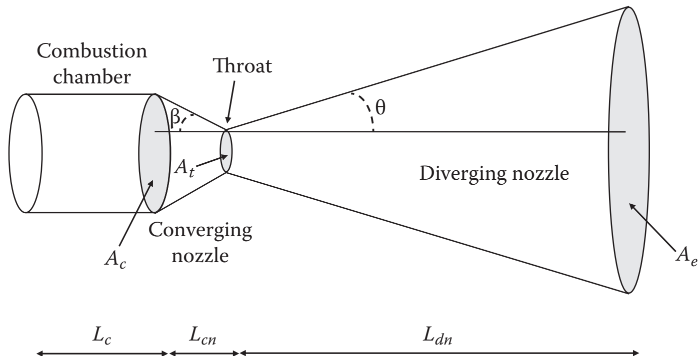


FIGURE 4.12
Converging–diverging nozzle design parameters are shown.

scope of this text. However, we will assume the median of that range is the standard. We can now use the expansion ratio and the nozzle divergence half-angle to determine the length of the diverging nozzle to be

$$L_{dn} = \sqrt{\frac{A_e}{\pi}} \frac{1}{\tan(\theta)}. \tag{4.53}$$

We can now determine the expansion ratio for our engine based on the properties of the engine propellants and performance we need, then we can use Equation 4.53 to determine how long the engine nozzle must be from the throat to the exit. What about the cone size from the inlet to the throat?

As shown in Figure 4.12, there is also a *nozzle convergence half-angle*, β, and it is optimal around 60°. Again, it is nontrivial to prove this, and we will accept it as standard here. There is also a *contraction ratio* of the inlet area to the throat area in that the pressure at the inlet should equal the combustion chamber pressure, and then it should converge to the throat area pressure. The standard rule for the contraction ratio is that the area of the inlet must be at least three times the area of the throat. The contraction ratio, χ, therefore, is

$$\chi = \frac{A_c}{A_t} \geq 3. \tag{4.54}$$

Or more simply put,

$$A_c \geq 3A_t. \tag{4.55}$$

With Equation 4.55 and the nozzle convergence half-angle, we can then determine the length of the converging cone between the combustion chamber and the throat. The length of the converging nozzle, L_{cn}, is

$$L_{cn} = \sqrt{\frac{A_c}{\pi}} \frac{1}{\tan(\beta)} = \sqrt{\frac{3A_t}{\pi}} \frac{1}{\tan(\beta)}. \tag{4.56}$$

It seems simple enough to basically take two appropriately sized nozzles and weld them together at their throats, thus giving us a convergent–divergent nozzle. The end result would look like two cones attached at their points. Unfortunately, the abrupt change from converging to diverging would create a shock wave as the flow went supersonic. The shock wave would likely disrupt the flow and put out the engine. At a minimal case, the shock wave would make the engine very inefficient. Sharp edges and supersonic flows are never a good combination.

Hence, the converging nozzle will start to curve gently into the throat; then, the curve will turn into a diverging one, smoothly transitioning to the cone. Actually, the design of two cones connected with a smooth throat transition is fine for a rocket nozzle. In fact, this is exactly how many modern rocket nozzles are constructed. The *de Laval* nozzle, as discussed in Section 2.4.1, is slightly different. The divergent part of the nozzle isn't a straight

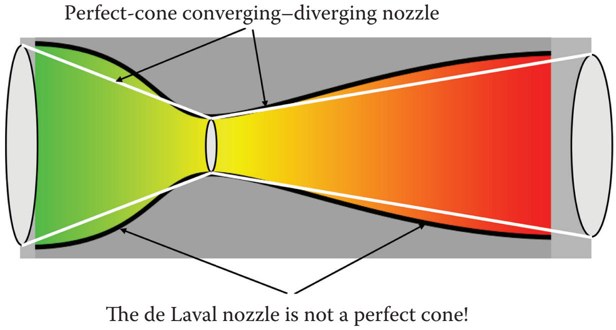


FIGURE 4.13
The converging–diverging nozzle compared to the de Laval nozzle. In actuality, the de Laval nozzle is usually shorter than the conical one.

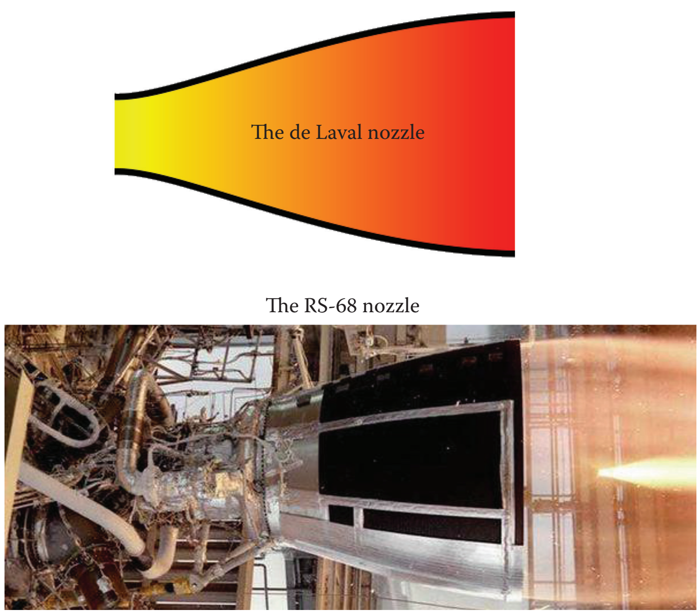


FIGURE 4.14
The RS-68 engine nozzle is a bell-shaped de Laval nozzle.

cone, as can be seen in Figure 4.13. It is a bell shape and is slightly shorter than the nozzles designed from the cone approach. The bell-shaped nozzle is slightly lighter than the cone simply because its construction can be shorter and uses less material. In some cases, the design complexity is not worth the trade in performance. In some cases, it is. Figure 4.14 shows the RS-68 engine nozzle. Notice that it is a bell-shaped system.

4.4.3 Properly Designed Nozzle

If a nozzle is designed for optimum performance, it will expand the flow so that the exit pressure of the nozzle is equal to the ambient pressure outside the rocket engine. In space, this would mean that the exit pressure should be zero. At sea level, the exit pressure should be 101,325 Pa. The exit pressure for the perfectly designed rocket nozzle would vary between the sea-level pressure to zero in real time as it ascended during launch. This would require the nozzle to change shape continuously throughout the flight, which is an impractical and extremely difficult engineering feat. Instead, rocket engines are designed for optimum exit pressures where it is expected that they will expend most of their fuel. If a thruster is planned only for use in space, it will be designed for a vacuum exit pressure. But why is this important?

Figure 4.15 shows three nozzles. The first nozzle is designed in such a way that its exit pressure is greater than the ambient pressure outside the rocket engine. In this case, the exhaust plume will expand in a diverging flow behind the engine. From simple vector physics, we can understand that some of the thrust is being converted to horizontal vector components and is not useful in lifting the rocket. This is called an *underexpanded* nozzle because the nozzle isn't large enough to allow the flow to expand to ambient pressure.

The second nozzle is similar in its inefficiency, but in an opposite way. In this case, the nozzle is designed for an exit pressure that is smaller than the outside ambient pressure. The exhaust flow plume will converge to a point behind the rocket. Again, some of the

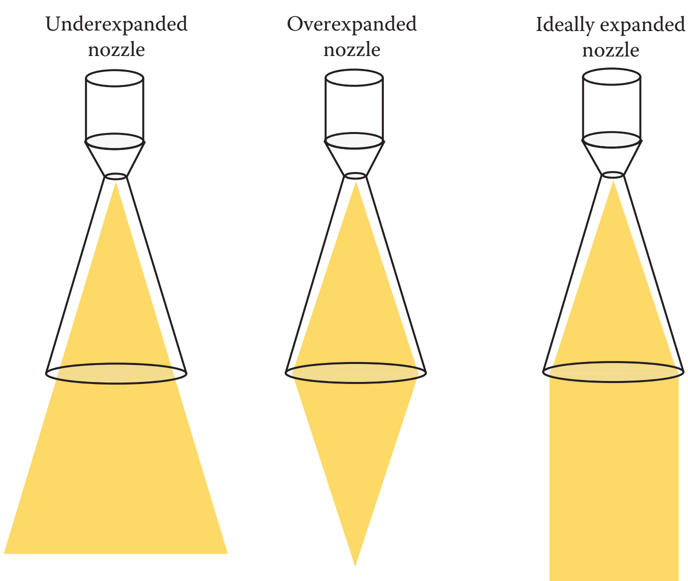


FIGURE 4.15
Proper nozzle expansion is imperative in order to efficiently generate thrust.

thrust is converted to horizontal vector components, and, therefore, the engine is not properly designed. This is an *overexpanded* nozzle.

If the nozzle is designed properly, the exhaust flow plume will exit the nozzle in a perfect cone with all of the thrust lifting the rocket. In this case, the exit pressure and the outside ambient pressure are equal. This is called an *ideally expanded* nozzle.

In some very extreme design cases, it is possible to create a nozzle that will create a thrust vector in the opposite direction of the flight path. If the nozzle is extremely underexpanded, the exhaust flow will exit the nozzle and converge so much so that it will actually turn 180° and flow in the direction of the rocket path—an extremely inefficient design, indeed.

4.4.4 Expansion Chamber Dimensions

Figure 4.16 shows the complete rocket engine flow path from the expansion or combustion chamber, through the converging nozzle, through the throat, and out the diverging end of the nozzle. To this point, we have learned how to calculate the size of the throat, the exit area of the nozzle, and the length of the nozzle. Another useful design parameter is the length of the expansion chamber. The combustion chamber of a liquid fuel engine is typically a cylinder. It is sized so that it will be large enough for the propellant liquids to fully mix and react together. The mixing length, L^*, of the chamber is found as

$$L^* = \frac{V_c}{A_t}, \quad (4.57)$$

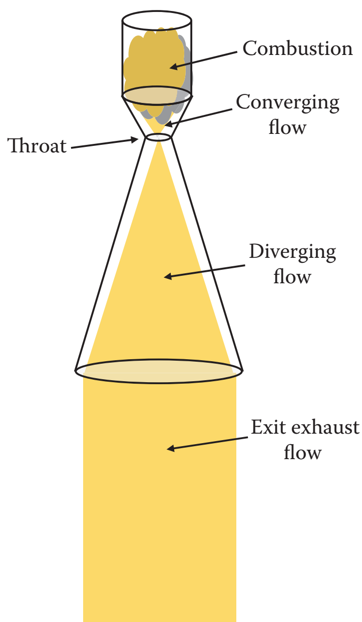


FIGURE 4.16
Shown is the flow through the rocket engine from the combustion chamber to the exit of the nozzle.

where V_c is the volume of the chamber and is found by calculating the volume of a cylinder, and, thus, Equation 4.57 becomes

$$L^* = \frac{V_c}{A_t} = \frac{\pi r_c^2 L_c}{A_t}, \tag{4.58}$$

where r_c is the radius of the cylinder. Solving for the length of the cylinder, L_c, is also the combustion chamber length and is

$$L_c = \frac{A_t L^*}{\pi r_c^2}. \tag{4.59}$$

This gives us the design length for the combustion chamber. We should realize here that the mixing length is a function of the types of propellant liquids used, but it ranges typically between about 0.5 and 1.5 m.

Because we learned how to calculate the area of the inlet nozzle, this tells us the cross-sectional area of the combustion chamber as they are equal. Equation 4.59 tells us how to determine the length of the chamber. The chamber must also be designed to withstand the high temperatures and pressures inside of it. In order to determine how thick the chamber should be, we need to understand what the stress will be in the cylinder walls. The stress, σ_c, is

$$\sigma_c = \frac{P_c r_c}{t_{wall}}, \tag{4.60}$$

where t_{wall} is the thickness of the combustion chamber wall. The stress is limited by the design material properties and is easily found in metal properties tables. Solving for the wall thickness results in

$$t_{wall} = \frac{P_c r_c}{\sigma_c}. \tag{4.61}$$

4.5 Rocket Engine Design Example

In this chapter thus far, we have developed the tools to completely design a rocket engine. We have skipped the examples up until now in order to spend the time to develop the tools necessary to perform a design of an engine. The equations we have developed are fairly complex and tedious and are best implemented through simulation and modeling. At this point, it would behoove the student to follow this design example using his or her favorite math-modeling software package like Mathcad®, MATLAB®, or equivalent, or even to write a program in a high-level computer language.

We will start with some givens. Figure 4.17 shows a complete rocket engine including the expansion chamber, convergent nozzle, throat, and diverging nozzle with all the geometrical variables marked and shown. So, to summarize, we are given the rocket engineering job to design an engine around these parameters:

$\gamma = 1.2$
$T_c = 3{,}500$
$M = 12$
$P_c = 20$ MPa
$P_e = P_o = 101$ KPa
$\varepsilon = 77.5$
$I_{sp} = 400$ sec at sea level
$F_{thrust} = 1.5$ MN
$\theta = 15°$
$\beta = 60°$
$\sigma_c = 55$ MPa

The variables we need to find are A_e, r_e, A_t, r_t, L_{dn}, A_c, r_c, L_{cn}, L_c, and t_{wall}.

The first step in our design process is to find some of the needed parameters for calculating the dimensions listed in Figure 4.17. Let us start by trying to find the exit area, the exit radius, and the throat area and radius. We start by finding the m-dot from the thrust equation:

$$F_{Thrust} = \dot{m}_{ve} + (P_e - P_o)A_e = \dot{m}C. \tag{4.62}$$

Because $P_e = P_o$, then the thrust is simply

$$F_{Thrust} = \dot{m}v_e = \dot{m}C. \tag{4.63}$$

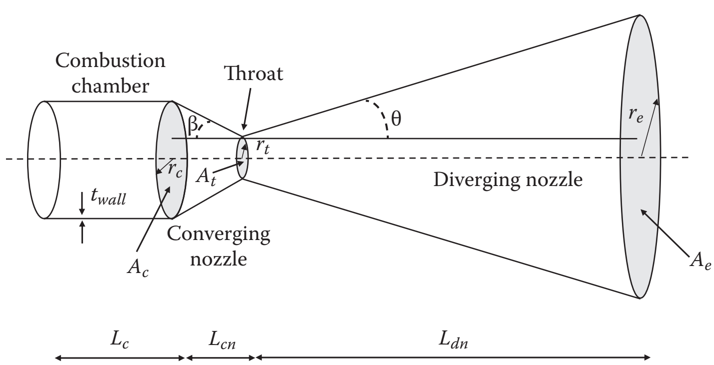


FIGURE 4.17
Design parameters for a rocket engine.

From Equation 4.63, it is clear that $v_e = C = gI_{sp}$ and, therefore,

$$v_e = (9.8\ \text{m/s}^2)(400\ \text{s}) = 3,920\ \text{m/s}. \tag{4.64}$$

The m-dot is then found by rewriting Equation 4.63 as

$$\dot{m} = \frac{F_{thrust}}{v_e} = \frac{1.5\ \text{MN}}{3,920\ \text{m/s}} = 382.65\ \text{kg/s}. \tag{4.65}$$

We can use Equation 4.47 and the exit pressure to calculate the exit area. At this point, it is a good idea to model the equation in a math-modeling software package. Figure 4.18 shows the exit area as a function of pressure. The point on the graph where the pressure equals the exit pressure is the exit area and is about 0.92 m^2. Figure 4.19 is a similar graph, but of the exit radius. From the graph, at the pressure equal to the exit pressure, we find that the radius of the nozzle is about 0.56 m.

At this point, we have the exit area and radius, and, from Equation 4.48 and the given expansion ratio of 77.5, we can find the throat area as

$$A_t = \frac{A_e}{\varepsilon} = \frac{\pi r_e^2}{\varepsilon} = \frac{(\pi)(0.56)^2}{77.5} = 0.0127\ \text{m}^2, \tag{4.66}$$

and

$$r_t = \sqrt{\frac{A_t}{\pi}} = \sqrt{\frac{0.0127}{\pi}} = 0.064\ \text{m}. \tag{4.67}$$

Calculating the length of the diverging cone is straightforward and is

$$L_{dn} = \sqrt{\frac{A_e - A_t}{\pi}}\frac{1}{\tan(\theta)} = \sqrt{\frac{0.92 - 0.0127}{\pi}}\frac{1}{\tan(15)} = 2.01\ \text{m}. \tag{4.68}$$

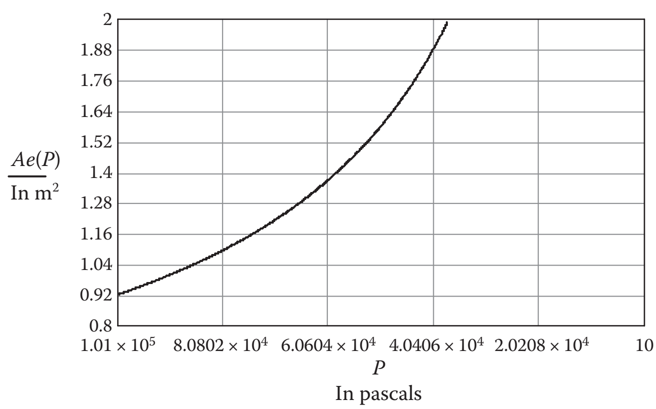


FIGURE 4.18
Nozzle exit area versus pressure.

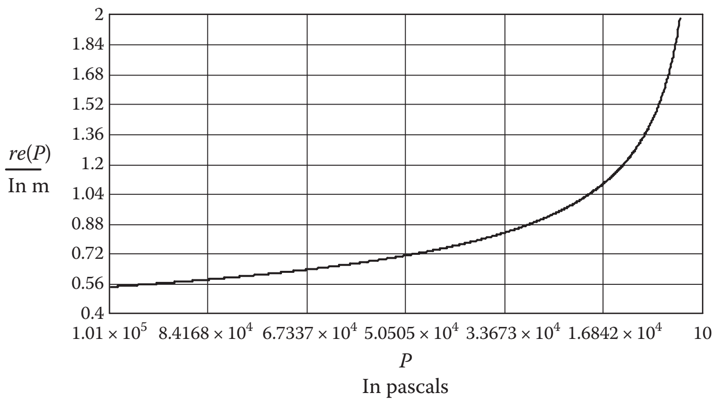


FIGURE 4.19
Shown is the nozzle exit radius versus pressure.

The area of the inlet at the combustion chamber is found from

$$A_c \geq 3A_t = 3(0.0127 \text{ m}^2) = 0.0381 \text{ m}^2. \tag{4.69}$$

And, likewise, the radius of the chamber is

$$r_c = \sqrt{\frac{A_c}{\pi}} = \sqrt{\frac{0.0381}{\pi}} = 0.11 \text{ m}. \tag{4.70}$$

The length of the converging nozzle is

$$L_{cn} = \sqrt{\frac{A_c - A_t}{\pi}} \frac{1}{\tan(\beta)} = \sqrt{\frac{0.0381 - 0.0127}{\pi}} \frac{1}{\tan(60)} = 0.052 \text{ m}. \tag{4.71}$$

Choosing the median L^* value of 1 m, the length of the combustion chamber is

$$L_c = \frac{A_t L^*}{\pi r_c^2} = \frac{(0.0127 \text{ m}^2)(1 \text{ m})}{\pi(0.11 \text{ m})^2} = 0.334 \text{ m}. \tag{4.72}$$

Finally, we can find the wall thickness of the combustion chamber if we assume the chamber is made of a nickel and copper alloy, which can withstand a stress of over 55 MPa. The minimum wall thickness of the combustion chamber is

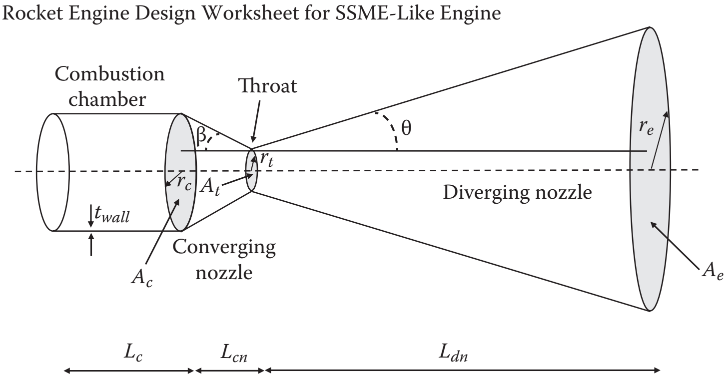
$$t_{wall} = \frac{P_c r_c}{\sigma_c} = \frac{(20{,}000{,}000 \text{ Pa})(0.11 \text{ m})}{2(55{,}000{,}000)} = 0.04 \text{ m}. \tag{4.73}$$

We have now developed our rocket engine. Figure 4.20 shows the engine design with all the parameters listed. This engine is within a few percent, give or take, of the SSME design. The nozzle exit area is a bit smaller because we designed our engine for sea level. The SSME nozzle is about 1.2 m in radius at the exit. Thus, using Figure 4.19, we can look at the graph where the radius is 1.2 m and determine the exit pressure the SSME is designed for, which is around 17 kPa. This corresponds to an altitude of somewhere between 12 and 13 km. The SSMEs are designed for optimal thrust at 12.5 km, so our model is right on target.

By the way, recall from Chapter 3 that the Space Shuttle goes through max-Q at about 11-km altitude. Is it coincidence or design that the engines are optimized for an altitude just past that? "Roger, Shuttle. You are go at throttle up!"

This section has given us the basic design tools for rocket engine design. The calculations used within it should be coded into a computer model so that all the user needs to do is to input the given components, and the model will give design output choices tabulated, as in Figure 4.20. That way, the readers can play around with the models and parameters and see what happens to the rocket engine designs as they are varied.

Rocket Engine Design Worksheet for SSME-Like Engine



Given parameters:

$\gamma = 1.2$
$T_c = 3{,}500$
$M = 12$
$P_c = 20$ MPa
$P_e = P_o = 101$ KPa
$\varepsilon = 77.5$
$I_{sp} = 400$ s at sea level
$F_{thrust} = 1.5$ MN
$\theta = 15°$
$\beta = 60°$
$\sigma_c = 55$ MPa

Calculated parameters:

$A_e = 0.92$ m^2
$r_e = 0.56$ m
$A_t = 0.0127$ m^2
$r_t = 0.064$ m
$L_{dn} = 2.01$ m
$A_c = 0.038$ m^2
$r_c = 0.11$ m
$L_{cn} = 0.052$ m
$L_c = 0.334$ m
$t_{wall} = 0.04$ m

FIGURE 4.20
Rocket engine design example parameters and calculated values. This engine design is very similar to the SSMEs.

4.6 Chapter Summary

In this chapter, we have built on our understanding of rocket science as formed in the previous chapters and have taken a step into rocket engineering. We started learning the basic components of the rocket engine in Section 4.1. After learning the details of the convergent–divergent nozzle and the combustion chamber, we began developing tools that would enable us to not only understand what goes on inside a rocket engine but also how to design one.

In Section 4.2, we learned of the all-important thermodynamic expansion properties of ideal gases that are truly the science behind rocket engine engineering. We learned that maintaining an isentropic flow within the engine enables us to make design calculations of the engine's physical dimension. We learned, in the subsequent sections, to manipulate that understanding.

In Sections 4.3 and 4.4, we took the isentropic process and thermodynamic properties a step further and developed equations for sizing the combustion chamber, converging nozzle, throat, and the diverging nozzle. These tools for rocket engine design finally allowed us to do some real rocket engineering in Section 4.5.

In the final section, we showed the complete design process for a rocket engine and, in essence, developed the design model for the SSMEs without realizing it. If any part of this book is learned in great detail, it should be the process in this section. A computer model of the design process should be made that will enable the student to make engineering decisions on rocket engine design that could be useful in a real-world day-to-day rocket engineer's job. Aside from all of that, this section was just plain fun.

Exercises

4.1 Define and describe the basic components of a rocket engine.

4.2 What is an exothermic reaction?

4.3 What is the stoichiometric ratio?

4.4 Calculate the number of moles of O_2 the SSMEs burn during launch.

4.5 Calculate the energy released by burning liquid H_2 and liquid O_2 together in the SSMEs during launch.

4.6 What type of engine doesn't need an igniter to spark the propellants to react?

4.7 Give some advantages and disadvantages of using a hypergolic engine.

4.8 What is thermal expansion?

4.9 What is an isentropic process?

4.10 What are the conditions for isentropic flow?

4.11 Define the isentropic expansion factor.

4.12 If a missile is traveling 3,000 km/h at an altitude where the speed of sound in the local atmosphere is 800 km/h, what is the Mach number?

4.13 A spacecraft slows down by atmospheric drag after reentry until it generates sonic booms at an altitude of 11 km where the vehicle's velocity is measured to be 1,063 km/h. What is the speed of sound at that altitude?

4.14 Discuss the importance of Equation 4.24 and how it determines the shape of a rocket nozzle.

4.15 What is choked flow?

4.16 Discuss the significant differences between Equations 4.36 and 4.37. What important fact does this difference tell us about the thrust of a rocket engine?

4.17 Define the expansion ratio.

4.18 The SSMEs have a nozzle exit radius of about 1.2 m. The expansion ratio is 77.5. What is the throat diameter of the rocket engine?

4.19 For an isentropic expansion factor of 1.3 and a Mach number at the exit of the rocket nozzle of 2.2, what is the expansion ratio?

4.20 In Exercise 4.19, assume a throat radius of 0.15 m. What is the exit nozzle radius?

4.21 In Exercise 4.20, what is the minimum combustion chamber radius?

4.22 In Exercises 4.19 and 4.20, determine the converging nozzle length. What is the diverging nozzle length?

4.23 In Exercises 4.19 through 4.22, determine the combustion chamber length and radius.

4.24 In Exercises 4.19 through 4.23, find the wall thickness of the combustion chamber if it is made of a material that can withstand a stress of 55 MPa.

4.25 Develop a computer model of the engine design process, as shown in this chapter. Generate graphs that describe the engine dimensions as functions of the pressure ratio and of the exit pressure. Use the model to learn how to optimize an engine design for a given external ambient pressure. In other words, learn how to optimize an engine for a particular altitude or in space operations.

5

Are All Rockets the Same?

Are all rockets the same? That is a loaded question. No two rockets are really the same unless they are manufactured in precisely the same way from exactly the same blueprints. But that really isn't the point of that question in the context used herein. The point of asking this question is to bring to light the fact that there are many different types of rocket engines, and they all do not necessarily function in the same way. Though it is likely that there will be very common components, such as an exit diverging nozzle, a combustion chamber, and some sort of propellant, it is just as likely that there will be components of the rocket engines that are completely specific to that type of engine. An example of this specific difference is that nuclear thermal rockets (NTRs) do not have a combustion chamber where chemical propellants are reacted together. Instead, they have an expansion chamber where a propellant liquid or gas is heated by the nuclear reactor core. That propellant expands as it is superheated, and then the common convergent–divergent nozzle approach comes into play.

Another example of how some components are specific to some engine types and not to others is the obvious difference of solid rocket motors and all others. Any rocket engines that require flowing between propellants as either fluids or gases will require pressure vessels or at least tanks, pumps, valves, and an assortment of other flow loop hardware. The solid motor has fuel and an oxidizer built in as a combustible solid material inside the housing, and the propellant itself already sits within the confines of the combustion chamber. No flow hardware is needed.

Thus, from the above two examples alone, we see immediate differences in the engine components and design. What we will do in this chapter is to discuss several different types of rocket engines to give the readers a flavor of how broad a range of knowledge the rocket scientist or the engineer must acquire.

5.1 Solid Rocket Engines

Perhaps the most widely understood and well known is the solid rocket engine. It is interesting to note that the rocketry community tends to refer to solid rocket engines as "solid rocket motors." They are referred to as *solid boosters* when the complete rocket system is being discussed. Solid rocket motors come in so many sizes and shapes that it would be difficult to discuss them all. They range from fireworks size to hobby rocket motors to upper-stage kick-motors to the solid rocket boosters (SRBs) on the Space Shuttle to every variation in between. Figure 5.1 shows some of these motors and the thrust they provide in comparison with each other. The figures are clearly not to scale, but their basic physical dimensions can be extrapolated from other scale references within the images.

Alliant Tech Systems
Hobby rocket motors ~ 10–30 N   Gem-40 ~ 500 KN   Space Shuttle SRBs ~ 12.45 MN

FIGURE 5.1
Solid rocket motors range from hobby rocket size to the Space Shuttle SRBs. (Images of Gem-40 and the Space Shuttle are courtesy of NASA.)

5.1.1 Basic Solid Motor Components

The key advantage to solid rocket motors is that they are fairly simple machines. There are no moving parts involved unless a thrust-vectoring control system is used. The propellant is typically stable and can be stored for years before use. Figure 5.2 shows a typical solid rocket motor and its basic components. At the top of the motor is an *igniter,* which is used to start the engine. Once the solid rocket is started, it can't be turned off until it burns itself out. Igniters can range from fuses like in bottle rockets to electrically activated components that generate enough heat quickly enough to spark the solid propellants to burn.

The propellant is known as the *grain* and is the bulk of the motor. The grain of typical solid rocket motors makes up about 85% of the rocket motor's total mass. The grain is mostly solid with a *burning surface* built into it. The burning surface is where the propellant is burned during operation. Some motors have a cylindrical channel along the central axis of the rocket, whereas the wall of the open cylindrical channel is the burning surface. Some engines have no hole in them, and they simply burn from the flat end of the grain. Others have more exotic burning surfaces, which we will discuss later.

Exterior to the grain is some sort of *thermal insulation barrier.* This barrier protects the outer *casing* of the motor from the extreme temperatures and pressures of the rocket motor. The casing is typically the only part of a solid motor that can be reused. The Space Shuttle's SRBs have a reusable casing that is recovered from the ocean after each launch (Figure 5.3). The casings are refurbished and then refilled with grain for future use. The Ariane 5 boosters have a similar design, but they are not reused. However, the cases do survive and are recovered for postflight inspection.

The part of the solid motor within the casing that houses the grain and the burning surface is the combustion chamber. As discussed in Chapter 4, we now understand the importance of the combustion chamber design and how it mates to the isentropic flow components of the rest of the rocket engine. Solid rocket motors implement various shapes and sizes to

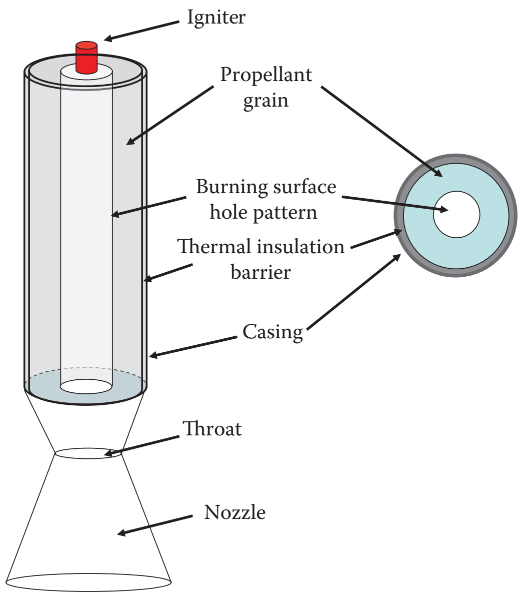


FIGURE 5.2
Schematic of the solid rocket motor.

FIGURE 5.3
Space Shuttle SRB reusable casing retrieval from the ocean. (Courtesy of NASA.)

optimize the combustion chamber for efficient burning of the propellant and for generating the desired thrust profile. Figure 5.4 shows several different solid motor configurations including the *cylindrical*, *spherical*, *conocyl*, and *finocyl* shapes. At the bottom of the combustion chamber (no matter which shape it is), the inlet to the convergent–divergent nozzle is connected where the flow is accelerated out of the engine to generate the desired thrust.

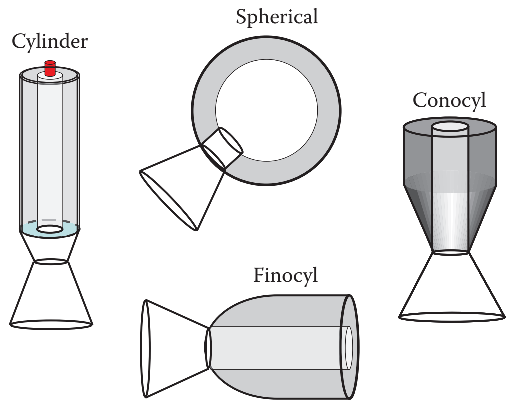


FIGURE 5.4
Shown are solid rocket motor grain configuration shapes.

5.1.2 Solid Propellant Composition

The grain of the solid rocket motor is an interesting mixture of materials that are practically the consistency of a rubber elastomer. In fact, the grain is a mixture of fuel, oxidizer, catalyst, some elastomer binder compound, plasticizer, curing agents, and, in some cases, other additives. The additives and binding materials may vary from manufacturer to manufacturer, but the most common fuel used is an elastomer binder and fuel combination. The two most common are *hydroxyl-terminated polybutadiene* (HTPB) and *polybutadiene acrylonitrile* (PBAN). HTPB is a clear viscous polymer belonging to the class known as polyols and is commonly used in the manufacture of polyurethane. PBAN is a copolymer and is less toxic during the curing process.

The binder, whether it be HTPB or PBAN, is mixed with an oxidizer. The most common oxidizer is ammonium perchlorate. Then, a catalyst and any other additives are mixed in, and the resulting compound is a solid rocket propellant. This mixture is commonly referred to as the *ammonium perchlorate composite propellant* (APCP).

The SRBs of the Space Shuttle are a good example of large-scale solid motors. According to the National Aeronautics and Space Administration (NASA) fact sheet for the SRB, their propellant composition is the following:

Ammonium perchlorate (oxidizer) = 69.8%,

Atomized aluminum powder (fuel) = 16%,

PBAN (binder and fuel) = 12%,

Epoxy curing agent = 2%,

Iron oxide powder (catalyst) = 0.2%.

The aluminum powder is added to improve the performance of the engine, and the iron oxide assists in the combustion process.

5.1.3 Solid Propellant Grain Configurations

Figure 5.5 shows diagrams of several grain configurations. As mentioned in Section 5.1.1, there is a burning surface where the propellant is ignited and burned, creating a

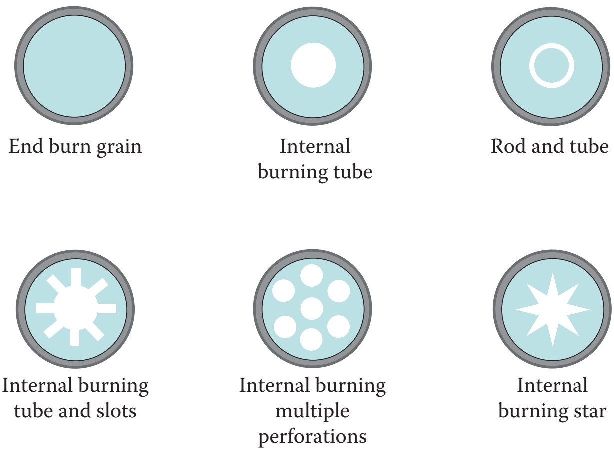


FIGURE 5.5
Images of solid rocket motor grain perforation configurations.

combustion chamber. Different geometrical configurations of the burning surface allow for different thrust profiles and performance capabilities. The burning surface can range from the flat end of the grain to a complex dendrite-shaped pattern. The Space Shuttle SRBs use an 11-point star shape.

The geometry of the channel is important in that the burning surface area is different. This channel is sometimes called the *perforation*. The burning surface area inside the perforation determines if the thrust increases, decreases, or remains constant during the rocket motor burn. There are three modes of burn:

1. *Regressive:* The thrust, pressure, and burning surface area decrease with burn time.
2. *Progressive:* The thrust, pressure, and burning surface area increase with burn time.
3. *Neutral:* Thrust, pressure, and burning surface area remain approximately constant throughout burn.

Figure 5.6 shows the profile of thrust as a function of time for the three types of grain burn modes listed above. Figure 5.7 illustrates several perforation designs and their respective grain burn modes.

Figure 5.8 shows the thrust profile of the Space Shuttle SRBs. Note that the burn is initially regressive, and then, at around 50 sec, the burn starts to increase again and is progressive until about 75 sec. Why is this? We have already discussed max-Q for the Space Shuttle. Well, this is the reason. The SRBs drop off to less thrust as the spacecraft pushes through max-Q, and then they burn with more thrust for a while until they turn back into a regressive thrust and finally burn out.

5.1.4 Burn Rate

The rate at which a solid propellant is burned inside the motor is mainly a function of the chamber pressure and follows Saint-Robert's law, which is

$$r = aP_c^n, \tag{5.1}$$

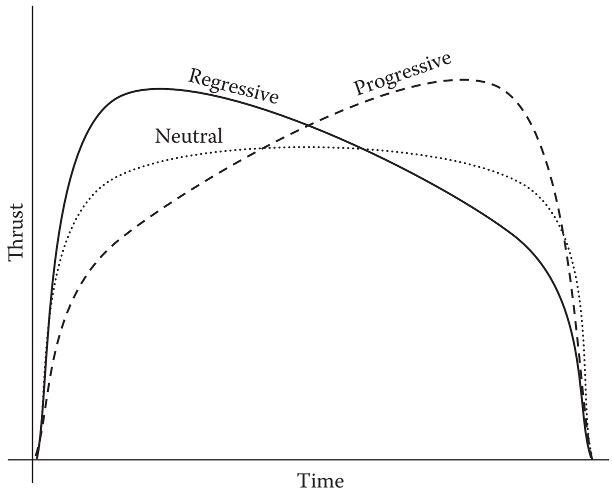


FIGURE 5.6
Solid rocket motor thrust versus time profiles.

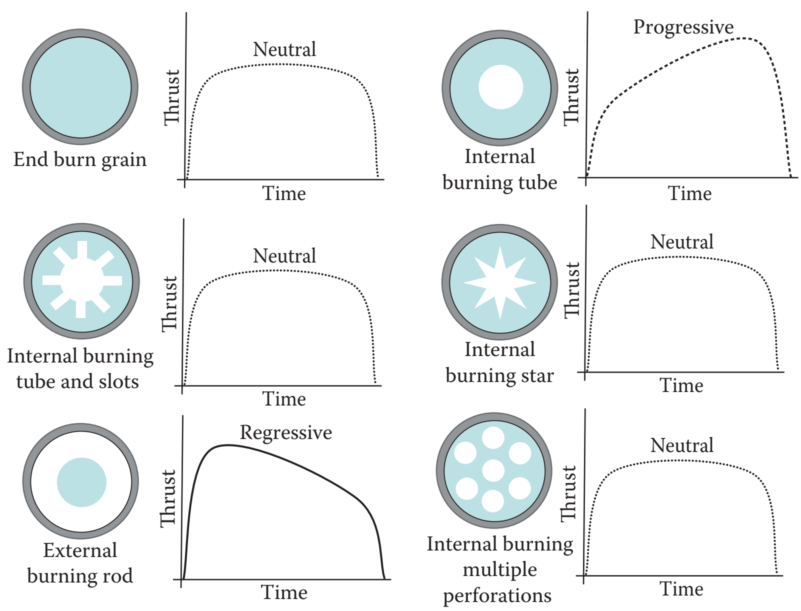


FIGURE 5.7
Solid rocket motor grain perforation configurations and their thrust versus time profiles.

where r is the burn rate, a is the *burn rate coefficient* and sometimes called the temperature coefficient and is based on the ambient grain temperature with units of mm/(sMPa^n), and n is the *pressure exponent*, also called the *combustion index*, and is dimensionless. Equation 5.1 tells us how fast the motor burns, but cannot be developed theoretically. The values of a and n are only found through measurement and are different for each propellant mixture recipe.

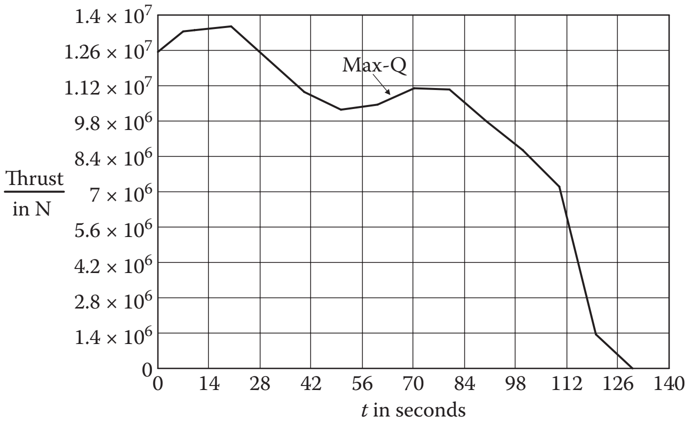


FIGURE 5.8
Space Shuttle SRB burn profile as a function of time after launch. Note that the profile is regressive, then progressive just around the max-Q, and regressive again until burnout.

5.1.4.1 Example 5.1: Burn Rate of the Space Shuttle SRBs

The Space Shuttle SRBs have a burn rate coefficient of $a = 5.606$ mm/(sMPa^n) and a combustion index of $n = 0.35$. Calculate the burn rate of the boosters if the chamber pressure of the booster is 4.3 MPa.

Using Equation 5.1, we see that

$$r = aP_c^n = \left(5.606\frac{\text{mm}}{\text{sMPa}^{0.35}}\right)(4.3\,\text{MPa})^{0.35} = 9.34 \ \text{mm/s}. \tag{5.2}$$

The burn rate of the propellant in the motor governs the m-dot as

$$\dot{m} = A_b r \rho_b = A_b \rho_b a P_c^n, \tag{5.3}$$

where A_b is the burning surface area, and ρ_b is the density of the solid propellant. Equation 5.3 can be used in conjunction with the rocket engine design equations given in Chapter 4 to develop a solid booster design. However, don't forget that the values defining the solid motor's burn rate are specific to an engine and have to be obtained from the manufacturer or through experiment.

5.2 Liquid Propellant Rocket Engines

We have already discussed liquid rocket engines to some degree throughout this book. Like the solid rocket engine is oftentimes called a *solid motor*, the liquid-fueled rocket engine is mostly referred to as a *rocket engine*. Figure 5.9 shows the basic components of a liquid rocket engine. The parts include fuel and oxidizer tanks, a gas generator, flow

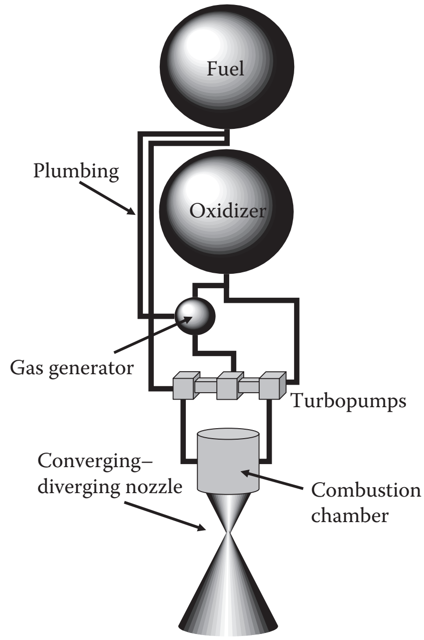


FIGURE 5.9
Schematic of a liquid fuel rocket engine.

plumbing, pump systems, a combustion chamber, and, of course, the nozzle. Because we have already discussed the pressure chamber and nozzles in great detail in Chapter 4, we will not repeat it here.

Some engines use cryogenic propellants, and some do not. The type of propellant liquids may add to the complexity of the tanks that hold them. In most cases, however, the propellants are fed out of the storage tanks by introducing a high static pressure into them. This is typically done by heating some of the propellant to its gaseous state and reintroducing it back into the tank. The propellant vapor pressurizes the tank, forcing it to flow out. Due to the high static pressure within the propellant tanks, they must be constructed of strong materials. Also, note that, in high-performance engines like the SSMEs or the RS-68 or the J-2X, a *turbopump* is used to flow the propellants from the tanks to the engines. The turbopumps are driven by propellant gas being ignited to spin the turbines within them. The turbopumps then force the propellants into the combustion chamber where the fuel and the oxidizer are mixed together through *injectors*.

The injectors are used to mix the propellants in the most efficient *stoichiometric ratio* for burning (see Chapter 4). Figure 5.10 illustrates a basic configuration for propellant injection into the combustion chamber. The propellants are forced through tiny nozzles and sprayed together in vapor streams where they mix and are then vaporized and combusted. In some cases, a premixer is used to mix the liquids together, and then they are sprayed into the combustion chamber.

In order to make the flow and mix of propellants fast and even, fairly complicated systems are sometimes required. Figure 5.11 shows the SSME propellant flow schematic. Both propellants are released from their tanks through an inlet valve and low-pressure pumps (the low-pressure fuel turbopump [LPFTP] and the low-pressure oxidizer turbopump [LPOTP]). The low-pressure pumps flow the propellants into two preburner chambers

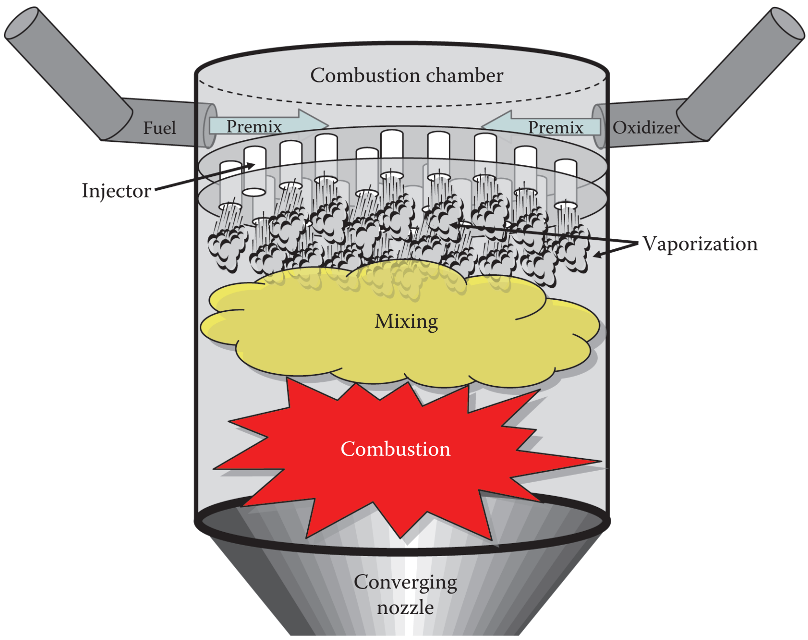


FIGURE 5.10
Schematic of a liquid fuel rocket engine injection, mixing, and combustion.

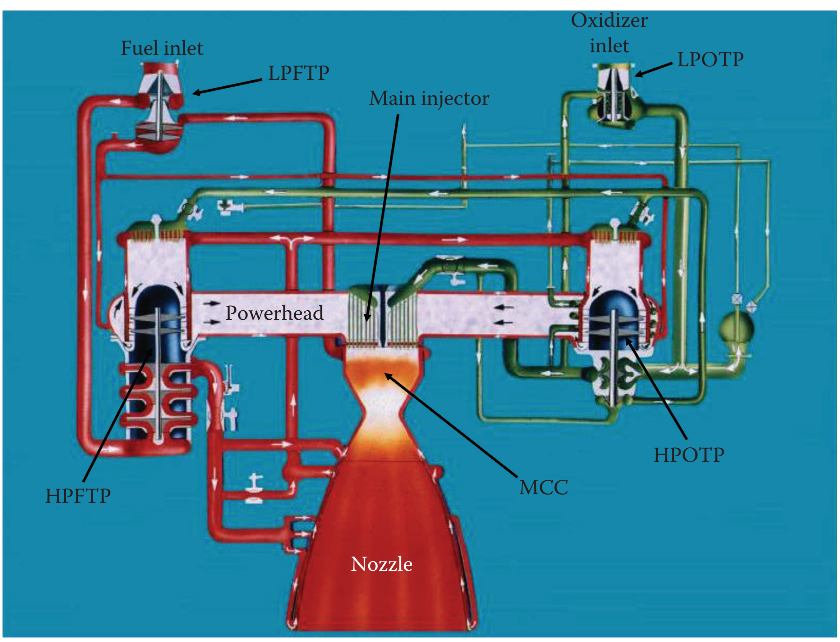


FIGURE 5.11
SSME propellant flow diagram. (Courtesy of NASA.)

that combust the propellants to drive high-pressure turbopumps. The high-pressure fuel turbopump (HPFTP) implements three turbines to force the liquid hydrogen fuel at high pressure through the rest of the flow system, as well as around the nozzle for cooling. The high-pressure oxidizer turbopump (HPOTP) forces the liquid oxygen through the engine systems as needed. The preburned propellants not only drive the turbopumps, but they also supply heat for the *power head* of the engine. This is where the oxidizer and fuel are heated and forced through the injectors, mixed, vaporized, and ignited into the *main combustion chamber*.

5.2.1 Cavitation

A problem with using high-pressure turbines in rocket engines that must be controlled is *cavitation*. This phenomenon occurs when propellers (or turbines in the case of a rocket engine) force a liquid to flow so fast near the surface of the turbine vane that it reaches a pressure level below its vapor pressure. Some of the fluid then boils off into vapor, forming a bubble. When the bubble flows into the cooler or higher-pressure region, it collapses back into a liquid state, which is much smaller in volume. The collapse of the bubble creates an acoustic wave within the flow. Depending on the flow characteristics, this acoustic wave can be quite intense. Figure 5.12 shows the blades of the SSME HPFTP after a test where the mix of oxidizer to fuel was too high, allowing cavitation to start. The blades of the turbine were damaged dramatically during the test.

5.2.2 Pogo

Another effect within liquid fuel rocket engines that can reduce performance and be damaging to the engine is called *pogo*. This effect is created when the propellant is accelerated through the pump inlet due to the thrust of the rocket. This can be detrimental in that the increase in pressure at the pumps will change the combustion process slightly because the flow rate changed. As a result, the thrust will change, once again placing a different acceleration at the pump inlet creating a different change in flow rate into the combustion chamber. It is clear that this is an uncontrolled feedback loop between the propellant flow and the thrust, which, in turn, can cause oscillations and even chaotic fluctuations in the

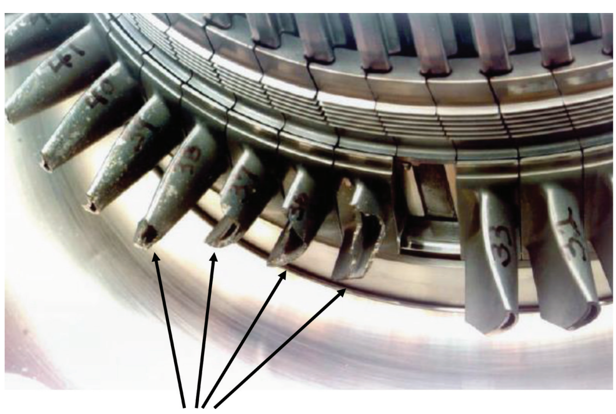


FIGURE 5.12
High oxygen/hydrogen ratio causes cavitation damage of SSME.

thrust profile. These oscillatory pressures within the engine can cause severe damage to the components. The way to fix this is to foresee the problem. Instead of leaving the uncontrolled feedback loop to run wild, a flow capacitor is placed into the system to create negative feedback. This flow capacitor is nothing more than a small volume of extra propellant. The capacitive volume either injects extra propellant at the inlet pump if the pressure is too low or sucks in extra propellant and, therefore, removes it from the flow if the pressure is too high at the inlet pump.

5.2.3 Cooling the Engine

Figure 5.13 shows a close-up of the SSME nozzle. Note that there are pipes running down the nozzle connecting to channels that are rings around the bell part of the nozzle. The pipes flow liquid hydrogen fuel into these rings, which are known as *cooling channels*. The cold liquid propellant flows around the nozzle to keep it cool for two main reasons. The simplest of the reasons is for structural integrity. The temperature and pressure inside the SSME nozzle are quite high, placing the material in a very extreme environment. Keeping the nozzle wall materials cool helps maintain the material strength. The other reason is to keep the temperature of the nozzle walls as constant as possible. Hot spots can cause the flow to be disturbed and, therefore, will make the engine less efficient. Cooling the engine this way is called *regenerative cooling*.

With some rocket engines, the nozzle walls are made up of a very-high-temperature material. Once the engine heats up, the nozzle will reach a state of thermal equilibrium, and it will glow white or red hot. The excess heat is radiated away into space. This is called *radiation cooling*.

