# EXPLORING IN AEROSPACE ROCKETRY

## 11. LAUNCH VEHICLES

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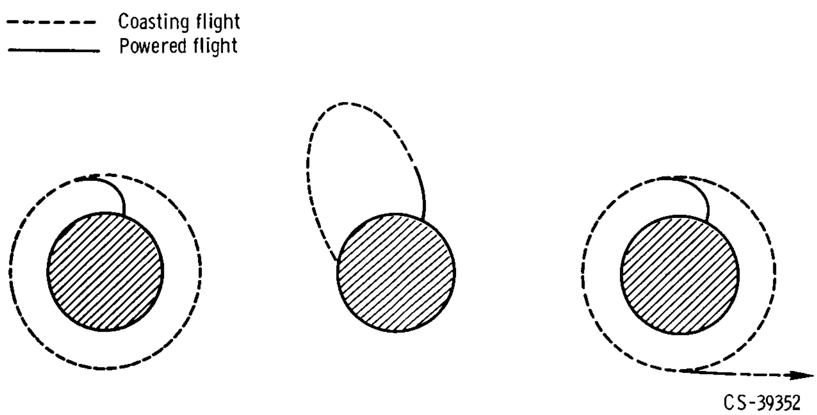
## INTRODUCTION

Investigation or exploration of space involves placing an instrument package or astronauts and their life support and return capsule into space. Placing these payloads into space is the job of the launch vehicle. Although this chapter discusses only the problems and characteristics of launch vehicles for placing a payload into an orbit about the Earth (fig. 11-1(a)), there are two other general classes of launch vehicle missions: sounding probes, and missions beyond the Earth to other bodies or regions of the Solar System. Sounding probes (fig. 11-1(b)) are generally lofted by relatively small launch vehicles (usually multistage solid rockets) to a high altitude above the Earth. Here, the space data are obtained quickly and the probe falls directly back to Earth. The other class of




(a) Earth orbit. (b) Sounding probes. (c) Beyond Earth.

Figure 11-1. - Launch vehicle missions.

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missions, that is missions beyond the Earth (fig. 11-1(c)), is really an extension of Earth orbit missions. The first step in going beyond the Earth is usually to place the payload and one or more propulsion stages into an Earth orbit. Then, this assembly coasts in Earth orbit until the proper position in space is achieved and the remaining stage or stages of the launch vehicle are fired. This firing accelerates the payload to the proper velocity and direction for ultimately reaching the target body.

Later, we will describe the NASA family of launch vehicles and give facts about the main vehicles. However, an appreciation of specific features of these launch vehicles requires an understanding of their general characteristics and some of the factors that determine their performance.

## TYPICAL TWO-STAGE LAUNCH VEHICLE

Most launch vehicles designed to establish an orbit around the Earth have more than one stage, usually two. The main reason for this is that an immense fuel weight is required to launch a payload into orbit, and the fuel containers or tanks are a large part of the hardware weight of a launch vehicle. Late in the flight, most of these tanks are empty and represent dead weight that has to be carried along. To be efficient, as the vehicle flies into orbit, it throws away or jettisons stages consisting of empty tanks and other no longer useful weight. A mathematical explanation of this will be presented later.

A sketch of a two-stage launch vehicle is shown in figure 11-2. Note that each stage is basically a complete vehicle in itself. Each stage has an engine system, fuel tank, and oxidant tank, all united by a structure. An interstage adapter is used to connect the second stage to the first. When the propellants of the first stage are consumed, the second stage is released from the forward end of the interstage adapter, and the second stage engine is started. The second stage and payload continue to accelerate to orbit, while the empty first stage falls back to Earth. The instrument compartment contains all of the electronic systems of the launch vehicle. This includes such things as the guidance and control systems, tracking and telemetry systems, electrical systems, batteries, etc. In figure 11-2, all these systems are neatly packaged into an instrument compartment. In practice, some or all of these systems are often scattered throughout the vehicle, alongside the tanks, on top of the tanks, between tanks, etc.

The payload is attached to the launch vehicle through a structure called a payload adapter. Upon reaching orbit, the payload is usually released from the payload adapter and separated from the launch vehicle. Since the payload usually consists of relatively delicate instruments and equipment, it must be protected from aerodynamic heating and

TABLE 11-I. - SYSTEM WEIGHTS FOR A TYPICAL LIQUID PROPELLANT SECOND STAGE

$$\left[\frac{\text{Hardware weight}}{\text{Propellant weight}} = \frac{4100}{30\ 000} = 0.1366.\right]$$

| System | Weight, lb |
|---|---|
| Structure and tankage | 1 000 |
| Propulsion and plumbing | 1 250 |
| Guidance | 350 |
| Control | 150 |
| Pressurization | 200 |
| Electrical | 250 |
| Flight instrumentation | 250 |
| Payload adapter | 150 |
| Residuals | 500 |
| Total hardware | 4 100 |
| Usable propellant | 30 000 |
| Total stage | 34 100 |

sheets. The sheets are formed and welded into cylindrical sections. The cylindrical sections of the vehicle are often strengthened by using a series of circumferential rings and longitudinal stringers.

Propulsion and plumbing. - Both liquid propellant and solid propellant rocket systems have been discussed in previous chapters, and most of the discussion here assumes the use of liquid propellant rockets. The most common liquid fuels used currently in NASA vehicles are RP-1 (essentially kerosene), liquid hydrogen ($LH_2$), and unsymmetrical dimethyl hydrazine (UDMH, a derivative of hydrazine $N_2H_4$). The most common oxidizers are liquid oxygen (LOX), used with RP-1 and hydrogen, and nitrogen tetroxide ($N_2O_4$) and inhibited red fuming nitric acid (IRFNA), both used with UDMH.

Guidance. - The purpose of the guidance system is to keep track of the vehicle position and velocity throughout the flight and to command the maneuvers required to reach the desired target or burnout conditions. In radio guidance, most of the tracking and computations are done on the ground, and the maneuvers are commanded through a radio link with the moving vehicle. In a full inertial system, all the position and velocity determinations and computations are done on board the vehicle itself. The guidance system consists of accelerometers to determine vehicle acceleration, gyroscopes to determine

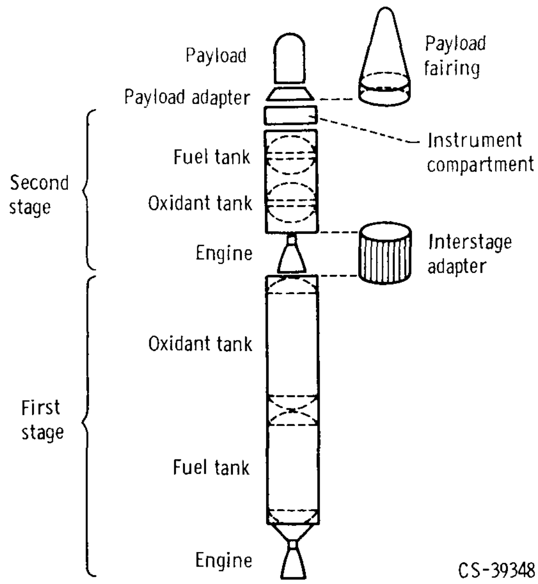


Figure 11-2. - Typical two-stage launch vehicle.

loads during the high velocity flight through the atmosphere. This is done by covering the payload with a large conical payload fairing. The payload fairing usually encloses the entire payload and is attached to the forward end of the second stage or instrument compartment. After the vehicle passes through the Earth's atmosphere on its way to orbit, the payload fairing is no longer required, and often, then, it is jettisoned by splitting it in two halves and allowing it to fall away while the vehicle continues to accelerate to orbit.

## Launch Vehicle Systems

Many of the systems in a two-stage launch vehicle are the subjects of other chapters in this book, and so they will only be discussed briefly here.

The weights of the major systems of the second stage of a typical launch vehicle are listed in table 11-I. This stage uses liquid propellants. Notice that the total empty or jettison weight for the stage shown is 4100 pounds. Since the stage has a propellant capacity of 30 000 pounds, the ratio of hardware weight to propellant weight is 0.1366. For high performance, stages must have as low a hardware weight as possible. Typical stage hardware weights range from 10 to 20 percent of the stage propellant weight.

Structure and tankage. - For many stages the propellant tanks also serve as part of the stage structure, and their respective weights cannot be readily separated. The stage structure and tanks are commonly fabricated from thin aluminum or stainless steel the vehicle orientation in space, and an on-board computer to perform the necessary calculations. Guidance and control systems will be the subject of chapter 12.