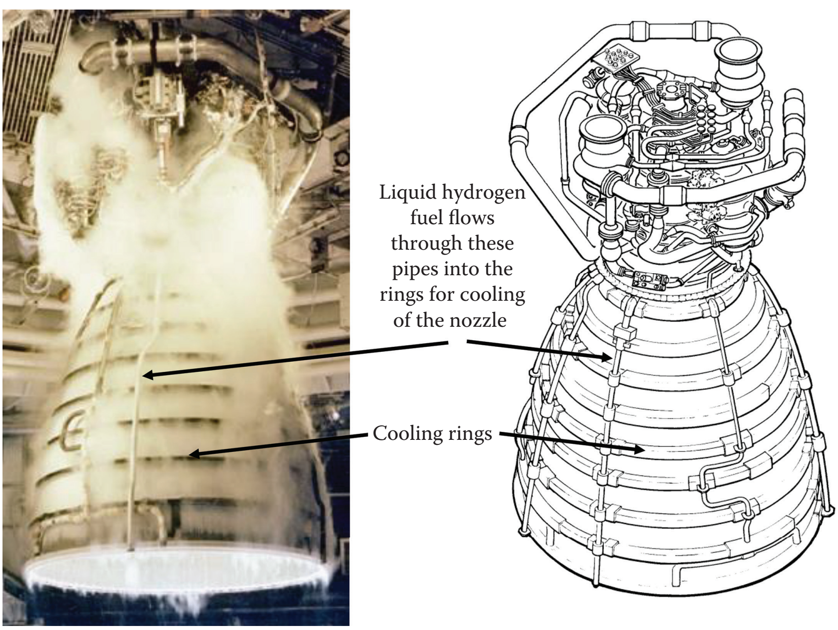


FIGURE 5.13
Regenerative cooling of the SSME nozzle. (Courtesy of NASA.)

5.2.4 A Real-World Perspective: The SSME Ignition Sequence

At the beginning of a Space Shuttle launch, sparks are seen shooting out of igniters near the base of the engine nozzles. These sparks at the base of the SSMEs are started up in order to keep the hydrogen propellant from pooling beneath them. The power head is fired up on each of the SSMEs to within a tenth of a second of each other in sequential order. They don't start up at the same time. Upon the start-up of each engine, they are pointed as far away from each other as possible. This is because their gimbals are set free from the hydraulic controls, allowing them to jump wildly around at first, and, if the nozzles collided, they could be damaged. They are set free so that the ignition and initiation of thrust don't damage the hydraulic thrust vector control mechanisms, which are called *steering linkages*.

Before full power is reached, the exhaust flow separates from the inside walls of the nozzle, disrupting the flow. It is this disruption in flow that causes the random vectors in the thrust, which jolt the nozzles around wildly. Once the engines have reached full-start status, the steering linkages are reconnected to the engines, and they are then under control of the thrust vectoring system. The engines are gimbaled to the optimum vector for liftoff, which causes the shuttle to tilt forward. Shuttle engineers often refer to this as the "twang." The twang motion settles, and the Space Shuttle is then in the proper orientation for launch. The SRBs are fired up, and the explosive *hold-down nuts* are blown free. If the nuts fail to blow free, the stress of the SSMEs and the two SRBs is enough to break the bolts off from their moorings. At this point, the Space Shuttle has lifted off.

5.3 Hybrid Rocket Engines

In Sections 5.1 and 5.2, we discussed the solid rocket and the liquid rocket engines, respectively. The solid uses a mixture of fuel and oxidizer that solidifies into the propellant material. The liquid engine uses a liquid oxidizer and fuel and mixes them together in a combustion process. It is possible to use a solidified fuel only and flow an oxidizer through the perforation. This type of engine is called a *hybrid* rocket engine. Figure 5.14 shows a schematic of a hybrid rocket. Gas pressurization is generated by heating some of the liquid oxidizer similar to the way it is done for a liquid engine. The oxidizer is flowed through the perforation of the solid fuel where it is ignited. Oxidizer is only present on the burning surface of the solid fuel, and, therefore, it will only burn when the oxidizer is flowing. This concept allows for the shutdown and restarts of the engine, which cannot be accomplished with a solid motor, as discussed in Section 5.2.4. Also, various perforation configurations can be implemented as with a typical solid motor to alter the burn rates and thrust profiles.

It is also possible to use a liquid propellant and a solid oxidizer. A rocket engine of this type is called a *reverse hybrid*. In such an engine, liquid hydrogen would be burned with solid oxygen.

A recent example of a hybrid engine being used successfully is that of SpaceShipOne (see Chapter 1). SpaceShipOne implemented a four-port perforated, solid fuel motor (HTPB) and nitrous oxide (N_2O) as an oxidizer. The rocket produced a thrust of 74 kN with a specific impulse of 250 sec. The engine had a burn time of 87 sec. The engine was of very simple design, as can be seen in the schematic in Figure 5.15.

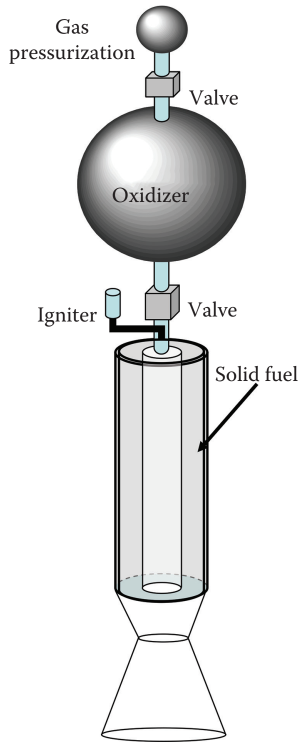


FIGURE 5.14
Schematic of a hybrid rocket engine.

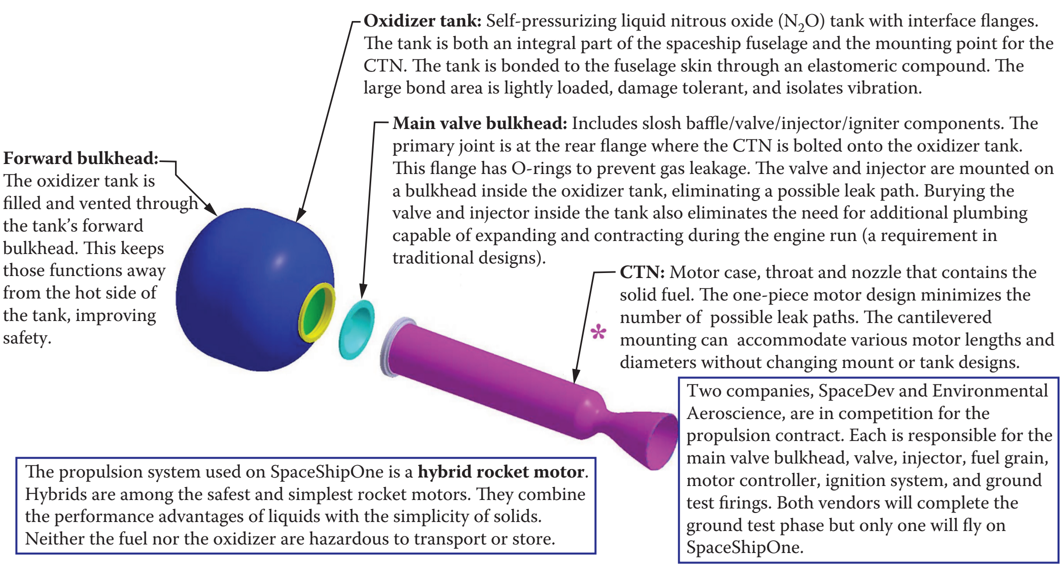


FIGURE 5.15
The hybrid engine of the SpaceShipOne. (Courtesy of Scaled Composites. SpaceShipOne is a Paul G. Allen Project.)

5.4 Electric Rocket Engines

To this point, we have discussed rocket engines that react or burn some type of propellant in order to generate thrust. Electric propulsion is a completely different concept. The electric rocket uses stored electrical energy in some clever manner to generate thrust. Most electric propulsion concepts are not designed for high thrust. Instead, they are very efficient engines that are used to generate very high specific impulses. The electric propulsion concept dates back to Robert Goddard who, in 1906, wrote about the concept in his personal notes, and, in 1911, when Tsiolkovsky actually published the idea.

5.4.1 Electrostatic Engines

Electrostatic engines make use of static electric fields in order to accelerate a propellant material. The driving physical force is the *electrostatic force,* which is governed by Coulomb's law. Coulomb's law is stated as

> The scalar magnitude of the electrostatic force between any two electric point charges is directly proportional to the product of the two charges and inversely proportional to the square of the distance between them. (Coulomb 1785a, pp. 569–577)

Mathematically, Coulomb's law is

$$F = k\frac{q_1 q_2}{r^2}, \tag{5.4}$$

where q_1 and q_2 are the charges in coulombs, k is the Coulomb's constant, and r is the distance between them. The electric field, E, in volts per meter created by a single point charge, q, is

$$E = k\frac{q}{r^2}. \tag{5.5}$$

If two parallel plates at a distance, d, from each other are charged (as shown in Figure 5.16) where one is positively charged and the other is negatively charged, the electric field strength between them is given as

$$E = -\frac{V}{d}, \tag{5.6}$$

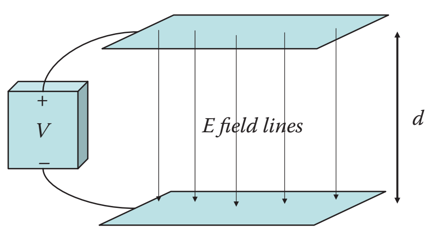


FIGURE 5.16
Electric field lines between two parallel conductor plates separated by a distance d, and connected to a voltage source as shown.

where V is the electric potential difference between the plates in volts, and d is the distance between them. If a charge, q, is placed between the plates in the electric field, the force on that charge is

$$F = qE = -q\frac{V}{d}. \tag{5.7}$$

The work done on the charge by the electric field is found as

$$W = Fs. \tag{5.8}$$

In this equation, s is the total distance the charged particle could move. Assume the charged particle is an ion with a positive charge and is released from the surface of the positively charged plate. The field will move the particle to the opposite plate. Therefore, the distance s becomes d, and we then have

$$W = Fs = Fd = qEd = -qV. \tag{5.9}$$

The kinetic energy of the electron can be equated to the work in Equation 5.9 to give

$$\frac{1}{2}mv^2 = W = -qV. \tag{5.10}$$

Solving for the velocity achieved by the particle results in

$$v = \sqrt{\frac{2(-qV)}{m}}. \tag{5.11}$$

Note that the minus sign will be taken care of by either the charge of the particle or the voltage drop, and all the values will multiply together to be positive before taking the square root. So, the velocity calculated in Equation 5.11 will be a real number, and, for simplicity, the minus sign can be dropped.

Now, consider the schematic shown in Figure 5.17. This is a schematic of an *ion thruster*. Gas particles are flowed into a chamber where they are bombarded with a stream of electrons. The gas is ionized, and a plasma mix of electrons and ions fills the chamber. The plasma is then flowed past a screen, which is the positively charged plate of a parallel plate pair. A distance, d, from that screen is a second screen, which is negatively charged. The potential between the two screens is V and is maintained by connecting a high-voltage power supply to them, as shown in the figure. Equation 5.11 tells us the exit velocity of the ions as they leave the system. We also have to place an electron gun just outside the system to fire electrons into the ion exhaust stream, or the entire system will eventually have a net negative charge. This is the description of an ion thruster.

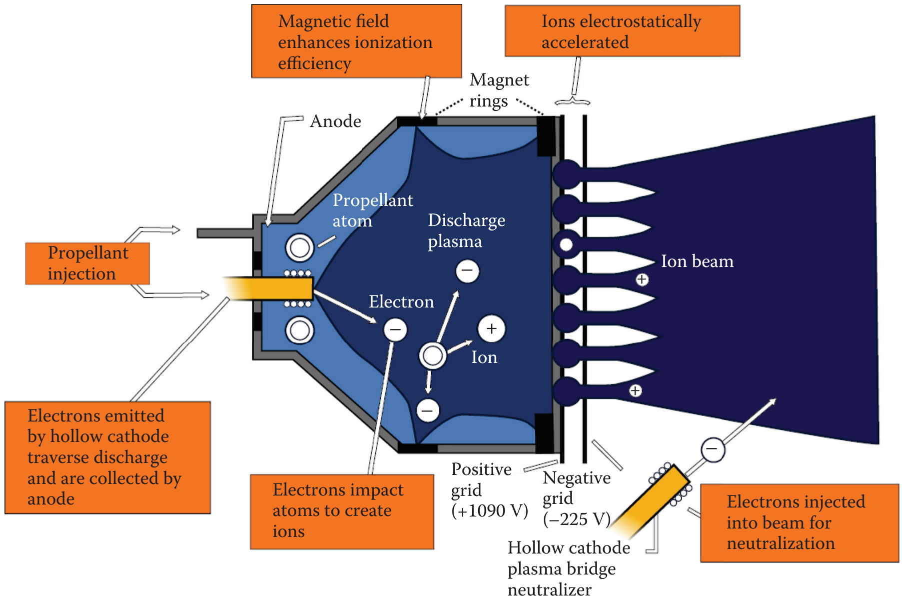


FIGURE 5.17
Schematic of the DS-1 ion thruster. The positive and negative grids shown are the conductor plates depicted in Figure 5.16. (Courtesy of NASA.)

If our thruster has a continuous mass flow rate of ions through it, then the thrust generated by the thruster is

$$F_{thrust} = \dot{m}v = \dot{m}\sqrt{\frac{2qV}{m}}\,. \tag{5.12}$$

In this case, the m is the mass of the individual ion, and its charge is $1.602 \times 10^{-19}\ C$.

5.4.2 Example 5.2: The Deep Space Probe's NASA Solar Technology Application Readiness Ion Engine

NASA's Deep Space Probe DS1 used a NASA Solar Technology Application Readiness (NSTAR) ion engine, as shown in Figure 5.18. The accelerator grids had a high-voltage potential difference of about 1,000 V and used xenon gas for propellant. What was the thrust and specific impulse supplied by the thruster if it continuously flowed xenon gas through the engine for 20 months? Assume a total fuel mass of 117.5 kg.

First, we must find the exit velocity of the exhaust material, which is xenon ions. The mass of a xenon ion is basically the mass of a xenon atom and, in kilograms, is found by dividing the molecular weight by Avogadro's number or about 2.18×10^{-25} kg per atom. The velocity is

$$v = \sqrt{\frac{2qV}{m}} = \sqrt{\frac{2(1.602\times 10^{-19}\,C)1,000\,\text{V}}{2.18\times 10^{-25}\,\text{kg}}} = 38,327\ \text{m/s}. \tag{5.13}$$

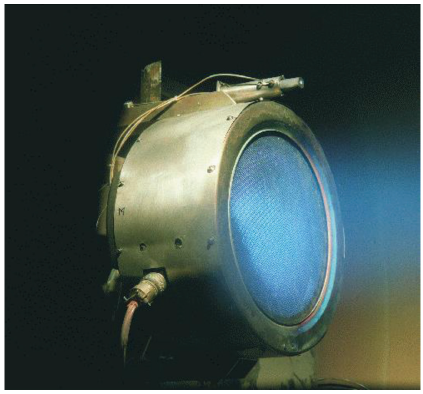


FIGURE 5.18
The DS-1 NSTAR ion engine. (Courtesy of NASA.)

Realizing that, in this case, the exit velocity is the equivalent velocity, the specific impulse of the engine is

$$I_{sp} = \frac{C}{g} = \frac{v}{g} = \frac{38,327\,\mathrm{m/s}}{9.8\,\mathrm{m/s^2}} = 3,910\,\mathrm{s}. \tag{5.14}$$

The average m-dot of the system is found by dividing the total mass of the fuel by the total thrust time of 20 months. This gives us an m-dot of about 1.6×10^{-6} kg/sec. The thrust, therefore, is found from Equation 5.12 to be

$$F_{thrust} = \dot{m}v = (1.6\times10^{-6}\,\mathrm{kg/s})(38,327\,\mathrm{m/s}) = 0.115\,\mathrm{N} = 115\,\mathrm{mN}. \tag{5.15}$$

Hence, the specific impulse and thrust of the NSTAR are calculated to be 3,910 sec and 115 mN (micronewtons), respectively. NASA references the values of specific impulse and thrust for this engine to be 3,100 sec and 92 mN. Why are our calculations different from the actual values?

In order to understand why, we must recall the actual physical construction configuration of the engine. A quick review of Figures 5.17 and 5.18 illustrates the answer. The xenon ions are accelerated between the charged screens. Some of these ions, however, do not make it out of the thruster. The negatively charged screen near the exit of the thruster captures some of the positively charged ions. An examination of the photo in Figure 5.18 of the engine shows that there is a significant portion of the exit nozzle blocked by the screen material. This is the main reason for the difference. The difference is even slightly worse because the actual voltage drop between the anode and cathode acceleration screens is about 1,300 V, as opposed to the 1,000 V we used in our calculations. The ions impinging on the molybdenum screens turn out to be the most detrimental force on this type of thruster, and this is called *screen erosion*. However, the NSTAR engine did function for more than a year and a half with little degradation in performance.

Another type of electrostatic thruster is the *Hall thruster,* which also uses an electrostatic field to accelerate xenon ions to high-exhaust velocities. Figure 5.19 shows a typical Hall thruster schematic. No grids are used. A strong magnetic field is supplied by

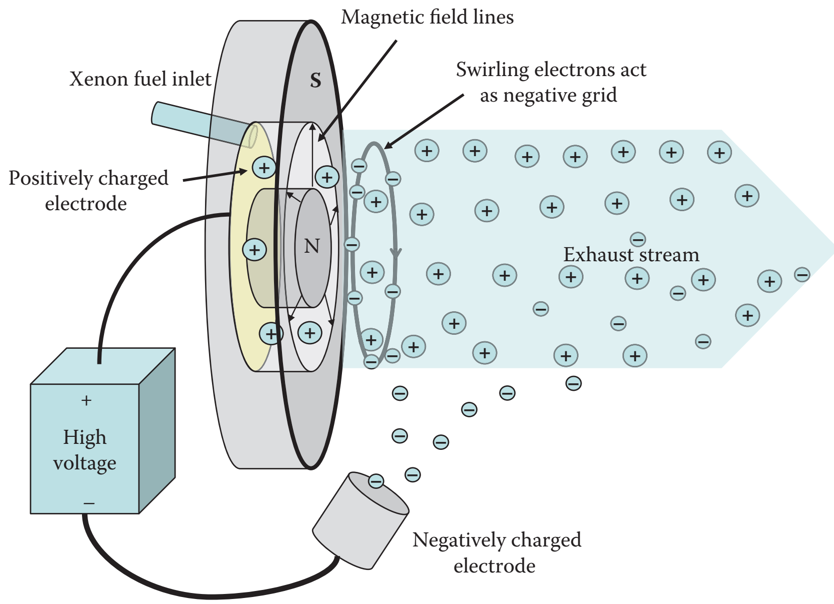


FIGURE 5.19
Schematic of a Hall effect thruster.

electromagnets that trap the electrons in place at the exit of the engine, acting as sort of a virtual negatively charged screen. This screen of electrons swirls about the axis of the thruster due to the interaction of their charge, the radial magnetic field, and the static electric field. The electrons swirling about are important to the thrust aspect of the Hall thruster as they act as a negatively charged screen with which to accelerate the ions. The swirling electrons are also very important in that they are used for charge neutralization as they recombine with some of the ions as they are thrust out of the engine through the area with the high density of swirling electrons. The ions are accelerated due to a potential across the anode and the swirling electron screen in basically the same premise as with the ion thruster discussed in Section 5.4.1. There is also a cathode just outside of the electron screen that adds to this effect and is also used for charge neutralization. Although the ions do have a spiraling motion imparted to them by the magnetic field, it is much less dominant to them as is the electric field simply due to the mass of the ions being much greater than the mass of the electrons. Therefore, the ions are not trapped by the magnetic field and are accelerated through the electron swirl outward from the engine. The quickly accelerated ions pull some of the electrons along with them, reducing the need for a lot of charge neutralization. There is a cathode neutralizer to account for any charge difference that may occur because the net change in charge of the system must be zero. A typical Hall thruster can deliver 80 mN of thrust with a specific impulse of over 1,500 sec and uses a few kilowatts of power. Figure 5.20 shows a 2-kW Hall thruster in operation. Note the brightness of the plasma within the anode ring and at the cathode neutralizer.

Another form of electrostatic propulsion is that of the *field emission electric propulsion* (FEEP) thruster. They are essentially the same as the aforementioned ion thruster except that the ions are supplied by a liquid metal source, such as cesium. FEEP thrusters supply very low thrust and are only useful for very-low-thrust applications on the order of micronewtons.

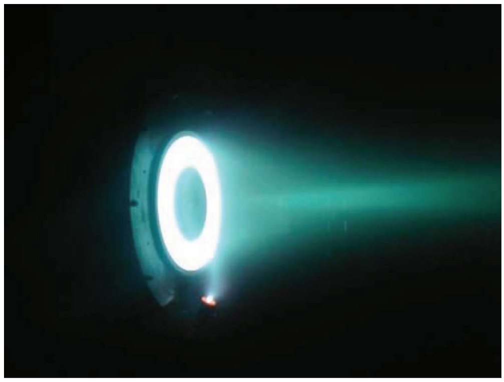


FIGURE 5.20
Image of a 2-kW Hall effect thruster in operation.

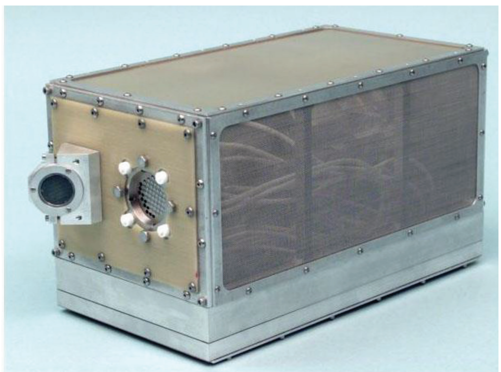


FIGURE 5.21
Image of a 20-μN colloid thruster. (Courtesy of NASA.)

A *colloid thruster* also works just like an ion thruster, but, instead of ions being used as propellant, a liquid spray is used. The liquid droplets are charged and are then accelerated by an electrostatic field. Figure 5.21 shows a picture of a colloid thruster. Like the FEEP, this type of thruster is a micronewton-class engine and only good for fine adjustment, station keeping, and attitude control.

5.4.3 Electrothermal Engines

Electrothermal engines use electric and magnetic fields in order to improve the performance of a propellant. This is done by increasing the thermal energy of the system by turning the propellant into a hot plasma by arcing an electric current flow through it, ionizing it with microwaves, or ionizing it with radio waves. The electrothermal engine might also make use of the electromagnetic fields to accelerate the ionized propellants. A typical example of an electrothermal engine is the Variable Specific Impulse Magnetoplasma Rocket (VASIMR) concept invented by astronaut Franklin Chang-Diaz. Figure 5.22 is a photo of the VASIMR tested and a diagram of the concept. The engine consists of superconducting magnetic cells, a plasma source, a radio- frequency (RF) booster, and magnetic

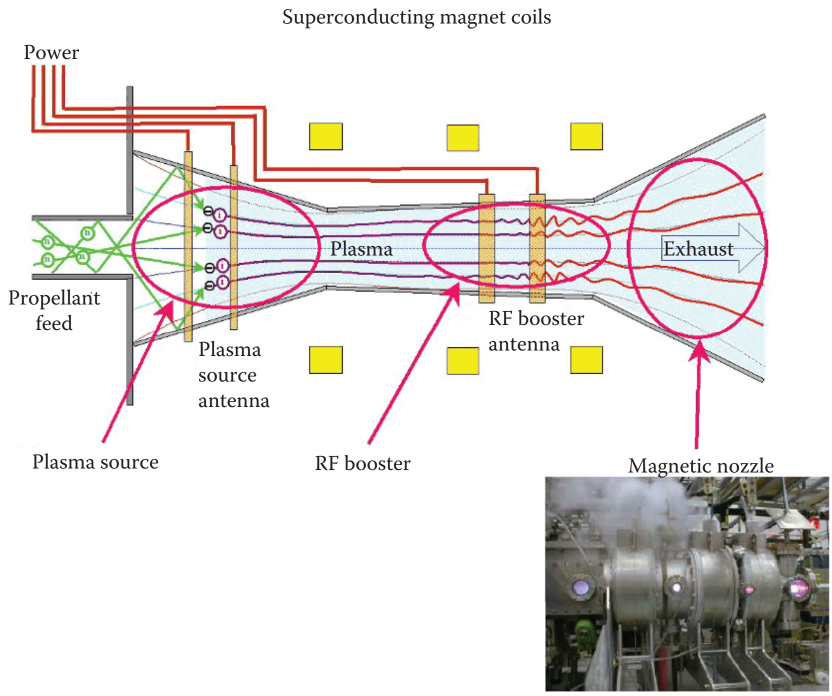


FIGURE 5.22
Schematic and image of the VASIMR concept invented by astronaut Franklin Chang-Diaz. (Courtesy of NASA.)

field lines shaped to act as a nozzle. A neutral gas is injected into an ionization chamber where the plasma energy is boosted by RF electromagnetic waves. The ionized plasma is accelerated by the magnetic nozzle to generate thrust. The VASIMR can generate specific impulses in a large range between 3,000 and 30,000 sec with thrust up to half of a newton.

5.4.4 Electromagnetic Engines

Electromagnetic engines operate mostly through the *Lorentz force* interaction between charged particles and electric and magnetic fields. The easiest-to-understand engine of this type is the *pulsed plasma thruster* (PPT).

The basics of the PPT are not at all unlike a rail gun. In fact, the function is practically identical. Figure 5.23 shows the basic schematic for an electromagnetic engine. A high-voltage power supply is connected across the electrodes of a capacitor to charge it. The capacitor is connected through a switch to electrode rails, as shown in the figure. When the switch is closed, the capacitor discharges rapidly, allowing a current flow between the rails, either through a physical piece of conductor, such as a metal bar, or a plasma arc that can be initiated in a propellant gas. The current loop created by the completed circuit generates a strong vector magnetic field, **B**, out of the plane of the circuit in the negative **z** direction, as shown in Figure 5.23. The force on the bar or plasma is due to the Lorentz force, $\mathbf{I}d \times \mathbf{B}$, and is calculated as

$$F = m\frac{dv}{dt} = Id \times B = IdB_o(x), \tag{5.16}$$

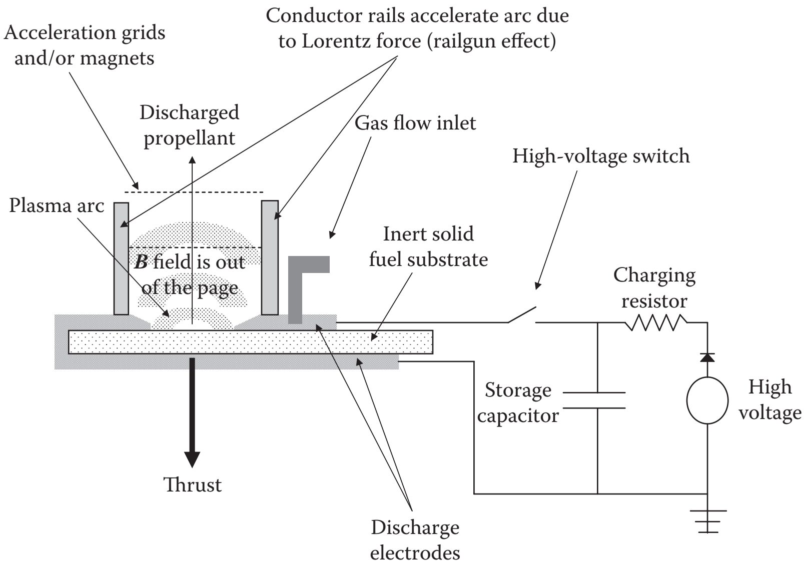


FIGURE 5.23
Schematic diagram of a pulsed plasma thruster.

where I is the current in amperes, B_o is the magnitude of the magnetic field, m is the mass of the bar of propellant in the current flow, d is the length of the bar, v is the velocity of the bar, and **x** is the vector direction out of the thruster along the axis of the rails. Integrating Equation 5.16 and realizing that the motion is all in the **x** direction, we can solve for the scalar velocity, which is

$$v(t) = \frac{IdB_o t}{m}. \tag{5.17}$$

Integrating Equation 5.17 gives us the position as a function of time:

$$x(t) = \frac{IdB_o t^2}{2m}. \tag{5.18}$$

Assume x to be the finite length of the electrode rails, and solving for t yields

$$t = \sqrt{\frac{2xm}{IdB_o}}. \tag{5.19}$$

Substituting Equation 5.19 into Equation 5.17 results in an equation for the exit velocity of the bar as a function of design space parameters only. The resulting equation is

$$v(t) = \frac{IdB_o t}{m} = \frac{IdB_o}{m}\sqrt{\frac{2xm}{IdB_o}}. \tag{5.20}$$

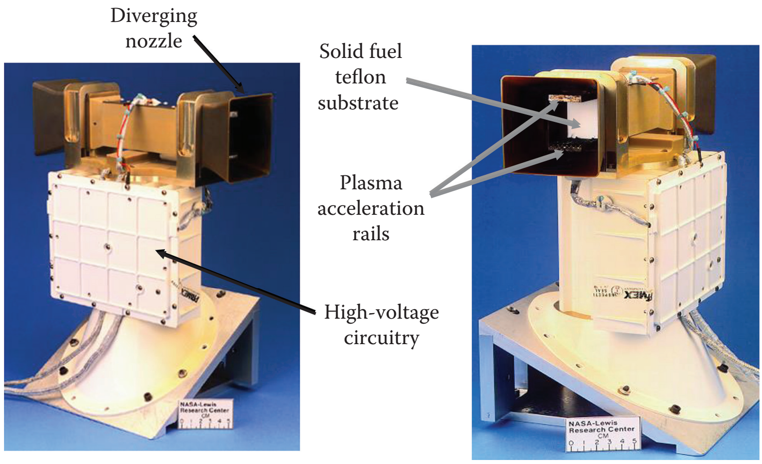


FIGURE 5.24
EO-1 PPT engine prototype. (Courtesy of NASA.)

In many PPTs, the propellant is a solid material like Teflon™ that is vaporized during the initiation of the arc between the rails. The m-dot of the thruster is based on how rapidly the capacitor can be recharged and fired again and on how much surface area of Teflon is burned off with each arc. Figure 5.24 shows a photo of the Earth Observer 1 (EO-1) PPT engine launched in 2000. The thruster was developed at NASA Glenn Research Center in Cleveland, Ohio, and demonstrated 860 μN of thrust, an exhaust velocity of 13,700 m/sec, an m-dot of 8.3×10^{-8} kg/sec, and an I_{sp} of about 1,400 sec.

Other configurations of this type of thruster implement varying electromagnetic fields, different geometrical configurations, and various propellant gases. But the premise of electromagnetic thrusters is that they implement the Lorentz force in some manner. Other thrusters of this type are often called *magnetoplasmadynamic thrusters*, *pulsed inductive thrusters*, and even *electrodeless plasma thrusters*. Each of these includes clever electromagnetic field configurations, but, in essence, is still based on the Lorentz force; however, we will not discuss them further here.

5.4.5 Example 5.3: The PPT Engine

Consider the PPT engines shown in Figures 5.23 and 5.24. If the thruster has an exhaust velocity of 13,700 m/sec, an average thrust of 860 μN, an average m-dot of 8.3×10^{-8} kg/sec, and an I_{sp} of 1,400 sec, the discharge capacitor is 1 μF, and the capacitor is charged to 2,000 V, what are the values of the charging and discharging resistors in order to maintain the thruster performing at this level of operation?

The first step is to determine how much of the Teflon fuel is ejected with each pulse of the high-voltage capacitor discharging across it. The way to do this is to equate the energy stored in the capacitor with the energy of the exhaust velocity. So, we have

$$\frac{1}{2}CV^2 = \frac{1}{2}mv^2. \tag{5.21}$$

In Equation 5.21, the C is capacitance in farads, V is electric potential in volts, m is the mass of the exhaust due to one capacitor discharge across the Teflon, and v is the exhaust velocity of the ionized Teflon plasma. Simplifying the equation and solving for the mass of the plasma discharge exhaust yield

$$m = \frac{CV^2}{v^2} = \frac{(1 \times 10^{-6} F)(2,000\ \text{V})^2}{(13,700\,\text{m/s})^2} = 2.13 \times 10^{-8}\ \text{kg}. \tag{5.22}$$

Because we know the average m-dot of the thruster, we can see how many discharges per second is required to maintain the average thrust and specific impulse by dividing the m-dot by the mass found above:

$$n = \frac{\dot{m}(1\,\text{s})}{m} = \frac{(8.3 \times 10^{-8}\text{kg/s})(1\,\text{s})}{2.13 \times 10^{-8}\text{kg}} = 3.9\,. \tag{5.23}$$

Thus, in order to maintain the average thrust performance of the PPT, we must have a minimum of four pulses per second. This means that the capacitor must be charged completely to 2,000 V and then discharged completely to 0 V at least four times per second.

Figure 5.25 shows a graph of charging and discharging voltage of a resistor–capacitor (RC) circuit versus time. The equation for the electric discharge circuit is

$$\begin{aligned} V(t) &= A\left(1 - e^{-\frac{t}{R_{charging}C}}\right) \quad \text{for } 0 \le t \le 5\tau \\ &= Ae^{-\frac{t}{R_{discharging}C}} \qquad\quad \text{for } 5\tau \le t \le 0 \end{aligned} \tag{5.24}$$

in which A is the amplitude of the voltage the capacitor is to be charged to (in this example, $A = 2{,}000$ V), $R_{charging}$ is the value of the charging resistor in ohms, $R_{discharging}$ is the value of the

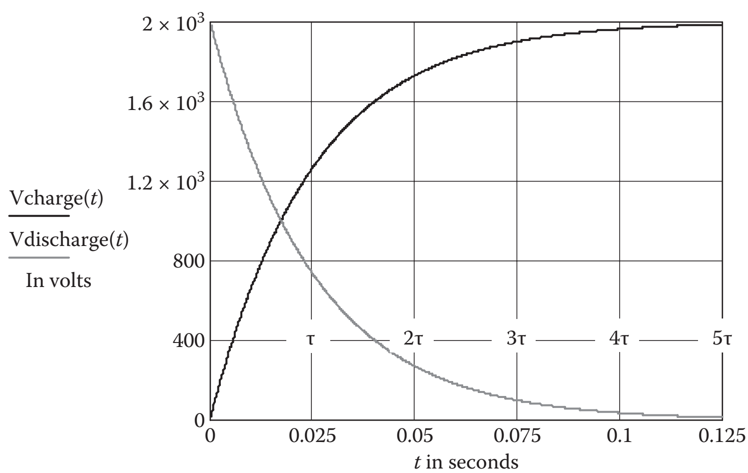


FIGURE 5.25
Charging and discharging of an RC circuit used to drive a PPT engine.

discharging resistor in ohms, t is time, and C is the capacitance in farads. We will assume that the charging and discharging resistor are the same value; therefore, we can drop the subscripts. Also, the constant τ is actually equal to RC and is known as the circuit's *time constant* and is measured in seconds. It takes five time constants for an RC circuit to charge or to fully discharge to more than 99% of the amplitude value desired. Now, we can determine the resistor values because we know the number of times the rocket thruster must fire per second. The engine must fire four times per second, and, therefore, it must charge up and then discharge down four times per second. Because a full charge requires 5 τ and a full discharge 5 τ, we see that we will need to allow for 40 τ per second. In other words,

$$\tau = \frac{1}{40}\,\text{s} = 0.025\ \text{s} = RC. \tag{5.25}$$

We were given the value of C, and thus R is

$$R = \frac{0.025\ \text{s}}{C} = \frac{0.025}{1 \times 10^{-6} F} = 25 \times 10^{3}\,\Omega = 25\ \text{K}\Omega. \tag{5.26}$$

We need a minimum of a 25-kΩ (kiloohm) resistor for the charging and one for the discharging circuits in order to maintain the average thrust desired for the PPT system in this example.

We have discussed the basic types of engines that are used for electric propulsion, but have not really discussed the source of power that the electric engines use. The engines can use anything, from batteries to solar panels to *radioisotope thermal generators* (RTGs) to nuclear fission reactors. The key is that the power source must supply enough power for the electric thrusters to fire as long as propellant is available. Figure 5.26 shows the basic

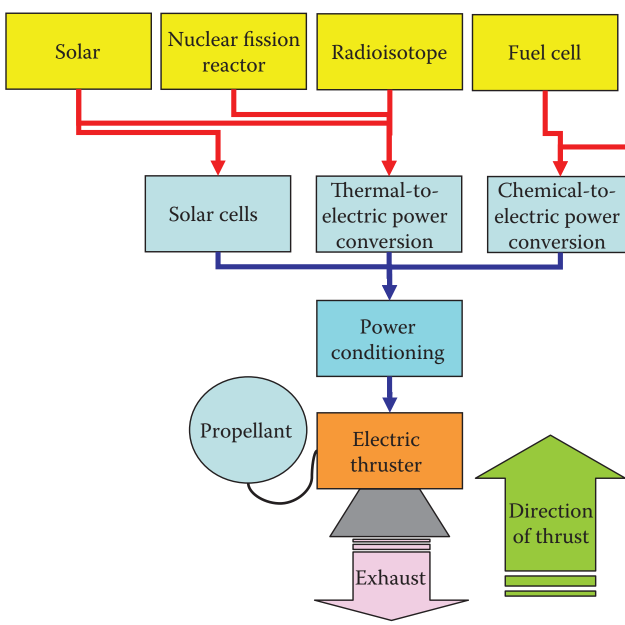


FIGURE 5.26
Components of an electric propulsion system.

components of electric propulsion and how the power source plays its role. Whether the source is the sun, batteries, or a fission reactor, the key to electric propulsion is that all the power sources are converted to electrical power, which is then used to drive the engine for thrust.

5.4.6 Solar Electric Propulsion

In the case of *solar electric propulsion* (SEP), the power source is actually our sun, Sol. At the surface of the sun, there is a *luminosity*, L_{Sol}, of about 3.86 × 10^{26} W of power from light leaving its surface. The *brightness* (also called *irradiance* by optical scientists and electrical engineers and is typically represented as *I*), b_{Sol}, of light energy per square meter at a given distance, *r*, from the sun is found by

$$b_{Sol} = \frac{L_{Sol}}{4\,\pi^2}\,. \tag{5.27}$$

Figure 5.27 shows a graph of the brightness in W/m^2 as a function of distance from the sun. Note that, at 1 AU from the sun, the brightness is about 1,355 W/m^2. This means that, for every square meter in a plane 1 AU from the sun, there is about 1 kW of light power continuously. (Actually, it is on a spherical surface of 1 AU in radius, but that is such a large sphere that it appears flat to the human scale.) Standard solar panels are anywhere from 8% to 15% efficient in converting that light power into electrical power; thus, a typical commercial solar panel that is a square 1 m on a side can supply about 100 W while in direct sunlight. The solar panels on spacecraft are a little bit more efficient than the ones used for powering houses here on Earth. For example, the solar panels on the International Space Station (ISS), as shown in Figure 5.28, cover over 375 m^2 and deliver about 100 kW. The efficiency of these panels is state of the art at about 19%.

The ISS doesn't use the solar power for propulsion, however. The Deep Space Probe DS1, as mentioned in Section 5.4.2 and shown here in Figure 5.29, was actually a successful

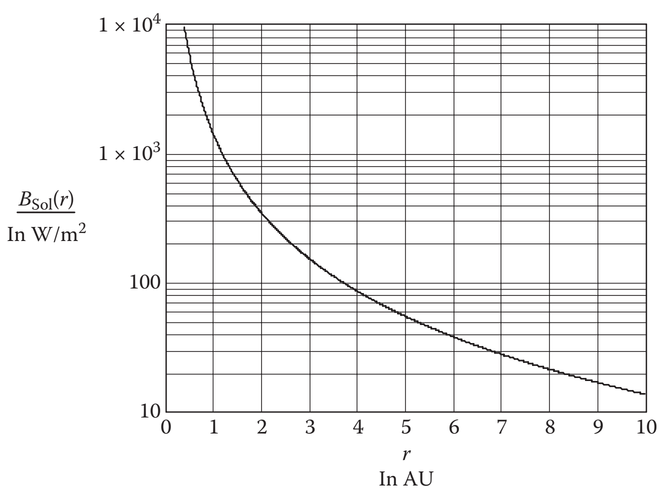


FIGURE 5.27
Brightness of the sun versus radius away from it.

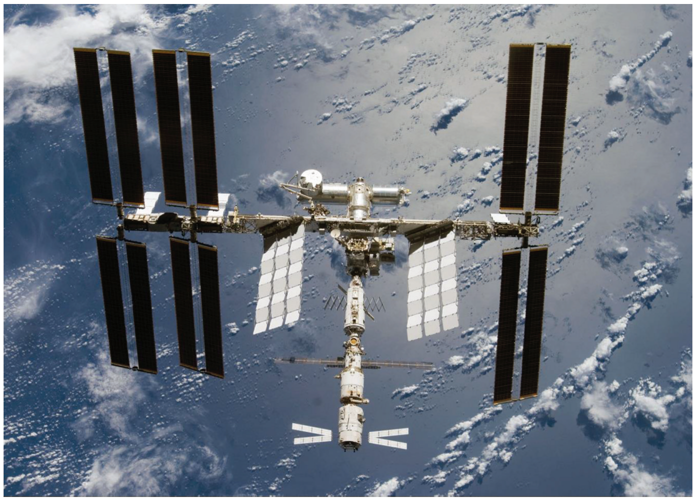


FIGURE 5.28
The solar panels of the ISS cover over 375 m^2 and generate over 100 KW. (Courtesy of NASA.)

FIGURE 5.29
The Deep Space 1 spacecraft demonstrated SEP. (Courtesy of NASA.)

demonstration of this concept. The DS1 used solar panels to power the NSTAR ion thruster electric engine on board.

5.4.7 Nuclear Electric Propulsion

Another way to power the electric engines is by using nuclear power. This can be done by using radioactive materials that decay slowly and generate heat that is then converted through special diodes into electrical energy. These systems are called radioisotope thermal generators and have been used for decades on space missions. Figure 5.30 shows the GPHS-RTG used to power the Cassini probe. The standard RTGs use plutonium oxide,

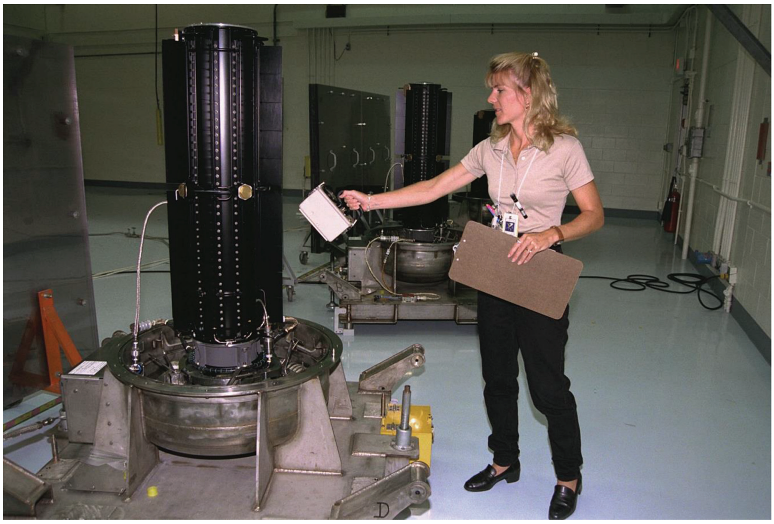


FIGURE 5.30
The GPHS-RTG power source used for the Cassini probe. (Courtesy of NASA.)

PuO_2, as the radioactive source. Similar power sources have been used on Pioneer 10, Pioneer 11, Voyager 1, Voyager 2, Galileo, Ulysses, Cassini, New Horizons, the Viking landers, and on several of the Apollo missions. The Russians have also launched spacecraft using RTGs. Many of them have used strontium, ^{90}Sr.

RTGs are well understood and supply stable power for very long periods of time. The power systems for the Voyager spacecrafts are still functioning at over 80% of their original designed performance. The disadvantage with RTGs is that they can only supply a few hundred watts of power. Therefore, RTGs are not a good candidate for high-power propulsion sources for electric engines.

Nuclear fission reactors, on the other hand, are great candidate power sources for electric propulsion engines. Using nuclear fission reactors isn't a new idea for powering spacecraft. In fact, both the United States and the Russians have considered the idea since the beginning of the space program, and each of them have flown reactors in space. The U.S. reactor that flew is called the System for Nuclear Auxiliary Power or SNAP-10A (Figure 5.31). The basic components of a *nuclear electric propulsion* (NEP) system are shown in Figure 5.32. A nuclear power plant creates heat from the radioactive fission reaction taking place within it. The heat is transferred through some means, such as heating liquid metal and flowing that liquid through pipes to a power conversion unit. Both the nuclear power system and the heat transfer system are typically designed to be within a radiation-shielded environment. The reason for shielding is to avoid having the radioactive decay particles escape and impinge on other spacecraft systems. This could influence sensitive measurements of onboard instruments or even damage some of the required spacecraft avionics.

The power conversion can implement a number of power conversion cycles, such as Brayton, Stirling, or Rankine. The most commonly studied conversion cycles make use of Brayton or Stirling generators. The nature of these generators is that moving parts are set in motion by the flowing liquids. These moving parts, in turn, generate electricity. Further detail is beyond the scope of this book.

Because the conversion cycles are only 20% to 30% efficient in converting heat to electrical power, there is a need to handle the excess heat of the system. If a reactor generates 300 kW

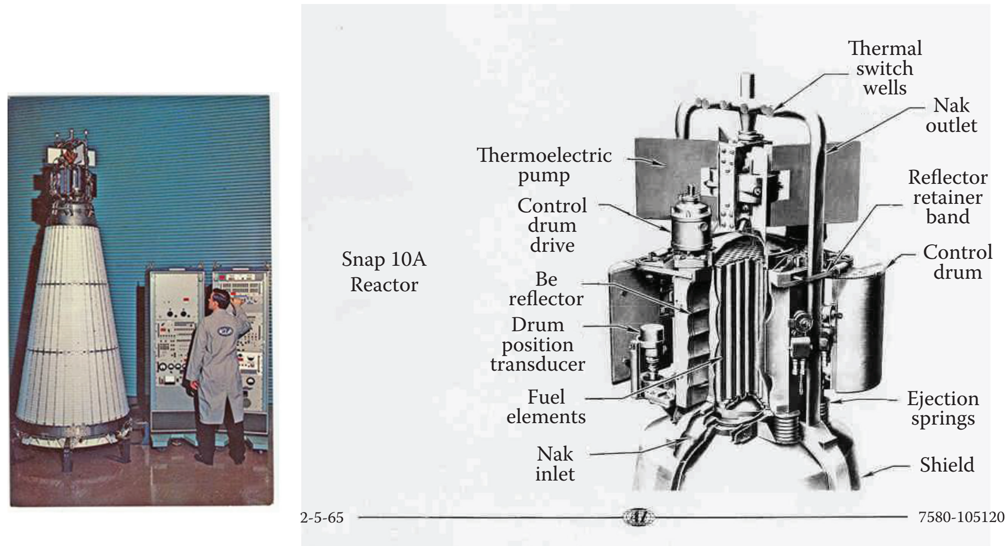


FIGURE 5.31
The SNAP-10A fission reactor system. (Courtesy of the U.S. Department of Energy.)

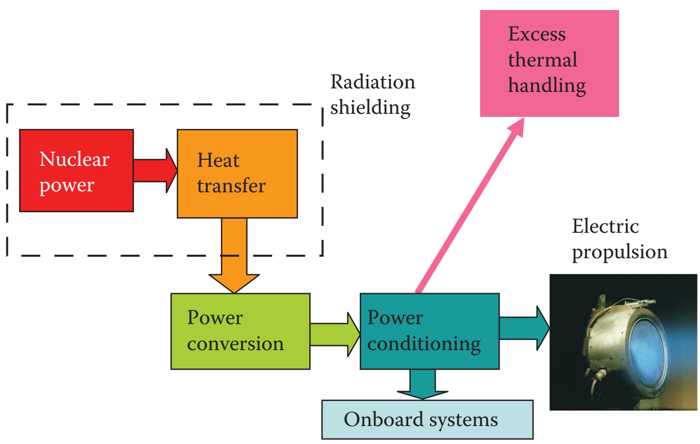


FIGURE 5.32
The basic components of an NEP system.

of thermal energy (usually denoted as 300 kWt), it could produce 100-kW electric (100 kWe), leaving an extra 200 kWt to deal with. In space, heat transfer becomes an issue as there is no air flowing around the spacecraft for which to transfer the heat. Hence, large radiators must be used to radiate the thermal energy into space.

Once the power is converted from heat to electricity, it is then conditioned to the voltage and current format that the electric thruster needs. The electrical power is then applied to the thruster. Propellant is flowed through the thruster, along with the electrical power, and thrust is generated.

In 2001, a new interest in NEP was created when a joint team from NASA Marshall Space Flight Center and Teledyne Brown Engineering began studying the concept and then proposed the *Tombaugh Orbiter* for a deep space probe to orbit and study the planetoid Pluto, its moon Charon, and then to move on into the Kuiper belt. The NEP-driven spacecraft is shown in Figures 5.33 and 5.34. The spacecraft design implemented a fission reactor developed by NASA, and the Department of Energy called the Safe Affordable Fission Engine (SAFE) reactor as it became to be known. The engine was discussed in many versions, ranging from a tested SAFE-30, which would produce 30 kWt, to a SAFE-400, which would produce 400 kWt. Figure 5.35 shows some schematics of the SAFE-30 testbed used in developmental testing.

The Tombaugh Orbiter was to implement a SAFE-300 reactor, Stirling cycle generators for power conversion, and six 30-kW ion thrusters under development at the time by NASA Glenn Research Center. The ion thrusters were scaled-up versions of the NSTAR engine used in the Deep Space DS1 mission mentioned in Section 5.4.2.

We should note here that, for the mission to Pluto, NASA did choose a different spacecraft design (New Horizons) that uses a 240-W RTG for power, but uses hydrazine thrusters for attitude control rather than any version of electric propulsion. Although the Tombaugh Orbiter was not chosen for the Pluto mission, it did reignite interest in the idea of NEP. Immediately following the Pluto–Kuiper mission design call from NASA, the Tombaugh Orbiter design was passed around the space community and was partly responsible for the beginning of a new NASA initiative called the Jupiter Icy Moons Orbiter (JIMO) and Project Prometheus.

The new NEP-driven initiative was to develop a spacecraft to travel to Jupiter and to orbit around the Jovian system to study all of the planet's moons. Figure 5.36 shows an artist's rendering of the final design concept for the JIMO spacecraft. It would have used a

FIGURE 5.33
Artist's rendering of the NEP spacecraft Tombaugh Orbiter. (Courtesy of Teledyne Brown Engineering.)

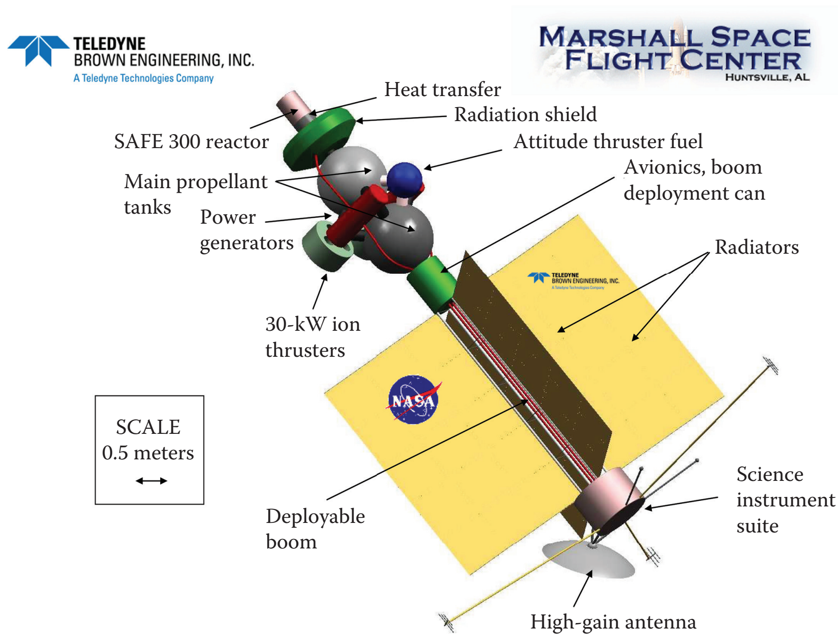


FIGURE 5.34
Schematic of the NEP spacecraft Tombaugh Orbiter. (Courtesy of Teledyne Brown Engineering.)