Control. - The control system is the on-board equipment that actually maneuvers and stabilizes the vehicle in response to the commands given by the guidance system. In many cases the attitude of the vehicle is controlled during main engine firing by gimballing the engine. During coasting, when the main engine is not firing, the control system consists of a series of small, low thrust rockets which are turned on and off to maintain and stabilize the vehicle attitude.

Pressurization. - The pressurization system provides pressurizing gas to the propellant tanks. This is required to force the propellants to the engine pumps or combustion chamber. Helium, a common pressurizing gas, is stored under high pressure in small, separate tanks on board the vehicle. During flight, a series of valves, regulators, and pipes are used to properly meter the gas to the propellant tanks.

Flight instrumentation. - The flight instrumentation system consists of on-board vehicle and engine instrumentation, radio-telemetry systems, range safety systems, and tracking systems. These on-board systems transmit vehicle data back to ground stations for range safety, tracking, and systems performance evaluation purposes.

Electrical. - The electrical system provides electrical power to the on-board guidance, control, and flight instrumentation systems. It consists of batteries, power conditioning equipment, wiring harnesses, etc.

Residuals. - The residuals consist of trapped liquid propellants and gases remaining in the tanks and feed lines after the main propellants have been consumed.

Payload adapter. - The payload adapter is the structure that unites the payload and the launch vehicle. Its weight and configuration depend, of course, on the size and shape of the payload.

## Launch Sites

Extensive ground facilities are required to prepare and launch a multistage vehicle to orbit. The Eastern Test Range (ETR) is used for launches that are predominantly eastward, and the launch sites are located at Cape Kennedy, Florida. The eastward launches are desirable, when feasible, since they take advantage of the rotation of the Earth to add velocity to that generated by the vehicle. The Earth's velocity at the latitude of Cape Kennedy is approximately 1350 feet per second eastward. The Western Test Range (WTR), with launch sites located near Vandenberg Air Force Base, California, are used for westerly and southerly launches. Southerly launches are desired for obtaining polar or near-polar Earth orbits.

The direction of launches from both ETR and WTR are limited by range safety con-

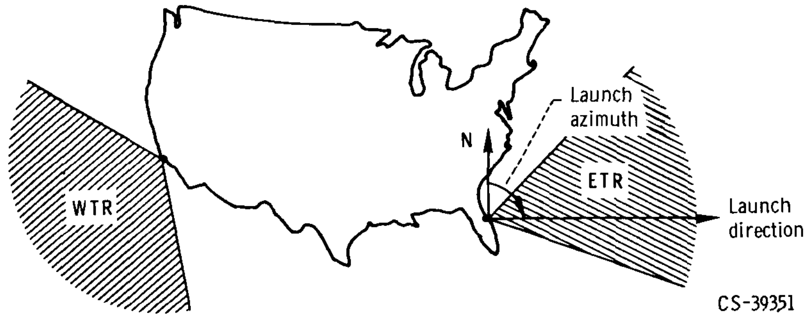

Figure 11-3. - Typical launch site restrictions.

siderations. That is, the vehicles are not generally allowed to fly over populated land areas. The direction of the launch is given by an angle called the launch azimuth. The azimuth angle is the angle measured clockwise from the geographical north around to the direction of launch (fig. 11-3). Launches from the ETR are generally limited to azimuth angles between 45° and 110° and launches at WTR from 170° to 300°.

## Launch Vehicle Performance

The computation of a launch vehicle trajectory and performance is a complicated procedure requiring the use of large electronic computers to obtain accurate solutions. However, some simplifying assumptions allow an approximate answer to be easily obtained. Assume that the launch vehicle will fly a 100-mile circular orbit. Actually, a wide variety of Earth orbits are required to accomplish the various NASA missions. However, almost all of these missions require minimum orbit altitudes near or above 100 miles. Below 100 miles, the Earth's atmosphere, although very thin, is sufficiently dense that, in combination with the high orbital velocities, it exerts a measurable drag on the payload. This may result in undesirable payload heating and, also, rapid decay of the orbit back to Earth. The problem, then, is to accelerate the vehicle from zero velocity and altitude at the launch site to orbital velocity at 100 miles. Assume that the Earth is not turning; this permits the initial velocity due to the Earth's rotation to be neglected and simplifies the calculations.

A sketch of a typical launch trajectory is shown in figure 11-4. Notice that the vehicle launches vertically and gradually turns over as it accelerates. For a circular final orbit, the orientation of the vehicle at burnout must be horizontal, or parallel to the Earth's surface. The required orbital velocity was determined in chapter 9 to be about 26 000 feet per second. Recall that in circular orbital flight the vehicle is in balance

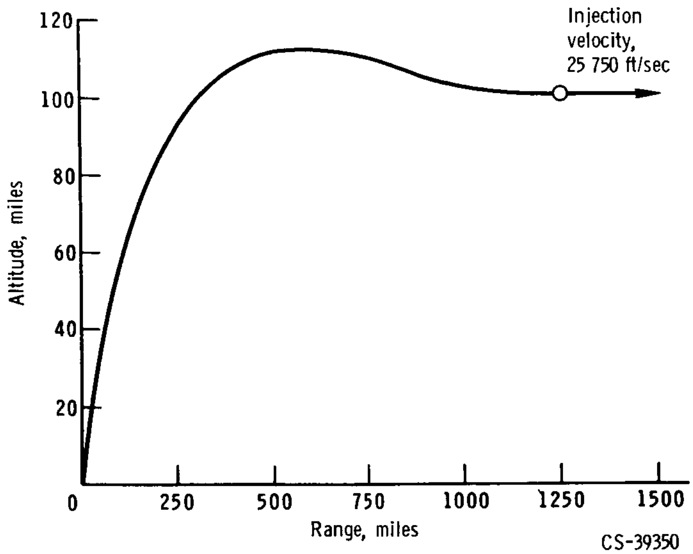


Figure 11-4. - Typical launch trajectory.

between the outward centrifugal force and the inward pull of gravity. The centrifugal force is given by

$$F_c = \frac{mv_c^2}{r_o} \tag{1}$$

and this is equal to the pull of gravity $mg_o$, so that

$$\frac{mv_c^2}{r_o} = mg_o \tag{2}$$

(All symbols are defined in the appendix.) The velocity in a circular orbit above the Earth $v_c$ is then found by rearranging equation (2) to give

$$v_c = \sqrt{g_o r_o} \tag{3}$$

The acceleration due to gravity at the surface of the Earth $g_e$ is 32.2 feet per second per second. Actually, $g_o$ gets smaller away from Earth (inversely proportional to the radius squared), and the $g_o$ at altitude is given by

$$g_o = g_e \frac{r_e^2}{r_o^2} \tag{4}$$

Substituting this into equation (3) gives

$$v_c = \sqrt{\frac{g_e(r_e)^2}{r_o}} \tag{5}$$

Using 4000 miles as the radius of the Earth and introducing the proper numbers into equation (5) give the circular velocity in a 100-mile orbit as

$$v_c = \sqrt{32.2\ \frac{4000^2}{4100}\ \frac{5280^2}{5280}} \approx 25\ 750\ \text{ft/sec}$$

The problem, then, is to accelerate the vehicle from zero velocity at launch to a horizontal burnout velocity of 25 750 feet per second at an altitude of 100 miles.

The performance of the launch vehicle will be computed by using the ideal or basic rocket equation discussed in chapter 2. This equation gives the burnout velocity of a vehicle as

$$v_b = g_e I_{sp} \ln \frac{W_i}{W_f} \tag{6}$$

It is called the ideal equation because it gives the maximum velocity that a vehicle can achieve flying in a vacuum in gravity-free space. It does not account for losses such as gravity losses and aerodynamic drag losses which will be discussed later. Nonetheless, equation (6) is of great use in determining launch vehicle performance, and it bears some detailed discussion. First, $g_e$ is the standard value of 32.2 feet per second per second. The specific impulse, $I_{sp}$, is a measure of the performance of the rocket engines as defined and discussed in previous chapters. The specific impulse of engines used today on NASA launch vehicles range from a little over 200 seconds to as high as 440 seconds, depending on the propellants used. The initial weight of the vehicle is $W_i$; the final weight is $W_f$. The weight of the propellant used can be determined from

$$W_p = W_i - W_f \tag{7}$$

Given the initial weight of a launch vehicle, the amount of propellant on board, and the specific impulse of its engines, the vehicle's burnout velocity can be determined by using equations (6) and (7). Conversely, if given a required burnout velocity, the final weight can be determined. These computations, however, require the logarithm of the initial- to