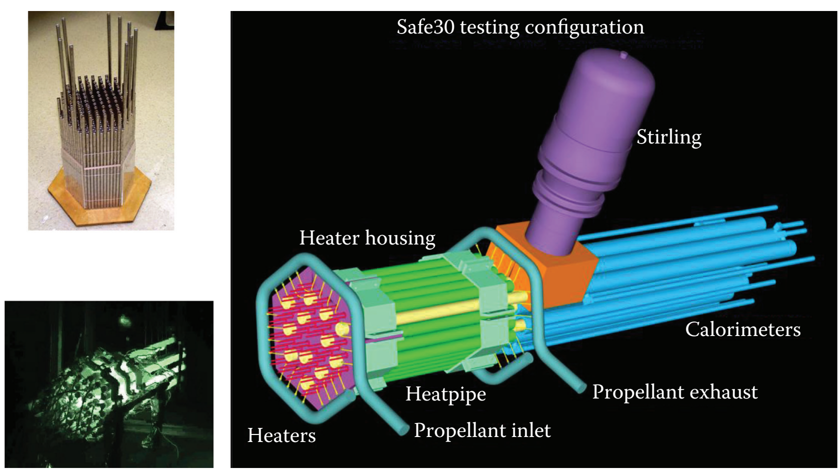


FIGURE 5.35
Schematics and images of the SAFE30 fission reactor design simulator and testbed. (Courtesy of NASA.)

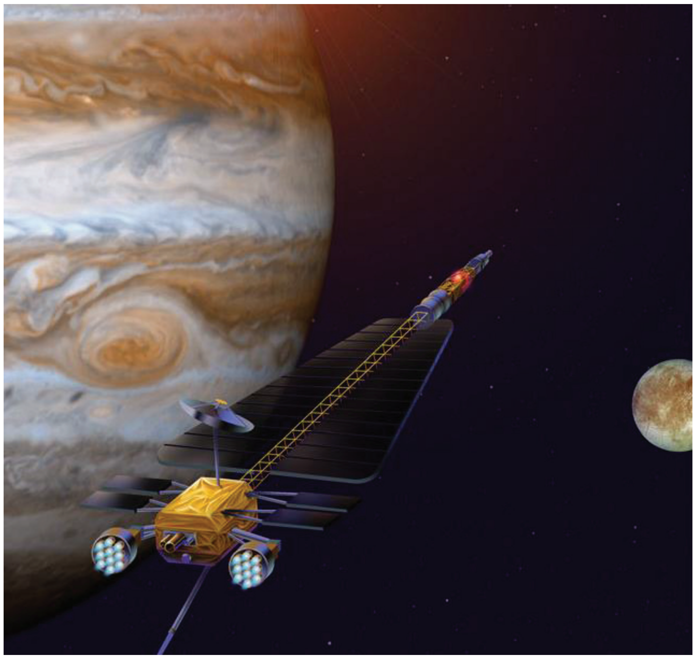


FIGURE 5.36
Artist's rendering of the JIMO NEP spacecraft. (Courtesy of NASA.)

SAFE-400 reactor for the power source, Brayton-cycle power conversion generators, a large radiator, and eight large ion thrusters. The project was killed in 2005 as NASA reorganized its internal funding plans and realigned for the new launch vehicle and Moon programs.

5.5 Nuclear Rocket Engines

Figure 5.37 shows a diagram of the NTR. Unlike the NEP concept discussed in Section 5.4.7, the NTR is truly a rocket engine. A nuclear fission reactor is the key component in this rocket engine that enables thermodynamic expansion of propellant gases. A fissile source, such as reactor-grade uranium (U^{235}), is used to generate heat. The level of the radioactive fission process is controlled by moderator control rods and by reflectors of the same material (typical graphite, boron carbide, and beryllium). The propellant is normally flowed through the fission reactor as coolant to the reactor system. In turn, the propellant is superheated and thermodynamically expanded in the expansion chamber of the rocket engine. At this point, the rocket engine functions just like any other inasmuch as the heated flow is forced from the expansion chamber into a convergent–divergent nozzle.

5.5.1 Solid Core

The diagram in Figure 5.37 illustrates an NTR system that uses solid fuel rods for the nuclear reactor core. The *solid core* is the most traditional. In fact, Figure 5.38 shows the

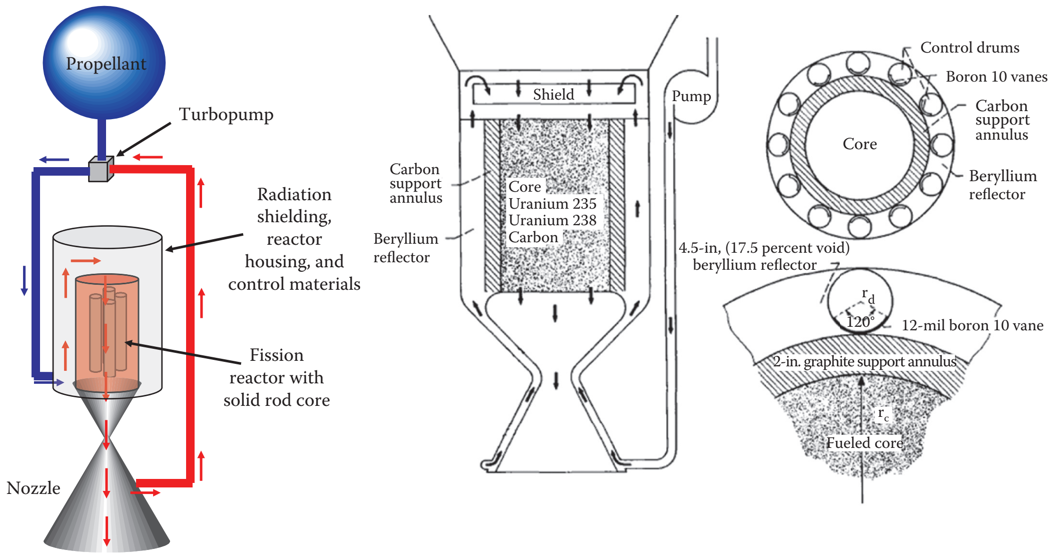


FIGURE 5.37
Schematic of an NTR engine.

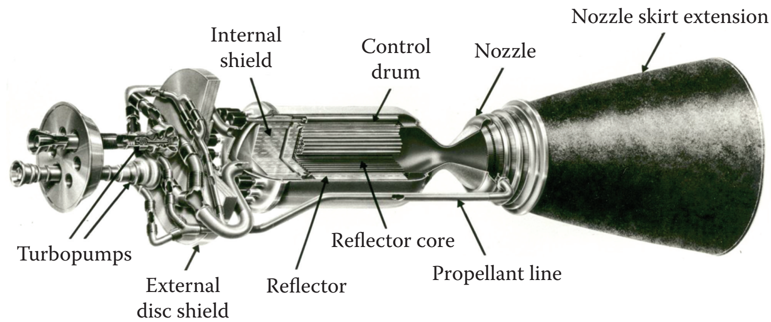


FIGURE 5.38
The NERVA rocket engine. (Courtesy of NASA.)

Nuclear Engine for Rocket Vehicle Application (NERVA). The NERVA engine was based on the Kiwi nuclear reactor shown in Figure 5.39, which is not unlike the SAFE reactor described in Section 5.4.7. The NERVA engine was developed in the 1960s by NASA and was originally investigated as a replacement for the J2 liquid engine upper stage on the Saturn V. NERVA produced 867 kN of thrust, an I_{sp} of 380 sec at sea level and 825 sec in vacuum, with a burn time of about 1,200 sec. It used liquid hydrogen as the coolant/propellant. At sea level, the engine did not perform as well as the SSMEs, but, in space, it outperformed them by a factor of two or more.

There are other solid core designs that use pebbles and dust of the fissile material as the heat source. These reactors have been shown to have the potential for improving the rocket engine performance to over 1,000 sec of specific impulse. There are still questions about the cost effectiveness of such designs.

FIGURE 5.39
The Kiwi reactor at the Nevada Test Site in the 1960s. (Courtesy of NASA and the U.S. Department of Energy.)

5.5.2 Liquid Core

A *liquid core* engine uses a liquid material as the fissile source. Because the core in these types of reactors is already in liquid form, they can be heated to temperatures above the melting point of the core materials, and, therefore, the heat source can grow much hotter. The limiting factors in how hot such a reactor can get is the stress the container wall can handle and the melting point of the reflectors and moderators. Liquid core engines could potentially deliver specific impulses as high as 1,500 sec. However, how to go about building such an engine safely is still in question. The radioactive fluids must be maintained inside the engine. The process of transferring the heat between the radioactive fluid and a propellant gas is a difficult one and has yet to be completely worked out. There are some concepts for liquid core engines; however, more research needs to be done.

5.5.3 Gas Core

A *gas core* engine would use a pocket of gaseous uranium as the fuel of the reactor. In order to prevent the gas escaping from the rocket engine, it must be housed in a very-high-temperature quartz container. This "nuclear lightbulb" would sit in the middle of the expansion chamber where hydrogen is flowed around it and superheated. The expanded hydrogen gas would then flow through a convergent–divergent nozzle. Studies suggest that such an engine could attain specific impulses of over 2,000 sec.

5.6 Solar Rocket Engines

Like the NTR concept, a similar rocket engine design is the *solar thermal* rocket (STR). In the case of the STR, a large lightweight mirror or lens is used to focus sunlight onto the thermodynamic expansion chamber. The focused sunlight then heats a propellant liquid and expands it until it is then forced out the convergent–divergent nozzle. Figure 5.40 shows a basic diagram of the STR concept. As was mentioned in Section 5.4.6, there is about 1,355 W/m^2 of sunlight at 1 AU from Sol. Thus, with a modest-sized, solar-collecting, optical element (lens or mirror), a significant amount of thermal energy can be transferred to the propellant gas. This concept is not unlike using a magnifying glass to burn paper. A small lens on the order of a few centimeters in diameter is more than ample size to collect enough sunlight and then to focus it onto a tiny spot. The number of watts of power within the beam stays the same, but the area it is contained within is decreased dramatically. The brightness of the spot, as we have already seen, is in W/m^2 or power per area. As the area goes down, the brightness goes up.

5.6.1 Example 5.4: The Solar Thermal Collector

Consider a lens of 5 m in radius in Earth orbit. The area of that lens would be 78.53 m^2. Thus, with the incident irradiance from the sun being 1,355 W/m^2, the lens can collect

$$P_{\text{collector}} = B_{\text{Sol}} A = (1,355 \text{ W/m}^2)(78.53 \text{ m}^2) = 106,421.5 \text{ W}. \tag{5.28}$$

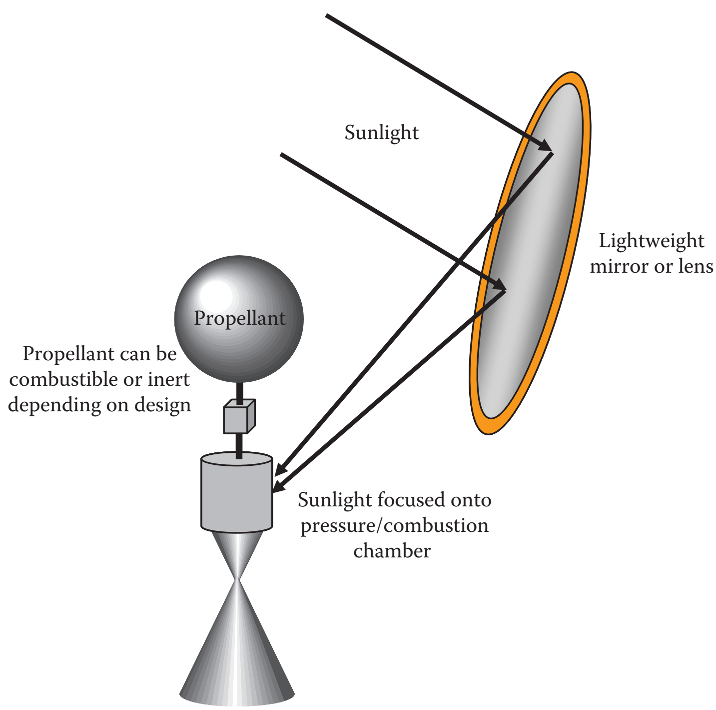


FIGURE 5.40
Schematic of solar thermal rocket.

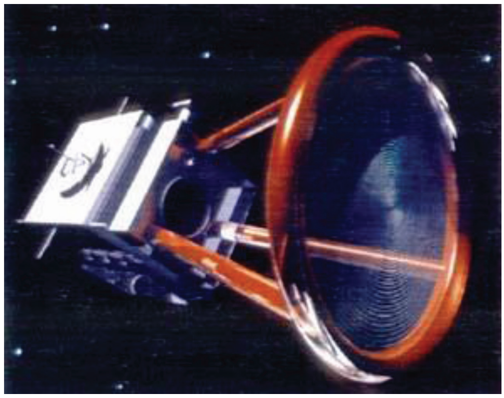


FIGURE 5.41
Artist's rendering of the STR experiment called Shooting Star. (Courtesy of NASA.)

We see that a modest lens can collect a tenth of a megawatt. Now, depending on the design of the lens, more than 87% can be put into a single spot at a very small diameter on the order of 10 mm. That makes the brightness of the spot at the focus

$$b_{spot} = \frac{P_{collector}}{A} = \frac{106,421.5\,\text{W}}{\pi(.005\,\text{m}^2)} = 1.355 \times 10^9\ \text{W/m}^2\,. \quad (5.29)$$

This is by far hot enough to weld metal.

If the expansion chamber is designed properly, the focused sunlight is absorbed by the expansion fluid propellant. The propellant can be water, hydrogen, hydrazine, or any other gas that is deemed appropriate.

Example 5.4 is a bit misleading in that putting a full gigawatt into an expansion chamber is not physically simple. Figure 5.41 is an artist's rendering of NASA's Shooting Star Experiment, which was an STR project. The concentrator was constructed of polymer materials and had several design possibilities. The concentrator could be inflated or rigidized depending on the manufacturing process chosen. Figure 5.42 shows the schematic of the coupling mechanism used to get the focused sunlight into the rocket chamber. The coupling mechanism was a refractive lens device that was used to spread the sunlight into the expansion chamber. Figure 5.43 shows the actual device, which was called the refractive secondary concentrator.

5.6.2 Example 5.5: The STR Exit Velocity, I_{sp}, and Thrust

Assume the STR design as given in Example 5.4. The propellant fluid is water. The m-dot of the rocket design is 0.001 kg/sec when in operation. Also, assume that 10% of the sunlight collected is actually converted to heat energy and that the material properties of the rocket engine will only allow it to be in direct sunlight for 0.1 sec at a time before a 10-sec cooling time is required. In other words, the rocket pulses with 0.1-sec pulse lengths with a minimum of 10 sec between pulses. Assume the isentropic expansion factor is 1.2. Also note that the mass of propellant in the expansion chamber at any given time is 10 kg. Determine the exit velocity, specific impulse, and thrust from one pulse of the engine.

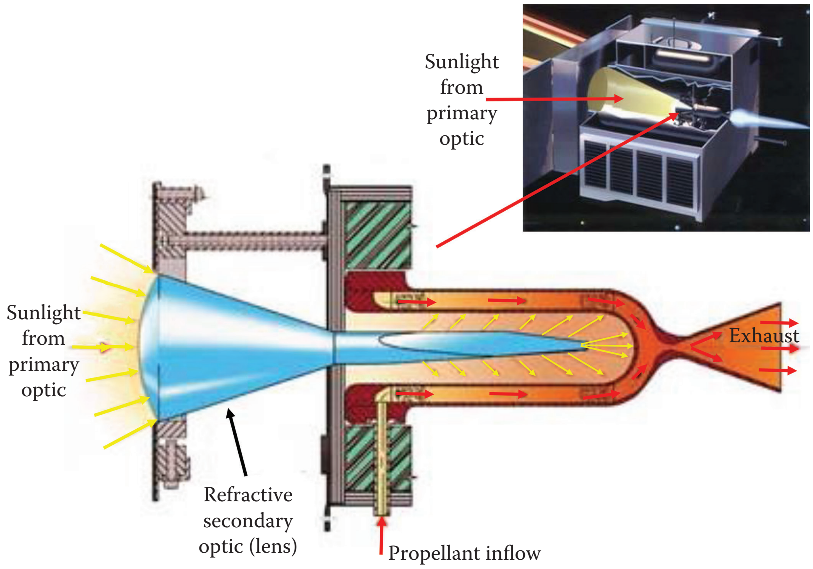


FIGURE 5.42
Schematics of the Shooting Star STR. (Courtesy of NASA.)

FIGURE 5.43
Image of the Shooting Star STR refractive secondary concentrator. (Courtesy of NASA.)

From Example 5.4, we saw that the solar concentrator in direct sunlight could deliver about 0.14 GW of power to the engine. If we use the 10% efficiency factor given in the problem definition, then there will be 0.014 GW of power converted to heat. Because the rocket can only handle direct sunlight exposure for 1 sec, then we can find the energy converted to heat by

$$H = (1.4 \times 10^8 \text{ W})(0.1 \text{ s}) = 0.14 \text{ GJ}. \quad (5.30)$$

The change in temperature can be determined from combining the heat equation and Equation 4.28:

$$H = mC_p\Delta T = m\frac{\gamma}{\gamma-1}\frac{R_u}{M}\Delta T. \tag{5.31}$$

From Equations 5.30 and 5.31 and the given information, we can find the change in temperature of the expansion chamber to be

$$\Delta T = \frac{\gamma-1}{\gamma}\frac{M}{mR_u}(H) = \frac{0.2}{1.2}\frac{18\,\text{kg/kmol}}{10\,\text{kg}(8,314.41\,\text{J/kmolK})}(0.14\,\text{GJ}) = 5,051\,\text{K}. \tag{5.32}$$

From Equation 4.37, we can find the exit velocity:

$$v_e = \sqrt{\frac{2\gamma}{\gamma-1}\frac{R_u}{M}T_c} = \sqrt{\frac{2(1.2)}{0.2}\frac{8,314.41\,\text{J/kmolK}}{18\,\text{kg/kmol}}5,051\,\text{K}} = 5,291.5\,\text{m/s}. \tag{5.33}$$

Calculating the specific impulse is straightforward at this point:

$$I_{sp} = \frac{v_e}{g} = \frac{5,291.5\,\text{m/s}}{9.8\,\text{m/s}^2} = 540\,\text{s}. \tag{5.34}$$

The thrust is a bit more confusing in that we assumed the m-dot was 0.001 kg/sec, but the rocket only pulses for 0.1 sec. So, for the duration of the pulse, the rocket will produce a thrust of

$$F_{thrust} = \dot{m}v_e = (0.001\,\text{kg/s})(5,291.5\,\text{m/s}) = 5.3\,\text{N}. \tag{5.35}$$

What we see from this example is that perhaps our concentrator is too large for our engine design because if it were kept in continuous sunlight for longer than a tenth of a second or more, the temperature inside the chamber would far exceed the melting point of any construction materials, and the rocket would burst open or simply destroy itself. A more efficiently designed system can be achieved by the equations given in this section and by careful choice of the proper materials, propellants, and component geometry.

5.7 Photon-Based Engines

This chapter wouldn't really be complete without discussing thrust that can be achieved from momentum transfer from light particles. We can describe light as if it were made of particles called *photons*. Although these photons have no discernable mass, they do have momentum that is due to an intrinsic property of photons called *spin angular momentum*. Further discussion of the quantum mechanical properties of photons is outside the scope of this text, but suffice it to say that an individual photon does have momentum, and it is calculated as

$$p = \frac{h}{\lambda}, \tag{5.36}$$

where p is the momentum, h is Planck's constant, 6.626×10^{-34} Js, and λ is the wavelength of the light making up the photons (e.g., the yellow-green light from Sol is about 575 nm). Each photon has momentum; although it is very, very small, it *is* there. Equation 5.36 is for one photon. If there are n photons, it is

$$p = n\frac{h}{\lambda}. \tag{5.37}$$

If a beam of photons is incident on a mass, m, we can see how much Δv it will impart to the mass by

$$p = n\frac{h}{\lambda} = mv, \tag{5.38}$$

or

$$v = \frac{nh}{m\lambda}. \tag{5.39}$$

Figure 5.44 shows a graph of the Δv for a 1-kg mass versus the number of photons. From that figure, we see that more than 10^{20} photons are required to generate a velocity of about

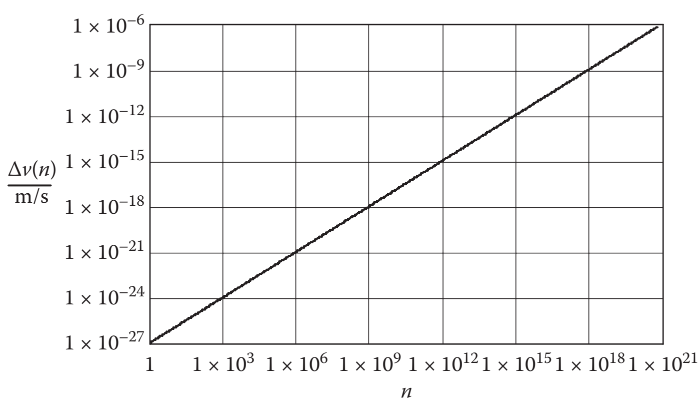


FIGURE 5.44
Delta-v of a 1-kg mass due to incident photon momentum transfer requires large numbers of photons for small velocity increases.

1 μm/sec. It is easy enough to determine how bright of a beam of light this is. The energy of a beam of light is found by

$$E = n\frac{hc}{\lambda}, \tag{5.40}$$

where c is the speed of light. Figure 5.45 shows the energy of a beam of light versus the number of photons in it. From that figure, it is clear that the energy in a beam of 10^{20} photons is on the order of about 50 J or so. It is also easy to show that the irradiance, I, of a beam of light (if we are talking about light from the sun, we will call it brightness, as mentioned in Section 5.4.6, but, for a generic beam of light, we will talk of irradiance) is

$$E = n\frac{hc}{\lambda} = IAt, \tag{5.41}$$

where A is the cross-sectional area of the beam, and t is the length of time the beam is incident on the surface of the mass. From Equation 5.41, we can find the irradiance of a beam of light if we know the area, duration, and the energy within it. Also, if we are considering the sun as the source, then the energy from it is practically constant over time. So, dividing Equation 5.41 by t gives the power, P, as a function of irradiance and area.

Another good relation to discuss is *light pressure* and the force due to light pressure. We have just shown that light momentum will actually impart a momentum to an object. We can also describe this effect in terms of pressure and force. The light pressure due to an incident beam of light on an object is

$$P = \frac{2I}{c}. \tag{5.42}$$

Equation 5.42 gives the pressure in pascals of an incident beam of light on an object. Note that there is a 2 in the numerator. This 2 is there because we actually get a momentum

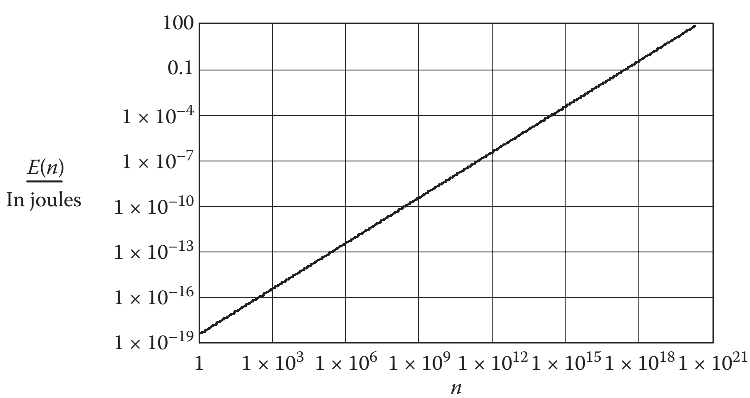


FIGURE 5.45
Graph showing energy as a function of the number of photons for 575-nm light.

transfer once when the photon hits the object and once if it bounces off the object due to Newton's Third Law. This implies that our object is reflective to the beam of light. We know that force is pressure incident on an area; thus,

$$F_{thrust} = PA = \frac{2IA}{c}. \tag{5.43}$$

Equation 5.43 shows the relationship between thrust on an object generated by an incident beam of light. This is important to note.

What we have just discovered is that a beam of light will actually push an object. It will push a reflective object twice as hard. Now, consider a very thin reflective mirror at 1 AU from the sun. Figure 5.46 shows the thrust generated by sunlight hitting the mirror as a function of the area of that mirror. Figure 5.47 shows the thrust on the mirror versus the length of a side, assuming a square mirror. In order to attain a thrust of 10 N at 1 AU from the sun, we would need a square mirror more than 1 km in length on a side. That is one big mirror.

If the mirror was much closer to the sun, however, we could get much more thrust or use a smaller area. Recall Equation 5.27 and Figure 5.27 where we showed that the brightness of the sun drops off with the inverse square of the distance. Very close to the sun, the irradiance on an object due to the brightness of the sun is quite significant. The inverse is true as well. If we look very far away from the sun (say, 5 AU or so), the sun's brightness is very much smaller. In the distant case, there is very little need in trying to gather thrust with a mirror because the size of the mirror required would be ridiculously large.

What we have described here is a *solar sail*. A solar sail is a rocket engine that uses light pressure to generate thrust. Another way of thinking of them is that they are photon thrusters, whereas the large mirror (called a sail for the same reason as a sail on a sailboat) acts as the rocket nozzle and redirects the photons in order to impart a thrust to the spacecraft. The idea of solar sailing has been around for a long time, and solar sail physics is fairly well understood, even though no solar sail spacecraft has yet flown. Figure 5.48 shows a schematic of a typical solar sail design. (Note that the design shown is most like the Halley's Comet Rendezvous design considered by NASA in 1977.) The sail uses smaller

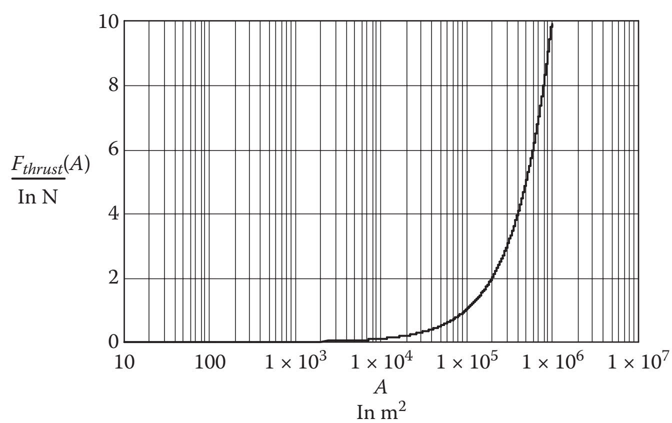


FIGURE 5.46
Graph showing the thrust on a mirror versus the reflective surface area of the mirror at 1 AU from Sol.

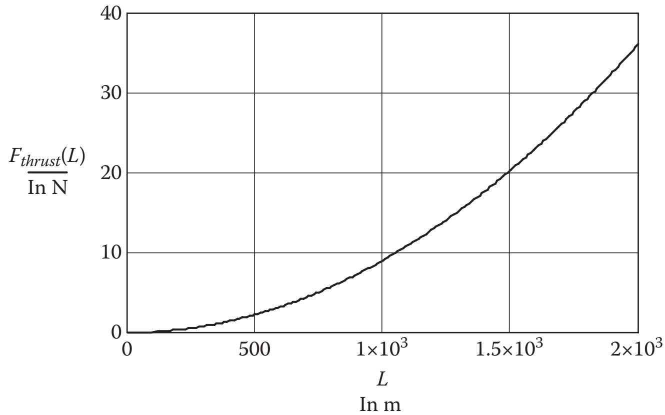


FIGURE 5.47
Graph showing the thrust on a mirror versus the mirror length at 1 AU from Sol.

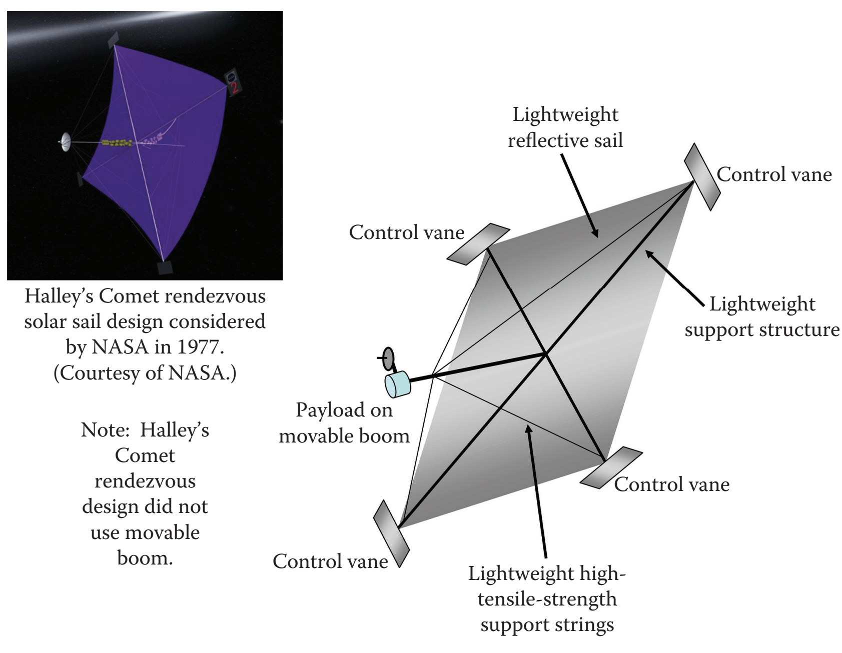
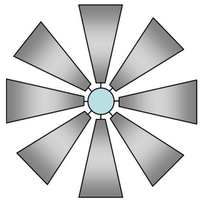


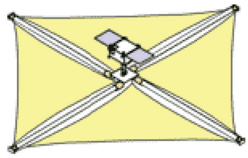
FIGURE 5.48
Typical solar sail configuration.

sails called *vanes* on the periphery for generating off-axis vectors for steering, just like a sailboat moves a rudder against the water or tilts a sail into the wind. The payload can be at the center of pressure, or it can be moved around (as shown in the figure) to place the center of mass at a different location from the center of pressure. This is another method of steering the solar sail. Note that the movable boom concept was not part of the Halley's Comet Rendezvous design.

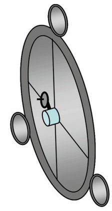
There have been several technology development efforts to fly solar sails both through governments and private enterprises, but none have yet flown. Figure 5.49 shows several



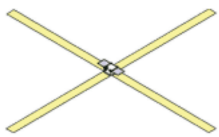
Steerable fan blade solar sail with payload at center. (Cosmos 1 design used by The Planetary Society.)



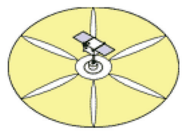
Square sail supported by booms only and movable boom for steering the center of mass. (Courtesy of NASA.)



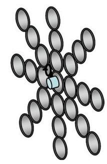
Hoop-supported sail with hoop control vanes and payload suspended on a web.



Spin-stabilized heliogyro sail uses centrifugal force to maintain sail rigidity. (Courtesy NASA.)



Spin-stabilized disk sail uses centrifugal force to maintain sail rigidity. (Courtesy of NASA.)



Multiple miniature hoop-supported sail array.

FIGURE 5.49
Shown are various modern solar sail design configurations. Note that none of the schematics are drawn to scale. The payloads would be too small, to say, at real scales.

different solar sail design configurations that have been proposed over the years. Between about 1997 and 2004, NASA had a significant solar sail propulsion project, but changes in the space administration's policy and budget have forced the program to be canceled.

We should also note here that the sail does not have to be driven by sunlight. Many technical studies have shown that large lasers could drive the spacecraft as well. The physics remains basically the same, except that, instead of the sun, lasers are used for the incident photons. At any rate, the idea is still that of a photon engine where a sail is used to redirect photons to generate thrust for a spacecraft.

In the near future, solar sails seem to be the only technology available (barring the invention of some new physics like warp drives from science fiction) that could propel a spacecraft into really deep space and maybe even the nearest star. Many studies have been performed to show that with very, very large and very, very lightweight solar sails, velocities approaching a hundredth or even a tenth the speed of light might be possible. However, there are some technological hurdles involved in developing, manufacturing, deploying, and flying such large structures in deep space.

5.8 Chapter Summary

In this chapter, we have really begun to see the details of rocketry and how an engineer or a scientist must understand a very broad spectrum of subject areas to truly become a practicing rocketeer. The rocket scientist and the engineer must understand the chemistry and construction of solid rocket engines, as shown in Section 5.1, as well as being well versed in the aspects of flow, turbo pumps, heat exchange, and liquid propellants for the liquid rocket engine, as discussed in Section 5.2.

Then, we showed in Section 5.3 that solids and liquids can be successfully married together to create a hybrid rocket engine that operates much like both solid and liquid engines. The aspects of solid fuel combined with flow of liquid oxidizer (or vice versa) come in to play.

In Section 5.4, we discussed electric propulsion. The different types of electrically driven engines require the rocket scientist and the engineer to have a deep understanding of electricity, magnetism, and electromagnetic theory. Plasma physics is also important in this type of engine. We also showed that, depending on where the electrical power is coming from, the rocket scientist and the engineer must have some working knowledge of nuclear reactors and of solar panel design and the nuances of the two power source technologies.

In Section 5.5, we discussed NTR engines that actually use a nuclear reactor to heat a propellant. We showed that there are several types of reactor and rocket design combinations possible. There are enough variations on the NTR theme that a rocket scientist or an engineer could make an entire career out of studying them.

The same goes for solar thermal rocket engines, as discussed in Section 5.6. An understanding of the solar brightness and material properties is a must with this type of engine. We showed that, very quickly, a poorly designed STR will destroy itself. In this section, we discussed the fact that optics is another area of physics and engineering required to understand modern rocketry.

Finally, in Section 5.7, we discussed photon-based engines. This type of rocket engine uses a different type of nozzle and propellant. The photon-based engine uses external incident photons of light from the sun or a laser source and redirects them via a large, lightweight reflector to generate thrust. The photon-based engines are solar and laser sails, and, indeed, they do redirect the flow of energy through the spacecraft in much the same way that nozzles are used to redirect the flow of thermal energy through a spacecraft to generate thrust.

Exercises

5.1 What is an igniter?

5.2 What is the propellant in a solid motor called?

5.3 Define burning surface.

5.4 What is the purpose of the thermal insulation barrier on a solid rocket motor?

5.5 Define HTPB.

5.6 Define PBAN.

5.7 What is APCP?

5.8 Describe the mix of the Space Shuttle SRB grain.

5.9 What is the perforation?

5.10 Describe several types of perforations and explain the type of thrust profile they produce.

5.11 Define the three solid motor modes of burn and sketch their thrust profiles as a function of time.

5.12 Using the data given in Example 5.1, model the burn rate of the SRBs and develop a sketch of the burn rate versus the burn rate coefficient and of the pressure exponent.

5.13 Use Equation 5.3 to show that the m-dot of a model rocket engine with no perforation is a function of the radius of the engine. What would the thrust profile of this engine look like?

5.14 What are injectors?

5.15 How are injectors and the stoichiometric ratio of fuel and propellant related?

5.16 What is cavitation?

5.17 What is pogo?

5.18 How is pogo controlled?

5.19 Discuss two types of liquid rocket engine cooling.

5.20 Describe a hybrid rocket engine.

5.21 Describe a reverse hybrid rocket engine.

5.22 Define the Coulomb force.

5.23 What is the Lorentz force?

5.24 An ion thruster has a grid voltage of 4,000 V and uses xenon for propellant. What is the exit velocity of the engine?

5.25 If the engine in Exercise 5.24 fires for 10 days continuous, what is the I_{sp} of the engine? Assume a total fuel mass of 117.5 kg.

5.26 In Exercises 5.24 and 5.25, determine the thrust of the ion engine.

5.27 In a Hall thruster, describe the purpose of the swirling electron field.

5.28 What is FEEP?

5.29 A PPT has an exit velocity of 14,000 m/sec, produces 1 mN of thrust, has an m-dot of 9×10^{-8} kg/s, and has an I_{sp} of 1,200 sec. If the engine has a 10,000-V charge on a 10-μF capacitor, what values of discharging and charging resistors are required?

5.30 What is NEP?

5.31 What is SEP?

5.32 Define luminosity.

5.33 How are brightness and irradiance different?

5.34 What is NTR?

5.35 What is STR?

5.36 Discuss how a solar sail is analogous to a rocket engine and why it is called a photon-based engine.

6

How Do We Test Rockets?

So far in the book, we have discussed a lot of details about rocket history, rocket concepts and architectures, rocket science, and rocket engineering. An extremely important aspect of rocketry is getting ready to fly. After all, the whole point of building a rocket to begin with is so that we can fly the thing.

This chapter will discuss testing rocket components, subsystems, systems, and complete products. It covers all the steps involved in taking a rocket concept, from the first drawings and calculations sketched on a whiteboard (or, in some cases, even a bar-room napkin) to flight readiness.

An actual example of a successful space mission starting with a bar-room napkin is the Clementine mission. In 1989, Stuart Nozette (soon to be at Lawrence Livermore National Laboratory), Pete Worden (from the White House National Space Council), and Geoff Tudor (a congressional staffer at the time) were having a drink and discussing ways to transition new technologies developed by the Strategic Defense Initiative Office (SDIO) to the National Aeronautics and Space Administration (NASA) and the civil space community. Nozette sketched out an idea on a napkin. Five years later, the ideas on that napkin were launched, and the spacecraft flew to the Moon, gathering some brilliant data with new instruments. There was a lot of work that took place between the napkin and the flight. Testing was a major part of the effort, as it is with most flight programs.

For a rocket vehicle program where new engines, flight bodies, moving parts, aerodynamic structures, and other flight avionics systems and subsystems are required, detailed analysis, modeling and simulation, and testing must be conducted in order to reduce the risk of vehicle failure on launch or throughout the mission. This analysis, modeling and simulation, and testing are conducted to gain detailed knowledge of how the design functions under simulated flight conditions. These three steps reveal weak aspects of the rocket's design, and, therefore, redesign is conducted. Then, the analysis of the test data leads to new modeling and simulation and verification that the design change should function properly. At that point, testing is redone. The new test data are analyzed to determine if the component performed properly. If it did, it is ready for the next level of integration with other parts and systems or maybe even flight testing. If the component did not perform properly, the analysis, modeling and simulation, redesign, and retesting process continues.

This performance refinement process is known as *systems engineering,* which is a major part of rocketry. The importance of this process becomes clear when a large launch vehicle development effort, such as the Space Shuttle, is considered. The Space Shuttle has over 2 million parts. Each of these parts must operate within particular standards throughout the flight profile of the vehicle, from launch to touchdown. Although there are built-in redundant parts in critical areas, the reliability of each of these parts must be extremely high. Each must also interact with all the other parts properly as to not cause an overall systems failure. A single part or even all 2 million parts might be built to design specifications, but, until they are put together as a working system, it is difficult to determine if they will function as a piece in a larger machine without first testing them together.

Thus, the rocket scientist and the engineer must learn how to conduct tests that will identify where critical items might fail when working with other components. Also, if one or more items might fail, the impact of that failure on the rest of the larger system must be understood. An analysis is performed to determine the level of severity of loss of functionality on the system, which, in turn, defines the part's criticality within the larger system. This type of analysis is called *failure mode effects analysis* (FMEA) and is a key tool in rocketry design and testing and the overall systems engineering of rockets.

In this chapter, we will discuss the basics of the systems engineering process implemented in most rocket programs today, and then we will discuss specifically how to conduct tests in order to measure the basic performance characteristics of rockets and components that will lead to design refinement and successful flight testing. We will discuss in detail NASA's Apollo and Constellation Program development efforts and the flight test programs in order to illustrate the complexity of such large rocket programs and the testing required. Also, it is in this chapter where we begin to see that there is more to rocket science and engineering than making a bunch of calculations, then slapping together a rocket, and lighting the fuse. Designs typically never work right the first time, or the second, or the third, and so on. In fact, there were over 30 tests of the Apollo program. This is why we test. And now, we shall discuss how we go about it.

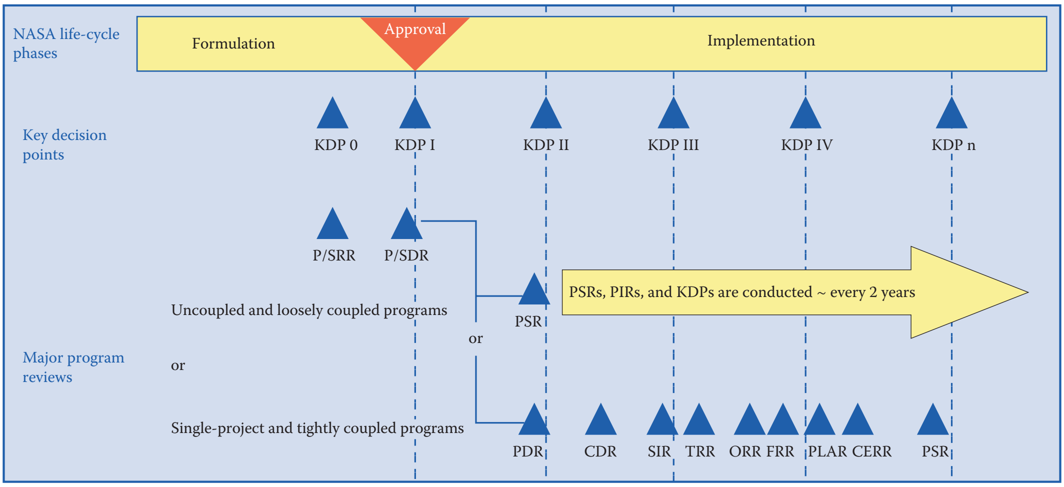
6.1 Systems Engineering Process and Rocket Development

As we have seen throughout this book, rockets are very complex machines. Developing a rocket and its subsequent test programs and operational life cycle is an even larger and more complex endeavor than the machine itself. Though we have talked mainly about the hard science and technical engineering aspects of rocket science and engineering thus far, we need to look at the *systems engineering* aspect of rocketry before we truly discuss the "nuts and bolts" of testing the hardware. According to the *NASA Systems Engineering Handbook (2007)*,

> Systems engineering is a methodical, disciplined approach for the design, realization, technical management, operations, and retirement of a system. A "system" is a construct or collection of different elements that together produce results not obtainable by the elements alone. The elements, or parts, can include people, hardware, software, facilities, policies, and documents; that is, all things required to produce system-level results. (p. 3)

In other words, a *system* is a complex thing made up of many pieces and functions that systems engineers often refer to as "elements." It is sometimes these millions of elements that make up the overall development effort that includes the hardware and software of the functioning device, all of the support infrastructure, the life-cycle elements, from conception to the end of life, and any other aspect involved with the project. NASA isn't the only organization that follows this philosophy. The Department of Defense (DoD) uses it. Software developers use it. Most large system commercial manufacturers use it. Scientists and engineers can make an entire career out of studying the ins and outs of systems engineering.

Figure 6.1 shows the NASA program life cycle. Programs are defined as the overall effort like putting a man on the Moon or Mars, or studying the outer planets, or creating a new



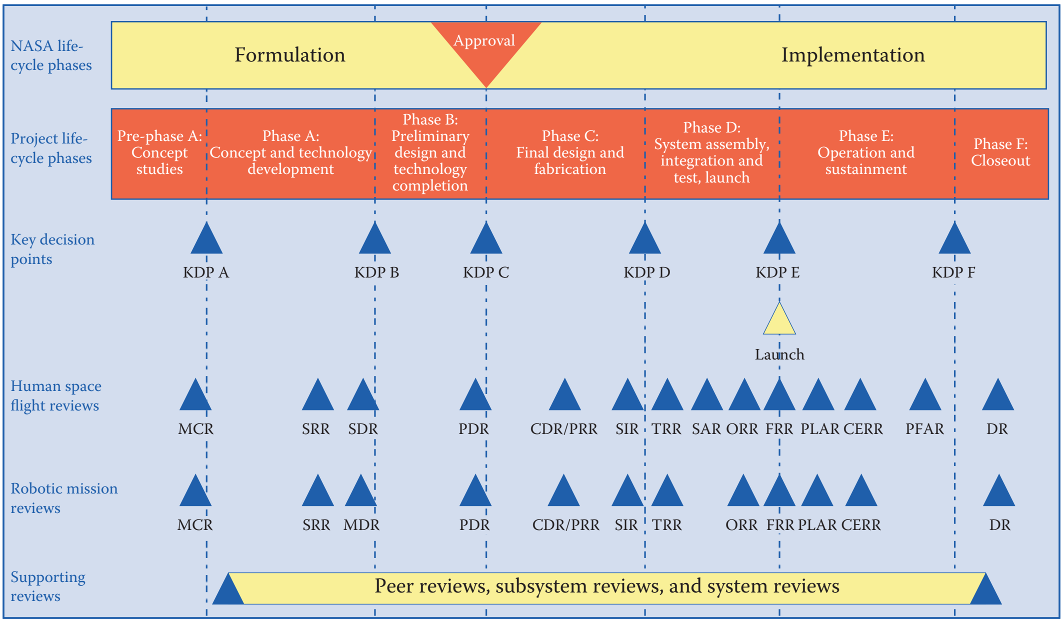
CDR	Critical design review	PLAR	Post-launch assessment review
CERR	Critical events readiness review	PRR	Production readiness review
DR	Decommissioning review	P/SDR	Program/system definition review
FRR	Flight readiness review	P/SRR	Program/system requirements review
KDP	Key decision point	PSR	Program status review
MCR	Mission concept review	SAR	System acceptance review
MDR	Mission definition review	SDR	System definition review
ORR	Operational readiness review	SIR	System integration review
PDR	Preliminary design review	SRR	System requirements review
PFAR	Post-flight assessment review	TRR	Test readiness review
PIR	Program implementation review		

FIGURE 6.1
NASA program life cycle shows the steps of a large-scale development space program. (Courtesy of NASA.)

access to space. Programs are big endeavors, and they include two major components: (1) *formulation* and (2) *implementation*. The formulation phase is the study and development of the ideas, while the implementation phase is the actual performance of the concept and achieving the end result. The Apollo Program is an example of such a program, and the more recent NASA Constellation Program is another. The Apollo Program included many projects, such as the development of the Lunar Excursion Module (LEM) or the Saturn V rocket or the Command and Service Module. Likewise, the Constellation Program is as equally complex. The development of the Ares rockets, Lunar Surface Access Module (LSAM), and the Orion capsule are examples of projects. Figure 6.2 shows the NASA project life cycle. It is this life cycle that is most relevant to the development of a rocket system.

The program and project life cycles enable the rocket scientists and engineers to categorize all the element goals of the mission program and the subsets of rocket development efforts that must be reached in order to reach a successful conclusion. The cycles include many so-called "key decision points" (KDPs), which is government speak for "go or no go." The project life cycle includes phases A through E, which are defined as follows:

- *Phase A*: Concept and Technology Development (i.e., define the project and identify and initiate necessary technology).
- *Phase B*: Preliminary Design and Technology Completion (i.e., establish a preliminary design and develop necessary technology).



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ORR	Operational readiness review	SIR	System integration review
PDR	Preliminary design review	SRR	System requirements review
PFAR	Post-flight assessment review	TRR	Test readiness review
PIR	Program implementation review		

FIGURE 6.2
NASA program life cycle shows the steps of a spacecraft development space program. (Courtesy of NASA.)

- *Phase C:* Final Design and Fabrication (i.e., complete the system design and build/code the components).
- *Phase D:* System Assembly, Integration and Test, Launch (i.e., integrate components, and verify the system, prepare for operations, and launch).
- *Phase E:* Operations and Sustainment (i.e., operate and maintain the system).
- *Phase F:* Closeout (i.e., disposal of systems and analysis of data).

The program and project life cycles really do offer an outline or a template for any large-scale technology development effort. In order to implement the life cycles, we must follow the *systems engineering process* (SEP).

The SEP is the process for describing the path for mitigating program and project risks. The risks can be cost, technical, managerial, safety, part availability, logistics, and a myriad other things. An example of cost risk is the collision of the International Space Station (ISS) and Space Shuttle programs' overall funding. The ISS development and construction continued to spiral out of control with never-ending budget overruns. Because NASA's overall budget was and is finite money from other programs, such as the Space Shuttle, upgrades were continuously cut in order to maintain the ISS schedule. The cost risk was even further increased with the Challenger and Columbia accidents (due to technical and safety risks). The overall ISS program, including its implementation, is now at risk because the Space Shuttles are being grounded immediately following the ISS construction finalization. Therefore, there will be no way to get crew and supplies up to and down from the station without relying on the Russian launch vehicles. Using the SEP has led NASA to the Ares I rocket development and the Orion space capsule to fill the void that will be left when the Space Shuttle program is grounded.

6.1.1 Systems Engineering Models

What does SEP look like, and how does it work? Figure 6.3 is the "standard V model" of systems engineering. It starts at the top of the left side of the V with a "top–down" view and is where the "big picture" is generated. Here is where the idea of the overall architecture for the system begins to take shape. System-level design requirements are defined but at a very top level in the *system functional review* (SFR). Then, the path of the SEP flows down the leg of the V where individual components' design requirements are developed in the *preliminary design review* (PDR). Once the design requirements of the complete system down to the component level are developed, then a *critical design review* (CDR) is held to make final adjustments to the blueprints before components are built and tested. The nomenclature here is important as any modern rocket scientist or engineer will often be working hard to meet the PDR or CDR deadlines.

Afterwards, the CDR fabrication of components begins. The components are integrated together into a larger system of subsystems, and testing begins following the *test readiness review*. Following rigorous testing, the system goes through the *system verification review* where the analysis of all the data of the SEP to date is conducted to determine if the rocket is ready to move forward into operational status. If the analysis suggests that more development is needed, then the process starts over again at the top of the left side of the V.

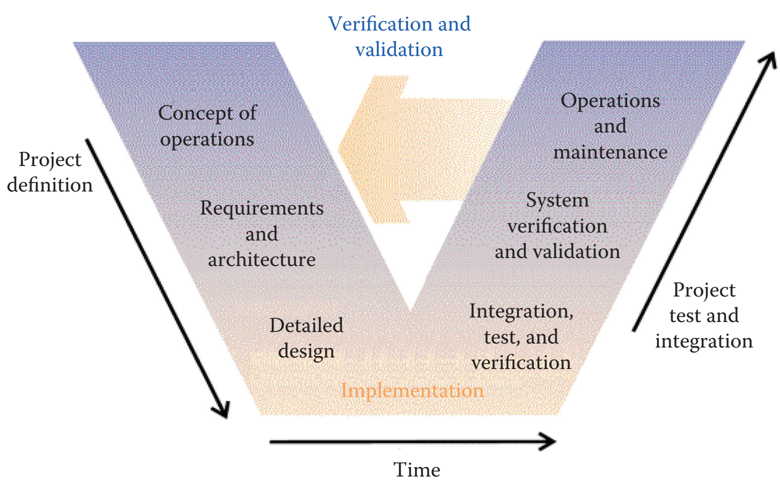

FIGURE 6.3
Standard systems engineering V is the template used by many programs to maintain best systems engineering practices. (Courtesy of the U.S. Air Force.)

We should also note here that NASA is making a step away from the V model and is implementing a *systems engineering engine* (SE engine), as shown in Figure 6.4. In the 1995 version of the *NASA Systems Engineering Handbook,* the V model was quite prevalent. In the 2007 version of the handbook, there is no mention of the V, and it is replaced by the SE engine. Even though the SE engine is not totally unlike the V, it is tailored more to NASA-type programs and projects. After all, the SEP is meant to be a living and update-able process and is not set in stone as the only way; rather, it is a template for a process. The SE engine is just the next step in refining the SEP. (Note here that the argument can be made that the SE engine is just a restatement of the V model, but displayed in a different manner.) A closer look at the SE engine reveals that it shows a bit of resemblance to an H, though nobody has yet started calling it that. DoD still uses the V model, and so do many other organizations. The point of this section isn't to debate which one is better, but merely to show that the two methods exist. There are other SEPs, such as "spiral development," which is again possibly just another way to display the SEP. Figure 6.5 shows the typical spiral development process. These SEP tools should be implemented to aid the rocket scientists and engineers in the rocket development efforts. One or all three or even others might be implemented, but, in reality, it is the fact that an SEP is put in place for the rocket development that is most important.