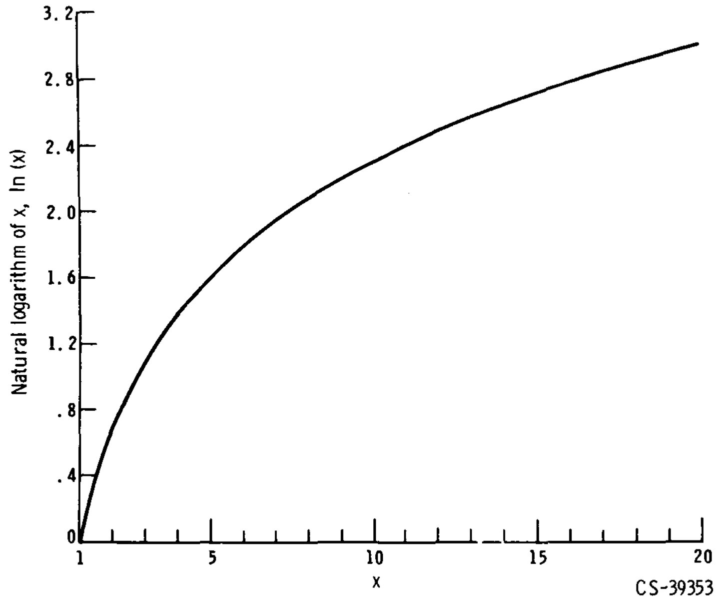

Figure 11-5. - Natural logarithm curve.

final-weight ratio. The logarithm used here is the natural logarithm (to the base e, where e = 2.7183), and the relation between a number and its natural logarithm is shown in figure 11-5. As an example, assume a launch vehicle whose burnout weight is one-fifth its initial weight. Then, $W_i/W_f = 5$, and from figure 11-5 the natural logarithm of 5 is 1.61. If the specific impulse of the vehicle engine is 350 seconds, the vehicle burnout velocity from equation (6) is

$$v_b = (32.2)(350) \ln 5 = (32.2)(350)(1.61) = 18\,150 \text{ ft/sec}$$

Before equation (6) will apply to a vehicle flying to orbit, the losses encountered in flying a real trajectory must be considered. There are three fundamental losses. First, some of the thrust of the engines will be lost in overcoming the aerodynamic drag imposed on the vehicle in flying through the atmosphere. This was discussed in chapter 9. Secondly, not only must the vehicle obtain a horizontal velocity of 25 750 feet per second, but it must also increase altitude from 0 to 100 nautical miles; this means that early in the flight (see fig. 11-4) all thrust is directed upward to gain altitude, and this thrust does not contribute directly into acquiring horizontal velocity. Finally, there are losses due to the gravitational pull of the Earth which are referred to as gravity losses. These exist because part of the engine thrust is used to overcome the Earth's gravitational pull on the vehicle, and only part of the thrust is then left to accelerate the vehicle. Consider the example shown in figure 11-6 which indicates the status of a vehicle that has

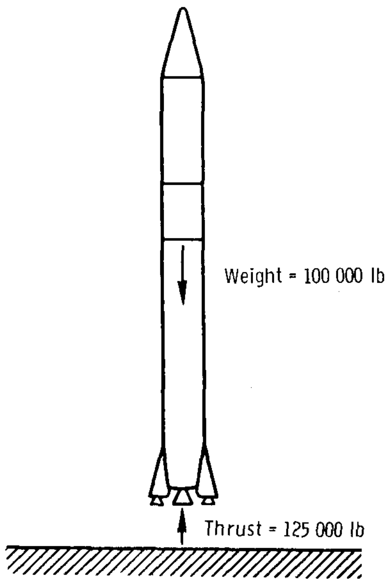

Figure 11-6. - Gravity loss early in the trajectory.

just left the launch pad. This vehicle weighs 100 000 pounds and has a thrust of 125 000 pounds. This initial thrust-to-weight ratio of 1.25 is typical of liquid propellant vehicles we are flying today. Notice that 100 000 pounds of thrust is used to support the vehicle (against the Earth's gravitational pull), and only 25 000 pounds is left to accelerate the vehicle. Thus, initially, 80 percent of the thrust is lost in overcoming gravity. Fortunately, this loss decreases rapidly as the vehicle continues along the trajectory since it is getting lighter as it consumes propellants. Also, as the trajectory begins to curve over (fig. 11-4), part of the gravitational pull of the Earth is counterbalanced by the centrifugal force of the vehicle. Indeed, when it reaches orbit, the entire gravitational pull of the Earth is balanced by the centrifugal force.