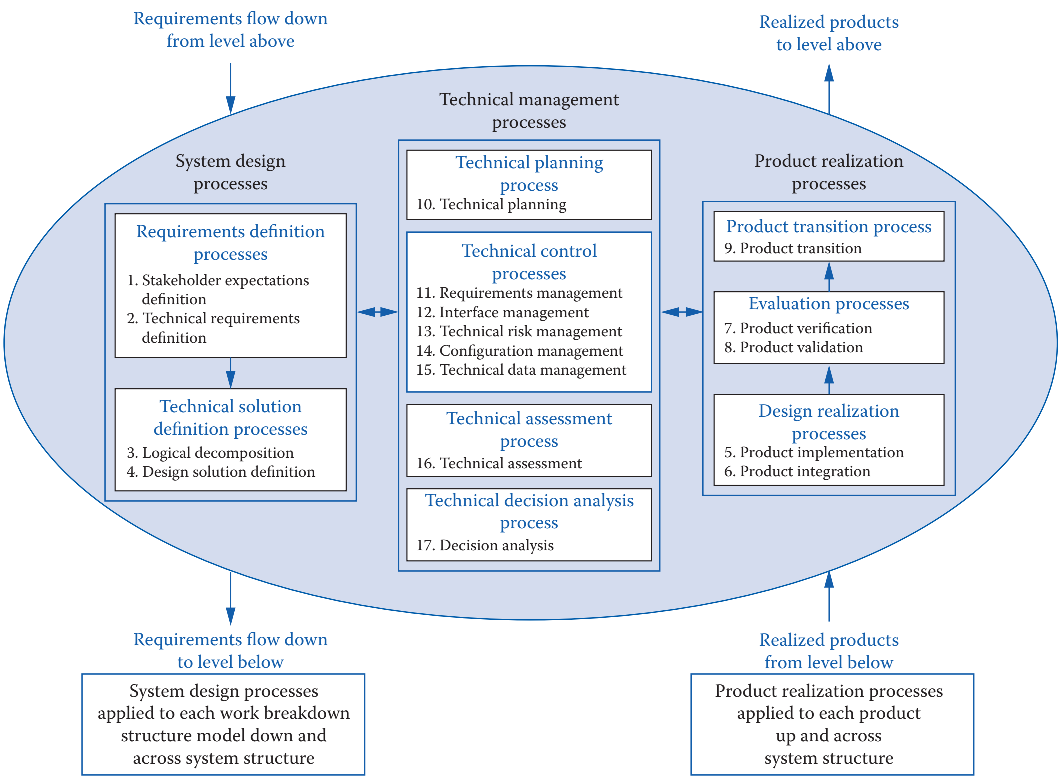

FIGURE 6.4
NASA systems engineering engine. (Courtesy of NASA.)

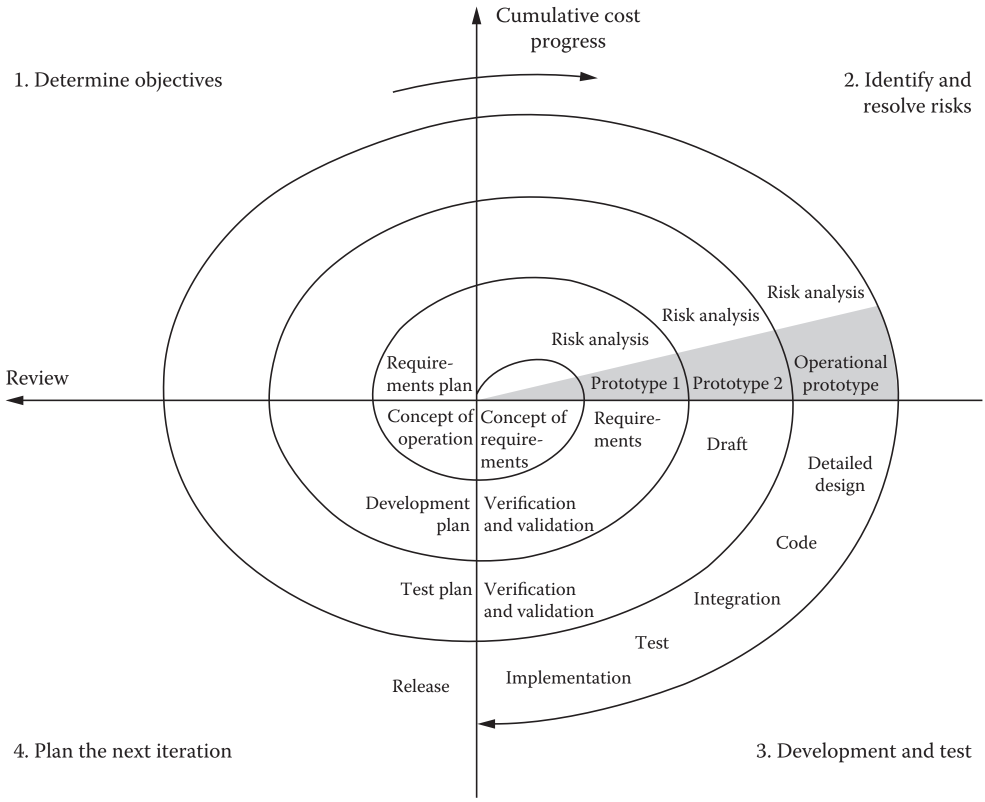

FIGURE 6.5
Spiral development systems engineering model. (GNU free document license image.)

6.1.2 Technology, Integrated, and Systems Readiness

Now that we have an understanding of the process for large system development, how do we gauge where the rocket is in the life cycle? The NASA and DoD community uses a concept known as *readiness levels* that describe the maturity of components and integration. The individual components are described by technology readiness levels (TRLs), which range from TRL 1 to TRL 9. Figure 6.6 shows the definition of the NASA TRLs. There are other definitions of TRL, but the NASA TRLs are most directly applicable to rockets as that is what they were designed for. The beginning rocket scientists and engineers must learn these definitions, as discussions, meetings, tests, and presentations of rocket development efforts always end up in an argument as to what TRL a particular component has matured. It is typically analysis, modeling and simulation, and testing of the SEP that lead to the particular rocket component being matured to the next TRL with the goal of reaching TRL 6, where a flight experiment can be conducted, and then reaching TRL 9 where flight operations can begin.

Figure 6.7 shows the NASA TRL definitions along with the *integrated readiness level* (IRL) definitions. Where TRL describes the maturity level of an individual component, IRL describes the maturity of multiple components working together as a subsystem. The development of a rocket system requires that we integrate all the components together and show that they function properly together without causing unwanted interactions

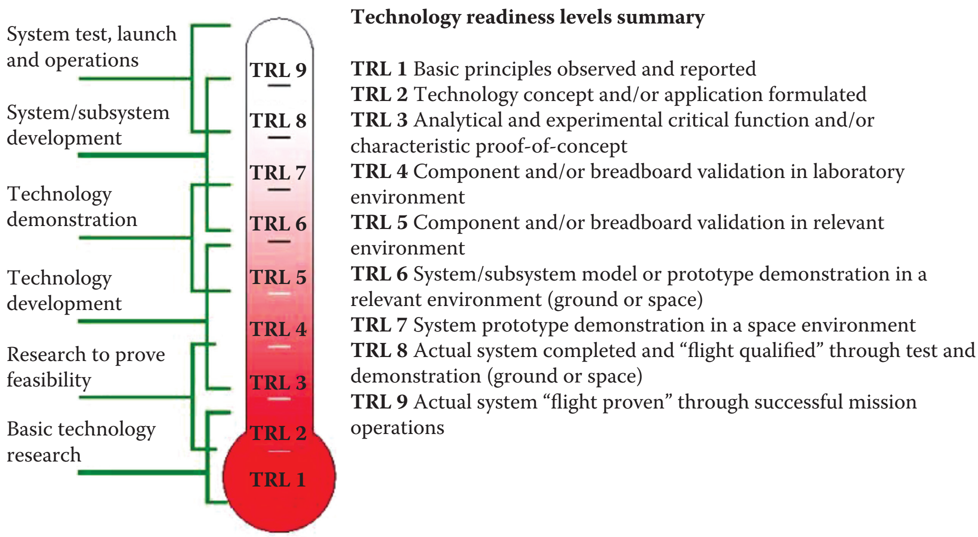

FIGURE 6.6
NASA Technology readiness levels. (Courtesy of NASA.)

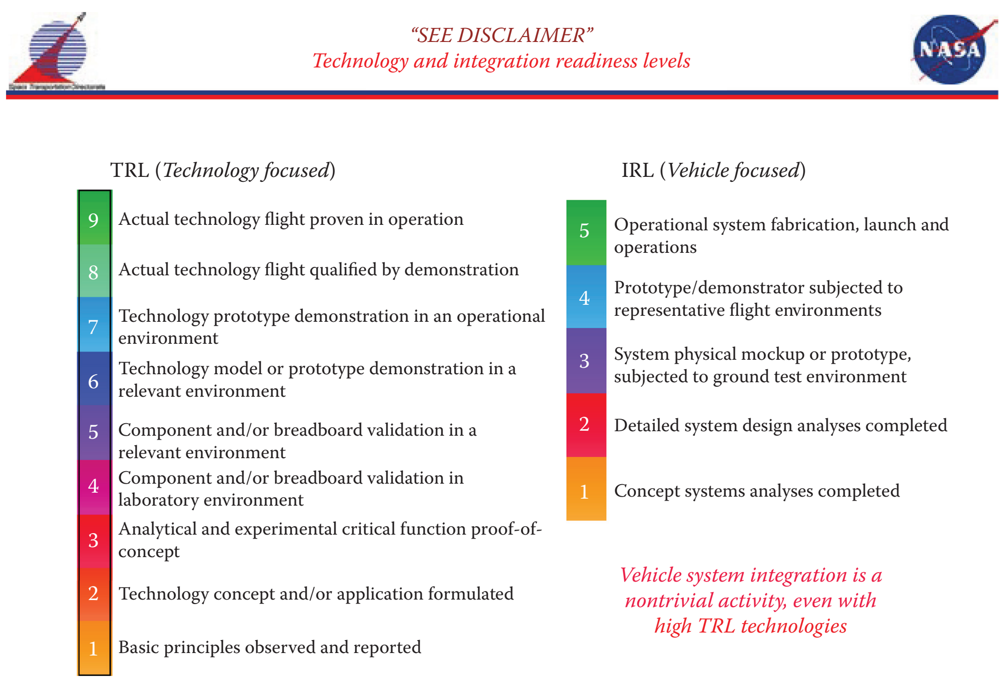

FIGURE 6.7
NASA Technology and Integrated readiness levels. (Courtesy of NASA.)

between the subsystems and components. An example of how TRL and IRL are important might be in the development of a spacecraft that is to implement a pulsed plasma thruster (PPT). The PPT generates enormous amounts of electromagnetic noise with each pulse. This noise could induce electrical energy onto the command-and-control avionics, which could be catastrophic to the vehicle. While both the command and control avionics and the PPT engine might be at TRL 6 individually, when integrated together in an integration test, we would see that the IRL is low. Once proper electromagnetic shielding or high-voltage pulse issues are addressed, then the subsystem IRL would increase, and the risk of flying the two components would decrease. Or, from a more optimistic description, the successful implementation of the PPT and the avionics would likely follow with both high TRL and IRL.

This leads us to another tool in the discussion of readiness. Figure 6.8 is a graph of IRL versus TRL and shows that there is a "pathway to success" that we will define as the *systems readiness level* (SRL). Where the TRL gives us a metric of the readiness of individual components and the IRL gives a metric of the readiness of the level to which the component has been tested with the larger system that the component is to be connected to, the SRL gives us a metric of the complete rocket system development effort. Systems with high TRL and low IRL or low TRL and high IRL will prove to be less likely to be successful than if they had both high TRL and high IRL. Also, the SRL pathway demonstrates that there is a more efficient route for the development resources. Spending too much of the project's resources too early on TRL improvement might be wasteful as later IRL developments (like with the PPT and the avionics example) force a redesign of components. The SRL is the diagonal line between the two. Because the SRL is along the diagonal of the TRL and IRL, it is related mathematically to them by

$$\text{SRL} = \text{IRL} - \frac{\text{TRL}}{2} - \frac{1}{2}. \tag{6.1}$$

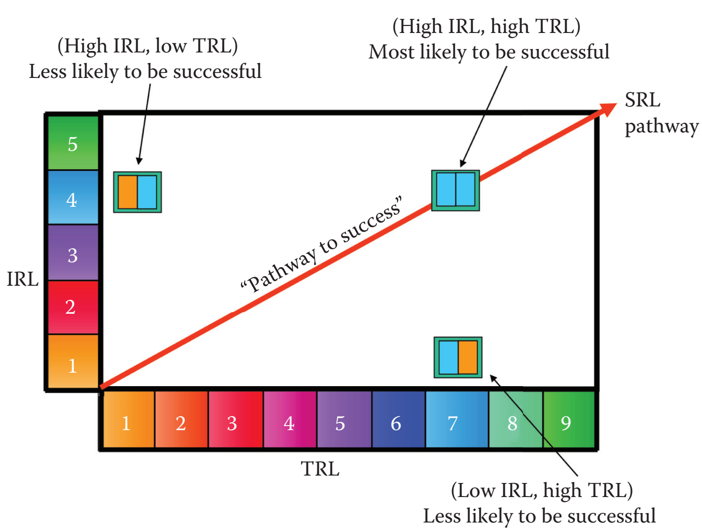

FIGURE 6.8
The systems readiness level is the pathway to success for rocket development.

Thus, an SRL of zero is the optimal readiness for the system. If the SRL is a positive value, then too much effort is being expended on integration development and not enough on component technology readiness. If the SRL is negative, then too much effort is being expended on the component development and not enough on the systems development.

The knowledge gained from Chapters 1 through 5 gives us the basis to define a rocket system and mission. With the program and project life cycles, the SEP, TRL, IRL, and the SRL pathway to success, the rocket scientists and engineers have the tools to implement a rocket development effort. Once the definition of the rocket is developed and the maturity level of the concept is determined, the refinement of the design takes place and the fabrication of test components begins. As the SEP unfolds, more and more tests are required; the analysis of test data and the modeling and simulation of new refinements lead to new more detailed and integrated tests. By applying a good SEP and the tools discussed in this section, more efficient tests can be devised that will identify many parameters in single tests that, without an SEP, would require multiple tests and, therefore, more budget and resources. A good SEP is a means of developing an optimum test program for a limited budget and time. The rest of this chapter will be involved with the types of testing required for rocket development, and some examples of large integrated test programs will be discussed.

6.2 Measuring Thrust

As we have shown throughout the previous chapters (most specifically Chapters 3 and 4), thrust is a force. The force is created by various mechanisms depending on the type of rocket engine used, but, in general, the force is created by exhaust gases escaping out the end of a nozzle at high velocities. In order to characterize a new engine to determine if it performs as predicted through the theory and design parameters, it must be tested. Clearly, one of the most important parameters of a rocket engine that needs to be well characterized is thrust. So, how is thrust measured?

Figure 6.9 shows a schematic of a basic *thrustometer,* which is a tool used to measure rocket engine thrust. It is often referred to as a "thrust gauge" or a "thrust meter," but, for our purposes, we will call it a thrustometer. A *test stand* is the main structure of the apparatus and holds the pieces together, including the engine, for the test purposes. The schematic shown in Figure 6.9 is for a vertical (downward thrusting) test. The rocket engine is placed upside down on top of a *load cell* or *scale*, which is the tool that measures the actual force due to thrust much in the same way that the bathroom scale measures the force on body mass due to gravity. The simplest amateur rocket thrustometers actually use a bathroom scale for this piece of the apparatus. More complex thrustometers use *hydraulic-* or *strain gauge*-type load cells that can handle much higher incident forces on them.

From the force-measuring component, a data-acquisition system is connected. The data-acquisition system might consist of fast video cameras; high-speed analog-to-digital voltage and current sensors; acoustic sensors (like microphones); accelerometers; cables; and a computer or multiple computers to capture, store, and analyze the data.

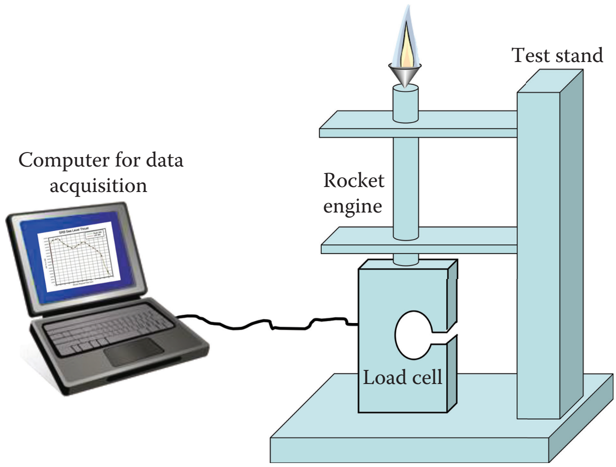

FIGURE 6.9
Basic thrustometer.

6.2.1 Deflection-Type Thrustometers

Figure 6.10 shows the simplest deflection thrustometer. A spring scale is used as the thrust-measuring component. A pencil is attached to the deflection needle of the scale, and, as the rocket is fired and thrusted downward, the needle deflects and makes a mark on a piece of paper, giving the peak magnitude of the thrust force delivered by the engine. If a video camera is added to this setup, then how far the needle is deflected versus time can be recorded as well, giving a thrust profile for the engine. The scale can be calibrated with known masses, and this is why this type of thrustometer is quite popular in the amateur

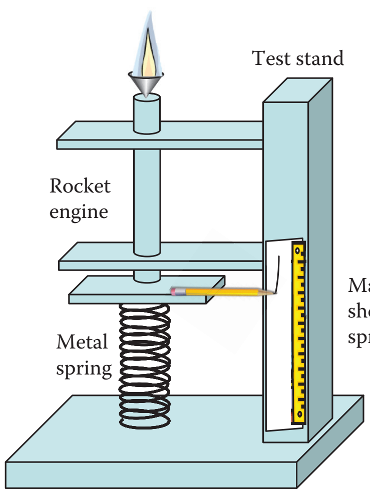

FIGURE 6.10
Simple spring and pencil thrustometer.

rocketry community. From a mathematical perspective, the spring-scale thrustometer is quite simple. The force on a spring can be found as

$$F = -kx, \tag{6.2}$$

where k is the spring constant and x is the deflection (compression or tension) distance of the spring. The spring scale is calibrated by determining what mass causes what deflection. This is calculated by

$$F = -kx = mg. \tag{6.3}$$

Figure 6.11 shows a graph of the spring constant versus mass for a 1-cm deflection of the spring. This graph is useful in designing the thrust scale because it tells us what the spring constant and, therefore, material properties the spring must have. Realizing that Equation 6.3 can also be rewritten in terms of rocket thrust, we see that

$$F_{thrust} = -kx = \dot{m}C. \tag{6.4}$$

Thus, with some knowledge of the engine to be tested, we can not only measure the thrust by watching the deflection of the spring, but we can also determine the mass flow rate if we know the equivalent velocity. Note that the negative sign only signifies the direction.

Figure 6.12 shows a *deflection bar* thrustometer. The deflection bar works similar to the spring scale discussed earlier in this section, except that the spring is replaced by a metal bar that spans across a gap between two supports. Like the spring, there is a fairly

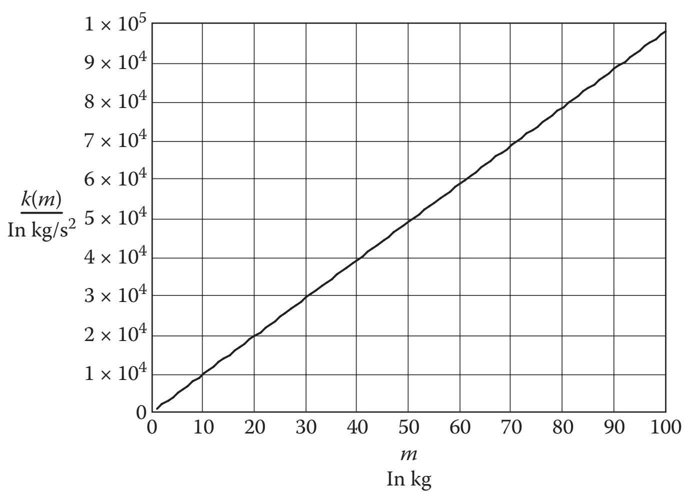

FIGURE 6.11
The spring constant versus mass required to deflect a spring by 1 cm.

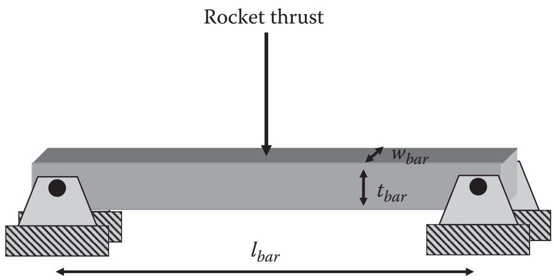

FIGURE 6.12
Deflection bar thrustometer.

straightforward mathematical description of the bar's deflection if a force is incident on it in its midspan, as shown in Figure 6.12. The thrust of the engine is given by

$$F_{thrust} = \frac{16 \times E w_{bar} t_{bar}^3}{l_{bar}^3}, \tag{6.5}$$

where x is the measured deflection of the bar, E is the *modulus of elasticity* of the bar, w_{bar} is the bar width, t_{bar} is the bar thickness, and l_{bar} is the bar length between the support points of the span. The technique for measuring the beam deflection is the component of this type of thrustometer that determines its sophistication. Something as simple as the moving needle and the video camera can be used, which is typical with the amateur community. More sophisticated force transducers can be implemented that enable digital collection of the deflection via a computer.

Equation 6.5 is used to design the thrustometer to match the level of thrust expected to be measured. If a steel bar is to be used, then the modulus of elasticity of the bar will be about 200×10^9 N/m^2. Therefore, the width, thickness, and length of the bar can be optimized for the thrust levels expected.

6.2.2 Hydraulic Load Cells

Figure 6.13 shows a different type of load cell for measuring thrust. This thrustometer makes use of a hydraulic system. The resistance to the thrust is due to compression against a hydraulic fluid in a piston. The fluid can be compressible or incompressible, but a compressible fluid adds complication to the calculations. If the fluid is a simple incompressible fluid, then the force on the hydraulic load cell is

$$F_{thrust} = P_{gauge} A_{piston} = P_{gauge} \pi r_{piston}^2, \tag{6.6}$$

where P_{gauge} is the pressure measured by the gauge, A_{piston} is the cross-sectional area of the piston, and r_{piston} is the radius of the piston. We can use Equation 6.6 to design the thrustometer to the scale we need to measure the expected thrust from the engine by sizing the radius for a range of expected pressure and thrust.

6.2.3 Strain Gauge Load Cells

The most common type of thrustometer used in modern rocket tests is based on the *strain gauge load cell*. Strain gauges measure the change in resistance of a material due to

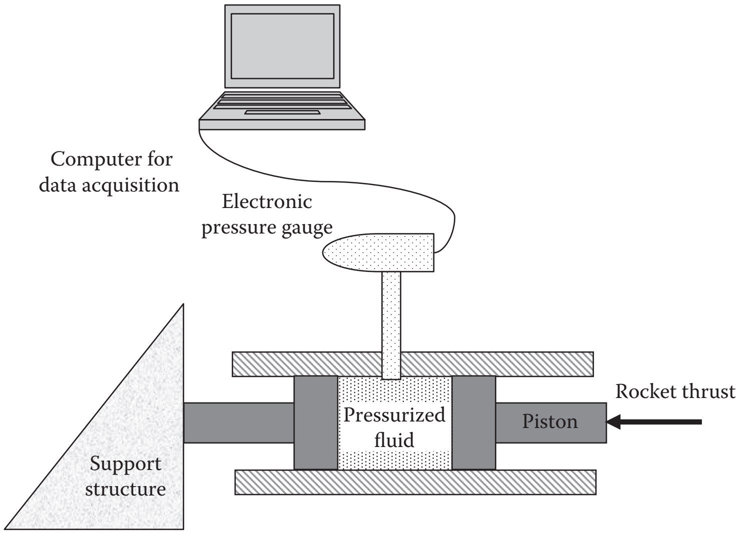

FIGURE 6.13
Hydraulic load cell thrustometer.

compression and bending. A tiny circuit is bonded to a beam or structure, and, as a load on that structure will generate stress and strain, the circuit is also stressed and strained. The mechanical stress causes a change in the circuit's electrical parameters, which is measured and enables a strain measurement. In fact, these types of gauges are commonly used because the output of them is an electrical signal that is easily measured as a function of time through an analog-to-digital data-acquisition system and computer.

Figure 6.14 shows a typical C-type load cell. It is called a C-type because it is in the shape of the letter C. A load cell of this type is designed such that the actual physical displacement of the cell components is very small. The electrical strain gauge has no difficulty in measuring small flexures, while, at the same time, no large dynamic changes in the test

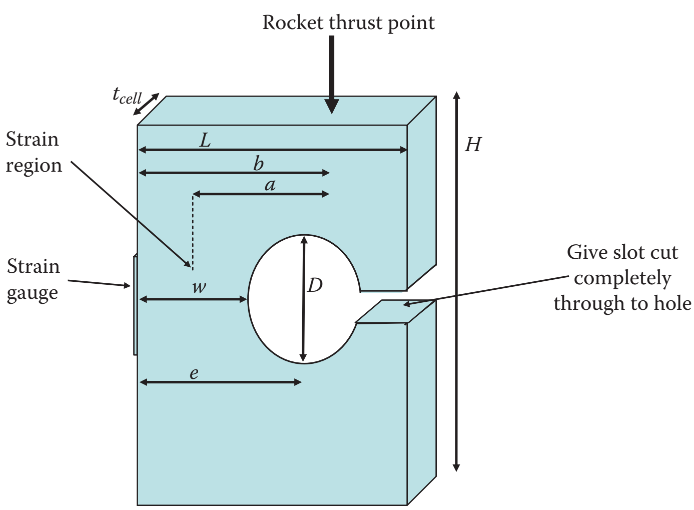

FIGURE 6.14
Schematic of a C-type load cell.

setup increase the safety and reliability of the test. The strain measured on the gauge must be calibrated against known force sources (such as known masses in known gravity or hydraulic pistons). The strain in the C-type gauge is given by

$$\varepsilon_{gauge} = \frac{8F_{thrust}(3b + D - L)}{A_{thrust}Et_{cell}(L - D)^2}, \tag{6.7}$$

where b is the distance from the edge of the load cell to the incident thrust point, D is the diameter of the hole, L is the length of the load cell, t_{cell} is the thickness of the cell, A_{thrust} is the cross-sectional area of the pressure point due to the thrust, and E is the modulus of elasticity for the material of the load cell. Equation 6.7 should be used to design the strain gauge load cell for the level of thrust expected during any tests planned.

There are other configurations of strain gauge load cells that range from simple in geometry to quite complex. Figure 6.15 shows a simple bar-type strain gauge load cell. Load cells can be as simple as the bar type to extremely complex.

Figure 6.16 shows a schematic of the T-97 Thrust Measurement System at ATK Alliant Techsystems in Utah. The T-97 test stand is used by NASA to measure the thrust

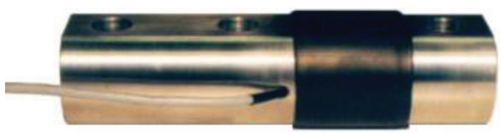

FIGURE 6.15
Bar-type load cell.

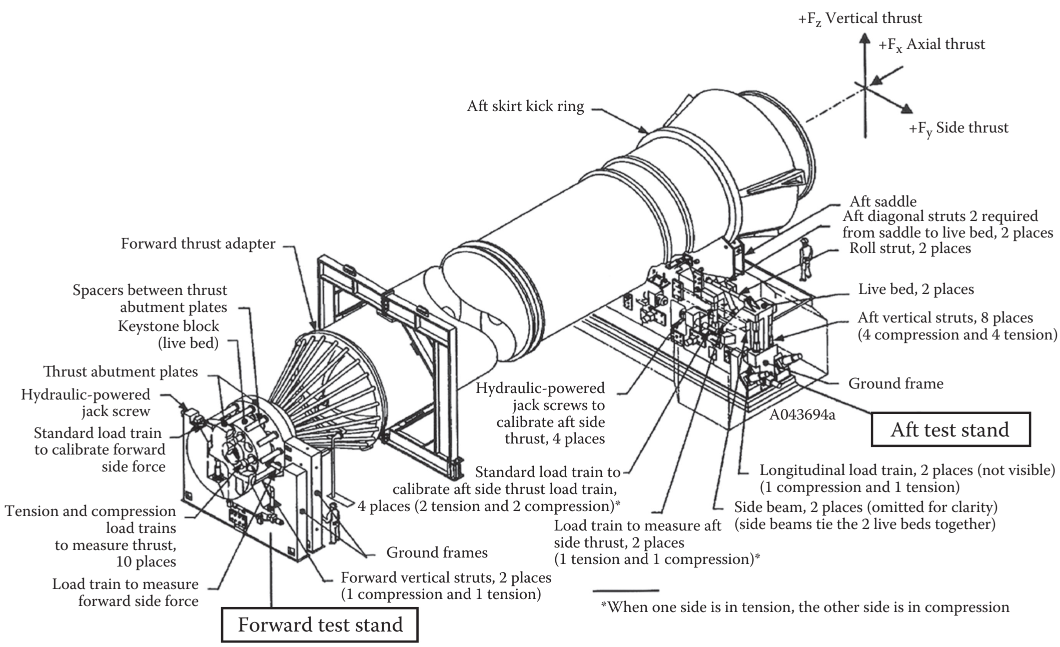

FIGURE 6.16
T-97 Thrust Measurement System for testing SRBs. (Courtesy of NASA.)

FIGURE 6.17
A solid rocket booster firing on the T-97 test stand. (Courtesy of NASA.)

FIGURE 6.18
Titan IV solid rocket motor test. (Courtesy of the U.S. Air Force.)

generated by the Space Shuttle solid rocket boosters (SRBs). The stand uses tension and compression load cells in 10 different locations to measure thrust. Other load cells are used to measure off-axis sliding forces. Figure 6.17 is a picture of the T-97 Thrust Measurement System with an SRB in place and firing. The SRB generates more than 12.45 MN of thrust.

Other test stands have been utilized throughout the history of rocket programs for thrust measurement. Figure 6.18 shows the Titan IV Solid Rocket Motor Test Stand while the motors are firing. The stand was designed to measure the more than 7.5 MN of thrust from the motor.

Figure 6.19 shows the Redstone rocket test stand in Huntsville, Alabama. The Redstone rocket was integral in the Mercury project that first launched men into space. The rocket was a liquid rocket system that generated over 350 kN of thrust.

Figure 6.20 shows the Saturn V engines on a test stand. The rocket used five F1 engines that generated over 34 MN of thrust. The rocket was the one that first landed man on the moon (see Chapter 1).

Figure 6.21 shows the Space Shuttle Main Engine (SSME) on the test stand at Stennis Space Center in Hancock County, Mississippi. Figure 6.22 shows the SSME thrusting

FIGURE 6.19
Redstone rocket on test stand in Huntsville, Alabama. (Courtesy of NASA.)

FIGURE 6.20
Saturn V rocket F1 engine test. (Courtesy of NASA.)

FIGURE 6.21
An SSME test. (Courtesy of NASA.)

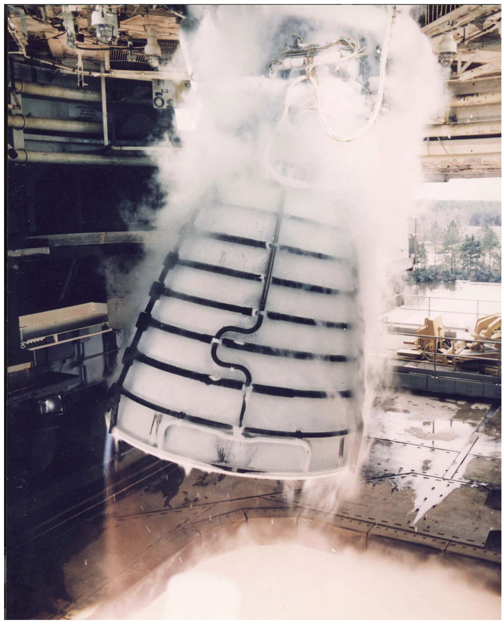

FIGURE 6.22
An SSME gimbal test. (Courtesy of NASA.)

while tilted. This photo was taken during a gimbal test. As was discussed in Chapter 5, Section 5.2.4, the SSMEs must be able to gimbal while thrusting to maintain thrust vector control. This test enabled the SSME engineers to determine the thrust vector incident on the gimbals and other control mechanisms.

Figure 6.23 shows the complexity of NASA's Stennis Space Center Test Complex. The complex is home to five separate test stands. The A1 test stand was designed for testing the Saturn V. A2 was designed for testing the SSMEs, and A3 is under construction for testing of the J-2X engine for the Ares rockets. The B-1/B-2 test stand was built to test the Delta IV rocket engines. And the E-Complex is used for testing of smaller rocket engines and support components.

The J-2X engine has also been tested at the Plum Brook Facility in Sandusky, Ohio. Figure 6.24 shows the J-2X being lowered into the vacuum chamber before a test firing at that facility. Figure 6.25 is a schematic of the entire test facility illustrating the complexity of test apparatus required to test modern rocket engines in space-like environments. The facility enables the firing of the engines while in a vacuum to simulate the vacuum of space. Recall from the definitions of TRL that there is a need to test in operational environments before flight readiness is achieved.

Figure 6.26 shows the NASA Ames Test Facility for test firing hybrid rocket engines. The test facility needs the complexity of testing both liquid engines and solids. Figure 6.27 shows an environmentally friendly wax-based hybrid engine on the stand while firing.

FIGURE 6.23
Multiple test stands at Stennis Space Center Test Complex. (Courtesy of NASA.)

FIGURE 6.24
The J-2X engine being lowered into a vacuum chamber for testing. (Courtesy of NASA.)

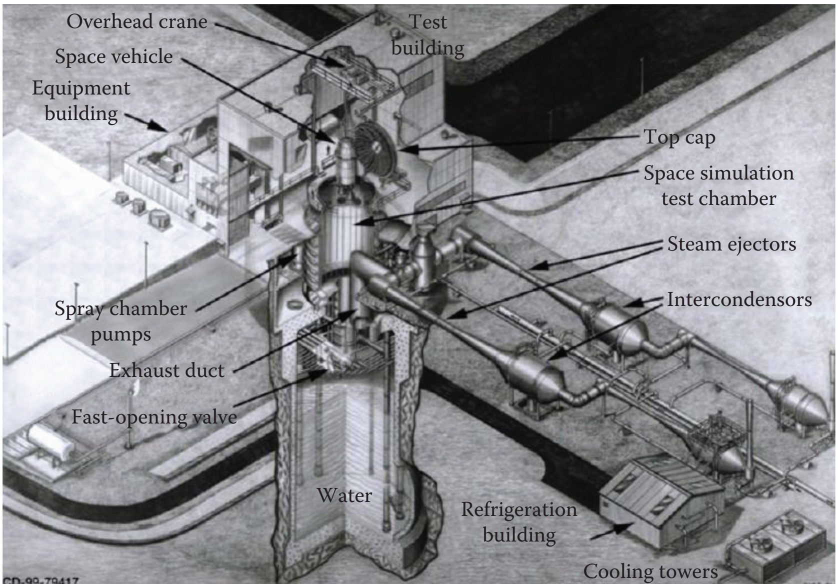

FIGURE 6.25
A schematic of the Plum Brook Facility in Sandusky, Ohio. (Courtesy of NASA.)

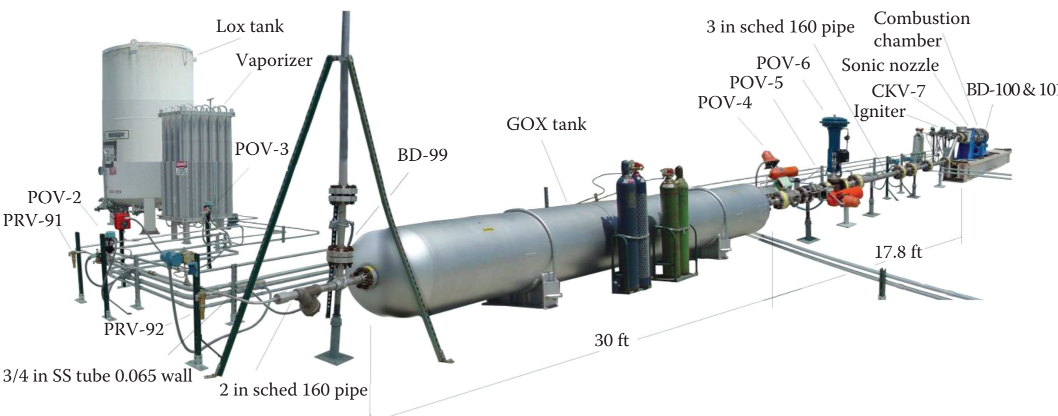

FIGURE 6.26
The hybrid rocket engine test stand at NASA Ames in Mountain View, California. (Courtesy of NASA.)

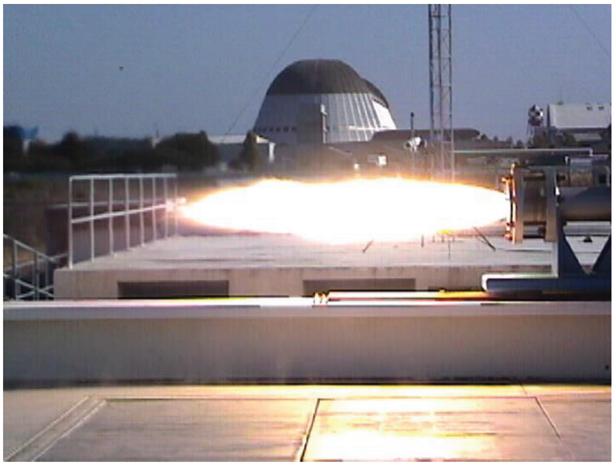

FIGURE 6.27
A wax-based hybrid rocket engine test at NASA Ames in Mountain View, California. (Courtesy of NASA.)

6.3 Pressure Vessel Tests

There are tanks on almost all rockets. In rocket scientist/engineer lingo, we say "tankage." The tankage onboard these rockets vary from liquid fuels and oxidizers to pressurized gases. As we saw in Chapter 5, Sections 5.2.1 and 5.2.2, sometimes, tanks are needed for nonoptimal performance control, such as the control of pogo and cavitation. There are tanks on hybrid engine rockets that hold either the oxidizer or the fuel, depending on if the engine is a reverse hybrid or not. Ion thrusters use tanks to store propellant gases such as xenon. And, on all engines, there is a combustion chamber of some sort.

All of these tanks and chambers are vessels for holding contents under potentially very high pressures. Therefore, these tanks are known as *pressure vessels*. As was shown in Chapter 4 in our rocket design example, we had to design the wall thickness of the combustion chamber to withstand the expected pressure within it during operation; likewise, we must do the same for the other pressure vessels on the rocket.

So, how do we test these pressure vessels? First of all, we must realize that there are many types of pressure vessels. There are metal tanks that are made with smooth walls and welded together to make a sphere or cylinder or some other desired shape. Some metal tanks have ribs and bands added to the outside for strength. There are composite tanks made of materials like fiberglass, Kevlar™, spun fibers and other epoxies, and many combinations of different materials (even carbon nanotubes). Some of these tanks are then slipped into a sock or web of extremely strong fiber materials to add tensile strength to it as the bands and ribs would on metal tanks. There are tanks made of metal and composite materials, as with the NASA X-33 program. And, it was where the metal hard points met the composite walls that the tanks would typically fail during testing. There are tanks that are a solid material for structure with a bladder on the inside to house the fluid. Pressurized gas is then forced between the tank wall and the bladder to squeeze it and, therefore, force the fluid propellants out of the tank and through the flow system. This type of tank/bladder configuration is often used for microgravity applications.

Once the type of tank to be tested has been identified, then the test plan can be configured. From a pressure vessel standpoint, there are several things that need to be tested. Figure 6.28 shows a typical pressure vessel test setup. The vessel is connected to propellant or at least a liquid propellant (or gas, if it is a gas tank) stimulant to be flowed into and out of the tank. The flow in and out is regulated in order to increase and decrease the pressure in a controlled manner. There are pressure gauges connected to the flow loop, as well as directly in line with the tank. Strain gauges are also typically bonded to the tank around its geometry and at potential weak points like weld joints or hard points and seams. Video, infrared (IR), and many other sensors are also typically implemented.

The simplest test to be performed on a pressure vessel design is the *burst test*. A burst test is exactly what it sounds like. The vessel is pressurized until it fails. This type of test can be dangerous because, when the pressure vessel fails, it might explode, or it might be as anticlimactic as springing a leak. However, consider a balloon as a simple pressure vessel test. If the balloon is filled too full, it pops, loudly. A balloon made of superstrong materials and pressurized to extremely high pressures might pop quite violently. Safety precautions must be taken with such tests. Figure 6.29 shows the NASA burst test facility at White Sands, New Mexico, and one composite tank after a burst test. The tank failed with an outward rupture of the material wall.

On the other side of extreme pressure for a tank is the *vacuum test*. This test gives another measure of the strength of the pressure vessel, as the extreme low pressure inside means that the outside ambient air pressure is putting great stress inward on the tank. This type of test allows for measurements of the stress the tank can handle from outside high-incident pressure.

A variant of the burst and vacuum tests requires a *drop test* of the tank beforehand. The tank is dropped from a high altitude, and then it is put through the rigors of the burst and vacuum tests again. This test enables the engineers to determine what type of impacts the tank can withstand and still function properly (or at least not fail catastrophically).

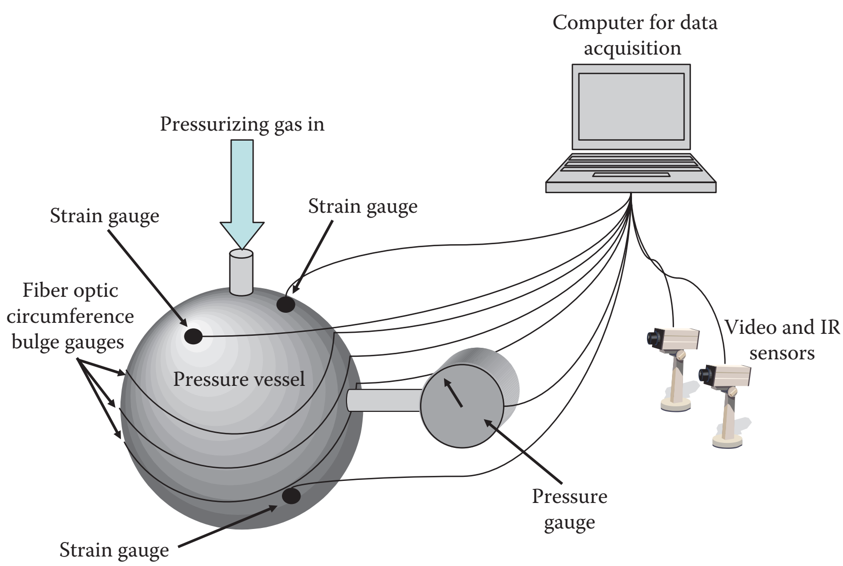

FIGURE 6.28
A typical pressure vessel test setup.

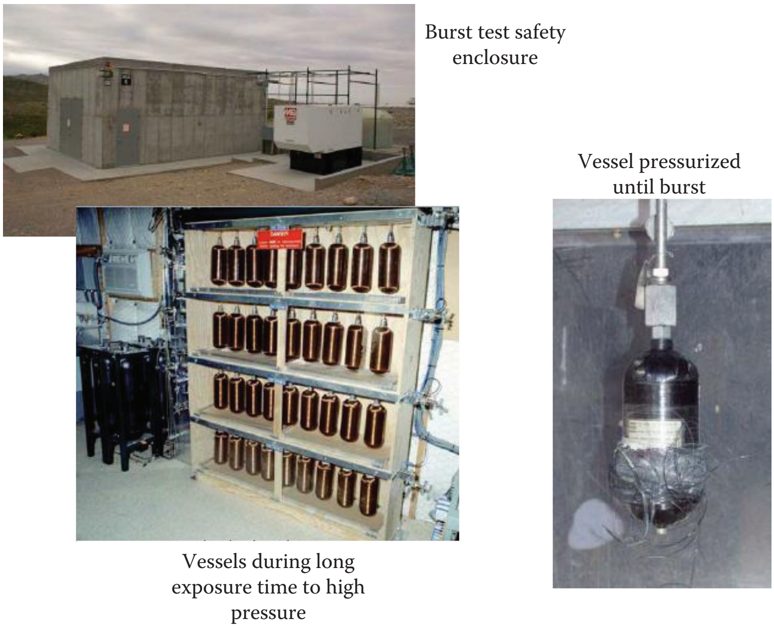

FIGURE 6.29
Burst test facility at White Sands, New Mexico. (Courtesy of NASA.)

Figure 6.30 shows some vessels after a *chemical compatibility test*. Rocket propellants can often be very nasty and reactive chemical compounds. The compatibility test is used to determine if the tank can withstand long-term exposure to such reactive chemicals. Clearly, the tanks in Figure 6.30 could not. Again, safety must be considered in all of these tests as the chemical reactions might be dangerous in many ways. The chemicals themselves can be corrosive or even explosive. Some of the reaction products might be hazardous as well.

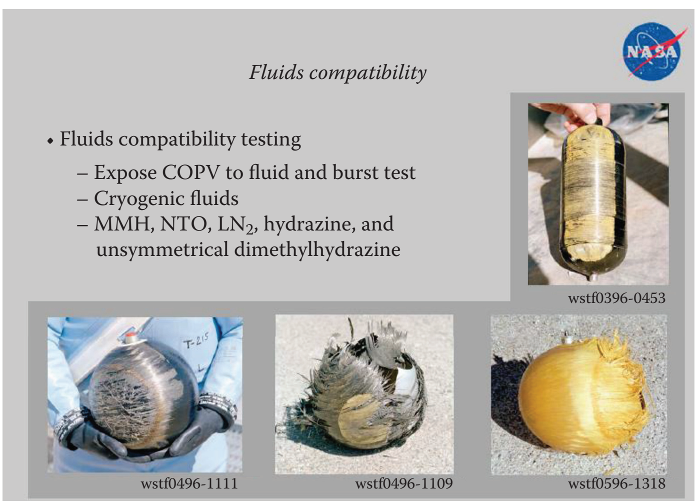

FIGURE 6.30
Materials compatibility test facility at White Sands, New Mexico. These tanks failed when exposed to rocket propellants under pressure. (Courtesy of NASA.)

Along with drop-testing the vessels, they might be put under other stress and strain tests where they are loaded with forces or tensions at certain points on their geometry. *Impulse response testing* is also conducted. This type of testing is sometimes referred to as "vibrational" or "modal analysis" and it consists of impacting the tank with a known impact force (an impulse like a hammer strike) and measuring how the tank "rings." All objects have natural acoustic frequencies, which they vibrate when once excited by an impulse. This test allows the engineers to determine the frequencies at which the tanks ring. If they vibrate too abruptly at particular frequencies, they might shake themselves loose from their connections to the rocket structure. Or, they might shake themselves to pieces much like the vibrato of a soprano can break the fine wine glass.

Because many of these vessels are expensive to construct, it is undesirable to destroy them every time they are tested. Thus, there are many types of *nondestructive evaluation* (NDE) tests that are used. Tanks can be x-rayed to look for faults in the material structures and to check welds and seams and microcracks. Neutron radiation is also used to measure variations in wall thickness and density. Ultrasound systems are used to look for acoustic attenuation properties of the tanks and for microcracking. Electrical properties of the tanks are studied to determine if there are abnormal electrical conductivity regions of the material, which might suggest other structural defects. Radio and microwave imagery of the vessels are implemented for measuring the dielectric properties of the vessels. Even magnetic resonance imaging (MRI) is used to possibly identify impurities and abnormalities in the chemical structure.

There is a large array of NDE tests being studied to find new ways of examining pressure vessels without destroying them or damaging them beyond reuse. Optical and thermal imagery can be manipulated in certain ways that enable the engineers to see where potential stress points might be on tank surfaces. Laser profilometry, interferometry, and speckle interferometry are used to test for strain.

A typical instrumentation and NDE test plan for a composite tank might include:

Visual Inspection: External overwrap inspection for visible damage.

Flash Thermography: Quickly heating surface and watching cooling as a function of time gives insight to subsurface layer delamination.

Borescope Inspection: Inspect the internal liner of the tank for damage and buckling signs.

IR Heat Soak Thermography: Fully heat soak vessel and observe heat signature decay will give insight to delamination.

Shearography: Uses laser image measurements before, during, and after testing and comparing them shows small shears in the surface.

Fiduciary Marking: Marking the tanks before, during, and after tests allows for seeing any tank surface movement.

Pressure, Temperature: Time, pressure, and temperature measurements.

Cabled Girth: Checking for circumferential displacement and disfiguration.

Boss Movement: Do boss hard points move during any/all tests.

Strain Gauge: Gauges are located all over surface to measure disfiguration due to stress.

Fiber Bragg Grating: A fiber optic cable is wrapped around the tank and an optical Bragg grating is set up inside the cable, which allows for detection of very small circumferential changes of the tank.

Acoustic Tests: Impulse response measurements.

Electromagnetic Properties Tests: Uses various electric probes to determine changes in the thickness of the vessel walls.

Chemical and Fluids Tests: Chemical compatibility and fluid flow measurements.

Drop Test and repeat all above.

Figure 6.31 shows a vessel under thermography testing and Figure 6.32 shows a close-up of some of the imagery from that test. The thermography illustrates where there are potential weak spots in the vessel.

Figure 6.33 shows a schematic of a Michelson-type *shearography interferometer* used to measure strain and surface changes in the vessel. Figure 6.34 shows a Space Shuttle Orbital Maneuvering System (OMS) tank under a shearography test. Figure 6.35 pictures some images from the shearography test showing composite layer delamination near the end boss of the tank as well as some displacement and deformation in the Kevlar materials.

FIGURE 6.31
Space Shuttle Orbiter OMS tank under heat soak thermography test. (Courtesy of NASA.)

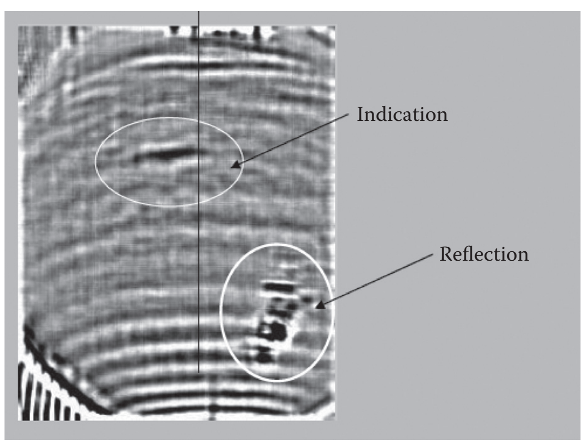

FIGURE 6.32
Space Shuttle Orbiter OMS tank thermography test results. (Courtesy of NASA.)

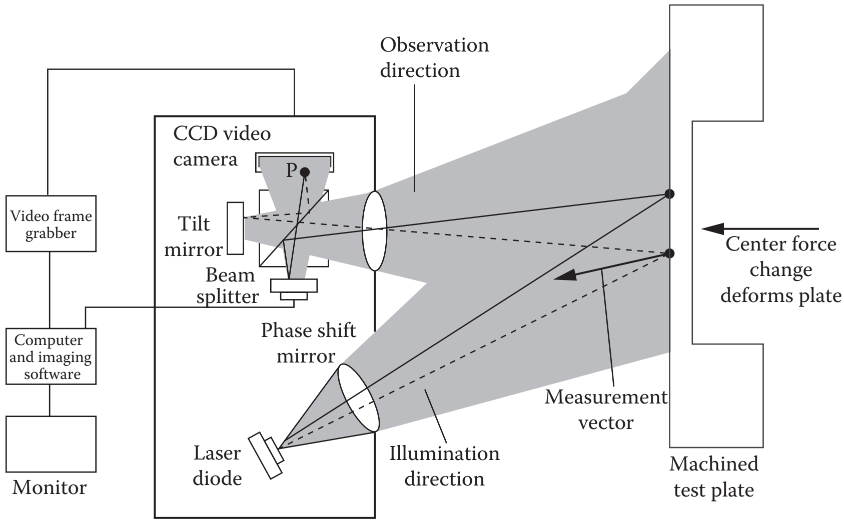

FIGURE 6.33
Schematic of a Michelson shearography interferometer. (Courtesy of NASA.)

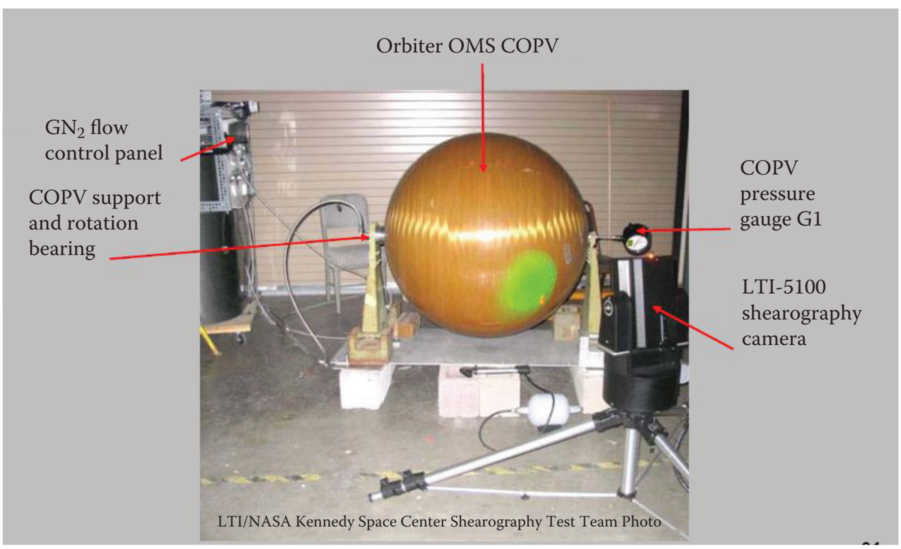

FIGURE 6.34
Michelson shearography interferometer test of Space Shuttle OMS tank. (Courtesy of NASA.)