Exact determination of the three losses is a complicated calculation requiring solution on an electronic computer. Experience indicates, however, that for a typical launch to orbit, these losses total about 4000 feet per second. Thus, the hypothetical launch vehicle whose performance is being calculated must have an ideal velocity capability of 30 000 feet per second: 25 750 feet per second to acquire orbital velocity and 4250 feet per second to account for the losses in an actual trajectory.

Now, the performance of a launch vehicle to orbit can be finally calculated. Assume that the initial weight of the vehicle is 110 000 pounds, its specific impulse is 390 seconds, and its hardware weight is equal to 10 percent of the propellant weight. Equation (6) will now appear as

$$30\ 000 = (32.2)(390) \ln \frac{W_i}{W_f}$$

or

$$\ln \frac{W_i}{W_f} = \frac{30\ 000}{(32.2)(390)} = 2.40$$

Using figure 11-5 to evaluate the logarithmic function gives

$$\frac{W_i}{W_f} = 11.0$$

and for the initial weight of 110 000 pounds,

$$W_f = \frac{110\ 000}{11.0} = 10\ 000 \text{ lb}$$

From equation (7), the propellant weight now becomes

$$W_p = 110\ 000 - 10\ 000 = 100\ 000 \text{ lb}$$

Thus, the vehicle has a burnout weight of 10 000 pounds in orbit and used 100 000 pounds of propellant getting there. To obtain payload we need to subtract the hardware or jettison weight of the stage from the burnout weight. Since the hardware weight was assumed to be 10 percent of the propellant weight, the hardware weight is 10 000 pounds which when subtracted from the burnout weight leaves no weight for payload. This example demonstrates that it is very difficult to deliver payloads to Earth orbit with a single-stage vehicle. In practice, then, most payloads are delivered to orbit by using more than one stage. To demonstrate the advantage of staging we will repeat the problem using two stages to reach orbit. The payload and burnout velocity of the first stage are the initial weight and velocity of the second stage. Again, the total vehicle weight is taken as 110 000 pounds, and each stage has a specific impulse of 390 seconds and a hardware percentage equal to 10 percent of the propellant weight. Finally, we assume that the 30 000-foot-per-second ideal velocity is divided equally between the two stages, that is, 15 000 feet per second each. For the first stage (using eq. (6)),

$$15\ 000 = (32.2)(390) \ln \frac{W_i}{W_f}$$

or

$$\ln \frac{W_i}{W_f} = \frac{15\ 000}{(32.2)(390)} = 1.194$$

Using figure 11-5 gives

$$\frac{W_i}{W_f} = 3.30$$

and the burnout weight of the first stage is

$$W_f = \frac{110\ 000}{3.30} = 33\ 300 \text{ lb}$$

The propellant weight is

$$W_p = 110\ 000 - 33\ 300 = 76\ 700 \text{ lb}$$

and thus the hardware or jettison weight of the first stage is

$$W_{hw} = (0.10)(76\ 700) = 7670 \text{ lb}$$

Subtracting this from the burnout weight of the first stage gives a payload equal to 33 300 - 7670 or 25 630 pounds. The initial weight of the second stage then is 25 630 pounds, and the second stage has to provide another 15 000 feet per second to reach orbit. For the second stage (using eq. (6)),

$$15\ 000 = (32.2)(390) \ln \frac{W_i}{W_f}$$

or

$$\ln \frac{W_i}{W_f} = \frac{15\ 000}{(32.2)(390)} = 1.194$$

and

$$\frac{W_i}{W_f} = 3.30$$

The final weight in orbit is, then, given by

$$W_f = \frac{25\ 630}{3.30} = 7770 \text{ lb}$$

and the second-stage propellant load is

$$W_p = 25\ 630 - 7770 = 17\ 860 \text{ lb}$$

The hardware weight of the second stage is, then, 1786 pounds which when subtracted from the burnout weight gives us a payload in orbit of 7770 - 1786 = 5984 pounds. Thus, whereas the single-stage vehicle can deliver essentially no payload to orbit, the two-stage vehicle can deliver over 5 percent of its initial weight to orbit.