Figure 6.36 shows a tank covered with sensors and strain gauges to measure stress and surface shape changes. Figure 6.37 shows the tank with electromagnetic sensors attached as well. Also note the fiber optic cable wrapped at the equator of the tank. This is a girth sensor that measures how much the tank expands while under pressure.

Figure 6.38 shows the tankage of SpaceShipOne's hybrid engine. The tank was a composite pressure vessel with a graphite epoxy overwrap. The figure demonstrates several of the tests that the tank underwent prior to being flown.

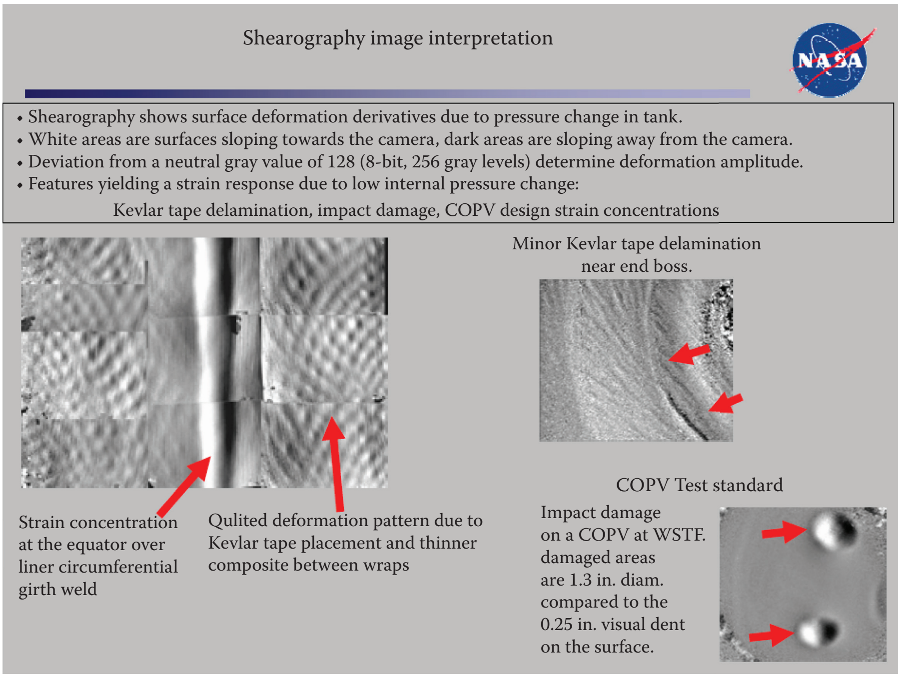

FIGURE 6.35
Michelson shearography interferometer test results of Space Shuttle OMS tank. (Courtesy of NASA.)

FIGURE 6.36
Space Shuttle OMS tank with stress sensors and circumference bulge gauges bonded to it. (Courtesy of NASA.)

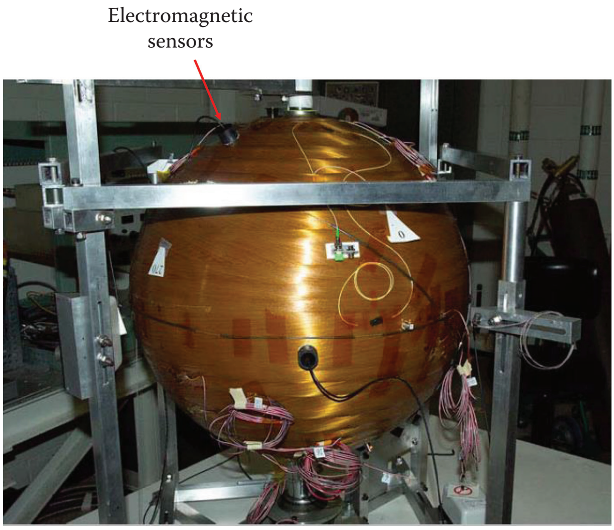

FIGURE 6.37
Space Shuttle OMS tank with electromagnetic sensors attached. (Courtesy of NASA.)

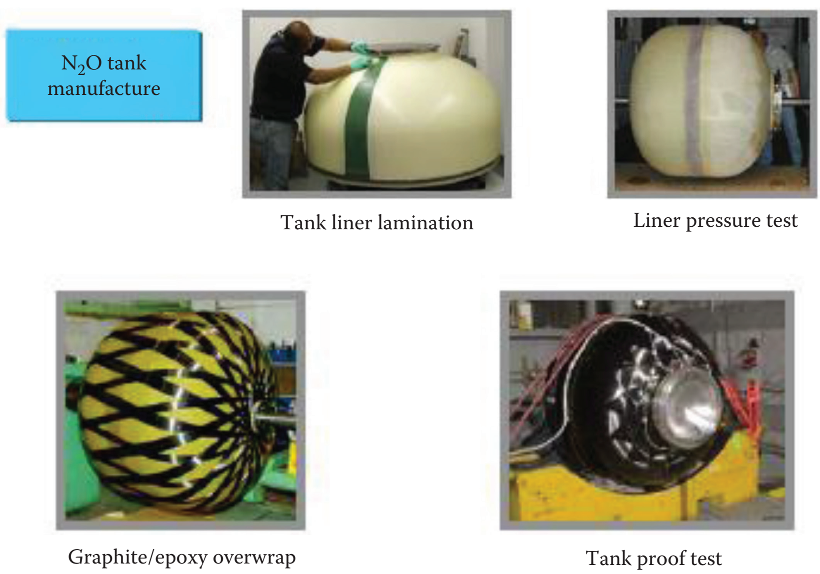

FIGURE 6.38
SpaceShipOne hybrid engine tank testing. (Courtesy of Scaled Composites. SpaceShipOne is a Paul G. Allen Project.)

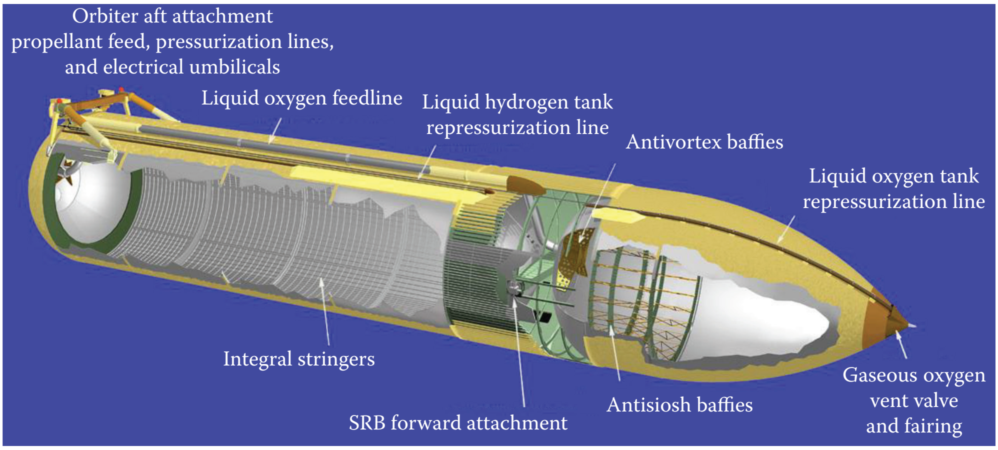

FIGURE 6.39
Cutaway view of the Space Shuttle External Tank. (Courtesy of NASA.)

Perhaps the most famous tankage is the Space Shuttle External Tank (ET). Figure 6.39 shows a cutaway view of the ET. There are actually two tanks inside the ET: one for the liquid oxygen and one for liquid hydrogen. The ET goes through rigorous testing before each flight.

6.4 Shake 'n' Bake Tests

Rocket scientists and engineers will often be overheard discussing the "shake 'n' bake" tests. These tests really are exactly what they sound like. They are designed to shake the rocket vehicle to test for vibrational modes that could prove destructive to the vehicle. And they are designed to bake the rocket vehicle to make certain that the system can handle the large temperature changes that it is likely to be exposed to during flight.

The ride from Earth to orbit is filled with dynamic forces that will shake a rocket to pieces if it isn't designed properly. Therefore, the rocket vehicle is placed on very large shake tables and put through a simulation of the launch and flight environment. This test is also referred to as a "ground vibration test" (GVT). Figure 6.40 shows the ET being loaded into a shake test stand where it was put through a GVT in 1977. Figure 6.41 shows the Space Shuttle Orbiter being loaded into the Dynamic Test Stand at Marshall Space Flight Center in Huntsville, Alabama. The test was called the Mated Vertical Ground Vibration Test and was the first time that all the shuttle elements (Orbiter, ET, and SRBs) were mated together for a vibrational test. The Dynamic Test Stand used for the Space Shuttle testing was originally designed for the Saturn V. Figure 6.42 shows the Saturn V in the Dynamic Structural Test Facility that was built in 1964. It was later upgraded and used for Space Shuttle testing and could potentially be used to test other vehicles, such as the Ares I or Ares V.

The bake test's name is a bit misleading in that it not only measures how the vehicle holds up to extremely high temperatures, but also to extremely low temperatures. In space, the rocket is likely to see temperature swings from as high as 122°C while in sunlight to –122°C while in shadow. Once the rocket is put through the bake test, it is examined with

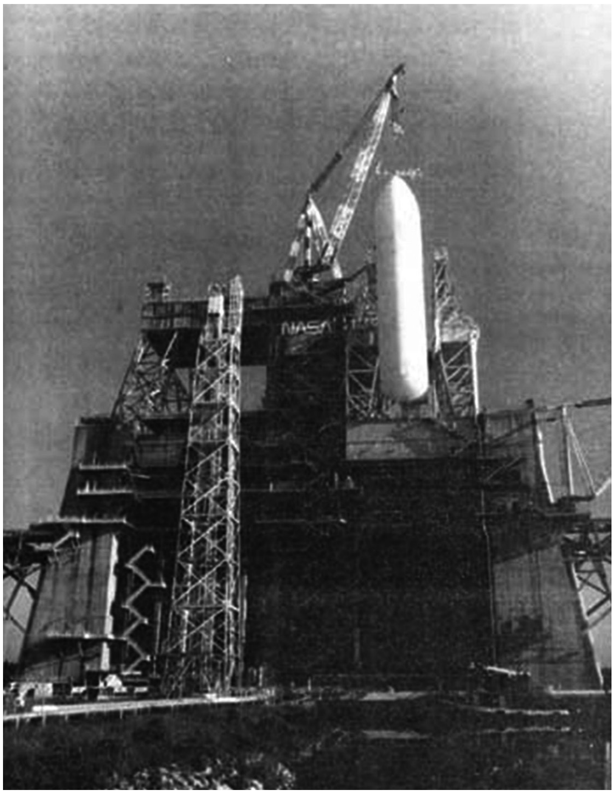

FIGURE 6.40
Space Shuttle External Tank being loaded onto test stand in 1977. (Courtesy of NASA.)

x-rays, neutrons, lasers, microwaves and radio, ultrasound, electrical, and other techniques (including visual examination) for any type of cracking or failure modes to determine if it stood up to the heat and cold extremes.

6.5 Drop and Landing Tests

We have mentioned drop-testing the pressure vessels already, but rocket components that will be exposed to high-impact forces also need to be tested. The best way to do this is through drop testing. Space capsules, such as those used in the Mercury, Gemini, and Apollo programs, landed in the ocean, but still were exposed to significant impact. The Russian spacecrafts typically landed on land and, therefore, were exposed to harder impacts. Drop testing enables the scientists and engineers to design the structure and force dampening couches and other systems to withstand these tremendous impacts. There are other components that are exposed to large impulses due to other mission needs. Stage separation often requires pyrotechnics, which impose an impulse on the spacecraft. Engine startup or restart might also impose an impulse on the rocket system. Drop testing allows for an easy way to measure the effects of these impulses.

FIGURE 6.41
Space Shuttle Orbiter being loaded onto the MSPC Dynamic Test Stand for Mated Vertical Ground Vibration testing. (Courtesy of NASA.)

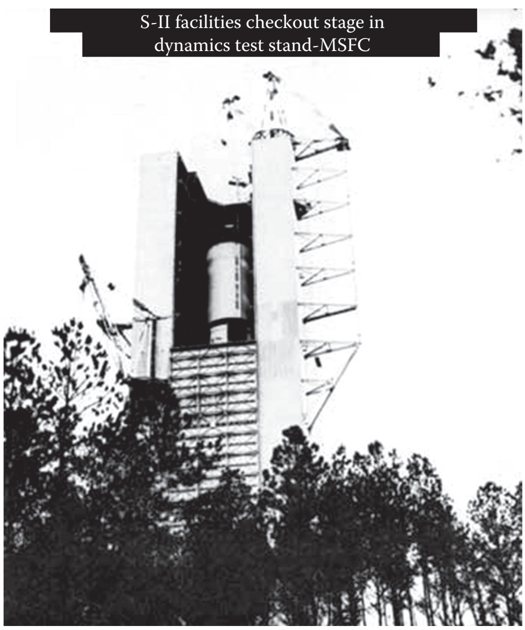

FIGURE 6.42
Saturn V on the Dynamic Test Stand at Marshall Space Flight Center in 1966. (Courtesy of NASA.)

Figure 6.43 shows the Orion spacecraft undergoing a drop test at NASA Langley (Hampton, Virginia). The Orion spacecraft is designed to land on the surface rather than in the ocean, and, therefore, it must be able to withstand the impact forces. The landing system of these vehicles can be tested during such tests.

For vehicles that land like an airplane or even like the LEM on the Moon missions, they need to be tested as well. Landing and drop test are specifically useful in experimentally verifying the design of the landing gear of these vehicles. Figure 6.44 shows a landing test taking place at the NASA Aircraft Landing Dynamics Facility at Langley. This particular test examines how landing gear interacts with the runway during touchdown at high speeds.

FIGURE 6.43
Orion vehicle drop test. (Courtesy of NASA.)

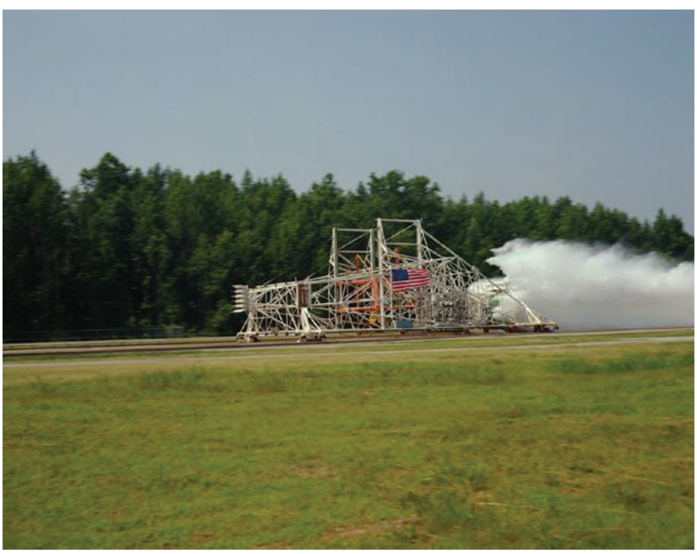

FIGURE 6.44
Aircraft Landing Dynamics Facility at Langley (Virginia). (Courtesy of NASA.)

6.6 Environment Tests

As we have already discussed in Section 6.1.2, part of the technology readiness path is to test the components, subsystems, and systems of the rocket in an environment that is like the environment in which the vehicle will have to operate. This requires test facilities that can simulate the vast envelope of environments seen by the rocket from the ground, through the large dynamic pressure of ascent to the vacuum and temperature extremes, micrometeoroid and particle bombardment, and electromagnetic field exposure to intense friction heating of reentry and the final descent to landing.

All of these extreme environments are simulated through vacuum chambers, particle accelerators, electromagnetic interference chambers, wind tunnels, and other types of test apparatus. Figure 6.45 is a photo from 1963 where a 1/10th-scale Centaur rocket was tested in the Supersonic Wind Tunnel at NASA Glenn Research Center (Cleveland, Ohio). The test was to determine if a safety system for venting fuel was designed properly. The fuel is being vented out of the rocket, and the wind tunnel data show if the vent is moving the fuel far enough from the body of the rocket to prevent it from igniting as it passes the exhaust of the rocket's engine. Figure 6.46 shows a high-velocity projectile impacting a solid surface in the NASA Ames Hypervelocity Ballistic Range (Mountain View, California). The test simulates impacts with orbital debris.

Figure 6.47 shows the Atmospheric Entry Simulator at NASA Ames. The large tank holds air under extremely high pressure and is forced down a trumpet-shaped nozzle that expands the air flow to simulate the change in density of the atmosphere versus altitude. There is also a gun at this facility that fires test models into the stream at reentry velocities near 25,000 km/h. The test facility enables rocket scientists and engineers to test reentry vehicle design performance. Figure 6.48 shows several different types of reentry vehicle

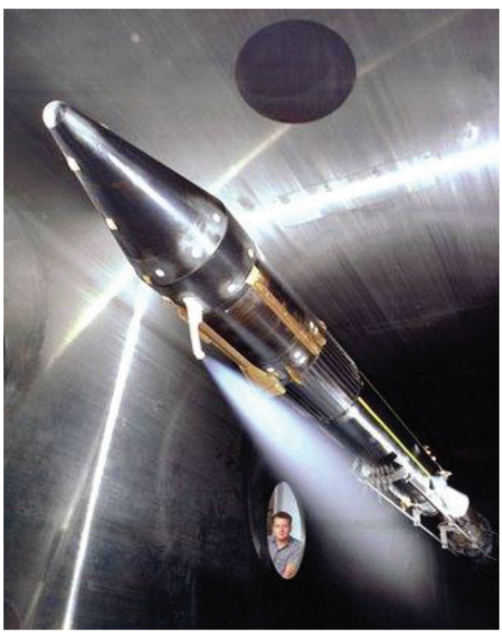

FIGURE 6.45
Supersonic wind tunnel testing of the Centaur rocket design. (Courtesy of NASA.)

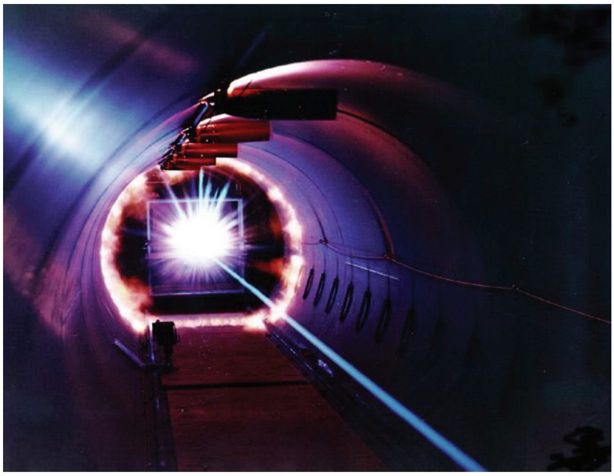

FIGURE 6.46
Supersonic wind tunnel testing of the Centaur rocket design. (Courtesy of NASA.)

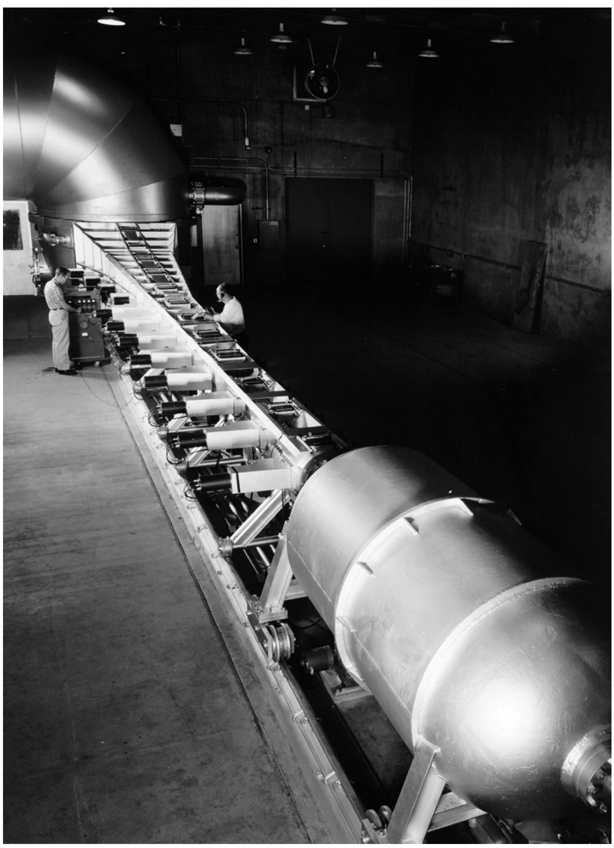

FIGURE 6.47
The Atmospheric Entry Simulator at NASA Ames (Mountain View, California). (Courtesy of NASA.)

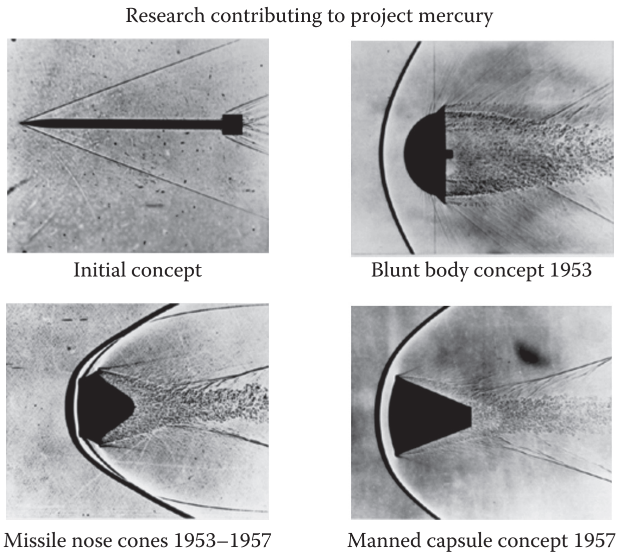

FIGURE 6.48
Shadowgraph data from reentry vehicle design tests. (Courtesy of NASA.)

concepts and how the air flows around them during reentry. These data were gathered as a part of the Mercury project. The test showed that a blunt body would remain cooler than sharp, pointy vehicles.

The Space Shuttle reentry protection uses a special thermal insulation tile. Figure 6.49 shows a Space Shuttle tile in a test at NASA Langley. The tile was tested under extreme temperatures and forces to simulate reentry.

Figure 6.50 shows the test of a recover parachute in a wind tunnel test. The parafoil is the Pioneer Aerospace Parafoil, also called the Advanced Recovery System II. The parafoil was tested in the NASA Ames' large wind tunnel facility.

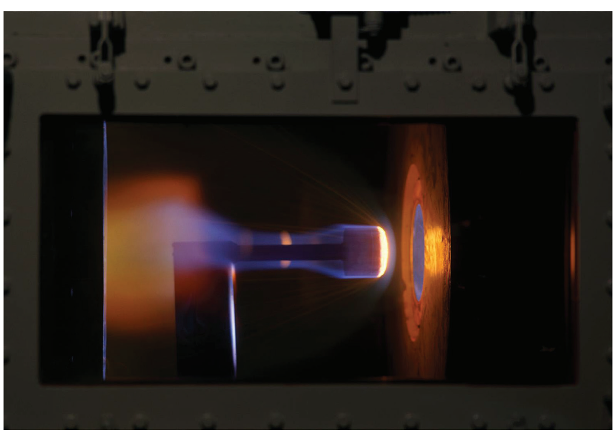

FIGURE 6.49
Space Shuttle Orbiter tile undergoing reentry heating test. (Courtesy of NASA.)

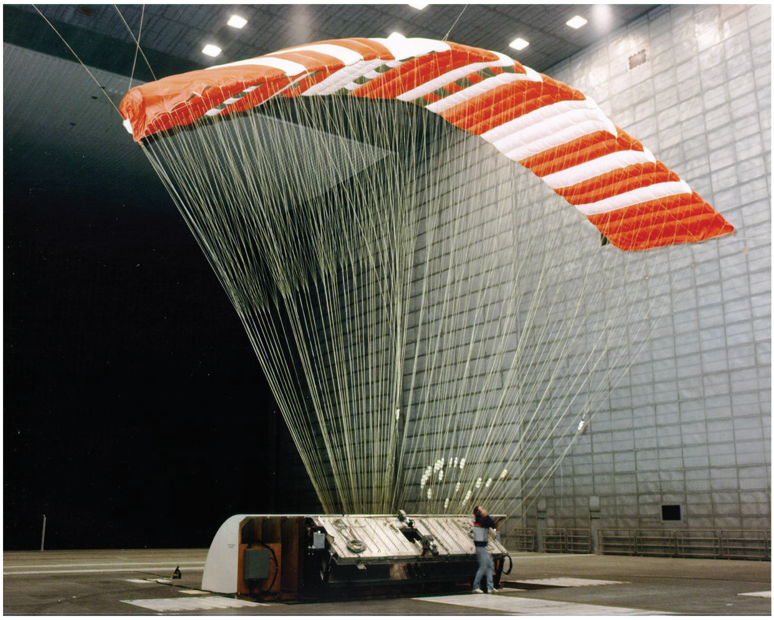

FIGURE 6.50
A large parafoil undergoing wind tunnel testing at the world's largest wind tunnel at NASA Ames Research Center, Mountain View, California. (Courtesy of NASA.)

6.7 Destructive Tests

We have already discussed the destructive and nondestructive testing of pressure vessels, but there are other aspects to testing a system to failure. Sometimes, the best way to determine all the failure modes of a system of components is to run the system under environmental conditions until it fails. This also is sometimes referred to as *life-cycle testing*. This type of testing is crucial for systems that have moving parts as they tend to wear out over time. If the system is a "use-once-and-throw-away" type of system like the RS-68 engines on the Delta IV, then life-cycle testing to destruction is not that critical. However, for a system like the NASA Solar Technology Application Readiness engine on the DS1 that flew for over 17 months, this type of testing is very important. For longer missions to the outer planets where the ion drive would have to fire for years, there is a potential for grid degradation. There are moving parts in the power-conversion units if they use generators of any sort. All of these components must be tested to failure. This process is part of the *failure mode effects analysis* that was mentioned in the introduction of this chapter.

Sometimes, a totally destructive test is the only way to measure the threat to safety and environmental impacts. What if a nuclear rocket is flown, and it fails in midascent? If the reactor is critical at the time and it falls to Earth, how hazardous would it be? Could it explode and scatter radioactive materials over the area and, in essence, be an unintentional dirty bomb?

Most modern reactor designs would prevent such an event, but the general populace is always afraid of nuclear systems they don't understand. The testing and dissemination of the test results help alleviate such fears. Figure 6.51 shows a destructive test of the Kiwi nuclear reactor at the Nuclear Rocket Development Station in Jackass Flats, Nevada. This

FIGURE 6.51
Destructive test of the Kiwi nuclear rocket engine reactor. (Courtesy of NASA.)

test was implemented as part of the NASA Nuclear Engine for Rocket Vehicle Application program for testing nuclear thermal rockets. The reactor was put into failure conditions in order to test rapid shutdown situations. The excess heat in the reactor forced it to burst apart in quite an exciting manner.

6.8 Modeling and Simulation

As part of the SEP and the goal to achieve system readiness, all and more of the tests mentioned in this chapter are performed. Data are collected and analyzed and compared to the original design requirements and performance criteria for the rocket vehicle. The new test data enable the rocket scientists and engineers to refine their mathematical models of the vehicle. These mathematical models can be run through simulations where external stimulus is simulated and applied to the vehicle model to calculate the possible outcome. More testing is done to verify the models, and then design changes are done, as they are deemed appropriate.

The advent of modern high-performance computers enables extremely detailed and complex analysis of the rocket vehicle to be conducted, whereas, in the Apollo era, the only

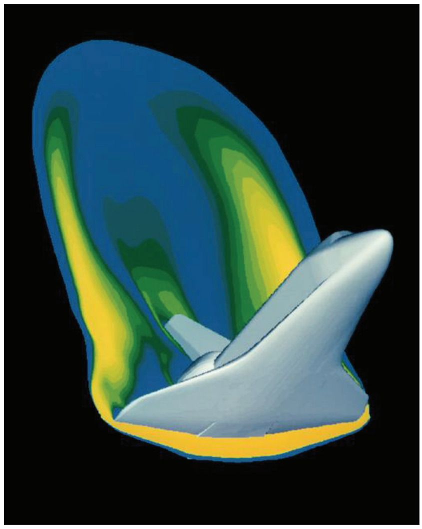

FIGURE 6.52
Computational fluid dynamics modeling and simulation of the Space Shuttle Orbiter reentry. (Courtesy of NASA.)

way to find an answer to certain questions was to build a test object and fly it. Figure 6.52 shows a computational fluid dynamics simulation of the Space Shuttle Orbiter on reentry. The calculations are very useful in design analysis and are much more cost effective than constructing scaled models and performing multiple wind tunnel tests. The simulations allow for the optimization of the test configuration, limiting the number of expensive tests that must be performed to achieve flight readiness.

6.9 Roll-Out Test

Figure 6.53 is a picture of a Saturn V test vehicle designated, the Apollo Saturn 500F, being rolled out to Launch Complex 39A from the vehicle assembly building (VAB) at the Kennedy Space Center. This rocket was not launched for the Moon, but instead was used to test the capability and processes of getting the large launch vehicle from the VAB to the launchpad. It was also used to verify that the launch facilities were designed and operating properly and to train launch crews. The test also enabled the development of checkout procedures. Similar tests are done with most other large rocket systems.

6.10 Flight Tests

Once the rocket design and development effort has gone from the whiteboard to the computer model to component and system testing in simulated flight environments, it is time to test it in the real environment. In other words, it is time to *flight-test* the rocket. In the

FIGURE 6.53
Saturn V test vehicle during roll-out test. (Courtesy of NASA.)

amateur rocketry community, this is quite simple. The amateur builds the rocket and then sets it up on the stand and launches it. Sometimes, they work, and, sometimes, they don't. Figure 6.54 shows just such a flight test of an amateur rocket actually launched by three rocket scientists. The figure shows NASA MSFC test engineer Vince Huegele, former NASA engineer and author Homer Hickam, and the manager of the Ares Projects Office, Steve Cook, launching a 1/100th scale of the Ares I rocket.

Even with amateur rockets, the flight test should follow some minimal protocols to ensure the safety of the test participants and viewers. There is a general safety code for amateur rocketry that has been initiated by that community. With larger rockets used for commercial and government use, there is actually a safety standard put in place by the Federal Aviation Administration, as well as a guidebook by the American Institute for Aeronautics and Astronautics. These protocols, at a minimum, should be strictly adhered to for safety. Most launch vehicle test ranges have far more stringent protocols than the minimum standards for the simple fact that rockets can be dangerous. This is part of the reason that most test facilities are in the middle of nowhere and why launch sites are many kilometers from populated areas. And, even when the safety protocols are adhered to, sometimes, accidents happen. In 2007, Scaled Composites was testing components of the hybrid engine for SpaceShipTwo when disaster struck, killing three and injuring others. This was in a ground test and not a flight test, but it does show the seriousness of safety and rocketry.

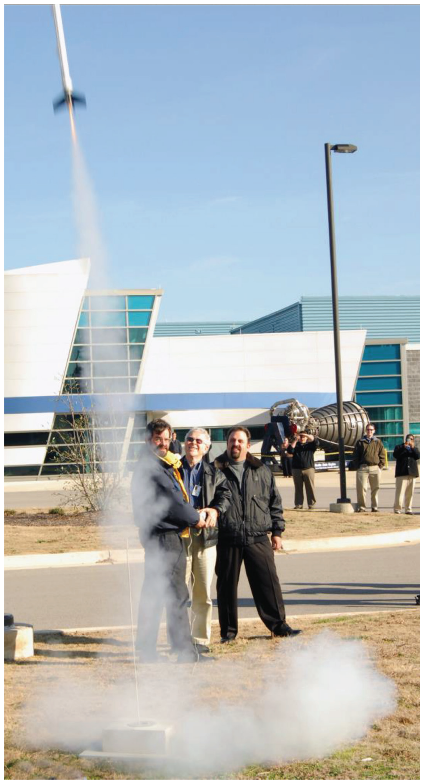

FIGURE 6.54
NASA MSFC test engineer Vince Huegele, former NASA engineer and author Homer Hickam, and the manager of the Ares Projects Office Steve Cook launching a 1/100th scale of the Ares I rocket. (Courtesy of NASA.)

In 1960, a Russian R-16 ICBM exploded on the launchpad, killing 126 people. In 1980, a Russian Vostok rocket exploded while being fueled, killing 48 people. In 1996, a Chinese Long March rocket veered off course a couple of seconds after launch and reportedly killed 56 people, according to the Chinese government. The U.S. intelligence officials estimated more than 200 were killed by the incident because the rocket crashed into a nearby village. No matter which government or private entity was responsible, all of these accidents were extremely serious. Safety cannot be emphasized enough to the rocket scientist or the engineer. So, to repeat the previous statement in this section, *rockets can be very dangerous*.

6.10.1 Logistics

Before the flight test occurs, many pieces of the complex plan must exactly fall into place, or there will be delays and difficulties in the test process. Parts for the support of the test will need to be available. Most such parts can't be bought at the local appliance or hardware store and might have long lead time to deliver. Planning ahead for such parts

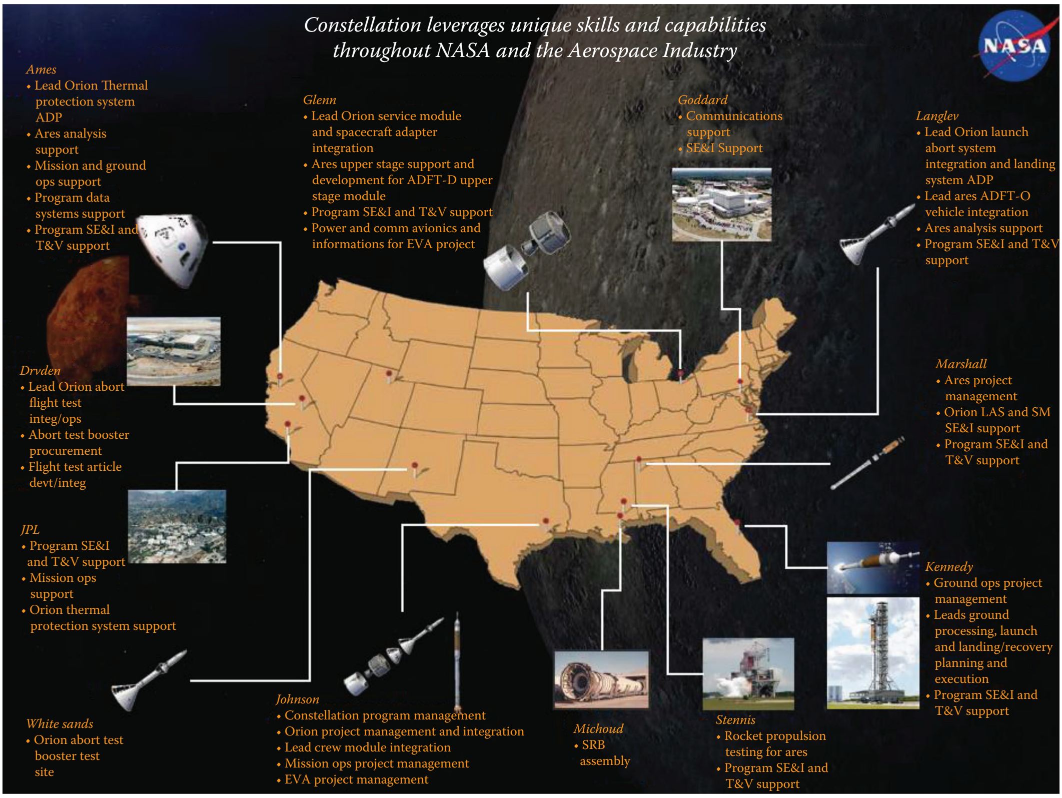

FIGURE 6.55
The Constellation Program requires serious attention to logistics! (Courtesy of NASA.)

is a must. There are also data from ground testing and simulation that must be available in order to plan the flight test appropriately. Making certain that all these pieces of the flight testing puzzle are in place on schedule is a *logistics* effort (and a complex one, indeed). Figure 6.55 shows a map of the United States with the major components of the NASA Constellation Program overlaid on it. There are major components constructed and tested all across the country thousands of kilometers distant from each other. In order to have all these components meet engineering and schedule requirements and to be at the launchpad in functioning order all at the same time is a logistics nightmare. Attention to the when, where, and how of all these parallel efforts taking place is critical to success in the flight-testing phase.

6.10.2 Flight Testing Is Complicated

Figure 6.56 shows the Apollo test program schedule, starting from the robotic precursor missions up to the second manned landing on the Moon. The test schedule was extremely detailed and expensive as it spanned over a decade and several rocket

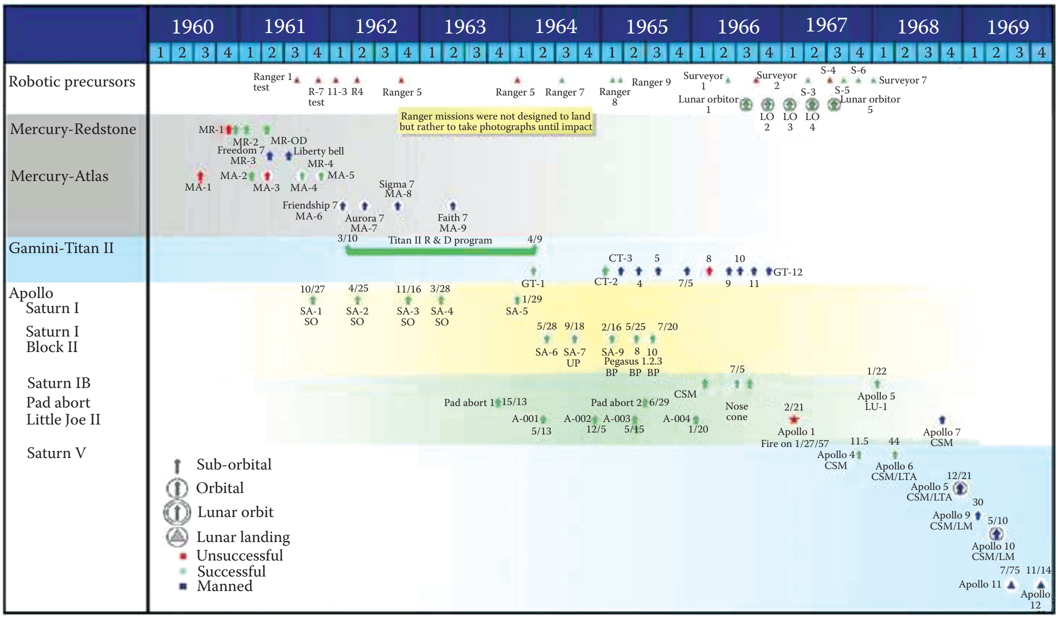

FIGURE 6.56
The Apollo test program schedule was tedious and included many flight tests over a decade. (Courtesy of NASA.)

designs. Figure 6.57 shows the Apollo flight test schedule as compared to the initial proposed flight test schedule for the NASA Constellation Program. Note that many of the Constellation Program flight tests will do the same type of testing that took several tests during the Apollo era. This is mainly due to modern knowledge of SEPs and how to implement those processes in large programs that were not available in the Apollo days. There is also a learning curve that NASA gained by the Apollo program and the Space Shuttle program and the development of the Delta IV and Atlas V vehicles that will not have to be relearned.

Unfortunately, as with any large development effort, there are some lessons that have been forgotten over the decades and will have to be relearned or were never uncovered and will be learned for the first time. The hope is that all of those lessons are foreseen, planned for, and not ones that will happen unexpectedly like the Apollo 1 fire, the Apollo 13 accident, and the Challenger and Columbia disasters. Figure 6.58 shows the flight test program summary for the Constellation Program.

Figures 6.57 and 6.58 were the early attempts at generating a flight test plan for the Constellation Program. The vernacular and acronyms have changed somewhat since then, and NASA has gained a better understanding of the type of flight testing that needs to be done at a much more detailed level. The refined plan for the Ares I vehicle is shown in Figure 6.59. Figures 6.60 through 6.62 are details of the ascent abort flight tests, the Ares I-X flight test, and the Ares I-Y test. From the test plans shown in this section, it is quite clear that the process of flight testing a new launch vehicle is a Herculean undertaking. This explains why such programs are always overbudget and behind schedule and why the end results are, in the end, spectacular.

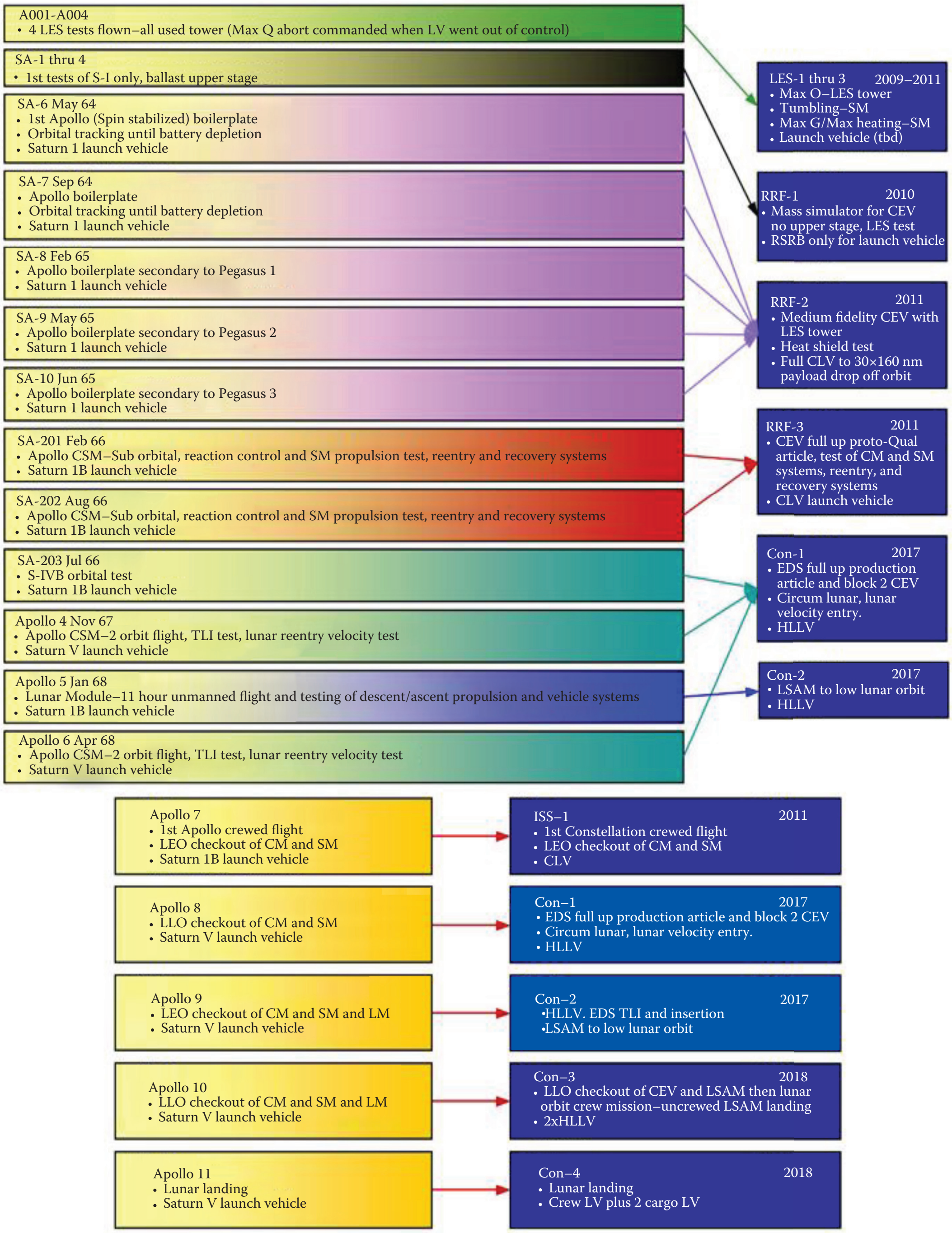

FIGURE 6.57
The Apollo test program compared to the Constellation test program. (Courtesy of NASA.)

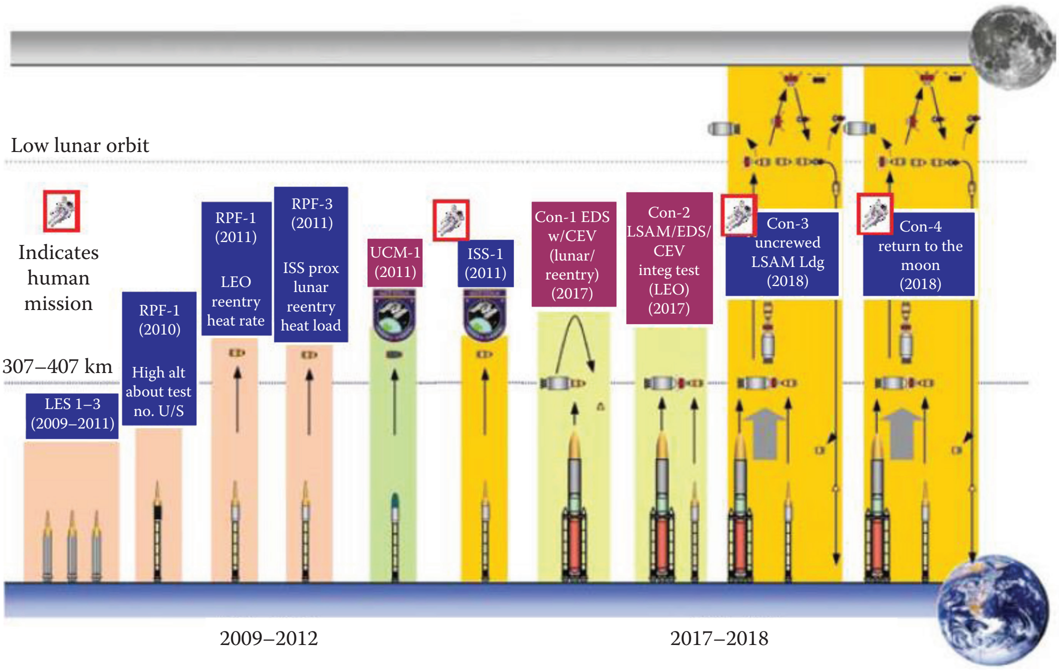

FIGURE 6.58
The Apollo test program compared to the Constellation test program. (Courtesy of NASA.)

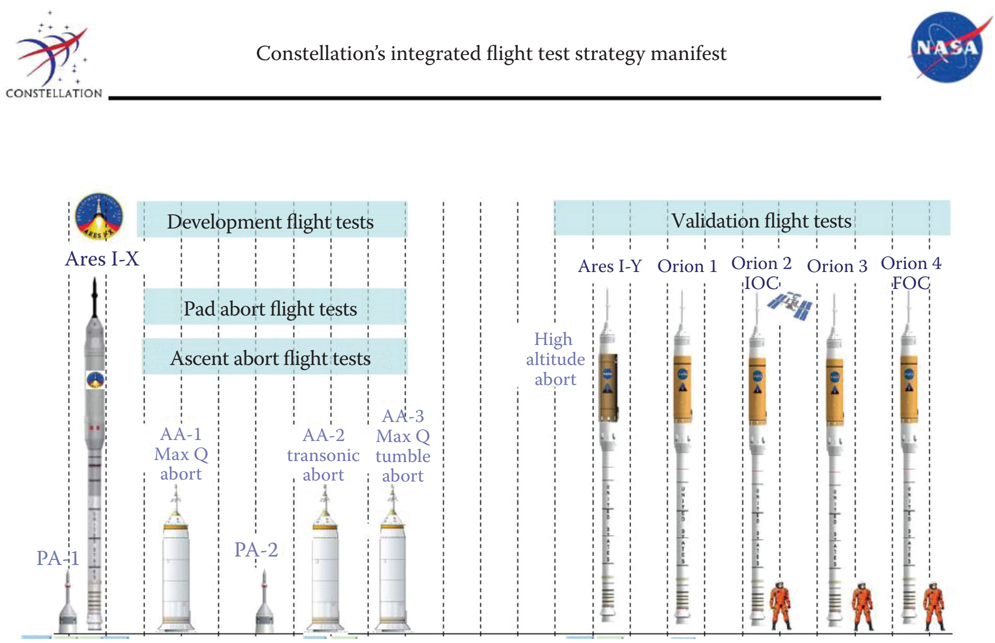

FIGURE 6.59
Constellation Program Ares I and Orion vehicle test program. (Courtesy of NASA.)

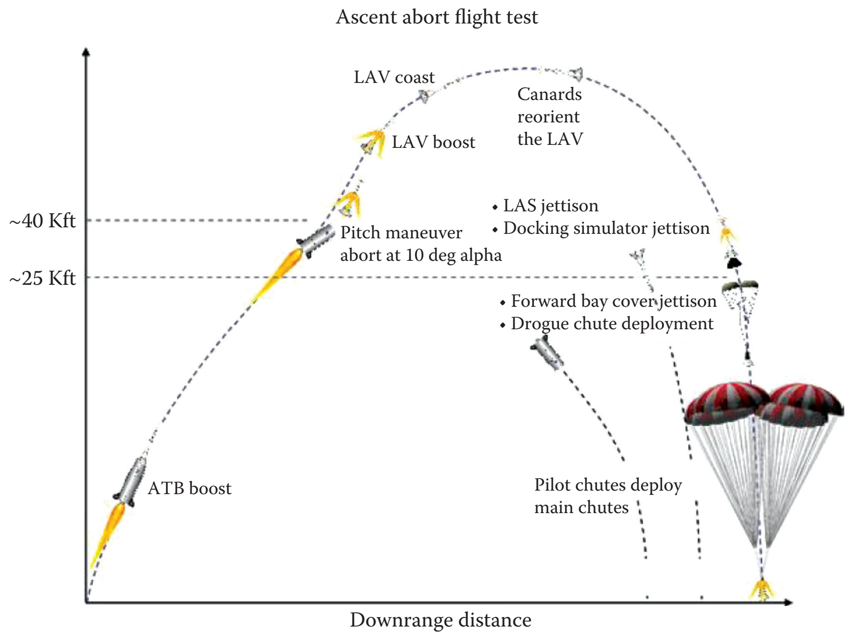

FIGURE 6.60
Ascent abort flight test profile for the Ares I. (Courtesy of NASA.)

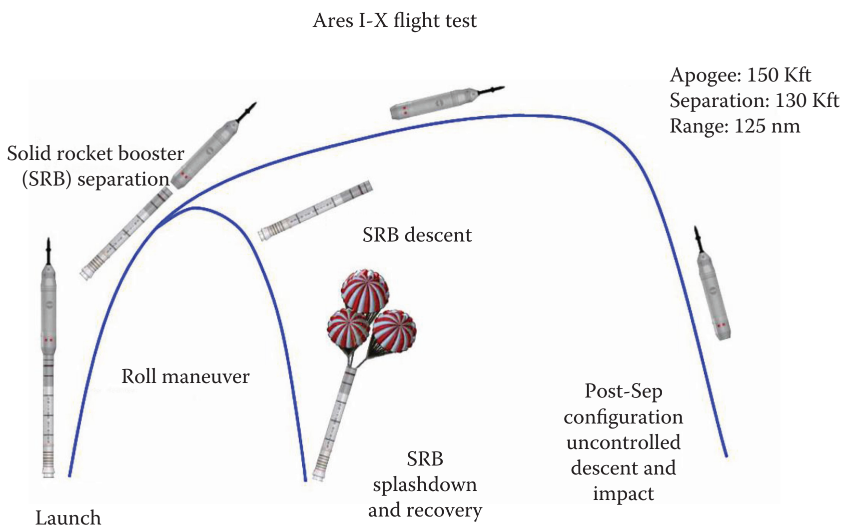

FIGURE 6.61
The Ares I-X flight test will be the first of many for the crew launch vehicle. (Courtesy of NASA.)