## NASA LAUNCH VEHICLES

It is impractical to use a large vehicle such as the Saturn V to launch a small instrument package that can be launched by a smaller, less expensive vehicle such as the Scout. On the other hand, to develop a new vehicle for each mission is expensive. Consequently, NASA has developed a family or "stable" of vehicles of various sizes, and tries to use each member for a range of missions within its capability. Moreover, the more experience we have with a few vehicles, the more reliable we can make them. With this in mind, NASA's aim is to develop the smallest number of vehicles consistent with the full scope of space missions now foreseen.

At present, NASA is actively using seven launch vehicles. They are the Scout, Delta, Thor-Agena, Atlas-Agena, Atlas-Centaur, Uprated Saturn I, and Saturn V. The Thor-Agena and Atlas-Agena were developed by the U. S. Air Force and are used jointly by NASA and the Air Force. The remaining vehicles were or are being developed by NASA. The Uprated Saturn I and Saturn V are man-rated vehicles and will be used for manned missions. The other vehicles are all used for unmanned missions. The characteristics of all the vehicles are summarized in table 11-II.

TABLE 11-II. - LAUNCH VEHICLE CHARACTERISTICS

| Vehicle (a) | Height, ft | Weight, lb | Payload to orbit | | Launch site | Program application | First stage | | | Second stage | | | Third stage | | | Fourth stage | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | Weight, lb | Altitude, n mi | | | Designation | Propellant | Thrust, lb | Designation | Propellant | Thrust, lb | Designation | Propellant | Thrust, lb | Designation | Propellant | Thrust, lb |
| Scout | 68 | 38 500 | 240 | 300 | Wallops, WTR | Explorer, reentry probes, ESRO, others | Algol | Solid | 88 000 | Castor | Solid | 61 000 | Antares | Solid | 23 000 | Altair | Solid | 5 800 |
| Delta[b] | 90 | 114 000 | 880 | 300 | ETR, WTR | Explorer, OSO, Tiros, Relay, ESSA, others | DM-21 | RP-1/LOX | 170 000 | ------- | UDMH/IRFNA | 7 500 | Altair | Solid | 5 800 | ----- | ---- | ----- |
| Thor-Agena[b] | 76 | --------- | 1 600 | 300 | WTR | Nimbus, Echo II, Alouette, others | DM-21 | RP-1/LOX | 170 000 | Agena | UDMH/IRFNA | 16 000 | ------- | --------- | ------- | ----- | ---- | ----- |
| Atlas-Agena | 91 | --------- | 5 950 | 300 | ETR, WTR | OAO, OGO, Mariner, Ranger, others | Atlas | RP-1/LOX | 388 000 | Agena | UDMH/IRFNA | 16 000 | ------- | --------- | ------- | ----- | ---- | ----- |
| Atlas-Centaur | 100 | 300 000 | 8 500 | 300 | ETR | Surveyor, Mariner | Atlas | RP-1/LOX | 388 000 | Centaur | $LH_2$/LOX | 30 000 | ------- | --------- | ------- | ----- | ---- | ----- |
| Uprated Saturn I | 225 | 1 300 000 | 40 000 | 100 | ETR | Apollo | S-IB | RP-1/LOX | 1 600 000 | S-IVB | $LH_2$/LOX | 200 000 | ------- | --------- | ------- | ----- | ---- | ----- |
| Saturn V | 365 | 6 100 000 | 285 000 | 100 | ETR | Apollo | S-IC | RP-1/LOX | 7 500 000 | S-II | $LH_2$/LOX | 1 000 000 | S-IVB | $LH_2$/LOX | 200 000 | ----- | ---- | ----- |

[a]All the vehicles are operational except the Saturn V, which is still under development.

[b]Currently being launched with three solid motors strapped to the first stage for increased launch thrust and payload capability (and increased vehicle weight). The thrust-augmented Delta is called TAD, and the thrust-augmented Thor-Agena is called TAT.

# APPENDIX - SYMBOLS

$F_c$ centrifugal force, lb

$g_o$ acceleration due to gravity at orbital altitude, $ft/sec^2$

$g_e$ acceleration due to gravity at Earth's surface, $ft/sec^2$

$I_{sp}$ specific impulse, sec

$m$ mass, slugs

$r_o$ orbit radius, ft

$r_e$ radius of the Earth, ft

$v_b$ burnout velocity, ft/sec

$v_c$ circular orbital velocity, ft/sec

$W_f$ final weight, lb

$W_{hw}$ hardware or jettison weight, lb

$W_i$ initial weight, lb

$W_p$ propellant weight, lb

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