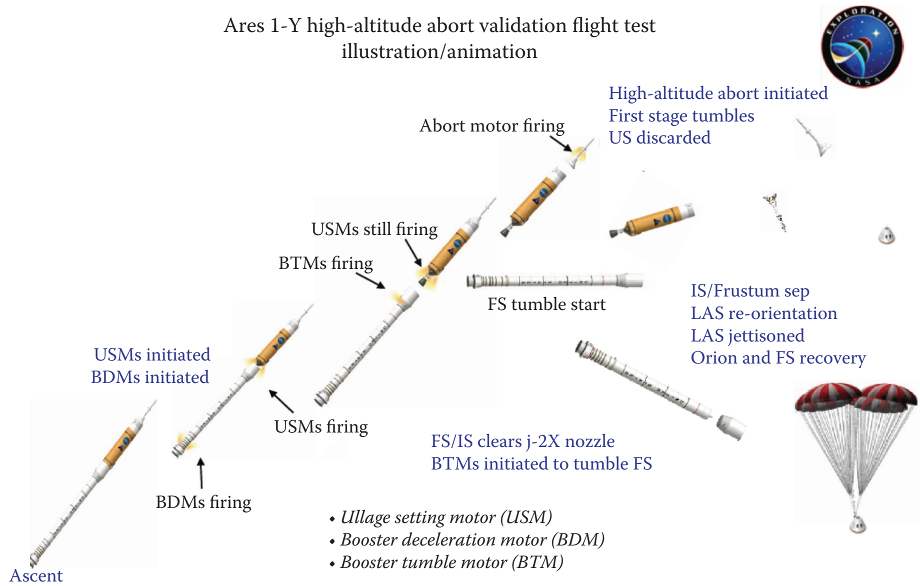

FIGURE 6.62
The Ares I-Y flight test will test high-altitude abort and Orion vehicle recovery. (Courtesy of NASA.)

6.11 Chapter Summary

In this chapter, we have been exposed to how rockets are tested and developed into a flight vehicle status. The process of development and testing starts with a good systems engineering process and an understanding of how to manage, develop, test, integrate, and fly within the confines of mammoth programs. After all, the development of a launch vehicle is a mammoth or Herculean effort. Good SEPs will make the development of the rocket system evolve more smoothly and efficiently. We also learned about TRLs, IRLs, and SRLs and why they are important to rocket development.

In Sections 6.1 through 6.9, we discussed specific types of testing and how those tests have been implemented in past and present rocket programs. These tests include the following:

- Thrust measurement tests
- Pressure vessel tests
- Shake 'n' bake tests
- Drop and landing tests
- Environment tests
- Destructive tests
- Modeling and simulation
- Roll-out tests

In Section 6.10, we discussed flight tests and their complexity. We learned that large-scale flight tests programs are extremely complex and require tremendous attention to logistics and planning in order for them to be successful. We also looked at the Apollo and Constellation Program flight test plans and compared them. Diving into the Constellation Program flight test schedule gives some insight into the events that are expected to occur at each new step of a flight test program.

This entire chapter is an introduction to modern rocket science and engineering vernacular, nomenclature, history, systems engineering processes, flight test planning, and safety protocols from a very hands-on perspective. Although many tests were discussed, a complete list of tests required for a flight test program is way beyond the scope of the chapter. However, we have shown that a lot of testing is indeed important and required. Also, within this chapter, the rocket scientist and the engineer start to see how broad a scope of knowledge is required to move forward with a successful rocket program and how to actually apply some of that knowledge to real hardware, software, and tests. Hopefully, this chapter answers the question (at least from an introductory level), "How do we test rockets"?

Exercises

6.1 Define FMEA.

6.2 What is systems engineering?

6.3 What is a system?

6.4 What are the many pieces and components of a system called?

6.5 What is a KDP?

6.6 What are the phases of a project life cycle according to the NASA systems engineering process?

6.7 Define SEP.

6.8 What is the PDR?

6.9 What is the CDR?

6.10 Define the V model.

6.11 What is the SE engine?

6.12 Define TRL.

6.13 If a rocket engine has been tested in the laboratory only and not in an environment chamber, at what TRL is it likely to be?

6.14 A concept that has only been developed as far as the laboratory whiteboard would be at what TRL level?

6.15 What is the TRL of the Space Shuttle?

6.16 Define IRL.

6.17 What IRL is the Delta IV rocket?

6.18 Define SRL.

6.19 What is the "pathway to success"?

6.20 A rocket system has an average TRL of 6 and an IRL of 2. Calculate the SRL.

6.21 Is the rocket system in Exercise 6.20 on the pathway to success? Should components or systems be focused on to fix this?

6.22 What is the SRL of the Atlas V rocket?

6.23 What is a thrustometer?

6.24 What is a test stand?

6.25 A rocket engine on a spring scale deflects the spring by 1 cm. If the spring constant is 0.1 N/m and the m-dot of the rocket is 1 mg/sec, what is the equivalent velocity?

6.26 Develop a computer model for the deflection bar thrustometer and study how variations in the bar geometry and the modulus of elasticity change the level of thrust that can be measured.

6.27 The pressure gauge on a hydraulic load cell thrustometer measures 10 MPa. The diameter of the thrust point is 0.25 m. Calculate the thrust measured by the device.

6.28 Develop a computer model for the strain gauge load cell thrustometer and study how variations of the load cell geometry (assume a C-type) change the thrust levels that can be measured.

6.29 What are pressure vessels?

6.30 Why is logistics important in flight test programs?

7

How Do We Design Rockets?

Up to this point, we have discussed the history of rockets, their various styles and configurations, how they are used, the basics of rocket engine design, the basics of rocket system design, and how they are tested. But we have yet to start with a blank sheet of paper and design a rocket to perform a task. So, how is that done? How do we design rockets for a specific task?

As we stated at the beginning of Chapter 2, the cost of a launch vehicle can reach as high as several hundred million dollars and require a small army of people to build, prepare, and fly. And, on top of that, it costs even more to design, build, and test a new design to bring it out of experimental status and up to operational flight status. There must be a good reason to expend such resources on such things; otherwise, people simply would not go to the trouble. So, typically, we start with a need such as putting a particular mass payload in a low Earth orbit at a particular altitude and inclination. Another example might be to send a particular payload mass on an escape trajectory toward the Moon or Mars. Or, perhaps, the need is to place a certain mass to a certain altitude or distance. Whatever the need, we are beginning to create the tools to develop such rockets based on the “requirements” for it.

In this chapter, we will walk through the process for creating a rocket design from mission requirements. The best way to do this is through examples, so we will choose a particular “design reference mission” (DRM) and design a rocket to achieve that task. We will make use of tools such as Mathcad or MATLAB®, whichever you prefer (I use Mathcad), and we will also use OpenRocket for the first time in this design effort.

7.1 Designing a Rocket

The first step in designing a rocket for a particular need is to determine what the actual need is. This part of the process is called the *requirements definition process*. It is also sometimes called the *stakeholder requirements definition process*. According to the International Council on Systems Engineering’s (INCOSE’s) *INCOSE Systems Engineering Handbook v. 3.2* (January 2010), the requirements definition process’ main purpose is to

> …define the requirements for a system that can provide the services needed by users and other stakeholders in a defined environment. (pp. 54–55)

Oftentimes, this part of the process comes from an idea for an experiment, or a deficiency in a system, or even a concept for offering a completely new capability. Sometimes, part of the process is to create what is known as a design reference mission, which is a made-up mission that might be similar to a future need. The DRM is often used to determine if rocket designers can actually develop a rocket concept to accomplish tasks that might have similar requirements as the DRM. Many times, the DRM never turns out to be the mission actually flown, but it is where the design effort started.

An example mission, we will call DRM #1, might be that a college electrical engineering and computer science research team desires to place a cell phone at a 60-km altitude to determine if it the data rates change and if the signal integrity is maintained. A side effort might be to determine if the phone could actually survive the environment and forces from whatever means needed to get it there, but the electrical engineers and computer scientists are typically not concerned with such matters as that is the job of the aerospace engineers and the rocket scientists.

In this case, the payload is the cell-phone experiment. The payload designers would soon realize that aircraft and high-altitude balloons would most likely not be able to support such a mission, and, therefore, a rocket would be the most likely choice. At this point, they would either turn to a rocket development team or become one themselves. Typically, developing a payload is a job within itself, and the team would turn to rocket developers. Either way, it is at this point that the team must sit down and start writing down the mission requirements in detail so that they can be relayed to the rocket development team.

From the rocket designer's standpoint, it matters not why the payload designers desire to put their payload in the position and environment; it only matters that they desire to get it there. There will be a need to interact as the design process continues such that each knows how to interface to the other component of the mission. The rocket designers will have to discuss how harsh the flight will be to the payload, and the payload designers will have to discuss how fragile or robust the payload is. If the rocket flight destroys the payload, then the reason for flying becomes moot. This is a "spiral" interaction between the integrated mission team.

7.1.1 Derived Requirements

Our college payload design team delivers a first set of "mission requirements" to the rocket design team. These DRM #1 requirements are as such:

Requirement #1: Deliver an iPhone 6 Plus to an altitude of 60 km.

Requirement #2: Return the iPhone 6 Plus back to Earth safely and undamaged.

These are the only two "mission requirements." From these requirements, the rocket design team must develop the "derived requirements." In other words, they will take requirements 1 and 2 and derive other features of the rocket design from there.

Sometimes, it is not obvious where to start in the process, and the best approach is to just jump in and tackle whichever design parameter comes to mind. In the case of DRM #1, we begin by writing down what we know.

Given: The iPhone 6 Plus's dimensions are 15.81 × 7.78 × 0.71 cm.

The iPhone 6 Plus's mass is about 0.172 kg.

From these known parameters, we can immediately derive two more requirements:

> Derived Requirement #1: Payload shroud must be larger than 15.81 × 7.78 × 0.71 cm with enough margin to house mounting hardware for the phone.
>
> Derived Requirement #2: The rocket's payload mass is a minimum of 0.172 kg, and sufficient margin must be allowed for support structure.

There also needs to have some durability information about the payload to start with. So our rocket design team either does some literature searching, or they buy some iPhone 6 Pluses, and they test them. Either will cost money as it takes time to do a literature search, and it takes money and time to buy and test the phones. But, after some quick implementation of "Google-fu," our rocket team finds papers suggesting that a drop of the phone from about 1 m is all it can take. In other words, it is suggested that, if the payload impacts anything with an impulse comparable to impacting the ground after a 1-m fall, then the phone is damaged. See Requirement #1.

Now, we have the ability to derive a third requirement—maximum acceleration and jerk. If a 0.2-kg mass falls from 1-m height, then the velocity of the mass under one gravity is

$$v = \sqrt{2gh} = \sqrt{2 \times 9.8 \frac{\mathrm{m}}{\mathrm{s}^2} \times 1\ \mathrm{m}} = 4.43\ \mathrm{m/s}. \tag{7.1}$$

With no safety margin, the maximum impact velocity is now known. Other information about the test must be used to derive more, but assume it was known that the phone impacted the ground and made a 1-mm divot, d, in the surface. We can therefore find the impact force by equating the work and energy of the test. In other words,

$$W = Fd = \frac{1}{2} mv^2 = \frac{1}{2} \mathrm{m}(2gh) \tag{7.2}$$

solving for *F*,

$$F = \frac{mgh}{d} = \frac{0.2\ \mathrm{kg} \times 9.8 \frac{\mathrm{m}}{\mathrm{s}^2} \times 1\ \mathrm{m}}{0.001\ \mathrm{m}} = 1{,}960\ \mathrm{N}. \tag{7.3}$$

We know that $F = ma$ and can solve for the maximum acceleration to be about 9,800 $\mathrm{m/s^2}$ or about 100 g. If we also know that the impact took approximately 1 msec, then the jerk, *j*, is found as

$$j = \frac{\Delta a}{\Delta t} = \frac{(9{,}800 - 9.8) \frac{\mathrm{m}}{\mathrm{s}^2}}{0.001\ \mathrm{s}} = 9.8 \times 10^6\ \mathrm{m/s^3}. \tag{7.4}$$

Now, we have several derived requirements for our rocket. For each of these, we need to add what is known as a "safety margin" or a "safety factor," which is a simple multiplier. For this experiment, a safety factor of 2 is most likely enough. For missions with more expensive payloads or with manned flights, the safety margin is sometimes as much as 10 or more.

So, our rocket must not accelerate our payload with more than 50 g or a jerk of roughly 4.9 mega $\mathrm{m/s^3}$ or 0.5 g/s. Also, our recovery system must land it softer than these requirements. This tells us that the parachute for the recovery system must be sized to land the payload mass at a velocity slower than 2.2 m/s, as found from Equation 7.1. From these derived requirements, we can begin to design our rocket. As the design, build, and test process continued in any rocket development efforts, new requirements are uncovered each day, but a complete rocket program is beyond the scope of this text. That being said, it is not necessarily beyond the scope of the readers.

7.1.2 OpenRocket

At this point, we will begin to design our rocket to perform the DRM #1 mission based on the requirements, both given and derived. We will implement a tool known as OpenRocket, which is a simulation open-source software program developed for designing and building rockets. Step one is to go to

http://openrocket.sourceforge.net/

and download the program. Once it has been downloaded and installed, the user should spend some time studying the examples that come with it and modifying them just to see what happens. Once you become familiar with the software, it will be time to start to work.

What the rocket design team will first have to realize is that it is unlikely that any commercially available motors will be able to push the DRM #1 payload to altitude with one stage. Our rocket will therefore be a multistage system. While the stage-optimization problem is a little outside what we want to do here, we can use years of rocket testing and historical knowledge to suggest to us that a two- or three-stage rocket might do the trick. But it might not either. We will start with a three-stage system and see if that works by changing various parameters within the program simulations. If two stages don't accomplish the requirements, we will simply add a stage. OpenRocket is very modifiable and easy to work with in this regard.

7.1.2.1 OpenRocket Step #1: Choose a Body Tube for the First Stage

Once we click the body tube icon, a window will open up with the parameters defining the type of materials to be used, as well as physical dimensions of the tube. In order to fly as high as required, we will need very large engines on the order of N or O sizes. The body tube will have to be sized in order to fit these engines, but don't worry about that yet because we can always adjust the sizes of parameters easily.

7.1.2.2 OpenRocket Step #2: Choose an Inner Tube Engine Mount and Engine for the First Stage

Choose the inner tube icon and place the tube inside the main body tube. We will need to click on the inner tube to make it a motor mount. We can also click the "Cluster" tab and make multiple tubes in case we decide to use multiple engines. Then, once we are back in the main screen, we can click the tab that says "Motors & Configuration" to choose from the list of motors.

7.1.2.3 OpenRocket Step #3: Fix the Center of Gravity (cg) and the Center of Pressure (cp)

This step will be iterative throughout your design effort. Recall here Rocket Design Rule #1, as discussed in Chapter 3.

Rocket Design Rule #1: For a rocket to have stable flight, the cg MUST be higher up the rocket than the cp

A good approach is to place a nose cone on the end of the tube and fins on the tail and see how the cg and cp move. Also, the stage can be simulated in the "Flight simulation" tab as a single-stage rocket. Do a new simulation and plot the results. The software also shows the apogee, the maximum velocity, and the maximum acceleration, along with other parameters on the main page in the design window. Keeping an eye on all these parameters as we build up our rocket is quite useful.

Also note that, at this point, we can add bulkheads and centering rings and all sorts of other components that will be required for a final rocket blueprint, but this tool is best used for the first simulation of performance. The more detailed design comes later, and likely uses a professional computer-aided design tool like SolidWorks™ or similar.

7.1.2.4 OpenRocket Step #4: Add New Stage

This step is just like the previous steps reiterated. Pick a body tube size, an inner tube, engines, fins, etc. If we decide to use a smaller-diameter stage body tube, we will also need to use a transition component.

7.1.2.5 OpenRocket Step #5: Add New Stage

Again, this step is just like the previous steps reiterated.

7.1.2.6 OpenRocket Step #6: Finish the Top and Place the Payload

Finally, choose a nose cone, a payload mass, and a parachute-recovery system. Once we do this, we keep an eye on the apogee in the bottom-left corner of the screen and see if we've reached our mission requirement goal. I doubt we did.

7.1.2.7 OpenRocket Step #7: Simulate, Modify, Simulate, Modify…

We need to note here that there are a significant amount of details left for the readers to figure out in each of the steps mentioned above. Some of these include choosing materials for each component, fin shapes, and sizes, when each engine in each stage fires relative to the previous-stage burnout, paint thickness, the nose-cone style, locations of components, and so on. Rockets have many parts, and optimizing each of them when coupled to all the others is difficult, complicated, and time consuming. The best way to do this is to run a new simulation for each modification, study the output, make a modification, run another simulation, study the output, and continue to iterate until we reach the design requirements. It is during this part of the process that we can truly see what is meant by our second rocket design rule:

Rocket Design Rule #2: If we touch the rocket ANYWHERE, we have touched the rocket EVERYWHERE.

This design rule means that, anytime we modify our rocket design, even in the smallest, and sometimes seemingly insignificant way, the rocket performance changes completely. It also means that changing one component on the rocket will change how that rocket interacts with all the other parts of the rocket and can actually cause unforeseen and unexpected deleterious effects. Once a change is made, new simulations and tests will be required.

At this point in the design effort, we should take a significant amount of time to vary the components in the OpenRocket model, run a simulation, and watch how these things change the rocket's performance. A common example that can be seen is to watch for a tumbling warning. Sometimes, moving or sizing a fin forward or backwards or up or down by a centimeter can make all the difference. Moving bulkheads, changing body-tube materials and sizes, and even changing the ignition delay after the burnout of the previous stage by a single second can cause vast improvements or disruptions in the rocket's flight performance.

Play around with all the toggles and buttons and tabs for a while. Then and only then will the importance of Rocket Design Rule #2 start to become clear. Until we actually build the rocket and test it, we will still only have a notion as to how important this rule is. But, we are learning this lesson right from the very beginning of our design effort.

7.1.2.8 OpenRocket Step #8: Realization

After multiple iterations, we finally realize that there isn't just enough "umpf" in a three-stage rocket using the engines in the OpenRocket database to push our smartphone payload to above 60 km. Once we come to this realization, we go back to Step #7 and modify our design. This time, we do so by adding more stages. We will continue the process until we find the right combination of engines, stages, fins, body tubes, etc., that will achieve our objectives and meet the mission requirements.

Figure 7.1 shows a final DRM #1 Rocket Design that meets the mission requirements. It is quite possible that many other designs exist. This is only a particular or point solution

Rocket design

Rocket
Stages: 7
Mass (with motors): 117338 g
Stability: 0.477 cal
cg: 687 cm
cp: 697 cm

Altitude	63731 m
Flight time	255 s
Time to apogee	170 s
Optimum delay	79.3 s
Velocity off pad	12.1 m/s
Max velocity	1023 m/s
Velocity at deployment	N/A
Landing velocity	N/A

-P; 2×N4800T-P; 2×21062-O3400-IM-P

Motor	Avg Thrust	Burn time	Max thrust	Total impulse	Thrust to wt	Propellant wt	Size
L1222	1218 N	3.03 s	2725 N	3694 Ns	17.82:1	3900 g	75/1340 mm
L200	188 N	13.9 s	312 N	2620 Ns	1.58:1	1594 g	111/724 mm
M1000 O	972 N	9.25 s	1482 N	9029 Ns	4.28:1	4174 g	111/1147 mm
M1000 O	972 N	9.25 s	1482 N	9029 Ns	2.90:1	4174 g	111/1147 mm
M1010 O	954 N	9.65 s	1473 N	9222 Ns	2.13:1	4276 g	111/1147 mm
N4800T (×2)	4119 N	4.66 s	6008 N	19274 Ns	5.30:1	9766 g	98/1201 mm
21062-03400-IM (×2)	3421 N	6.14 s	4699 N	21041 Ns	2.97:1	11272 g	98/1239 mm
Total:				114224 Ns	5.95:1	60195 g	

FIGURE 7.1
OpenRocket design parameters for DRM #1 rocket.

for the design. Note that the solution required seven stages, with motor sizes varying from type O to L. Figures 7.2 and 7.3 show the design components for the DRM #1 Rocket Design. Using Figures 7.1 through 7.3, the rocket can be reproduced and simulated.

Figures 7.4 through 7.9 show the thrust curves for each of the motors used in the design. The types of curves used varied from progressive to neutral to regressive, and some of them were quite complex. It is very interesting to change the motors, and therefore the thrust curves, and see how the rocket performance changes. As an example of this, the M10100 and M10000 thrust curves are very, very similar, but swapping one for the other on their respective stages has a significant impact on the rocket's performance.

Parts detail
Sustainer

Nose cone	Fiberglass (1.85 g/cm³)	Conical	Len: 48 cm	Mass: 269 g
Unspecified		Diaout: 8.5 cm		Mass: 300 g
Transition	Fiberglass (1.85 g/cm³)	Fore dia: 10 cm Aft dia: 8 cm	Len: 12.5 cm	Mass: 128 g
Body tube	Carbon fiber (1.78 g/cm³)	Diain: 7.6 cm Diaout: 8 cm	Len: 138 cm	Mass: 1204 g
Trapezoidal fin set (4)	Carbon fiber (1.78 g/cm³)	Thick: 0.3 cm		Mass: 53.4 g
Parachute	Ripstop nylon (67 g/m²)	Diaout: 30 cm	Len: 2.5 cm	Mass: 7.98 g
Shroud lines	Elastic cord (round 2 mm, 1/16 in) (1.8 g/m)	Lines: 6	Len: 30 cm	
Bulkhead	Carbon fiber (1.78 g/cm³)	Diaout: 7.6 cm	Len: 0.2 cm	Mass: 16.1 g
Booster stage				
Transition	Fiberglass (1.85 g/cm³)	Fore dia: 8 cm Aft dia: 12 cm	Len: 11.5 cm	Mass: 133 g
Body tube	Carbon fiber (1.78 g/cm³)	Diain: 11.6 cm Diaout: 12 cm	Len: 81 cm	Mass: 1069 g
Trapezoidal fin set (4)	Fiberglass (1.85 g/cm³)	Thick: 0.3 cm		Mass: 55.5 g
Booster stage				
Body tube	Carbon fiber (1.78 g/cm³)	Diain: 11.6 cm Diaout: 12 cm	Len: 135 cm	Mass: 1782 g
Trapezoidal fin set (4)	Fiberglass (1.85 g/cm³)	Thick: 0.3 cm		Mass: 130 g
Inner tube	Fiberglass (1.85 g/cm³)	Diain: 7 cm Diaout: 7.1 cm	Len: 105 cm	Mass: 215 g
Booster stage				
Body tube	Carbon fiber (1.78 g/cm³)	Diain: 11.6 cm Diaout: 12 cm	Len: 135 cm	Mass: 1782 g
Trapezoidal fin set (4)	Fiberglass (1.85 g/cm³)	Thick: 0.3 cm		Mass: 130 g
Inner tube	Fiberglass (1.85 g/cm³)	Diain: 7 cm Diaout: 7.1 cm	Len: 105 cm	Mass: 215 g

FIGURE 7.2
OpenRocket components list (part 1) for DRM #1 rocket.

Booster stage

Body tube	Fiberglass (1.85 g/cm³)	Dia in: 11.6 cm Dia out: 12 cm	Len: 145 cm	Mass: 1989 g
Trapezoidal fin set (4)	Carbon fiber (1.78 g/cm³)	Thick: 0.3 cm		Mass: 167 g
Inner tube	Fiberglass (1.85 g/cm³)	Dia in: 7 cm Dia out: 7.1 cm	Len: 105 cm	Mass: 215 g

Booster stage

Transition	Cardboard (0.68 g/cm³)	Fore dia: 12 cm Aft dia: 21 cm	Len: 7.5 cm	Mass: 60.8 g
Body tube	Carbon fiber (1.78 g/cm³)	Dia in: 20.3 cm Dia out: 21 cm	Len: 134 cm	Mass: 3117 g
Trapezoidal fin set (4)	Fiberglass (1.85 g/cm³)	Thick: 0.3 cm		Mass: 159 g
Inner tube	Fiberglass (1.85 g/cm³)	Dia in: 7 cm Dia out: 7.1 cm	Len: 105 cm	Mass: 430 g

Booster stage

Body tube	Fiberglass (1.85 g/cm³)	Dia in: 20.6 cm Dia out: 21 cm	Len: 133 cm	Mass: 3216 g
Trapezoidal fin set (4)	Carbon fiber (1.78 g/cm³)	Thick: 0.3 cm		Mass: 254 g
Inner tube	Fiberglass (1.85 g/cm³)	Dia in: 7 cm Dia out: 7.1 cm	Len: 105 cm	Mass: 430 g

FIGURE 7.3
OpenRocket components list (part 2) for DRM #1 rocket.

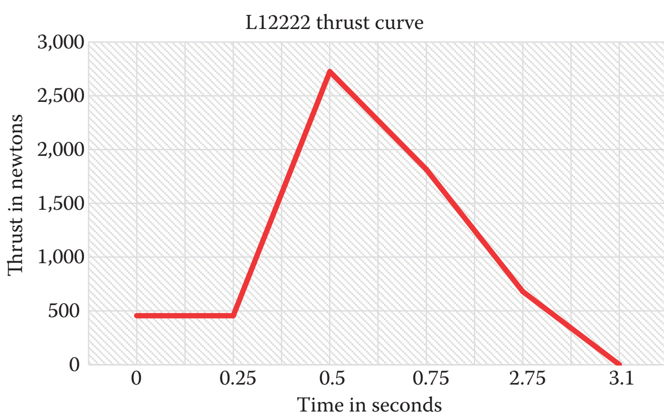

FIGURE 7.4
Stage 7 motor thrust curve for DRM #1 rocket.

It is at this point that we should emphasize that the design reaches a fairly high velocity and is greater than Mach 1. OpenRocket and other design tools that are available typically have accuracy problems in the supersonic regions. So, we must realize that this design is a starting point. More detailed analysis and testing will be required. There are other codes out there that will handle the super and hypersonic regions of flight, such as RASAero II, which would be a good tool to graduate to once we become truly proficient with OpenRocket. There are others as well, and, of course, the students are always encouraged to explore new tools and to create their own.

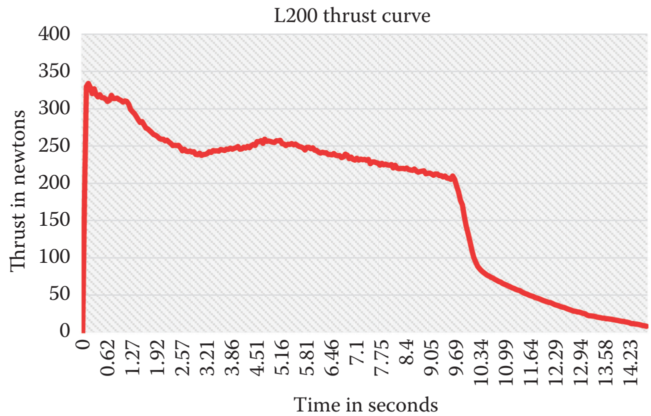

FIGURE 7.5
Stage 6 motor thrust curve for DRM #1 rocket.

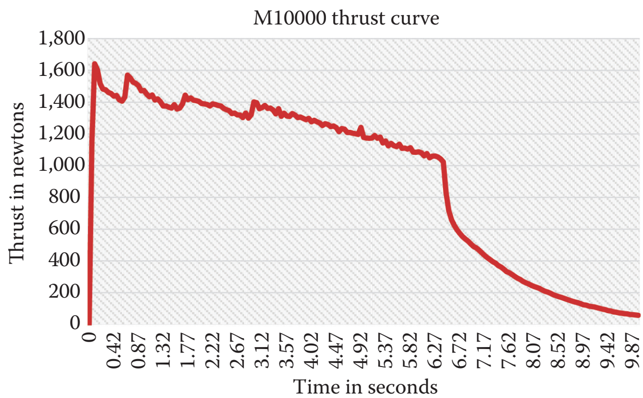

FIGURE 7.6
Stages 5 and 4 motor thrust curve for DRM #1 rocket.

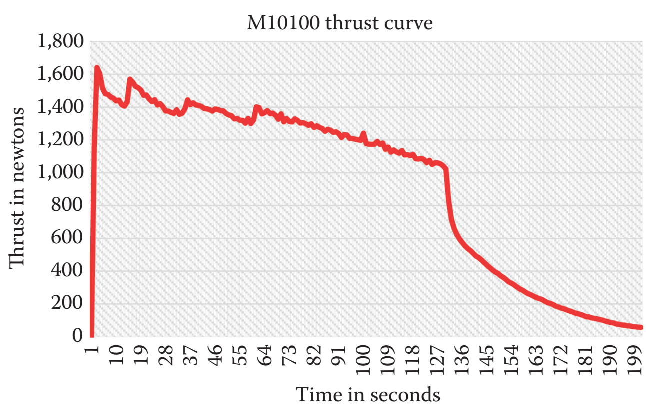

FIGURE 7.7
Stage 3 motor thrust curve for DRM #1 rocket.

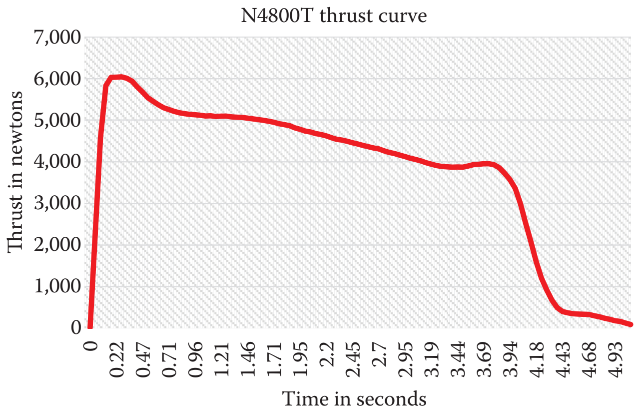

FIGURE 7.8
Stage 2 motor thrust curve for DRM #1 rocket.

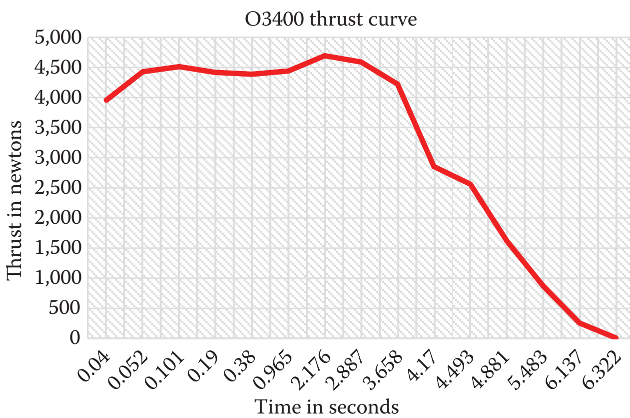

FIGURE 7.9
Stage 1 motor thrust curve for DRM #1 rocket.

7.1.3 From OpenRocket to Real Design

At this point, we will begin to design our rocket to perform the DRM #1 mission based on the OpenRocket simulation. This is where we draw the detailed blueprints and diagrams and truly figure out how one component is mated and attached to others. During this phase of designing a rocket, we determine where to put avionics computers, staging charges and/or igniters, ignition-control circuits, recovery systems, thrust structures, launch lugs, and other components required to keep the rocket upright during the buildup of the stack. This is the phase where we realize that the motors are bolted into something: a thrust structure bulkhead or similar. Only once we figure out these details will we have a good mass model of the rocket design, and only then will we truly have a robust simulation. Then, we will build and test. And as we do so, we MUST keep in mind Rocket Design Rule #'s 1 and 2 along the way. At this point, we should also realize the need for a third design rule, and that is the following:

Rocket Design Rule #3: Keep it simple. Simple is almost always better than complex and less likely to fail.

By following the three main rules we have now, we will be moving in the right direction toward a successful rocket design, build, test, and flight. The details of a complete rocket design is a book within itself. At this point, there is no better teacher than looking at other designs and building techniques through a literature search and then actually building a rocket for yourselves. The rocket design for DRM #1 is likely too expensive as some of the motors themselves will cost several thousand dollars. But, now would be a great time for a lab exercise to design a smaller rocket and then actually go through the design, build, test, and fly process. A good trial would be to take the requirements for the NASA Student Launch Initiative and design a rocket to meet those needs. Information for the international high school through college rocket competition can be found at

http://www.nasa.gov/audience/forstudents/studentlaunch/handbook/index.html.

Whether or not the NASA competition is for you, at this point, in your path to becoming a rocket scientist or engineer, you need to build some rockets. At least build one. Even if the rocket is a low-cost cardboard hobby rocket, going through the design process above and seeing it take shape and then take flight are of immeasurable importance in gaining insight into how to design and build a rocket for a purpose.

7.1.4 Fineness Ratio and Structural Design

We need to discuss a bit about a parameter known as the "fineness ratio" of a rocket body. We will denote this ratio as f_b:

$$f_b = \frac{L}{D}. \tag{7.5}$$

Here, L is the length of the rocket body, and D is the diameter of the tube. Figure 7.10 is a graph of the fineness ratio showing for a fixed L. The figure gives us insight into how the fineness changes with diameter of the rocket. The total length of our DRM #1 rocket

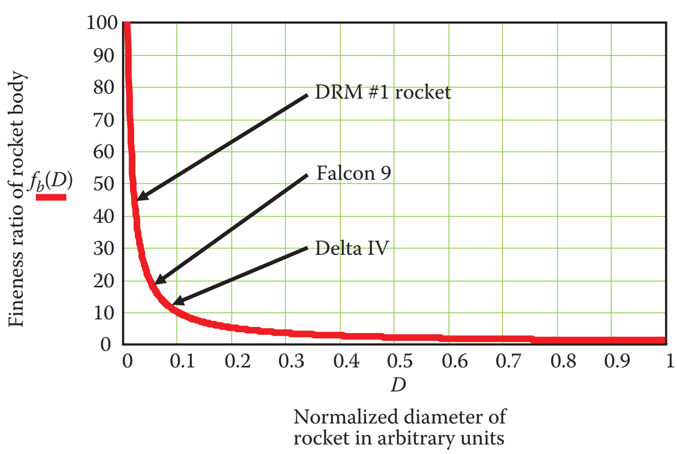

FIGURE 7.10
Fineness ratio of rocket body versus diameter.

is 9.83 m with a maximum diameter of 0.21 m. The minimum fineness of this rocket is 46.81. The minimum diameter is 0.08 m, giving a maximum fineness of 122.88. The longest portion of our rocket is the middle section, which is 5-m long and 0.12 m in diameter. The fineness of this section is 41.67. The smaller diameter is only at the top stage of the rocket, in a 2-m long section. The fineness of the top stage is only 25. The average of these gives us an average fineness ratio of 59.09. For perspective, the fineness ratio of the SpaceX Falcon 9 is about 18, and the Delta IV is about 12. High fineness ratios suggest very "bendy" rockets that could fail through buckling or become difficult to control because they "wiggle" and "flop" during flight.

Our DRM #1 rocket has a very high fineness for a rocket and is likely to fail because of this. However, a detailed buckling analysis would have to consider the metal cases for the rocket motors and other internal structures that will improve structural integrity. Assuming our rocket is a hollow carbon-fiber composite cylinder, we can determine if it will buckle from a critical pressure, P_{cr}:

$$P_{cr} = \frac{\pi^2 EI}{L^2} \tag{7.6}$$

where E is the Young's modulus for the material, and I is the moment of inertia of the rocket. If we assume the rocket is a thin-walled cylinder of mass, M, and radius, R, then $I = MR^2$ and Equation 7.6 becomes

$$P_{cr} = \frac{\pi^2 EMR^2}{L^2}. \tag{7.7}$$

It is at the dynamic pressure incident on the rocket during flight equal to the critical pressure calculated from Equation 7.7 where the rocket will buckle and fail. This type of buckling is called "Euler buckling." If simulations suggest that the dynamic pressure on the rocket during flight exceeds this critical pressure, then we will need to rethink our design.

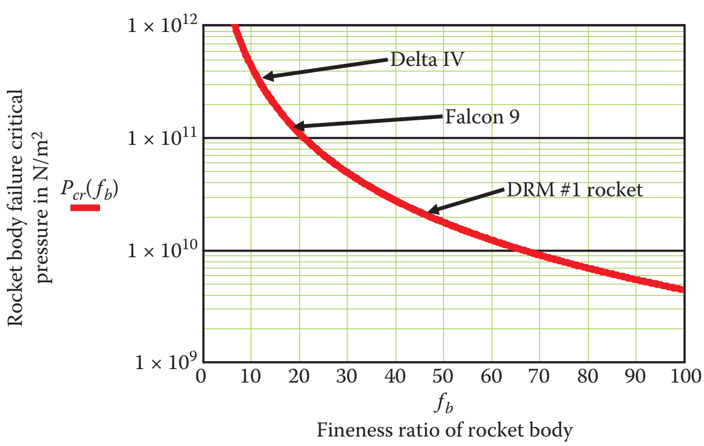

FIGURE 7.11
Rocket body failure critical pressure versus fineness.

Again, don't forget that the rocket has structure on the inside to help with this. If we look at Equation 7.8 closely, we realize it can be rewritten as a function of fineness:

$$P_{cr} = \frac{\pi^2 EM}{4 f_b^2}. \tag{7.8}$$

The modulus, E, for carbon fiber is about 150 × 10^9 N/m^2. From Figure 7.11, we can see that the critical pressure that will cause Euler buckling with our rocket is about 20 G N/m^2. If the Delta IV and Falcon 9 were made of the same material, they would be able to handle an order of magnitude more pressure simply because of their smaller fineness values.

7.2 Designing Bigger Rockets

In Section 7.1, we discussed designing rockets using commercially available enthusiast rocket motors and parts and designing such a rocket using the open-source code called OpenRocket. While OpenRocket, and other tools, are very useful as both teaching/learning and design tools, it does have its limitations. There are other codes available; some are expensive; others are open source, but, in some cases, if we have learned rocket science fundamentals, we can create our own design tools that are just as effective.

In this section, we will consider a larger rocket problem. What if we need to design an orbital rocket or larger? We first need to learn how to size our rocket motors, and then we can look for commercially available ones or use the information given in Chapters 4 and 5 to design and build a new one.

7.2.1 DRM #2: Orbital Liquid-Fueled Rocket

In Section 7.1, we designed a rocket based on commercial hobby/enthusiast solid rocket motors. These types of rockets will be useful and are great learning tools, but they are limited in capability.

Consider the problem of placing a 200-kg payload into a minimum circular orbit of 600 km (our mission requirement). Using Equation 2.73 from Chapter 2, we can determine that the velocity required to maintain such an orbit is about 7.56 km/s. Let's also assume that we have nine available liquid oxygen and methane (LOX/LCH_4) liquid fuel engines for our design effort. Each of these engines will deliver 250 kN of thrust with I_{sp} of 275 s. We are also given engine test data that tell us that, with the proper nozzle configuration, our engines will have an exhaust velocity at sea level, vex1, of 3,000 m/s, in midrange altitudes, vex2, of 3250 m/s, and at upper altitudes to space, vex3, of 3,500 m/s. The engines require a fuel/oxidizer mixture ratio of 3:1. Each of the engines has a mass of 100 kg. We have also been given a decommissioned Peacekeeper missile nose cone and asked to use this for our payload shroud. The shroud is made of aluminum, is 4.07-m tall, and is 2.34 m in diameter with 1.7-cm thick walls. We have also been asked to keep the rocket to a maximum of three stages due to launchpad limitations.

Have we developed enough tools thus far to at least determine the feasibility of such a rocket? Our OpenRocket exercise taught us some of the intricacies, difficulties, and complexities of rocket designing. However, we do have the tools and information within the previous chapters of this book to at least build ourselves a design tool to solve this problem. We will design this rocket using Mathcad.

First let's define our given parameters for the design calculations. These are shown in Mathcad Window 7.1.

Mathcad Window 7.1********************************

dnoseinner := 2.323

hnose := 4.07

dnoseouter := 2.34

$r1outer := \frac{dnoseouter}{2}$

r1inner := r1outer − 0.005 outer skin is 0.5 cm thick

r1outer = 1.17

r1inner = 1.165

Mpayload := 200

ρAl := 2810

Mnose := 352

MEngine := 100

mixtureratio := 3

ρfuel := 422.36

ρlox := 1141

****

The inputs are shown in Mathcad Window 7.1, and most are self-explanatory. All units are based on the meter–kilogram–second (mks) system of units. Note that r1outer and r1inner are the values assigned as the inner and outer radii of stage 1 and are calculated to mate exactly with the Peacekeeper nose-cone shroud. We make the design choice assumption here that the skin of the rocket will be 0.5-cm-thick aluminum. We will build our rocket body from aluminum, which has a density of 2,810 kg/m^3. The density of LOX is 1,141 kg/m^3, and the density of our fuel LCH_4 (liquid methane) is 422.36 kg/m^3.

Now, we need to develop a calculation for the mass of the propellants in the first stage. The following Mathcad code windows will have boxes within them. The values within these boxes are calculated by the code. These enclosed values are our sought-after design parameters. Also note that, in order to begin a design effort, some design choices have to be made to start with. We will start with a design choice of a first-stage propellant mass of 20,000 kg. We can always change the parameters later if we need to.

Mathcad Window 7.2********************************

Mpropellantsstage1 := 20,000

$Mloxstage1 := \frac{Mpropellantsstage1}{mixtureratio}$ Mloxstage1 = 6.667×10^3

Mfuelstage1 := Mpropellantsstage1 − Mloxstage1 Mfuelstage1 = 1.333×10^4

$Volumefuelstage1 := \frac{Mfuelstage1}{\rho fuel}$

$Volumeloxstage1 := \frac{Mloxstage1}{\rho lox}$

rfuelouter1 := 1.0025 rloxouter1 := 1.0025 tanks are 1/4 cm thick

rfuelinner1 := 1 rloxinner1 := 1

$$hfuel1 := \frac{Volumefuelstage1}{\pi\ rfuelinner1} \cdot 1.1 \qquad hfuel1 = 11.053$$

$$hlox1 := \frac{Volumeloxstage1}{\pi\ rfuelinner1} \cdot 1.1 \qquad hlox1 = 2.046$$

$$Vfueltank1 := \pi(rfuelouter1^2 - rfuelinner1^2)hfuel1$$

$$Mfueltank1 := \rho Al \cdot Vfueltank1$$

$$Vloxtank1 := \pi(rloxouter1^2 - rloxinner1^2)hlox1$$

$$Mloxtank1 := \rho Al \cdot Vloxtank1$$

****

Mathcad Widow 7.2 shows the next step in the code. Here, we calculated from the total propellant mass and the known mixture ratio the mass of the LOX and the mass of the liquid methane fuel. With known densities for the propellants, we also calculated the volume of each of them. We made an assumption that the propellant tanks were aluminum and have a wall thickness of 0.25 cm. We also made a design choice of limiting the tank's inner radii to 1 m. Note the 1.1 multiplier factors on the height of the fuel and LOX. This is a multiplier used to allow for structural components and mounting for the propellant tanks and ullage volume if needed.

Next, we will need to account for any mass required from structural ribs and supports for the exterior of the rocket. We will use thin-wall aluminum rectangular tubing. And we will use one support rib per meter of circumference. So there will be, N, supports found by the algorithm shown in Mathcad Window 7.3.

Mathcad Window 7.3**********************************

$$Lsupportout := 0.05$$

$$Wsupportout := 0.05$$

$$Lsupportin := 0.0475$$

$$Wsupportin := 0.0475$$

$$N := \frac{2\pi\ r1inner}{1}$$

$$N = 7.32$$

****

So there will be on the order of seven or eight supports inside the rocket stage running from top to bottom.

At this point, we can begin to calculate the height of our first stage using the calculations completed thus far. The height of the first stage is found by implementing Mathcad Window 7.4.

Mathcad Window 7.4************************************

$$hbody1 := (hfuel1 + hlox1)1.1$$

$$hbody1 = 14.409$$

****

Again, note the 1.1 multiplier is for structure and attachment componentry. Our rocket now has first-stage dimensions identified and calculated. From these, we can determine the "dry mass" of the first stage and the total mass.

Mathcad Window 7.5************************************

$$Vbodytube1 := \pi\,(r1outer^2 - r1inner^2)hbody1$$

$$Mbodytube1 := \rho Al \cdot Vbodytube1$$

$$Mbodystructure1 := hbody1 \cdot [[(Lsupportout - Lsupportin) \cdot [(Wsupportout - Wsupportin) \cdot N]]]$$

$$Mdrystage1 := (MEngine \cdot 3 + Mbodytube1 + Mbodystructure1 + Mfueltank1 + Mloxtank1) \cdot 1.2$$

$$Mdrystage1 = 2.837 \times 10^3$$

$$Mtotalstage1 := Mdrystage1 + Mpropellantsstage1$$

$$Mtotalstage1 = 2.284 \times 10^4$$

****

The multiplier in the case of the total dry mass of stage 1 is 1.2. This gives more design margin for structural elements. Mathcad Window 7.5 completes the analysis for designing the mass properties and basic physical dimensions of the first stage of the rocket.

We now repeat the process for stages 2 and 3. The process is shown in Mathcad Windows 7.6 through 7.9. Note that the design choices made for the second and third stages for total propellant mass were 7,000 kg and 4,000 kg, respectively.

Mathcad Window 7.6************************************

$$Mpropellantsstage2 := 7000$$

$$Mloxstage2 := \frac{Mpropellantsstage2}{mixtureratio}$$

$$Mfuelstage2 := Mpropellantsstage2 - Mloxstage2$$

$$Volumefuelstage2 := \frac{Mfuelstage2}{\rho fuel}$$

$$Mloxstage2 = 2.333 \times 10^3$$

$$Volumeloxstage2 := \frac{Mloxstage2}{\rho lox}$$

$$Mfuelstage2 = 4.667 \times 10^3$$

$rfuelouter2 := 1.0025$ $rloxouter2 := 1.0025$ tanks are 1/4 cm thick

$rfuelinner2 := 1$ $rloxinner2 := 1$

$hfuel2 := \frac{Volumefuelstage2}{\pi\ rfuelinner2} \cdot 1.1$ $hfuel2 = 3.869$

$hlox2 := \frac{Volumeloxstage2}{\pi\ rfuelinner2} \cdot 1.1$ $hlox2 = 0.716$

$Vfueltank2 := \pi(rfuelouter2^2 - rfuelinner^2)hfuel2$

$Mfueltank2 := \rho Al \cdot Vfueltank2$

$Vloxtank2 := \pi(rloxouter2^2 - rloxinner2^2)hlox2$

$Mloxtank2 := \rho Al \cdot Vloxtank2$

****

Mathcad Window 7.7**********************************

$hbody2 := (hfuel2 + hlox2) \cdot 1.1$

$Vbodytube2 := \pi\ (r1outer^2 - r1inner^2)hbody2$

$hbody2 = 5.043$

$Mbodytube2 := \rho Al \cdot Vbodytube2$

$Mbodystructure2 := hbody2 \cdot [[(Lsupportout - Lsupportin) \cdot [(Wsupportout - Wsupportin) \cdot N]]]$

$Mdrystage2 := (MEngine \cdot 3 + Mbodytube2 + Mbodystructure2 + Mfueltank2 + Mloxtank2) \cdot 1.2$

$Mdrystage2 = 1.227 \times 10^3$

$Mtotalstage2 := Mdrystage2 + Mpropellantsstage2$

$Mtotalstage2 = 8.227 \times 10^3$

****

Mathcad Window 7.8**********************************

$$\text{Mpropellantsstage3} := 4000$$

$$\text{Mloxstage3} := \frac{\text{Mpropellantsstage3}}{\text{mixtureratio}}$$

$$\text{Mfuelstage3} := \text{Mpropellantsstage3} - \text{Mloxstage3}$$

$$\text{Volumefuelstage3} := \frac{\text{Mfuelstage3}}{\rho\text{fuel}}$$

$$\text{Volumeloxstage3} := \frac{\text{Mloxstage3}}{\rho\text{lox}}$$

$$\text{Mloxstage3} = 1.333 \times 10^3$$

$$\text{Mfuelstage3} = 2.667 \times 10^3$$

rfuelouter3 := 1.0025 rloxouter3 := 1.0025 tanks are 1/4 cm thick

rfuelinner3 := 1 rloxinner3 := 1

$$\text{hfuel3} := \frac{\text{Volumefuelstage3}}{\pi\ \text{rfuelinner3}} \cdot 1.1$$

$$\text{hfuel3} = 2.211$$

$$\text{hlox3} := \frac{\text{Volumeloxstage3}}{\pi\ \text{rfuelinner3}} \cdot 1.1$$

$$\text{hlox3} = 0.409$$

$$\text{Vfueltank3} := \pi\ (\text{rfuelouter3}^2 - \text{rfuelinner3}^2)\text{hfuel3}$$

$$\text{Mfueltank3} := \rho\text{Al} \cdot \text{Vfueltank3}$$

$$\text{Vloxtank3} := \pi\ (\text{rloxouter3}^2 - \text{rloxinner3}^2)\text{hlox3}$$

$$\text{Mloxtank3} := \rho\text{Al} \cdot \text{Vloxtank3}$$

****

Mathcad Window 7.9**********************************

$$\text{hbody3} := (\text{hfuel3} + \text{hlox3}) \cdot 1.1$$

$$\text{Vbodytube3} := \pi\ (\text{r1outer}^2 - \text{r1inner}^2)\text{hbody3}$$

$$\text{hbody3} = 2.882$$

$$\text{Mbodytube3} := \rho\text{Al} \cdot \text{Vbodytube3}$$

$$\text{Mbodystructure3} := \text{hbody3} \cdot [[(\text{Lsupportout} - \text{Lsupportin}) \cdot [(\text{Wsupportout} - \text{Wsupportin}) \cdot \text{N}]]]$$

$$\text{Mdrystage3} := (\text{MEngine} \cdot 3 + \text{Mbodytube3} + \text{Mbodystructure3} + \text{Mfueltank3} + \text{Mloxtank3}) \cdot 1.2$$

$$\text{Mdrystage3} = 855.362$$

$$\text{Mtotalstage3} := \text{Mdrystage3} + \text{Mpropellantsstage3}$$

$$\text{Mtotalstage3} = 4.855 \times 10^3$$

****

Now, we need to allow for interfaces between the stages, componentry, and subsystems of the rocket. This is done by using a multiplier again. We simply take the combined weights of all the stages and multiply it. We will use a multiplier of 5% or 0.05. Using all of the given and calculated mass values, we can determine the mass ratio of the rocket stages. These calculations are given in Mathcad Windows 7.10 and 7.11.

Mathcad Window 7.10**********************************

$$Minterfaces := 0.05\cdot(Mdrystage1 + Mdrystage2 + Mdrystage3 + Mnose)$$

$$Mtotaldry := Mdrystage1 + Mdrystage2 + Mdrystage3 + Mnose + Minterfaces + Mpayload$$

$$Mtotaldry = 5.739 \times 10^3$$

$$Mtotalpropellants := Mpropellantsstage1 + Mpropellantsstage2 + Mpropellantsstage3$$

$$Mtotalpropellants = 3.1 \times 10^4$$

$$Mtotalwet := Mtotaldry + Mtotalpropellants$$

$$Mtotalwet = 3.674 \times 10^4$$

$$MRstage1 := \frac{Mtotalwet}{Mtotalwet - Mpropellantsstage1}$$

$$MRstage1 = 2.195$$

****

Mathcad Window 7.11**********************************

$$MRstage2 := \frac{Mtotalwet - Mdrystage1 - Mpropellantsstage1}{Mtotalwet - Mpropellantsstage1 - Mpropellantsstage2 - Mdrystage1}$$

$$MRstage2 = 2.015$$

$Mpropstage1 := Mpropellantsstage1$

$Mpropstage2 := Mpropellantsstage2$ Convert to shorter variables

$Mpropstage3 := Mpropellantsstage3$

$$MRstage3 := \frac{Mtotalwet - Mdrystage1 - Mpropstage1 - Mdrystage2 - Mpropstage2}{Mtotalwet - Mpropstage1 - Mpropstage2 - Mpropstage3 - Mdrystage1 - Mdrystage2}$$

$$MRstage3 = 3.394$$

****

Note that, in Mathcad Window 7.11, some of the variables were converted to variables with shorter names for convenience.

Given the exhaust velocities, I_{sp}, the propellant masses, and the mass ratios, we can determine the mass flow rate, m-dot, and the time to burnout, tbo, for each stage. These are calculated in Mathcad Window 7.12.

***************************************Mathcad Window 7.12**************************************

Isp1 := 275

Δvstage1 := 9.8·Isp1·ln(MRstage1)

$\Delta vstage1 = 2.119 \times 10^3$

ThrustperEngine := 250,000

vex := 3000

$mdot := \frac{ThrustperEngine \cdot 3}{vex}$

mdot = 250

$tbo1 := \frac{Mpropellantsstage1}{mdot}$

tbo1 = 80

Isp2 := 275

Δvstage2 := 9.8·Isp2·ln(MRstage2)

$\Delta vstage2 = 1.888 \times 10^3$

ThrustperEngine2 := 250,000

vex2 := 3250

$mdot2 := \frac{ThrustperEngine2 \cdot 3}{vex2}$

mdot2 = 230.769

$tbo2 := \frac{Mpropellantsstage2}{mdot2}$

tbo2 = 30.333

Isp3 := 275

Δvstage3 := 9.8·Isp3·ln(MRstage3)

$\Delta vstage3 = 3.293 \times 10^3$

ThrustperEngine3 := 250,000

vex3 := 3500

$mdot3 := \frac{ThrustperEngine3 \cdot 3}{vex3}$

mdot3 = 214.286

$tbo3 := \frac{Mpropellantsstage3}{mdot3}$

tbo3 = 18.667

****

The final step of the design analysis is to determine if we can achieve the altitude required, as well as the orbital velocity required. Recall from the description of DRM #2 that our requirements are to place a 200-kg payload into a 600-km orbit, which requires about 7.56 km/s velocity to maintain. From the given information and all of the calculated information in the previous Mathcad Windows, we can now use the rocket equation as defined in Equation 3.32 to calculate the burnout velocities after each stage. The final velocity will be the burnout velocity after stage 3. The final calculations of the design are shown in Mathcad Window 7.13.

Mathcad Window 7.13**********************************

$vbo1 := vex \cdot \ln(MRstage1) - 9.8 \cdot tbo1$

$vbo1 = 1.575 \times 10^3$

$ymax1 := \frac{vbo1^2}{2 \cdot 9.8}$

$ymax1 = 1.265 \times 10^5$ in meters

$ymax1km := \frac{ymax1}{1000}$

$ymax1km = 126.518$ in kilometers

$vbo2 := vex2 \cdot \ln(MRstage2) - 9.8 \cdot tbo2 + vbo1$

$vbo2 = 3.554 \times 10^3$

$ymax2 := \frac{vbo2^2}{2 \cdot 9.8} + ymax1$

$ymax2 = 7.71 \times 10^5$ in meters

$ymax2km := \frac{ymax2}{1000}$

$ymax2km = 771.014$ in kilometers

$vbo3 := vex3 \cdot \ln(MRstage3) - 9.8 \cdot tbo3 + vbo2$

$vbo3 = 7.648 \times 10^3$

$ymax3 := \frac{vbo3^2}{2 \cdot 9.8} + ymax2$

$ymax3 = 3.755 \times 10^6$ in meters

$ymax3km := \frac{ymax3}{1000}$

$ymax3km = 3.755 \times 10^3$ in kilometers

$\Delta vtotal := vbo3$

$\Delta vtotal = 7.648 \times 10^3$

****

Our Mathcad program enabled us to develop a concept architecture for a launch vehicle that could deliver about 7.65 km/s of Δv with a payload of 200 kg. The final altitudes and orbital velocity calculated are sufficient to perform the required mission. Our concept has three stages with three engines per stage. There are mass design parameters calculated and sizes and volumes for the components and structure, as well as the total propellant needs. As the details of the rocket become more and more defined, these initial calculated parameters will be used as design envelopes. If the actual final rocket components grow in size or mass, then the flight analysis will need to be recalculated and iterated with each change. Also note that this is just the concept for the design. For a final real design, aerodynamic flight models and simulations must be completed. Vibrational and mechanical analyses must be completed. And, at some point, the control electronics and avionics will have to be designed, built, and tested. This is truly the very first step just to determine if building such a rocket to meet particular requirements is possible. At this point, it would appear that such a rocket could be constructed. Figure 7.12 shows a drawing of what our DRM #2 rocket concept looks like.

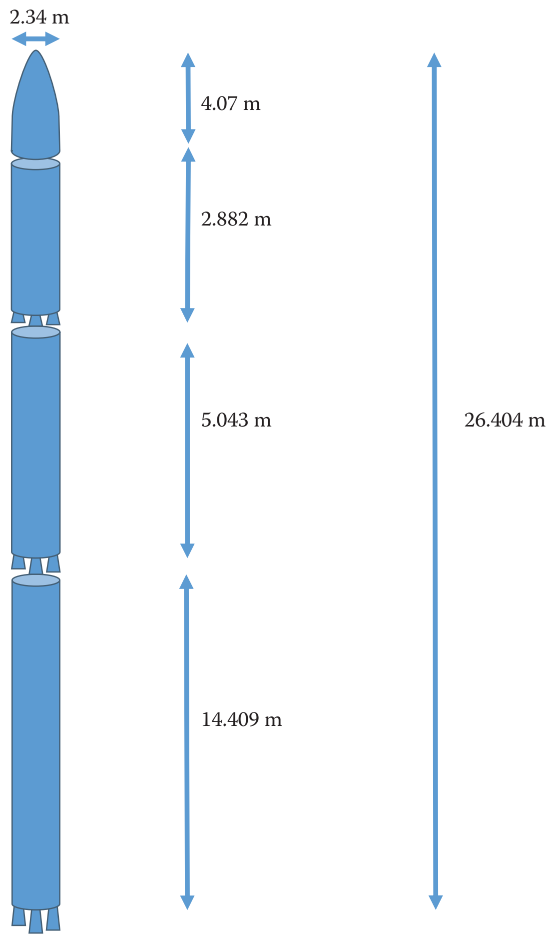

FIGURE 7.12
Rocket design for DRM #2.

7.3 Reverse Bifurcation Designing

In Sections 7.1 and 7.2, we discussed designing rockets using open-source software and Mathcad. We also talked about aspects and approaches to designing a rocket based on mission requirements and how to derive further requirements from them. The rocket design process can be quite complicated. When starting with a simple set of mission requirements and a blank page, it can be overwhelming as to where to start. The problem is like a simile for eating an elephant. How does one eat an elephant? One bite at a time! The concept sounds simple until we look at the elephant and realize that we have no idea which part we should eat first. This is exactly how the rocket scientist and the engineer can be overwhelmed while staring at that blank page. There truly is no right place to start. We simply must pick a point and start chewing.

While the design process might seem as overwhelming as our pachyderm delight must be to a lion, years of rocket development programs and systems engineering processes developed during them does lead us toward the means of narrowing down our design parameters. Figure 7.13 shows how we might go through a process of elimination and create a backward flowing or reverse bifurcation design tree that will lead to a final design concept.

Assume we desire for whatever reason to design a rocket much like the one used in the DRM #2, but a new mission requirement is to make the rocket as low cost as possible. We must start by eliminating some design possibilities by making some design choices. A first choice might be to decide if we plan to build a reusable or expendable vehicle. Then, we can choose the types of engines, propellants, control methods, and so on. The reverse bifurcation allows us to look at all the available possibilities and then eliminate some and keep others. Once we reach a final concept, we can then simulate them with our design tools and make decisions whether or not to back up the bifurcation and take a different design choice branch.

Note that the bifurcation shown in Figure 7.13 is notional and not intended to be all encompassing of every possible design choice available to mankind. Feel free to make your own design chart and see what other components might be added to it. Your end result will be a good starting point for your next rocket design concept.

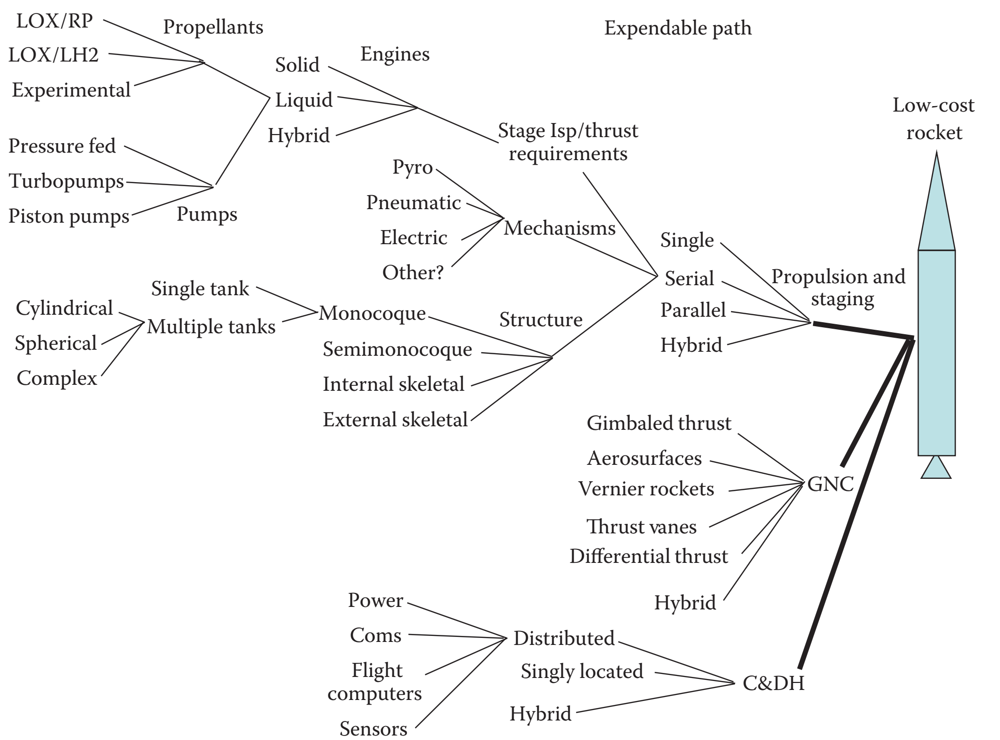

FIGURE 7.13
Reverse bifurcation design process.

7.4 Chapter Summary

In this chapter, we have finally put all of the information in the book thus far to work for us by designing two rocket concepts for two completely different purposes.

In Section 7.1, we began by discussing the requirements defining process and how to "derive requirements" from given mission requirements. We continued through the development of our understanding of requirements by developing a launch vehicle based on a set of given mission requirements. At this point, we introduced the open-source software rocket-simulation tool called OpenRocket. We worked through our example using the OpenRocket code and developed a concept for a sounding rocket using all commercially available hobby rocket parts to perform a mission need. We also discussed the fineness ratio of a rocket body and how it can potentially impact rocket performance and failure risk. We then extrapolated the fineness ratio with the critical pressure for Euler buckling and showed how the two are related.

Likewise, in Section 7.2, we discussed the details of designing even larger rockets. In this section, we introduced a new set of mission requirements, as well as a new means for calculating and simulating rocket performance. A Mathcad code was given and implemented to derive a concept rocket architecture that could place a 200-kg payload into a 600-km orbit. The Mathcad code could easily be modified to analyze other rocket designs for other needs.

Chapter 7 has demonstrated for us how to start in a rocket design effort. We learned the three basic rules for rocket design. At some point, we actually have to put some numbers in place and begin thinking through our rocket designs. With the tools discussed in this chapter, we can now do more than just think about our designs; we can actually begin to design the rockets. Once several iterations of simulation with OpenRocket and/or the Mathcad code have been completed, it will then be time to develop even more detailed engineering models and eventually blueprints and mass models with details of every last nut and bolt in the vehicle.

Use the tools in this chapter for educational, experimental, and developmental purposes. The more we design rockets of different types for different needs, the more proficient we will become at understanding the intricacies and complexities required in rocket science and engineering.

Exercises

7.1 What is INCOSE?

7.2 In your own words, give a definition for the requirements defining process.

7.3 What is a DRM?

7.4 What is a derived requirement?

7.5 The U.S. Army has a call for proposals to deliver a 400-kg payload to an altitude of 40 km with a downrange distance impact point of at least 70 km. You are evaluating the proposals for the army. One proposed solution "requires" solid rocket motors only. Would this be a derived requirement, a mission requirement, or a design choice?

7.6 Give the three rocket design rules and discuss what they mean.

7.7 Use OpenRocket to design a hobby rocket using an F-class motor. The mission requirement is to reach the highest altitude possible and deliver the rocket safely back to Earth. Show all components of the rocket including centering rings, motor tubes, thrust structure/bulkheads, parachutes, nose cones, engines, couplers, etc. Simulate and plot the flight to ensure it is stable.

7.8 A rocket is 7-m tall and 1.4-m in diameter. Calculate the fineness. How does it compare to a Delta IV?

7.9 If the rocket in Exercise 7.8 is made of 6061 aluminum that has a Young's modulus, E, of 69 GPa (69×10^9 N/m^2), and a density of 2,700 kg/m^3, and we assume after structure and tanks that it has an effective wall thickness of 25 mm, calculate the critical pressure for the buckling of the rocket.

7.10 Use a computer program (such as Mathcad) to graph the critical pressure of the rocket in Exercises 7.8 and 7.9 versus a wall thickness ranging from 1 to 50 mm. Also, plot the mass of the rocket versus the wall thickness.

7.11 Use the Mathcad algorithm/code given in Section 7.2 to reanalyze DRM #2, but this time with a 300-kg payload. Note: feel free to convert the Mathcad code to MATLAB or any other software you prefer.

7.12 In Exercise 2.8, we studied the following rocket:

A rocket is launched with a burnout velocity of 75 m/sec, a burnout altitude of 300 m, and a burnout range of 100 m. Assuming a flight path angle of 75°, calculate the final range of the rocket when it impacts the ground.

Use OpenRocket to simulate as best you can the rocket in this exercise. Simulate and plot your results.

7.13 What does OpenRocket calculate for the maximum altitude reached by the rocket in Exercise 2.8?

7.14 Design, build, and fly a rocket using any motor and materials of your choice.

8

How Reliable Are Rockets?

So far, in this text, we have discussed many details about the complexities of rockets and how to design, build, and test them. One consideration that every rocket scientist and engineer must make from the start is that rockets can fail. They might just sit on the pad and do nothing. They might also catastrophically explode, causing serious damage and possibly injuries or death, as has been the case in many experimental rocket programs throughout history. Recall from Chapter 1 the story of the Chinese official Wan-Hu who attempted to build a rocket chair and, in the process, vanished in a puff of smoke. Since rockets are so volatile, like holding onto the tail of a fire-breathing dragon, how do we plan for and determine the likelihood of a rocket failing or being successful? Just how do we determine the "reliability" of the rocket we are designing?

As we design rockets, it is of utmost importance to consider reliability from the beginning. If our rocket development program requires us to deliver a rocket that will perform the mission objectives 99% of the time without failure, or if it only requires 80% completion of objectives, this requirement will drive the type of components to be used in the design, as well as the amount of testing required to achieve the desired reliability.

If our rocket is to carry a manned crew, then it is most likely to require better than 99% reliability to be built into the design. Such high reliability will require multiple redundant systems, subsystems, and components, as well as very exhaustive and extensive testing. The required increased number of redundant systems and testing will drive the cost of the vehicle up. This is why Space Shuttles and other manned systems are so expensive.

If the rocket is to deliver inexpensive nanosatellites to low Earth orbit and, to do so at very low cost as per our design reference missions (DRMs) in Chapter 7, then it is likely that a significant amount of the redundancy and testing might be forgone, provided that the user will accept the riskier possibility of the rocket failing catastrophically. It is important for us to understand a bit about the probability of failure based on parts count, the number of redundant systems, and the amount of testing to be performed. In this chapter, we will see the coupling between system complexity, testing, and reliability and why we need at least a rudimentary understanding of this coupling in order to design a launch vehicle (LV) successfully.

8.1 Probability and Parts Count

The complexity of any rocket design in general continues to grow as the concept architecture matures, and more detail about systems, subsystems, and components becomes clearer. While the complexity of the design becomes more defined and the nonredundant parts count increases, the risk of the vehicle design failing will increase accordingly. This is the point during the design process when, with good design practices, the rocket designers should begin to shrink the rocket design space. It is very likely that there will still be many "black-box" or "placeholder" components and subsystems still in the design and therefore leaving many risks that are simply not quantifiable based on the available level of analysis and design maturation, and, therefore, the said risks remain unknown. In the early design process, there is likely to be a large number of items of unknown maturity, and, likewise, it is very difficult to determine the projected reliability of the test articles and, ultimately, the flight vehicle. However, it is possible to calculate the rough order of magnitude reliability. And, for rocket scientists and engineers, sometimes, the rough order of magnitude reliability calculation is enough to tell us if an idea is a "go" or a "no-go" concept.

8.1.1 The Probability of Success and Quality Control

Figure 8.1 demonstrates this phenomenon of reliability versus the number of single-point-of-failure articles. The articles can be modeled as systems, subsystems, or even single components if that single component will cause the loss of mission. By definition, in this example, the loss of an item means the loss of the vehicle/mission, so these items are the critical performance technologies such as an engine combustion chamber, a turbopump, a guidance, navigation, and control (GNC) system, or a main thrust structural component. It is straightforward to calculate the reliability of this type of system based on Bernoulli trial statistics and processes discussed in MIL-STD 882C and other reliability sources. At this point, it is recommended for readers to go to the Internet and download a copy of the MIL-STD 882C as it is a good reference for how the U.S. Defense Department implements reliability calculation standards.

Consider a system with N parts. Each part has a reliability or quality control, QC, value ranging between 0.0 and 1.0. A $QC = 1$ means it never fails, and a $QC = 0$ means it always fails. If all parts are the same part with the same QC value, then the probability of the system being successful is

$$P_{success} = QC^N. \tag{8.1}$$

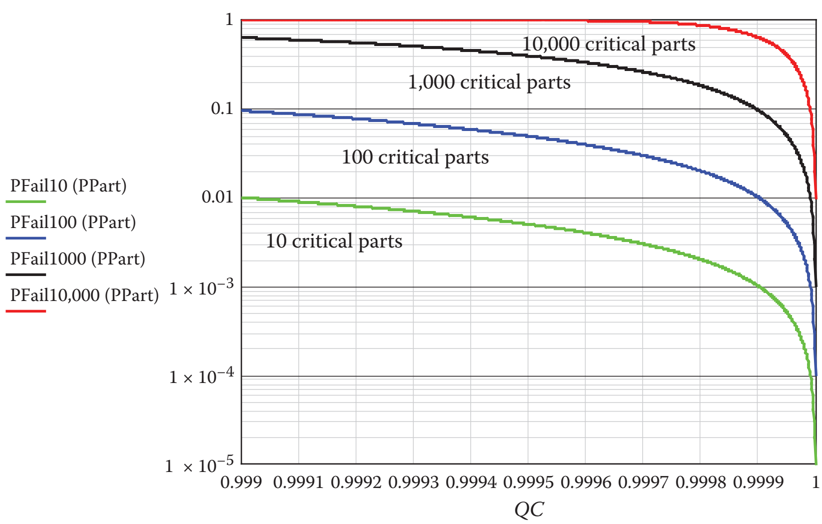

FIGURE 8.1
Probability of rocket failure versus quality control factors and the number of parts.

And, likewise, the probability of system failure is given by

$$P_{failure} = 1 - P_{success} = 1 - QC^N. \tag{8.2}$$

From Equation 8.2 and Figure 8.1, it becomes clear that the greater number of single-point-of-failure items leads to a high probability of system failure.

An understanding of the system parts count; the quality control of each part; and an assessment of single-point-of-failure components, subsystems, and systems will enable the rocket designers to determine if their design space is growing in risk. While it may be impossible to prevent the parts count of single-point-of-failure items from growing into something as complex as an LV, tracking the reliability with this growth will illuminate if there is a need to add redundant systems to the design.

8.1.2 Single Point Failure

Let's consider an example of single-point-of-failure items in our rocket design. Assume for now that our rocket design is to use semimonocoque liquid propellant tanks for each of three stages. There is an oxidizer and a fuel tank on each stage. Each tank requires two structural ribs (hence, the semimonocoque construction), and each rib is a single point of failure component of the propulsion system of the rocket. In other words, there are four single-point-of-failure ribs on each stage. And, therefore, there are 12 of these items with no redundancy. It is highly likely that adding redundant structural members would increase the mass of the rocket and, in turn, increase the thrust required from the engines. Recall our number-two rule of rocket design:

Rocket Design Rule #2: If we touch the rocket ANYWHERE, we have touched the rocket EVERYWHERE.

Using Figure 8.1 and allowing for 2 of the structural ribs to be perfect (which is nonsense), leaving 10 with some QC value less than unity, it becomes clear that even if the single-point-of-failure items are successful with a $QC = 0.999$, the probability of the rocket failing based on 10 structural ribs alone is only 1%. And this is only the tankage support members (and only 10 out of 12 of them). There will clearly be many other components of the rocket system design that are single point failures. It is extremely important for the rocket scientists and engineers who are designing the rocket to keep the parts count of single-point-of-failure items in mind and to maintain appropriate documentation expressing the number of single- or even multiple-point-of-failure systems (systems with redundancy) from the beginning to the end of the design process.

A "probability and reliability assessment" (PRA) needs to be conducted only at the top-level rocket block diagram with the onset of the rocket development effort. And the PRA should be continuously updated as the rocket design is matured. Doing so will aid the rocket development team in determining what analyses must be conducted in order to understand failure risks, which, in turn, will drive the types and number of component, subsystem, and system tests to be implemented during the test phase.

The only way to gather some of the system information that will lead to a meaningful PRA is to perform analysis and/or tests until the details of the vehicle are understood. In many cases, the complexity of the system is too great for analysis, and a detailed test effort must be conducted. Most manned rocket development efforts have historically ended up with vehicles consisting of millions of parts. That's right, MILLIONS of parts.

With the advent of modern computational systems and an understanding of reliability probability mathematics, it is possible to track the reliability of so many items. However, analysis will only take the design team so far, and there are always unknowns that can't be calculated. This is where a rigorous and well-conducted test program is required. Neither analysis nor testing by themselves is adequate. As is usually the case, a well-documented PRA is often a requirement for range testing and is certainly a requirement for Federal Aviation Administration (FAA) flight safety. In other words, if we want to actually be able to fly our rocket that we build, we had better be continually updating the PRA and documenting it along the way. Otherwise, the FAA and most flight test ranges will not let us fly it.

8.2 Testing Our Rockets for Reliability

While an increasing parts count decreases rocket reliability, sometimes, there is just no way around having a complex arrangement of multiple parts. In that case, there is only one real way to drive the risk of failure down. We test, and test, and test, and test some more.

An increase in testing will improve the reliability. This is due to the testing enabling a better understanding of potential failure modes and improving and verifying component quality control knowledge. The more we test our rocket design, the more we will understand how it might fail and how to keep it from doing so.

8.2.1 Reliability versus Testing

Figure 8.2 shows a historical perspective of liquid-fueled rocket engine development efforts versus the number of tests required to quantify the reliability of the engines.

Note that Figure 8.2 shows heritage liquid propellant engines that have been used recently or are in use to date, ranging from the Space Shuttle Main Engines (SSMEs) to the

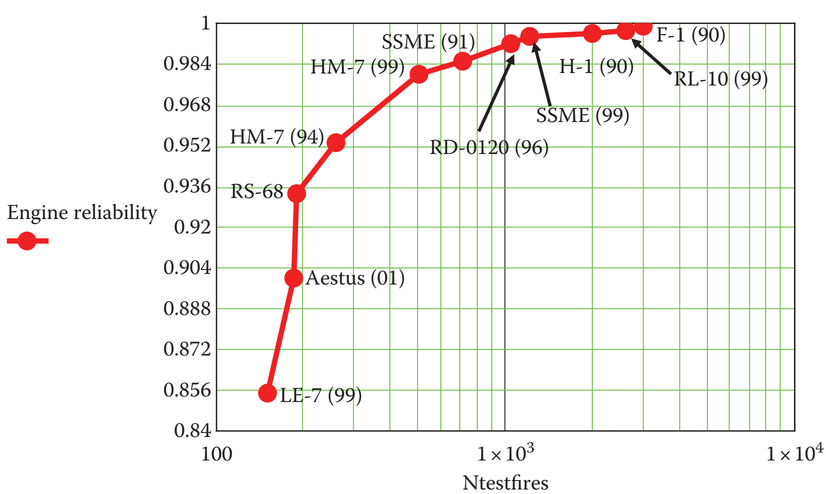

FIGURE 8.2
Engine reliability versus number of test fires.

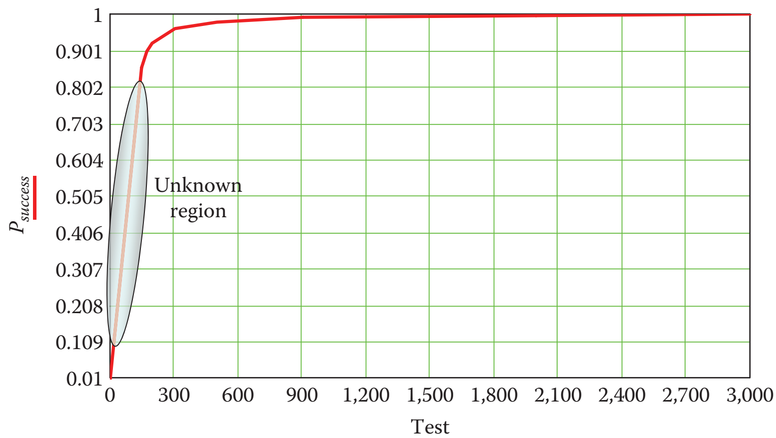

FIGURE 8.3
Probability of successful engine performance is unknown for low test fire numbers.

RS-68s used on the Delta IV rocket. The figure shows that testing with over 150 hot fires, as in the case of the LE-7 engine, only leads to a reliability assessment of the engines to about 85%. It takes upwards of thousands of tests to lead to greater than 99% engine reliability.

Figure 8.2 suggests that a new rocket engine design effort will require hundreds of hot fire engine tests that are likely to be outside of the scope and available budget for the program. Figure 8.3 shows the same graph as given from Figure 8.2, but extrapolated to the regime of testing where a lower-budget small rocket design program might reside. In the region with fewer than 150 or so hot fire tests, an understanding of the reliability of the engine is in a heretofore unknown region. Since there are no real data of rocket engine development with so few engine hot fire tests, it is unclear if the curve extrapolation truly means anything.

What it does suggest, however, is that so little testing leaves our engine reliability as an unknown. The standard approach with the large-budget programs allows for design–build–test–fail–fix–retest and then to continue along this path for many iterations. The lower-budget programs certainly cannot afford the luxury of such a test program.

Hence, it is likely that a true assessment of the engine reliability will only be illuminated after much analysis, the test program, and many operational flights in the future. This graph illustrates a significant risk due to having an unquantifiable engine reliability. This is a risk with lower-cost development efforts that the community and users will have to determine for themselves if they are willing to accept the risk. Projects and companies such as Virgin Galactic, Xcor, and others in the commercial space industry must deal with this type of unknown risk and plan their design space around it or simply accept it as a high and potentially very dangerous risk.

8.3 Redundant Systems and Reliability

The best way to begin a PRA for a rocket design is to start early with the most top-level block diagram or picture of the rocket system available. There are basic components of any rocket such as the payload shroud, stages, and engines. Starting with only a handful of top-level systems and applying a reliability analysis will offer insight as to critical subsystems and redundancy requirements.

8.3.1 Reliability Is Costly

Figure 8.4 shows the probability of our DRM #2 rocket design (from Chapter 7) success if the only failure items considered are the engines on each stage and the payload shroud separation (Pshroud probability assumed 99.99% successful). There are, of course, many other subsystems and components that could fail, but this figure emphasizes the point of needing to understand the reliability of the engines (and other subsystems and components). Assume for now that we stay with the design choice of three engines per stage as in our DRM #2 design from Chapter 7. Let's add a mission requirement that our rocket is 80% reliable. That means 8 out of 10 times, it will deliver our payload to the prescribed orbit.

The figure shows that, in order to reach the DRM goal of 80% launch reliability, the engines must be about 97% reliable each. According to Figure 8.2, our development effort would require thousands of hot fire tests (and/or flights) to have this type of reliability assessment knowledge. That would be very expensive.

The total probability of a successful flight based on only the parameters in Figure 8.4 is calculated by

$$P_{total} = P_{stage1} P_{stage2} P_{stage3} P_{shroud}. \tag{8.3}$$

$$P_{total} = QC^3_{engines1} QC^3_{engines2} QC^3_{engines3} P_{shroud}. \tag{8.4}$$

What is being suggested and emphasized here is that extreme care and due diligence must be maintained in all aspects of the rocket design program in order to reduce as much

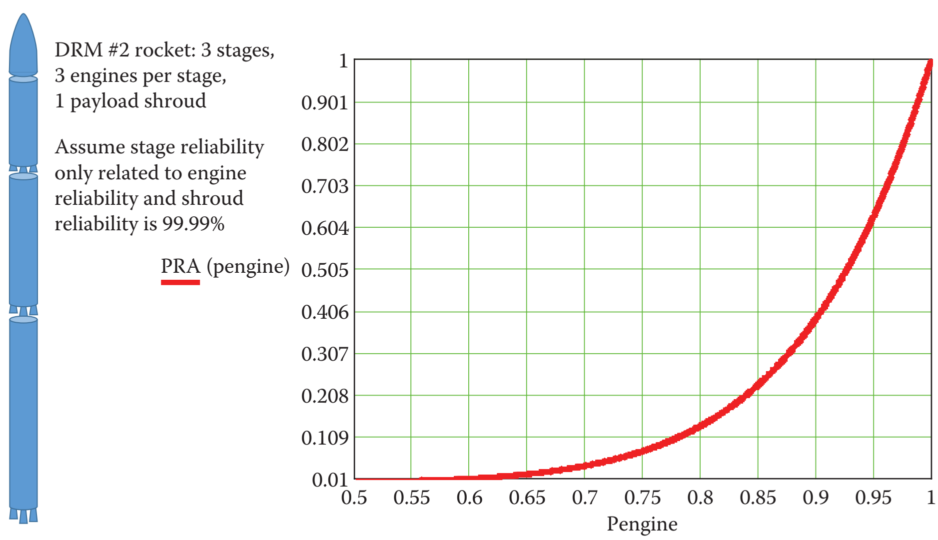

FIGURE 8.4
Probability and reliability assessment of DRM #2 rocket based on the probability of engine success.

risk as possible and to reveal the unknown and unquantifiable risks. It should be reiterated here that our DRM model is very much like the fledgling commercial space industry, whereas the small companies do not have the budget to identify all the unknown variables through testing. The emphasis that should be taken away from this analysis is that while the low-budget effort-developed rocket design might one day reach 80% reliability, during the testing phases, the reliability is for all intents and purposes unquantifiable. That is why developing new rockets on a shoestring budget is indeed such a trailblazing and Herculean effort, and the development teams and their customers should not lose sight of this fact. I'm not so certain I would want to be a passenger on a rocket that is only going to reach its destination 8 times out of 10. Imagine how much harder, meaning expensive, it is to push the reliability upwards to 99% and higher!

8.3.2 Reliability and Series Systems

Once the top-level PRA, such as shown in Section 8.3.1, is complete, the next step is to drive downward deeper into the block diagram of the LV. At this point, we begin to add more details of the subsystems and determine which subsystems are single point of failure ones. If there are too many single-point-of-failure subsystems, it will be necessary to consider the addition of redundant subsystems. Again, realizing that any additional systems, subsystems, and components will impact overall rocket performance and increase the requirements on the thrust and I_{sp} of the engines, with each additional redundant system added, a new rocket performance analysis must be conducted.

As shown in Equation 8.3, single-point-of-failure items impact reliability as a multiplicative value. These types of items are known as *series* items. These items can be components, subsystems, or entire systems as long as the overarching "black box" describing them is represented as a single reliability value.

Figure 8.4 shows a typical reliability block diagram (RBD) for a series system. The items are represented schematically in the same manner as series components in an electrical circuit diagram. The probability of success of the system is calculated as

$$P_{total} = P_{success1} P_{success2} P_{success3} P_{success4} \tag{8.5}$$

$$P_{total} = \prod_{i}^{N} P_{successi} \tag{8.6}$$

Each of the items represented in the RBD in Figure 8.5 is a single-point-of-failure item.

8.3.3 Reliability and Parallel Systems

For items that are redundant, they are placed in parallel, as shown in Figure 8.6, and the probability of success is calculated differently. The system is then known as a *parallel*

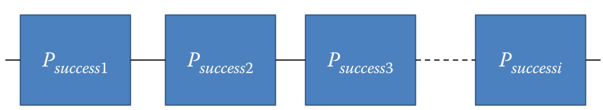

FIGURE 8.5
Probability of success for series components.

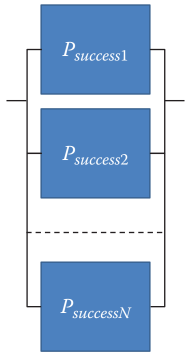

FIGURE 8.6
Probability of success for parallel components.

system. The parallel system is successful unless all the items in it fail. The probability of failure of the system in Figure 8.6 is therefore

$$P_{failure} = (1 - P_{success1})(1 - P_{success2})(1 - P_{success3})\cdots(1 - P_{successN}) \tag{8.7}$$

$$P_{failure} = \prod_{i}^{N}(1 - P_{successi}) \tag{8.8}$$

And, likewise, the probability of success is

$$P_{success} = 1 - P_{failure} = 1 - \prod_{i}^{N}(1 - P_{successi}) \tag{8.9}$$

Placing items in parallel is the process of adding *redundancy*.

8.3.4 Reliability and Mixed Series and Parallel Systems

While redundancy is beneficial, in many cases, it is not required every time. Some items are not single-point-of-failure items, are not in the critical function path, or are extremely unlikely to fail.

So, in most realistic cases, a system will be a mix of parallel (redundant) and series (nonredundant) items. If, as in Figure 8.5, for example, we wanted to make one of the items redundant in order to improve the PRA value, then the redundant item would be placed in parallel with the initial one, as shown in Figure 8.7. The probability of success of the hybrid system shown in Figure 8.6 is given by

$$P_{success} = 1 - P_{failure} \tag{8.10}$$

$$P_{success} = 1 - P_{success1}[1 - (1 - P_{success2a})(1 - P_{success2b})]\prod_{i=3}^{N} P_{successi} \tag{8.11}$$

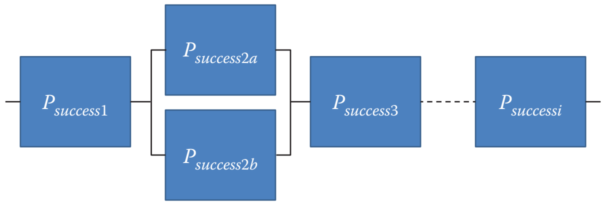

FIGURE 8.7
Probability of success for series and parallel components.

Example 8.1

Calculate the probability of success (reliability) of the complex system shown in Figure 8.8. For the system, use the following values: R1 = R3 = RN = 0.99, R2 = 0.98, M = 6, and N = 5.

$$P_{success} = P_1\left[1-\prod_{i=1}^{M}(1-P_{2i})\right]\prod_{j=3}^{N}P_i \tag{8.12}$$

$$P_{success} = (0.99)\left[1-\prod_{i=1}^{6}(1-0.98)_i\right]\prod_{j=3}^{5}(0.99)_j \tag{8.13}$$

$$P_{success} = (0.99)[1-(0.02)^6](0.99)^3 = 0.96. \tag{8.14}$$

So, the system shown in Figure 8.8 has a probability of success of 96%.

Now that we have seen how to calculate the probability of success of a complex system, the process can be used to evaluate the likelihood of success of a rocket design based on the knowledge of the components. As the rocket design becomes mature and the block diagram is filled in with more details, estimates of the PRA become more realistic.

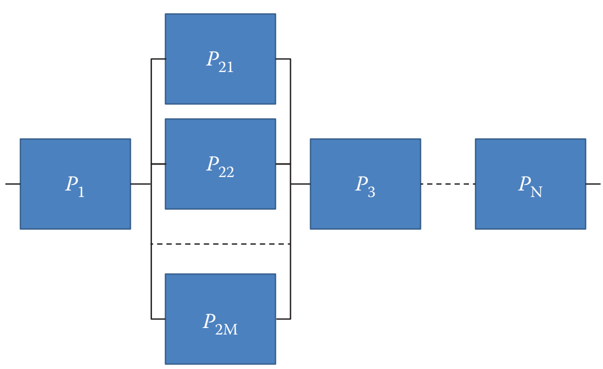

FIGURE 8.8
Probability of success for N series and M parallel components.

The proper approach to developing a PRA is to begin as we have already discussed here. The starting point should always be to use a system RBD and build the PRA calculations from a "top–down" perspective. During the design process, there will be occasions when detailed information about all parts on the rocket will be available and will likely be kept in a spreadsheet fashion. The rocket scientists or engineers must be cautious not to start from the "bottom–up" perspective, and they must use the industry reliability values given for each component and allow the spreadsheet to generate a PRA value. The value will be erroneous at best from the start but will improve as the design matures. A PRA must follow the proper flow of redundancies and account for serial and parallel items from the topmost RBD level down to the component level in order for it to be a reliable calculation. A spreadsheet format could be implemented for such a calculation, provided that the appropriate formulation based on the RBD is used.

8.4 Chapter Summary

In this chapter, we developed an understanding of how parts count plays a part in the reliability of a rocket design. The design engineer must take into account how increased complexity of the rocket and supporting systems will, in turn, decrease the likelihood of success. In order to improve the reliability of the rocket, past development programs show that increased testing is very important. As budget allows, the rocket design effort should plan to test as much as possible. However, in some cases, there is not enough budget to test exhaustively, and the rocket scientists and engineers must take this into account when making choices on design parameters versus acceptable risks.

A major point to take away from this chapter is the importance of starting off in the design process with a block diagram of the rocket. Without a block diagram of the rocket, it would be almost impossible to develop the RBD. In fact, the rocket block diagram is the top-level RBD. The RBD therefore becomes a very useful tool for the rocket design engineer to use to maintain a detailed understanding of critical items in the rocket that can singularly cause mission failure. If the RBD is developed appropriately, it is truly the complete picture of the rocket design and will give a good and reliable estimate on the rocket's chances for success or failure.

Exercises

8.1 What is a PRA?

8.2 In your own words, give a definition for the RBD.

8.3 What is the difference between series and parallel components when it comes to calculating reliability?

8.4 What does *QC* stand for?

8.5 If a rocket system is determined to be extremely low reliability because of complexity in design, how can the reliability risk be reduced?

8.6 The Russian Proton rocket has six engines on the first stage, three on the second, and one on the third. Assuming no other risky components exist on the rocket (engine-based assessment only), determine how reliable each engine must be to make the Proton 90% reliable.

8.7 Explain how Rocket Design Rule #2 is important when it comes to improving a rocket's reliability.

8.8 Calculate the probability of success (reliability) of the complex system shown in Figure 8.8. For the system, use the following values: R1 = R3 = RN = 0.91, R2 = 0.8, M = 8, and N = 9.

8.9 If a rocket system has been tested less than 150 times, should its PRA be reliable, unreliable, or unknown?

8.10 For the hobby rocket you designed in OpenRocket for Exercise 7.7, draw an RBD to the maximum level of detail available.

8.11 For the RBD developed in Exercise 8.10, calculate the PRA. Make reliability assumptions as best you can based on test data, personal experience, and literature available on the Internet and within OpenRocket.

9

Are We Thinking Like Rocket Scientists and Engineers?

Throughout this text, we have been developing a knowledge base from historical to future rocket programs, orbital mechanics, the laws of rocketry, types of rockets, systems engineering, testing, and modeling and simulation, and various other fields of science and engineering. What we have learned thus far is merely scratching the surface of the complexity of rocketry. We have barely even discussed the chemistry of rocket fuels, for example. Rocket chemistry for solids, liquids, hybrids, and gels is an amazingly complicated topic within itself and beyond an introductory text. There is cutting-edge materials science for rockets that a scientist could study for an entire career. The point is that most rocket scientists and engineers never really become experts at all facets of the field. Instead, they become experts at a particular subset while maintaining a generalist's knowledge of the field as a whole. So, from an introductory standpoint, rather than attempting to become experts in a particular subset of rocketry, we will learn to think in general about rockets from a big picture, and start learning to think like rocket scientists and engineers must.

In order to think like rocket scientists, we will start using what we've learned to identify some unusual quirks of rockets that must be considered when designing, developing, or implementing them. By no means will this chapter be exhaustive of the nonobvious aspects of rockets. In fact, scratching the surface would be an optimistic description. We will be scratching the scratch on the surface in the topic of unexpected and unusual rocket problems. Also, by no means will the descriptions and calculations in this chapter be of great detail or the complete story. Instead, they will be "back-of-the-envelope" calculations that will be conducted to offer insight into the issues.

By pointing out some of these more devilish concepts, ideas, and aspects, we will begin to see that the rocket scientist and the engineer must have a very open mind, be very clever, and truly think "out of the box" because the array of multidiscipline problems one must face is vast. Sometimes, unforeseen combinations of the laws of nature will occur that can be catastrophic to rocket systems, and, therefore, the rocket scientist or the engineer must learn to think of the unusual, unlikely, and unthinkable.

9.1 Weather Cocking

We will start our foray into thinking like a rocket scientist by considering what happens to rockets upon launch if there is a prevalent crosswind. A simple experiment with hobby rockets makes a good demonstration of crosswind impact on ascent trajectory. Following liftoff in a crosswind, the model rocket can be seen clearly turning into the wind, but the model rocket has no active control surfaces. How does the rocket make such a maneuver? How does it know?

This maneuver is caused by the aerodynamic forces from the crosswind on the rocket surfaces in the same way that a weather vane on the rooftop of a barn turns into the wind or a windsock at an airport turns into the wind. The phenomenon is called *weather cocking*. The aerodynamic forces, lift and drag, on the rocket increase with the square of the velocity of the rocket, as shown in Chapter 3, Equations 3.51 and 3.52. In a perfectly stagnant atmosphere, the path of the rocket would be a perfect vertical line. With a crosswind present, an overall pressure against the rocket's body is generated at the center-of-pressure point. Because our rocket was designed properly, the center of gravity will be above the center of pressure (see Chapter 3), and, therefore, the crosswind pressure will create a torque on the rocket body rotating it into the wind, as shown in Figure 9.1.

Also shown in Figure 9.1 is that a new flow path is generated around the rocket once it has completed the weather-cocking maneuver, and the torque on the rocket due to the crosswind becomes zero. At this point, the flight path of the rocket is inclined at an angle θ with the vertical and is calculated by

$$\tan(90° - \theta) = \frac{v}{w}, \tag{9.1}$$

where v is the velocity of the rocket and w is the velocity of the wind. With the uncontrolled rocket, weather cocking limits the maximum altitude the rocket can achieve, as shown in Figure 9.2. The lost altitude, Δy, is determined by

$$\Delta y = y_{max}(1 - \cos(\theta)), \tag{9.2}$$

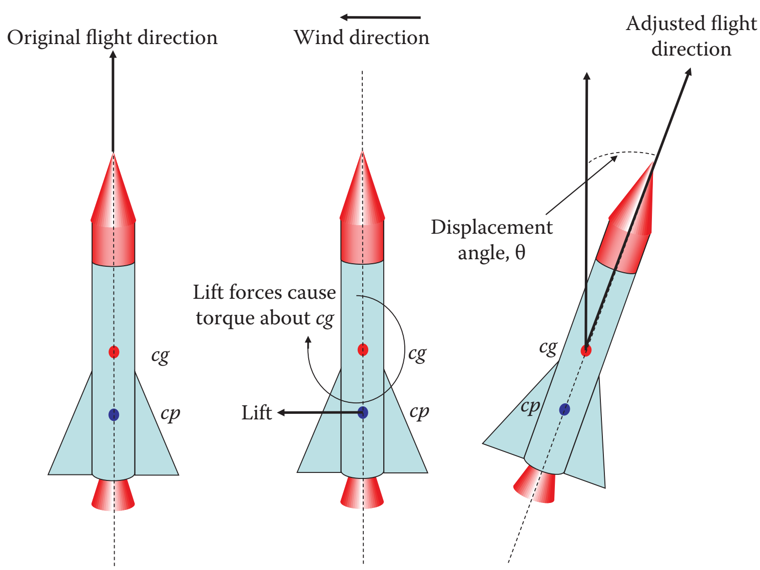

FIGURE 9.1
A rocket flying through a crosswind will experience a self-induced maneuver called weather cocking.

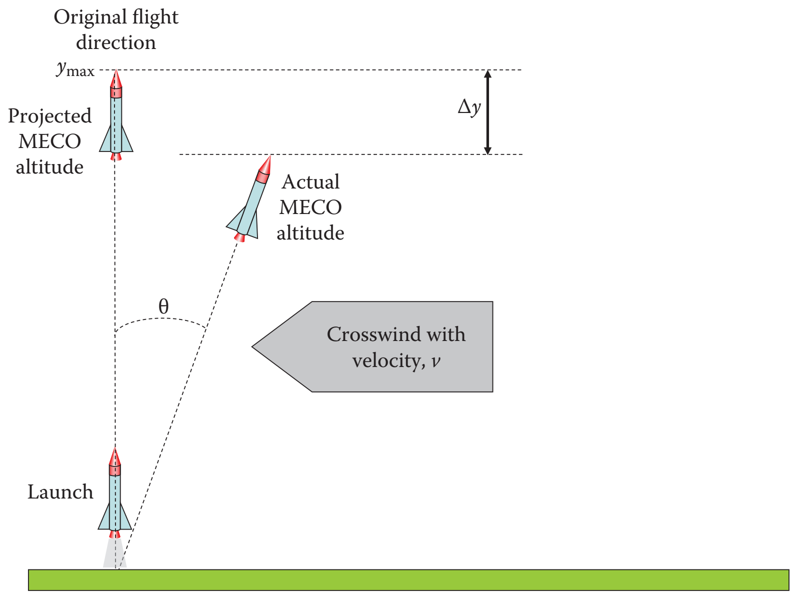

FIGURE 9.2
Weather cocking limits the maximum altitude the rocket can reach.

Thus, something as simple as a crosswind will limit the altitude of an uncontrolled rocket. For a controlled rocket, trade-offs must be made as to maximum altitude needed versus fuel and/or other energy spent to maintain vertical flight. If the rocket's application is not maximum height oriented, then the loss due to the crosswind might not be of any concern. But, how does the crosswind affect the overall trajectory?

The crosswind will force the rocket naturally into a trajectory that flies into the wind. If the rocket is a missile intended to deliver a payload to a target that is not directly upwind, then some expenditure of control will be required simply due to the weather-cocking phenomenon.

Using Equations 9.1 and 9.2, the max altitude loss as a function of the crosswind velocity can be found as

$$\Delta y = y_{max}\left(1 - \cos\left(90 - \arctan\left(\frac{v}{w}\right)\right)\right). \tag{9.3}$$

Figure 9.3 shows a graph of the loss in altitude versus the crosswind velocity for a hobby rocket with a velocity of 100 m/sec and a max altitude of 1,000 m. Note that crosswinds of up to 100 m/sec will decrease the max altitude by about 300 m. But, a 100-m/sec crosswind is present in extreme conditions, and the rocketeer should be at home in a storm shelter rather than out launching hobby rockets.

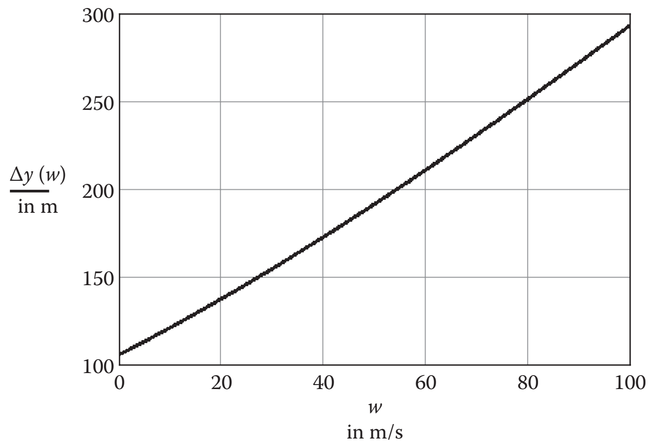

FIGURE 9.3
Weather-cocking altitude limitation as a function of wind speed.

9.2 Propellant Sloshing

Consider the liquid fuel rocket as shown in Figure 9.4. The rocket shown in the figure shows the propellant tanks during flight and partially full. Note that if the rocket is flying on a trajectory that is off the vertical axis, then the propellant will flow to one side. This is an obvious scenario and is as simple as tilting a glass partially filled with water and seeing that the liquid level remains level with the horizontal due to acceleration from gravity (and the rocket engines). With the propellant more to one side of the tanks, as shown in Figure 9.4, then the center of gravity for the rocket is shifted off the vertical axis through the rocket because the mass distribution is no longer symmetric about the rocket's axis.

The thrust force acting on the off-axis center of mass of the rocket will create a torque about the center of gravity (*cg*). This torque will force the rocket to rotate further increasing

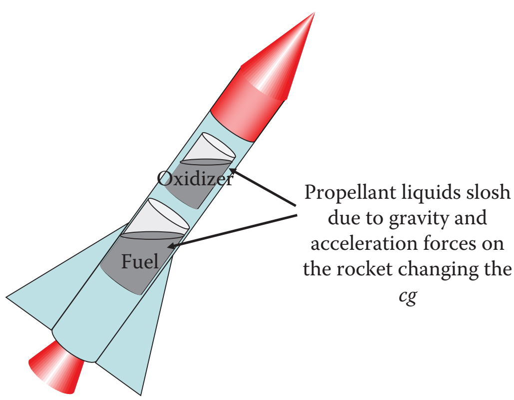

FIGURE 9.4
Propellant slosh makes the rocket mass distribution asymmetric and continuously changing as propellants are burned and sloshed.

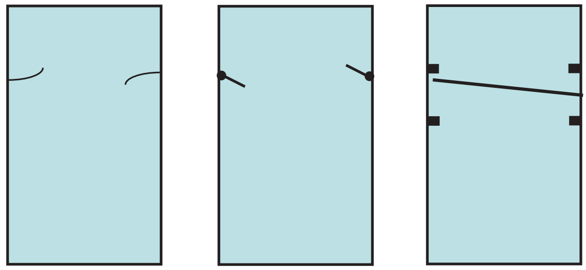

FIGURE 9.5
There are many types of slosh baffles. Shown here are three common types.

the off-axis *cg* problem as the liquid propellants gather to one side of the vehicle. A control action must be taken to force the rocket back to vertical orientation, but, when this happens, the liquid fuel will flow to the other side of the tanks as a reaction causing the vehicle *cg* to shift to the opposite side and off-axis, again. This problem is known as *propellant sloshing*. The sloshing of the propellant creates a significant wobble to the rocket that must be accounted for and corrected.

In Chapter 6, Figure 6.39 shows a schematic of the Space Shuttle External Tank (ET). Note that the liquid oxygen (LOX) tank at the top of the schematic shows slosh baffles. Slosh baffles are usually annular ring structures within a tank that suppress the sloshing of the fuel. Because they are rigidly attached to the internal structure of the tank, there must be a number of them at different height levels in order to suppress sloshing as the propellant level decreases as it is burned. The Saturn V rocket used many baffles with a total weight of several tons. Hence, rigid slosh baffles cause a severe mass penalty, but are a necessary evil because the effect of the sloshing on rocket stability is large. In fact, slosh was one of the problems that occurred in the second flight test of the Space-X Falcon 1 launch vehicle, which failed before it could complete its flight plan.

The Space Shuttle actually adjusts the thrust vector of the three Space Shuttle Main Engines (SSMEs) so that they all thrust through the *cg* as it changes due to propellant use. Any sloshing of the propellant within the ET that doesn't get baffled can be adjusted for by the SSMEs if need be.

Figure 9.5 gives some examples of other possible slosh baffle designs. These range from free-floating covers to elastic cantilever fingers. Slosh has been studied in great detail over the years of rocket development, but no perfect solution has yet been developed.

9.3 Propellant Vorticity

Just as sloshing of the fuel can cause undesirable effects on a rocket's stability, so can *vorticity*. Anybody who has ever flushed a bathroom toilet has seen vorticity, which is the swirling motion the water makes as it drains out of the bottom of the bowl. A liquid rocket tank is not entirely unlike a toilet bowl (or more precisely a bathtub, but the toilet analogy is funnier as amateur rocketeers have probably flown one at some point because they seem to enjoy strapping rocket engines to nearly anything). Like the tub, the propellant tank is a large reservoir filled with liquid, and it has a drain orifice in the bottom of it. When the drain is opened, the fluid flows down and out of the tank in a swirl with the drain at the center of the swirling motion. In the case of rocket tanks, pumps and positive pressure are sometimes used to suck the fluid propellants out of them at high speeds.

Consider the diagram shown in Figure 9.6 of a cylindrical propellant tank. As the fuel is pulled out of the bottom of the tank, a vortex is created. The details of why vorticity occurs are beyond the scope of this text, but suffice it to say that it does indeed occur. The liquid propellant begins to swirl with an angular velocity, $\omega_{propellant}$, about the cylindrical tank of radius, R_{tank}. The moment of inertia, I_{cyl}, of the fluid is

$$I_{cyl} = \frac{1}{2} m_{propellant} R_{tank}^2. \tag{9.4}$$

The kinetic energy of the swirling propellant is

$$K_{propellant} = \frac{1}{2} I_{cyl} \omega_{propellant}^2 = \frac{1}{4} m_{propellant} R_{tank}^2 \omega_{propellant}^2. \tag{9.5}$$

The propellant is inside the rocket, and, therefore, the rocket itself will react to the angular moment of inertia in Equation 9.4 by spinning in the opposite direction, as shown in Figure 9.7. (Note that this analysis neglects fluid friction with the inner tank surfaces.) The rocket body can be described as a hollow cylinder, which is a good approximation of it geometrically. Therefore, the rocket's moment of inertia is

$$I_{rocket} = m_{rocket} R_{rocket}^2. \tag{9.6}$$

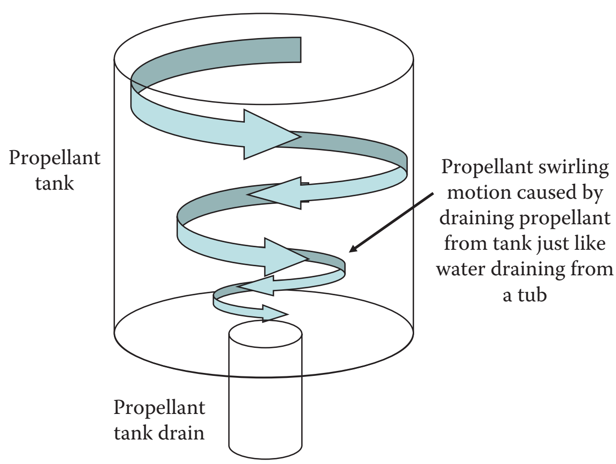

FIGURE 9.6
A swirling motion is induced into the propellant when it is drained or pumped into the rocket engine. The swirling motion is called vorticity.

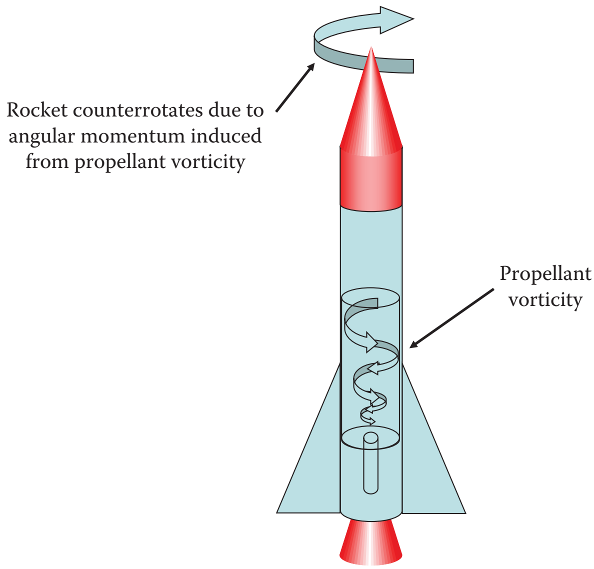

FIGURE 9.7
Vorticity in the propellant tank adds unwanted angular momentum to the rocket body.

Note that the ½ is dropped because we are considering the rocket to be a hollow cylinder. Applying the law of conservation of energy to the swirling propellant and the now-spinning rocket, we see that

$$K_{propellant} = K_{rocket} \tag{9.7}$$

$$\frac{1}{4} m_{propellant} R_{tank}^2 \omega_{propellant}^2 = \frac{1}{2} m_{rocket} R_{rocket}^2 \omega_{rocket}^2 . \tag{9.8}$$

Solving for the angular velocity of the rocket gives

$$\omega_{rocket} = \sqrt{\frac{1}{2} \frac{m_{propellant} R_{tank}^2 \omega_{propellant}^2}{m_{rocket} R_{rocket}^2}} . \tag{9.9}$$

The vorticity inside the propellant tank will induce upon the rocket vehicle a spin with the angular velocity given in Equation 9.9. What if we don't want our rocket to spin?

Even worse than spinning, there is a chance that precession could occur. Consider the rocket shown in Figure 9.8. While in the ascent phase of its trajectory, the rocket is in the Earth's gravitational field. The swirling propellant will induce a *gyroscopic precession* to the rocket, as shown in Figure 9.8. The precession angular velocity is calculated by

$$\omega_{precession} = \frac{m_{propellant} gh}{I_{propellant} \omega_{propellant}} = \frac{m_{propellant} gh}{\frac{1}{2} m_{propellant} R_{tank}^2 \omega_{propellant}} = \frac{2gh}{R_{tank}^2 \omega_{propellant}} , \tag{9.10}$$

where h is the distance between the center of the tank and the cg of the rocket.

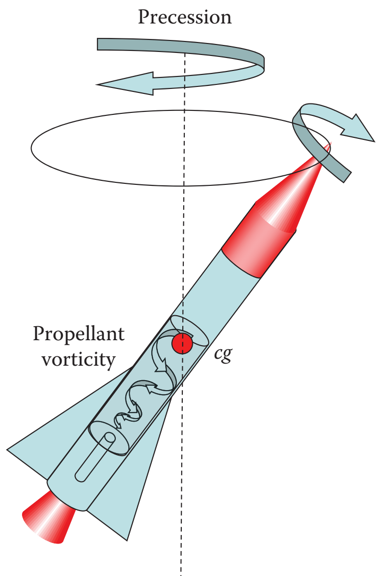

FIGURE 9.8
Vorticity in the propellant tank adds unwanted angular momentum to the rocket body and might cause precession.

Equation 9.10 shows us that the vorticity inside the propellant tanks can cause the rocket to precess about the vertical through the *cg*. If this precession is too large for the control system to dampen out, the rocket could be forced into an unstable tumble and fail catastrophically. Hence, we definitely don't want vorticity occurring in the tanks.

In order to prevent a vortex from forming around a drain, *vorticity baffles* can be installed. These baffles are generally constructed of grids of holes over the drain orifice. The size and spacing of the holes are calculated in complex fluid dynamics simulations of the propellant flow and are optimized to prevent a vortex from forming. Sometimes, multiple baffles must be implemented with a slight distance separation and horizontal offset between them to create more disruption to the fluid flow parameters.

Pogo and cavitation can also be the results of both sloshing and vorticity. Most modern liquid rockets implement baffles to prevent (or at least minimize) these two potentially devastating fluid flow phenomena. Again, there has been no perfect solution for the problem to date.

9.4 Tornadoes and Overpasses

At first glance, this section seems to be out of place in this book. However, it is a good test to see if we are thinking like rocket scientists and engineers yet. It is a very common misconception that taking shelter underneath an overpass while a tornado passes overhead is the safest place to be. This is, in fact, the absolute worst place to be, and a good rocket scientist or engineer should be able to tell us why.

Figure 9.9 shows a typical overpass configuration. The first thing that should literally jump off the page at the rocket scientist is that the area where the overpass meets the edge

FIGURE 9.9
A rocket scientist or engineer should realize why an overpass is NOT the place to be during a tornado!

of the span looks an awful lot like the converging end of a rocket nozzle. So, what happens when subsonic airflow (as with a tornado) is forced through a converging nozzle? We discussed in detail in Chapter 4 that the converging nozzle will accelerate subsonic flow. Thus, the overpass actually makes the air flow faster underneath it than outside and away from it. And, the apex of the overpass where it meets the edge of the span acts like a nozzle throat area where the acceleration is at a maximum. The apex area is most definitely not safe. In fact, anywhere underneath the overpass the flow will converge and accelerate and, therefore, is a very unsafe place to be during a tornado. A good rocket scientist or engineer should understand this and be able to explain it.

Don't feel bad if you didn't get this at first. While this book was being written, the tornado and overpass question was posed to a room full of NASA Marshall Space Flight Center rocket scientists and engineers ranging from fresh out of college to older, graybeard status, and not a single one of them could answer the question. The instant the answer was explained to them, there was a simultaneous smacking of foreheads and the remark, "duh."

9.5 Flying Foam Debris

What better time to discuss flying foam debris than following a brief discussion of tornadoes. There are all sorts of things that the rocket scientist and engineer must be able to analyze, and, in some cases, the analysis is following a mission failure. The Space Shuttle Columbia failed upon reentry on February 1, 2003, as a result of flying foam debris during launch ascent a few days earlier. We will roughly analyze this problem to understand how this happened.

The ET is covered with an insulation foam that, much like the Orbiter's heat tiles, has had a tendency to peel loose during launch. Unfortunately, and occasionally, this foam would fly off the upper parts of the ET and impact into the Orbiter. Why is this important? The insulation foam is, well, foam. How could foam damage a Space Shuttle Orbiter to the point that it would fall apart upon reentry? After all, it's just foam, right?

Figure 9.10 is a photo of the left bipod foam ramp of the ET. It is this piece of foam that is suspected of flying off the ET and hitting the left wing of the orbiter. The foam culprit is about 0.7-m long and 0.3-m wide. (These are approximations made from looking at the photo in Figure 7.10 and from other newscasts about the accident.) The Space Shuttle memorial Website states that this foam piece weighed about 2.5 lbs. in English units. That is about 1.13 kg. Also, the investigation shows that the impact occurred at 81 sec after launch (not long after max-Q—think about that). The velocity of the Space Shuttle at that point in the ascent trajectory was about 900 m/sec. At that altitude, the density of the air is around 0.3 kg/m^3. Using Equation 3.53 in Chapter 3, the force on the block of foam due to the atmospheric drag can be calculated. We should note here that the foam starts out at the same velocity as the Space Shuttle; thus, the relative velocity between them is zero when the foam flies free, but the airflow velocity relative to the free piece of foam would be 900 m/sec. If we use 0.8 as the drag coefficient (which is the coefficient for a cube, so it is a close approximation), we find the drag force on the block to be

$$D = C_D A \frac{\rho v^2}{2} = 0.8(0.7\text{ m})(0.3\text{ m})\frac{(0.3\text{ kg/m}^3)(900\text{ m/s})^2}{2} = 20{,}412\text{ N} \approx 20\text{ kN}. \tag{9.11}$$

The block fell approximately 15 m before it impacted the orbiter (again, an educated guess made from watching videos of the foam block falling). The work done on the block by the drag force is, therefore,

$$W = Ds = 20\ \text{kN}(15\ \text{m}) = 300\ \text{kJ}. \tag{9.12}$$

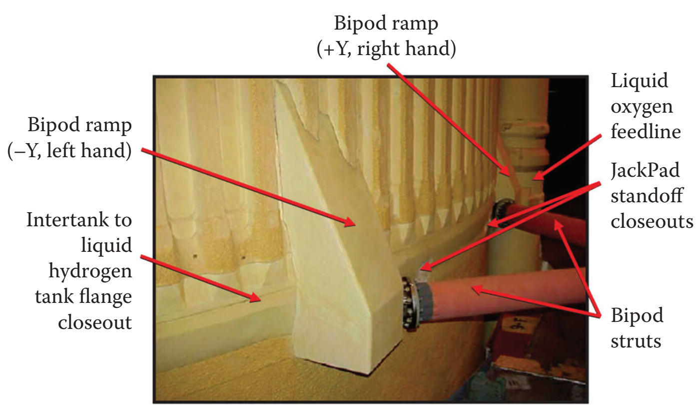

FIGURE 9.10
The left bipod foam ramp is suspected of being the piece of foam that impacted the Columbia Orbiter and subsequently causing the heat tile failure.

The velocity of the block after it traveled that far is found by equating the work done in Equation 9.12 with the kinetic energy of the block of foam just before impact and then solving for the velocity:

$$W = Ds = 20 \text{ kN}(15 \text{ m}) = 300 \text{ kJ} = \frac{1}{2}mv^2. \tag{9.13}$$

Thus, v is

$$v = \sqrt{\frac{2(300 \text{ kJ})}{\text{m}}} = \sqrt{\frac{2(300 \text{ kJ})}{1.13}} = 710 \text{ m/s}. \tag{9.14}$$

In other words, the foam collided with the Columbia's wing with a relative velocity of 710 m/sec and a kinetic energy of over 300 kJ. That, in relative terms, would be the kinetic energy of a 1,000-kg car traveling at about 25 m/sec. For those of you who haven't converted from English speeds yet, that is a sports car doing about 55 mph. The Shuttle did an amazing job withstanding the impact as long as it did, and it is incredible that it didn't fail on ascent. Even if the foam (because it was, well, foam) absorbed 99% of the impact (which is unlikely), more than 3 kJ would still have been imparted to the Orbiter's wing, which is still like being hit by the sports car at 8.8 km/h (or 5 mph) or a bowling ball at about 278 km/h (173 mph). The bowling ball analogy is more realistic because it is more the size of the foam block. Even if only 1% of the energy of the foam block was imparted to the Columbia, we see that it was a tremendous blow. When the Columbia reentered the Earth's atmosphere a few days later, the damage to the left wing where the foam hit was enough that superheated air was vented into the inner workings of the wing. The superhot plasma weakened the wing spar, and, eventually, it failed completely under the extreme aerodynamic loading of reentry. Once the wing collapsed, the Orbiter fell apart, killing the crew. As was mentioned in Chapter 6, flying rockets (and riding on top of them) are extremely dangerous.

9.6 Monocoque

Monocoque (French for "single shell") is a structure design technique where the structural integrity is supplied by the skin of the structure. A beer or soda can is an example of a monocoque structure. Because the structure of the can is nothing but the thin walls of the cylinder (its skin), the only structural integrity is supplied in these thin walls. A simple experiment of standing on a beer can will tell us some important information about how monocoque structures function. If the can is empty, it will collapse under much less loading than it will when it is full and unopened. The beer inside the can offers much more resistance to external pressure than unpressurized air in the empty can. So, why do rocket scientists care about this?

Two immediate examples are the SpaceX Falcon 1 shown in Figure 9.11 and the Atlas rocket shown in Figure 9.12. Each of these rockets use monocoque structures of thin metal skin pressurized with gas and propellants to make them very stable and rigid. When the rockets are unfueled and the tanks are empty, they cannot support the weight of the rocket structure and payload. In fact, Wernher von Braun often referred to the Atlas rocket as a "blimp" or

FIGURE 9.11
Falcon 1 uses monocoque propellant tanks.

FIGURE 9.12
The Atlas rockets used monocoque propellant tanks to the dismay of the German rocket scientists. (Courtesy of NASA.)

as the "inflated competition" to the Army Redstone rocket. But the "stainless-steel balloon" Atlas rocket performed well and was even used to put the first American into orbit. The monocoque design was used on the Atlas II and Atlas III launch vehicles' propellant tanks as well. The tanks were designed as so-called "balloon tanks" of very thin stainless steel and reduced the need for structural mechanisms and, therefore, weight, in the tankage.

The cleverness of the monocoque rocket and/or tankage designs is in the fact that far less material is needed for structural integrity. With less mass in the structure and more mass in the fuel, as we learned in Chapter 3, improves the mass fraction and, therefore, the performance of the rocket. The reason that the balloon tanks are no longer used on most rockets in the United States stems from a historical German fear of them. The German rocket scientists brought to this country after World War II never could come to grips with the concept. Even after they were allowed to beat on the pressurized tanks with a mallet and produced no damage to them, the Germans still had a prejudice against the balloon tanks. Though NASA and the U.S. Air Force no longer are using the monocoque rockets, other commercial groups, such as SpaceX, are.

9.7 Space Mission Analysis and Design Process

As the rocket scientists and engineers develop their skills, they will transition from just developing components and rocket systems to complete missions and applications of the rockets. This aspect of rocket science will follow a larger systems engineering approach, as shown in Chapter 6, but will carry the program through not just development but also to operation and even closeout, as discussed in the program life-cycle discussion. The overall mission design for a space mission requires detailed analysis and design iteration and is often referred to as the *space mission analysis and design* process. (Some also refer to it as the "smad" or SMAD process based on the fact that they likely learned the process from the textbook of the same name by Wertz and Larson.)

The SMAD process truly does follow a mixture of the NASA systems engineering process, the systems engineering (SE) engine, the V model, and the spiral development models we discussed in Chapter 6. It also includes elements of the program and project life-cycle models. These are just the programmatic components. Also included in the SMAD process is all of the rocket science and engineering calculations required to successfully define, develop, and implement a space mission. This may include developing a brand new rocket technology or pulling an already available rocket off the shelf for the mission. The basics of the SMAD process are the following:

- *Kick-off:* The actual point in time in which the program officially starts.
- *Space mission requirements definition:* The requirements for a successful mission completion are defined.
- *Identification of concept mission architecture candidates:* What candidate elements could be used to meet the mission requirements.
- *Characterization of concept candidates:* The individual candidates are considered in design reference missions (DRMs, sometimes the same as "case studies").
- *Concept candidate element definitions:* The individual candidates are defined at the subsystem level for the characterization study in the DRMs.

- *Trade candidates:* The larger number of concept candidates are downselected to a few leading competitors.
- *System requirements review:* Details of the system's design are developed to trade against the downselected candidates.
- *Process iteration:* Perform next level of analyses and/or experiments to scrutinize downselected candidates.
- *System definition review:* The final mission design is defined, and a final candidate set is chosen.
- *Process iteration:* The final system definition requirements are used to update the blueprints of the downselected system.
- *Preliminary design review (PDR):* Last chance for design changes of a medium scale; otherwise, program risk increases.
- *Process iteration:* Fix any problems in designs found at PDR and correct design.
- *Critical design review:* Last chance to make changes before hardware and software are constructed.
- *Production and deployment:* It is at this phase where the actual mission systems are manufactured and loaded into position for launch.
- *Mission operational:* Launch.

Figure 9.13 shows the flow of the SMAD process for a general space mission. Figure 9.14 is a typical organizational and work breakdown structure for a space mission. The

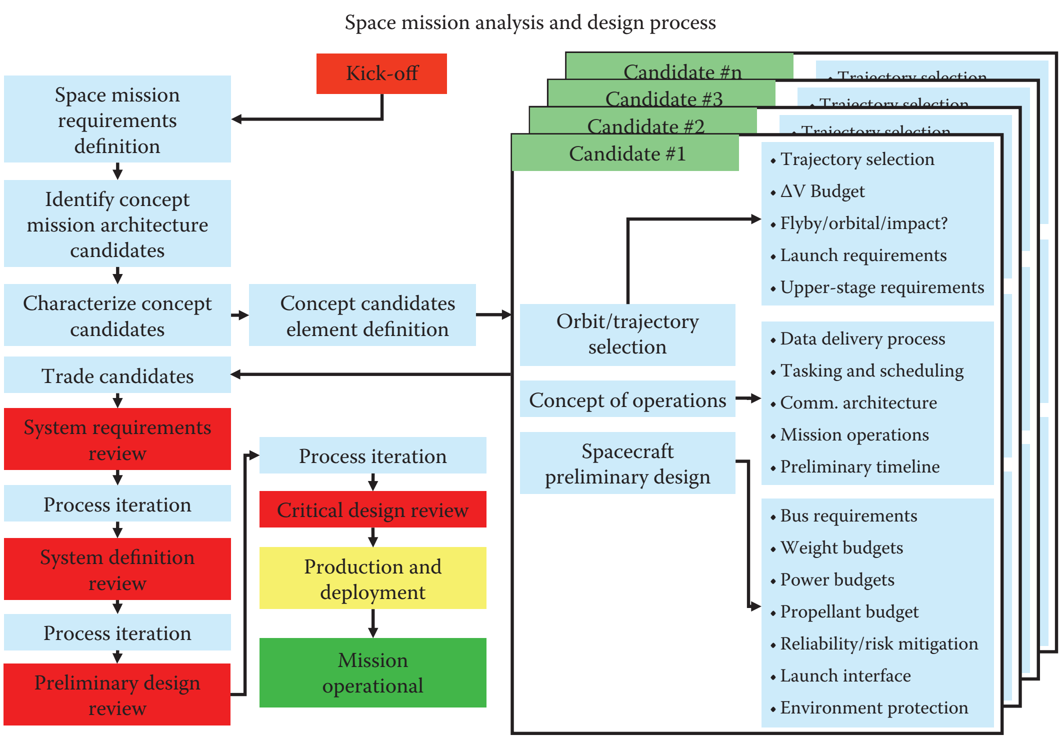

FIGURE 9.13
SMAD process.

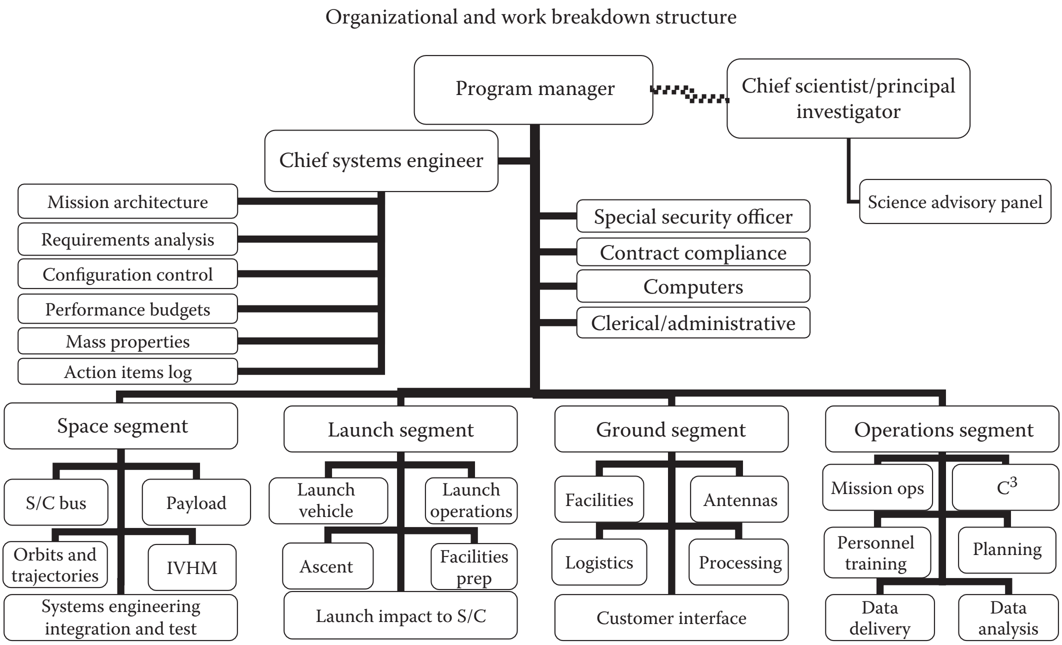

FIGURE 9.14
The WBS for a typical space mission or development effort.

process for mission design is quite complex with almost as many moving parts as a rocket itself. This is why some rocket scientists and/or engineers are required to become program managers of the rocket programs. They truly need to have been rocket scientists at some point in order to manage the rocket development or space mission program. Two very good examples of such rocket scientist/program managers were Sergei Korolev and von Braun. Both of these men dealt with day-to-day problems involving complex rocketry to mind-boggling program management to the constant fight with politicians and the public in order to maintain program funding.

9.8 "Back to the Moon"

As we discussed in Section 9.7, there are many aspects and talents needed in order to become a rocket scientist or engineer. Not all of these are purely technical training, mathematical understanding, and engineering prowess. Indeed, as was just shown, some of the talents needed to become a rocket scientist involve the skill to complete a research and development effort on budget and/or schedule, or to overcome major logistics hurdles as discussed in Chapter 6, Section 6.1, or to maintain program funding by educating the general public on the topic, and, of course, there is always politics. The rocket scientists and engineers who aspire to run BIG rocket programs like the Soviet Soyuz program or the Chinese Long March development or the NASA Apollo, Shuttle, Constellation Programs, and now the Space Launch System. A rocket scientist in a program, such as in the now-canceled Constellation Program where the Ares I and Ares V vehicles were being developed in order to send man back to the Moon, must learn how to not only be brilliant at the concepts of rocketry, but at the larger programmatic picture aspects of rocket science as well. And all of the elements of the program from nuts to votes are part of the overall holistic subject of rocket science and engineering. And, alas, the rocket scientist and engineer must be a dynamic proponent of the program or it will go the way of the dinosaur just like the Constellation Program. This is a perfect example of how not having a von Braun or Korolev driving an effort, it is likely to go away. We will discuss this further at the end of this next essay.

The following essay is an article that was first written after the announcement of the Constellation Program and was written by (an excited) Travis S. Taylor and published in the popular science fiction e-magazine, *Jim Baen's Universe*. Some of the vernacular, nomenclature, and acronyms might have changed since that time, but the general concept of the Constellation Program and how NASA was then planning to go back to the Moon are covered in the article. It also discusses in detail why the then new approach was a good idea and how it might save time and money based on previous program heritage. The essay is included here to offer the new rocket scientist or engineer some insight into the "big-picture" things with which they might have to deal one day. More discussion on the "big picture" will follow the essay.

Back to the Moon!

History Repeats Itself?

"As I take these last steps from the surface for some time in the future to come, I'd just like to record that America's challenge of today has forged man's destiny of tomorrow. And as we leave the Moon and Taurus–Littrow, we leave as we came, and, God willing, we shall return, with peace and hope for mankind." These are the words said by astronaut Gene Cernan (see Figure 9.15) the commander of Apollo 17 as he stepped from the Moon in preparation to return to Earth.

FIGURE 9.15
Gene Cernan was the last man on the Moon. (Courtesy of NASA.)

On December 14, 1972, astronauts Harrison Schmitt and Eugene Cernan climbed aboard their Lunar Excursion Module (LEM) and humanity left the Moon not to return for at least 40 years. Due to the Cold War, lingering aspects of the Viet Nam era, political, socio-economical, and public opinion issues the general public in America seemed to lose interest in any return to our closest celestial neighbor, the Moon. The three decades that followed the Apollo program saw a floundering and almost dying American space program. The days of "Better, Faster, Cheaper" removed the hope of mankind ever returning to altitudes much higher than Low Earth Orbit (LEO).

Atop the failing space program initiatives were also the failing NASA budget and the failing of its leadership. Poor leadership led to the horrible tragedies of both the Challenger and Columbia accidents. These tragedies all but devastated the already lackluster American space efforts.

But as Apollo 17 astronaut Harrison Schmitt is wont to say, "We do things in fits and starts." And that is exactly where humanity is today—at the beginning of an all new fit… an all new start.

On January 14, 2004, President George W. Bush said:

> Our … goal is to develop and test a new spacecraft, the Crew Exploration Vehicle, by 2008, and to conduct the first manned mission no later than 2014. The Crew Exploration Vehicle will be capable of ferrying astronauts and scientists to the Space Station after the shuttle is retired. But the main purpose of this spacecraft will be to carry astronauts beyond our orbit to other worlds. This will be the first spacecraft of its kind since the Apollo Command Module.

The following is a statement made by the newly appointed NASA Administrator Michael Griffin on the second anniversary of President Bush's announcement of the plan to return to the Moon, travel to Mars and destinations beyond—a Vision for Space Exploration.

> Two years ago this week, President Bush committed our nation to the Vision for Space Exploration. This Vision commits America to a journey of discovery and exploration with new and exciting plans to return astronauts to the moon. From there, to voyage to Mars and beyond, while continuing to engage in groundbreaking space science and pioneering advances in innovation, creativity and technology. Together with the partnerships we have in the International Space Station program, our nation has the tremendous opportunity and solemn responsibility to lead the way toward the dawn of a new space age.

There is a whole lot more history that took place over the next couple of years between contractors, internal NASA issues, and contractor selection. The project originally started under NASA Administrator [Sean] O'Keefe. He had his way of doing things—a way that was most apparently the status quo.

Issues began to arise with the contractors and the teams. One of which was that Burt Rutan's team was basically "run off" from the competition due to the "high paperwork burden" required. Burt Rutan and his Scaled Composites team had built the first commercial manned and reusable space vehicle, but NASA's approach somehow led to Rutan's team leaving the competition.

The final competition came down to the usual suspects, Lockheed Martin on one side and Northrop Grumman and Boeing on the other. Lockheed Martin's team basically tried to resell the dead penguin lifting body design that killed the X-33 program and Boeing's design was more like the old Apollo approach with some modifications.

These programs were to go through a spiral development approach following then NASA Administrator Sean O'Keefe's direction. O'Keefe put Rear Admiral (retired) Craig Steidle in charge of the development program. Steidle had used the spiral development effort—quite successfully—for the F-35 Joint Strike Fighter development program. However, that program was a Department of Defense large acquisition program that operates quite differently than the spacecraft development community is accustomed to. The spiral approach was beginning to bog down when the new NASA Administrator Griffin took over.

On June 28, 2005, Griffin made his distaste for the previous management approach quite clear to Congress.

> You asked, what will we be doing different? First of all, I hope never again to let the words spiral development cross my lips. That is an approach for large systems very relevant to DoD acquisition requirements, but I have not seen the relevance to NASA and I have preferred a much more direct approach, and that is what we will be recommending and implementing.
>
> … I hope that you will see … a straightforward plan to replace the shuttle and a very straightforward architecture for a lunar return, that, on the face of it, will seem to you that if we are to do these things, that the approach being recommended is a logical, clean, simple, straightforward approach.

So, we now have a new Presidential initiative to return to deeper space as we did for the Apollo era. And we have a new NASA Administrator that is fired up to make some changes to the old ways and to move forward—and back—to the Moon. Do we have a plan? How will we do it?

How We Will Make it Back to the Moon?

The new approach at NASA has been a complete change to the previous development approach. In the summer of 2004, Griffin, while at Johns Hopkins Applied Physics Laboratory before he was named O'Keefe's successor, participated in a study for NASA called *Extending Human Presence into the Solar System*. The study suggested three stages.

Stage 1: Develop the crew exploration vehicle (CEV), finish the International Space Station (ISS), retire the Shuttle Orbiter as soon as possible.
Stage 2: Develop an uprated CEV capable of multiple month-long manned missions, components required to enable human flight to the Moon and Mars, Lagrange points, and various near-Earth asteroids.
Stage 3: Develop human-rated planetary landers, such as the LEMs of the Apollo era.

The new program is called Project Constellation and President Bush's budget request in 2005 was for $428 million and $6.6 billion over the next five years. The budget request was for the development of the CEV and, in fact, was confirmed by Congress with the full amount of funding requested by the President.

So, what to do now? Well, NASA, under the new Administrator Griffin, set up a study to determine what would be the best way to really get started back to space. The *Exploration Systems Architecture Study*, affectionately referred to as "ESAS" in NASA-speak, was initiated. In large, the ESAS study derived similar conclusions as the study effort previously done by the *Extending Human Presence into the Solar System* effort.

The ESAS study has led to the development of some new space vehicles. These vehicles are known now as the Exploration Launch Vehicles. The Exploration Launch Vehicles Office has developed the scope of the development effort as such:

Crew Launch Vehicle (CLV, which is now the Ares I): A single five-segment reuseable solid rocket booster that is human-rated (RSRB/M) and has an upper stage that is powered by a single engine derived from the old Saturn V J-2 rocket engine.

Cargo Launch Vehicle (CaLV, which is now the Ares V): A system that has a core stage derived from the Space Shuttle External Tank with five Space Shuttle Main Engines (SSMEs) powering it (note that the engines now to be used will be the J-2X engines rather than SSMEs). Atop the core stage is a large cargo container. Also attached to the core are two of the five segment RSRB/Ms.

Earth Departure Stage: This component of the Exploration Vehicles scope is the upper stage that is attached to the CaLV and will be the all important system for getting out of Earth's orbit and to the Moon. The upper stage component uses tankage derived from the Space Shuttle's External Tank and is powered by a single J-2X engine.

The concept is actually brilliant from a paperwork and reinventing the wheel perspective. In order to put a human being on top of any spacecraft, a literal mountain of paperwork must be completed. Most of the paperwork involves proving that each individual component of the spacecraft down to the screws, nuts, and bolts have flown before and are of a quality that they have an extremely low risk of failure. A spacecraft of the CLV or CaLV stature will have as many as two million separate parts. If each of those parts has a handful of forms to be filled out, checked off, and so on, the paperwork nightmare becomes apparent.

But, what if there were a whole bunch of parts that have already had the paperwork completed on them? In that case, there would be no need to reinvent the wheel and fill out all that paperwork again. So, the ESAS group developed the brilliant Exploration Launch Vehicles plan.

The CLV is based on the SRBs flown with the shuttle and an upper stage engine flown in the Apollo program. The CaLV and Earth Departure Stage follow the same approach. But were there not problems with the shuttles that caused the Challenger and Columbia incidents?

Of course there were, but again this is really clever, those components are left out. The problems that caused the Challenger incident were due to the [solid rocket boosters] SRBs having thrust exhaust leaks around the segments of them. This hot exhaust heated up the External Tank and caused it to explode. That problem was due to the old SRB design and the operation protocols being violated. That problem was fixed long ago.

The Columbia accident was due to foam falling off the External Tank and damaging the Orbiter's heat shield tiles. That problem was solved by there no longer being an Orbiter and all of the crew and payload components are above the tankage. Therefore, nothing can fall off the tankage and damage the crew components. Oh, and by the way, the crew will be returned in a capsule and reenter just like the Apollo astronauts did except that they will land on land instead of water the way the Russians do it.

Brilliant!

Sounds a lot like the old Apollo doesn't it? Well, Apollo worked well and the SRBs in the shuttle program have worked well. So, the new plan is to take the best of both worlds and marry them together with modern computers, modern design and fabrication techniques, and new flight systems and avionics.

The Mission Profile

So, here is how a mission might go. The crew of three to six astronauts will climb aboard the crew exploration vehicle (CEV now the Orion capsule) atop the CLV. They will launch about the same time the unmanned CaLV is launched. Atop the CaLV in the cargo compartment is the Lunar Surface Access Module or LSAM, which is an updated version of the Apollo LEM.

The RSRB/Ms will fall back to Earth to be refurbished for future launches just as the SRBs do with the shuttle. The CLV upper stage will meet and dock with the CaLV upper stage, which contains the Earth Departure Stage and the LSAM. The docking will be much like the Agena module and the Gemini spacecraft docked or the same as the Apollo Command Service Module (CSM) and the LEM docked in LEO.

Now all mated together, the Earth Departure Stage fires its modernized J-2X engine. The thrust from the engine places the CEV and the LSAM into a translunar insertion trajectory and the Earth Departure Stage is then jettisoned.

As the CEV/LSAM approaches the Moon, a burn of the LSAM engine is made to put the spacecraft into a lunar orbit. This is called a lunar orbit insertion maneuver. Then the CEV and the LSAM separate just as the CEV and the LEM of the Apollo program did. The CEV will continue to orbit the Moon while the LSAM descends to a lunar landing.

At this point the LSAM is on the Moon. Whatever the lunar mission of the day is will be undertaken. Once the mission is completed, the crew will climb back into the LSAM and fire the Ascent Stage. The Ascent Stage portion of the LSAM lifts the crew back up to meet with the CEV. Once the CEV and the Ascent Stage dock the crew will leave the Ascent Stage. The CEV is then sealed up and the Ascent Stage is jettisoned.

The CEV then fires its engine in a transEarth injection maneuver. Once the CEV engine is used up, it is jettisoned leaving just the Crew capsule. The Crew capsule then reenters Earth's atmosphere directly and will land with parachutes at a predesignated land-based landing zone.

Mission completed and everything is A-OK!

How Does the New Spacecraft Compare to the Apollo?

The CLV is the smallest of the two new spacecraft systems. It will be about 309 feet tall with a total lift-off mass of 2 million pounds. It will be able to lift about 55 thousand pounds to LEO. Recall that this spacecraft will implement one five-segment RSRB/M with an upper stage that uses the modified J-2 engine (J-2X). The J-2X engine uses liquid oxygen and liquid hydrogen for oxidizer and fuel.

The CLV will stand 358 feet tall and will have a total lift-off mass of about 6,400,000 pounds. It can lift 121,000 pounds to a trans-lunar injection. This spacecraft uses two of the RSRB/Ms and five SSMEs for the core stage (now 5 J-2Xs) and a single J-2X engine for the upper stage.

The original Apollo spacecraft was the Saturn V. It stood 364 feet high and had a total lift-off mass of about 6,500,000 pounds. It consisted of three stages. The first stage consisted of five F-1 engines that ran off of liquid oxygen and rocket propellant. The second stage was five J-2 engines. The third stage was one J-2 engine.

When we consider the combination of the CLV and the CaLV spacecraft designs and compare them to the Apollo spacecraft, we can realize that the new system is indeed an upgrade and not simply a copy of the old ideas. The CLV/CaLV (Ares I/Ares V) combination will enable a larger payload to be delivered to the Moon. This means more crew and more science will be enabled.

There is another need for the two different spacecraft: the CLV and the CaLV. The CLV will be needed immediately to carry crew and small amounts of supplies to the International Space Station. The CLV will be the first system developed to flight readiness most likely.

The CaLV has a complete other use that most people have yet to realize. We no longer have any Titan rockets and, if the Space Shuttle is decommissioned, the United States will have lost its capability to place heavy payloads into Earth orbit. An example of these payloads might be the Hubble Space Telescope. Only the Space Shuttle or a Titan could lift such a payload to the proper orbit. If the shuttle is gone before the James Webb Space Telescope is completed, how do we expect to get the thing into orbit?

What about other national assets that are needed for defense purposes and intelligence gathering purposes? It is likely that those payloads are large as well. What about commercial, very large, relay systems like the Tracking and Data Relay Satellite System or TDRSS? How will we get next generation systems up without the Shuttles or Titans?

The CaLV can do it! We will not need the upper Earth Departure Stage. Instead of that part of the vehicle, we can place the heavy payloads. The CaLV might even offer us the capability to launch systems with payloads larger than Delta IVs and Atlas Vs can handle to higher orbits, such as geosynchronous ones.

So, in the near term as the shuttles are decommissioned we might have to take these new NASA spacecraft and implement them with a dual use. That is a good idea. That is one of the smarter things NASA could do or would have done in the last few decades. At this point, it is unclear if NASA has thought of this potential dual use of the Exploration Launch Vehicles. On the other hand, it is likely that the Air Force has. And, with Griffin's previous ties to DoD and the intelligence community, it is most likely that he has considered this as well. (In fact, NASA has since set up an office to study dual use capabilities of the Ares V rocket.)

So What Are the Long-Term Goals? Why Should We Go Back?

An overview of the program does not really reveal any hard technology problems. Most all of the technologies being considered for the Exploration Launch Vehicles are flight tested from heritage spacecraft, such as the Shuttles and the Apollo programs. The biggest hurdle appears to be maintaining enthusiasm for the mission. What do we do once we get to the Moon?

We are no longer in a Cold War era space race with the Soviets—although many would argue that we are in a Cold War-like space race with the Chinese—so getting there first cannot be our goal. NASA Administrator Griffin has created a team of high-ranking NASA officials to investigate our long-term Moon goals. Why are we going back?

Well, to start with, the Moon is a lot closer to Mars and is a good place to practice leaving Earth and going to another space body with manned systems. If we can't go back and forth between the Moon, how do we expect to go to Mars? It will be good practice and an excellent method of flight testing our concepts and technologies.

We have no idea what the Moon is all about. We have studied the Moon with probes and a few manned missions and from telescopes, but there is a lot about the Moon that we simply do not know. There are deep craters near the poles that have perpetual shadows over the floor and some of these have given confusing readings to various probes. Some of the probes have detected high levels of hydrogen and other substances that seem out of place. We simply do not fully understand what the Moon is, how it got there, and what we can do with it. We never knew there was gold in California until we got there and started digging around in the dirt. Perhaps the Moon will hold similar riches. Keep in mind that the riches will have to be large as to overcome the cost of the expedition through space to the Moon.

What about for other scientific purposes? The far side of the Moon is an ideal place for radio astronomy as there is no "noise" from terrestrial radio communications there. It would also offer a platform for other astronomical observation posts as the Moon has no atmosphere to interfere with the electromagnetic signals coming from outer space.

Finally, there should be a military outpost there. What? A military base on the Moon? Why not? Think of it this way. What if global diplomacy collapsed and China or Russia or any other country decided to destroy the United States of America's defense capabilities. If somehow all of our bases and military resources were wiped out, then we would be defenseless. But, if there was a contingent of forces on a base at the Moon, they would offer us a last resort. As with Heinlein's *The Moon is a Harsh Mistress,* we could implement a railgun on the Moon that could hurl projectiles to Earth, which would cause destruction of enemy targets far better than nuclear devices without the undesirable radiation fallout. Of course, there are some major technical hurdles for such a system, but it is feasible.

Also consider that same railgun system as a possible defense for asteroids, meteors, and comets that might be on an impact trajectory with Earth in the future. This could be a major reason for having a military base on the Moon. As it stands currently, we have no line of defense for such impacts.

And then there is the other big science fiction possibility—mathematically, it is a finite probability—that the Earth is invaded by aliens. Having our military in multiple locations might be useful in that situation. Having humanity spread out in multiple places wouldn't be a bad idea either.

Well, one thing for certain though, militarization of the Moon is a long way off. So, if you are one of those types that are opposed to such an idea, then don't panic. There is plenty of civilian exploration to be had on the Moon. There is plenty of science to discover and uncover on the Moon. Perhaps some smart entrepreneur will develop an economically viable business model for Moon missions. Maybe there will be a Club Med Tranquility Base in the not so distant future.

Whatever the outcome is the thing to remember is that there is a big, bright future for space exploration that starts on the Moon. And, if there are ideas that you have for reasons of going and staying on the Moon, by all means don't keep them to yourself. NASA is looking for great ideas and applications for space travel. What to do once we get to the Moon is such a question that Administrator Griffin had these words to pass along in an e-mail to his upper echelon advisors:

> The next step out is the Moon. We're going to get, and probably already are getting, the same criticisms as for ISS. This is the 'why go to the Moon?' theme.
>
> We've got the architecture in place and generally accepted. That's the 'interstate highway' analogy I've made. So now, we need to start talking about those exit ramps I've referred to. What ARE we going to do on the Moon? To what end? And with whom? I have ideas, of course. (I ALWAYS have ideas; it's a given.) But my ideas don't matter. Now is the time to start working with our own science community and with the Internationals to define the program of lunar activity that makes the most sense to the most people. I keep saying—because it's true—that it's not the trip that matters, it's the destination, and what we do there. We got to get started on this.
>
> … and the International Partners to get started down the track on pulling together an international coalition. They are annoyed and impatient with our delays since the Vision speech. We need to be, and be seen to be, proactive in seeking their involvement. We need to work with them, not prescribe to them, regarding what we can do together on the Moon.
>
> Beyond the Moon is Mars, robots first. Most of the Internationals are at present more interested in Mars, as I hear the gossip. Fine, we can't tell them what to be interested in. But our road to Mars goes through the Moon, and we should be able to enlist them to join on that path.
>
> Everyone … wants to be part of making Exploration what NASA does. It won't survive if all we worry about is getting there. That was the essential first step. But it has to sell itself on what it is that we DO there.

So When Are We Going?

As the program currently stands, NASA plans to be testing the systems for the CLV as early as this year. Design studies and reviews are to begin no later than 2008. Suborbital flight testing of the spacecraft is to begin sometime around 2009 to 2010. There are at least three so-called "risk reduction flights" scheduled between 2010 and 2012 [see Section 6.10]. The hopes are to have the CLV flight proven and ready for operation by 2012. This will allow decommissioning of the shuttles as the CLV will be able to transport crewmembers to the ISS.

The heavy launch vehicle, CaLV, will be developed parallel to the CLV. However, the flight readiness of the CLV seems to have priority status. The current NASA plan is to implement what Griffin refers to as the "Lunar Sooner" plan that will see flight testing of the CaLV sometime between 2013 to 2016 with flight readiness soon after. The "Lunar Sooner" plan optimistically has the CLV and CaLV ready for the first manned Moon mission by March of 2017! That is only 11 years away and is three years ahead of the original schedule suggested by President Bush. So just be patient, we are liable to make it back to the Moon within the lifetimes of the majority of people that are reading this article!

Travis S. Taylor

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9.9 A Perspective on the Big Picture, Rockets, and Dinosaurs

During the Apollo era, there was always a fear of budget being pulled and the program being cancelled. Having the Soviets in a race helped keep the program alive most assuredly; however, there was a lot of money being spent, and, at times, the American public wasn't sure why. Von Braun was clever enough to realize that public sway was extremely important. He then went to Walt Disney for help (see Figure 9.16).

Together, they devised short movies and cartoons to play on television and before other Disney movies that explained the space program and how important it was in a way that excited not just adults but also the children. Once the children were excited, the parents became excited, and parents vote. This was not a side effect, it was intentional and planned. The two visionary men used their holistic understanding of how things work to help keep

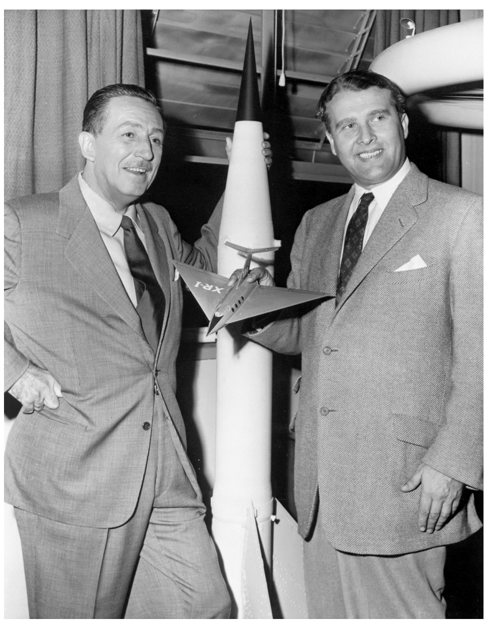

FIGURE 9.16
Walt Disney and Wernher von Braun. (Courtesy of NASA.)

one of the greatest endeavors of mankind alive. This is an aspect that has been lost on the space era for some time—pretty much since the deaths of the two men.

There are other visionaries and personalities doing what they can to keep the dream alive and to keep mankind building rockets and exploring, but it is a difficult task, and it is indeed one of the most difficult aspects of rocket science. Consider the fledgling commercial rocket programs (not SpaceX) such as Virgin Galactic, Bigelow Aerospace, and Xcor to name a few. SpaceShipOne was built on a budget of a few tens of millions of dollars. As we saw in Chapter 8, the reliability of a rocket is directly coupled to the number of times it has been tested. Rocket testing is extremely expensive, and, therefore, it is difficult for the small aerospace endeavors to fully test their rockets and determine if they are reliable enough. These efforts are truly risky, daring, and quite dangerous simply because of the lack of budget available to do more testing and better reliability assessments.

As we discussed in Chapter 1, SpaceX has at least had more budget for testing, which is why they have been more successful. Some might argue that Elon Musk is the new "big personality" for the modern space era, and, indeed, he has spent a considerable amount of his own money. But, as discussed in Chapter 1, SpaceX would still be much like the other fledgling aerospace companies, were it not for the U.S. government putting nearly $2 billion into the Falcon 9 development. Perhaps, Elon Musk learned from von Braun and Disney in a sense; however, he has captured political help, not the excitement of the general public—at least not in the way von Braun and Disney did.

We as rocket scientists and engineers simply cannot lose sight of the big picture because we never know until it is too late what the ramifications will be. A perfect example of this is the cancellation of the Space Shuttle program. The general public has very little comprehension as to how the cancellation of this program has impacted space exploration and science.

As we discussed in previous chapters, the Space Shuttle used liquid hydrogen and LOX as propellants. The tanks aboard the orbiter were quite large and held a significant volume. When the shuttle would carry crew and cargo up to the ISS it would dock there for some time. What most people don't realize is that the shuttle didn't need all of the remaining LOX in the tanks so it would "top off" the LOX tanks used for breathing on the ISS each time it would visit. Other rockets that are filling in for the shuttle now that it has been decommissioned do not have this capability. In other words, the ISS tanks do not get topped off with fresh oxygen each time a new crew or cargo arrives. So what was the "big-picture" impact? Fewer extravehicular activities (EVAs).

That's right. Each time astronauts conduct EVAs outside the ISS, some amount of oxygen is lost in the air locks and used up in the suits. During the shuttle era, there was plenty of oxygen on the ISS, and EVAs were not limited, at least due to lack of oxygen. Now, there are much fewer EVAs allowed because there simply is not enough oxygen available up there, and we don't have a big-enough rocket to take it up often enough. EVAs are vital to our repairing the ISS, conducting experiments, and learning how to work in space. Who knew that the cancellation of a rocket program would hinder us in such a way? Who was thinking of the big picture?

This brings us to the status of SLS. Who knows if it will go the way of the dinosaur as Constellation did? As political administrations come and go, so will these programs unless there are public excitement and drive to maintain them. The programs will continue to start and stop based on political change as opposed to technical change. Here is where the modern rocket scientist and engineer must not only learn from history and step up to do more than just the devilish details within the rocket calculations but also be mindful of the big picture and be the next von Braun or Korolev. Otherwise, the rocket programs will go the way of the dinosaurs. We all know what happened to the dinosaurs. And, we know why what happened to the dinosaurs happened to them. The dinosaurs didn't have any rocket scientists or engineers!

9.10 Chapter Summary

This chapter has been an overall mix of rocket science and/or engineering concepts ranging from the details of aerodynamics and fluid flow to the large-scale holistic view of large program management and public opinion improvement. All of these components are key pieces of rocket science and engineering. In some cases, the rocket scientist or engineer will desire to remain in the lab and develop new twists on an interesting piece of hardware for his or her entire career. Or, he or she might desire to develop new methods of modeling and simulating the esoteric components of rockets or orbits or thermal management or any of a thousand other things. Or, he or she might be interested in learning how to keep large development programs in motion and how to make certain that such Herculean efforts are successful through the implementation of systems engineering and program management techniques. A lot about rocket science is personality, and it all depends on what part of rocket science or engineering you are interested in. There are, for example, rocket scientists whose jobs are to be staffers to politicians and explain in layman's terms what the rocket scientists and engineers need from the community in order to do their jobs. And there are rocket scientists and engineers who ride in the Space Shuttle and who will go back to the Moon and to Mars. No matter what type of rocket scientist or engineer you will become, this book was meant as a starting line. The starting pistol has been fired, and it is now up to you to continue on your own race in whichever direction you desire. The important part is that you use the tools within this book to build on your *Introduction to Rocket Science and Engineering* and that you learn to think like rocket scientists and engineers.

Exercises

9.1 What is weather cocking?

9.2 If a rocket has a vertical velocity of 17 m/sec and the crosswind velocity is 1 m/sec, what is the weather-cocking angle?

9.3 In Exercise 9.2, what is the lost height due to the crosswind?

9.4 Why did the Saturn V rocket have several tons of baffles installed inside the propellant tanks?

9.5 Define slosh.

9.6 Write a computer code to model the precession of a rocket due to vorticity in the propellant tanks. Simulate various designs and graph the results.

9.7 In our discussion of vorticity, we neglected the friction of the propellant fluid with the inner tank surface. What effect would this friction have if not neglected?

9.8 We discussed tornadoes and overpasses. With that discussion in mind, why is it usually not a good idea to open both the front door and the balcony door of a beachfront condo at the same time?

9.9 How much energy would be imparted to a rocket if it had a head-on collision with a sea gull weighing about 0.5 kg? Assume the rocket velocity is 300 m/sec.

9.10 Why was the mass fraction of the Atlas rocket better than its competition at the time?

9.11 What does monocoque mean?

9.12 Discuss the systems engineering processes in Chapter 6 in comparison with the SMAD process. Be sure to explain why and where the SE processes fit in the SMAD and vice versa.

9.13 Why is using heritage designs from the Apollo- and Space Shuttle– era programs a good idea for the Constellation program? (*Hint:* paperwork)

9.14 How does the Ares rocket design immediately improve safety by reducing the falling foam risk of the Shuttle if it still plans to implement ET-like components?

9.15 Discuss the overall holistic subject matter and talent pool required for the rocket scientist and/or the engineer.

9.16 Name at least one impact on other programs and/or technologies cancellation of the Space Shuttle program had.

9.17 Why are dinosaurs extinct?

9.18 MOST DIFFICULT PROBLEM IN THE BOOK: Use all the information in this book and at https://en.wikipedia.org/wiki/Falcon_9_full_thrust to create a calculation and simulation of the Falcon 9 Full Thrust. Determine the fineness of the rocket, as well as an estimate on the critical pressure. Make assumptions where information is not available. Also, complete a PRA for the rocket at the block diagram level and thus creating an RBD. Determine, using techniques like in the Mathcad code in Chapter 7, how the loss of engines will impact the delivery of maximum payload to orbit from 1 to 9.

Suggested Reading for Rocket Scientists and Engineers

There are plenty of books out there about rocketry, rocket science, and rocket engineering, but there are few of them that read like an introductory text. Most books on rocket science and engineering concepts are reference style and assume the readers already have a working knowledge of the field. The following is a list of books (in no particular order) that should be within reach for any practicing rocket scientist or engineer. The list is by no means exhaustive but is a good starting point following the information in this book. And, as always, if there is a bit of information that can't be found in these books, the best place to start is at http://www.google.com. The Internet has a vast source of data on rocketry out there. The trick, though, is finding, compiling, and absorbing them all in a useful manner.

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