Module – 4

LAUNCH VEHICLE DYNAMICS AND ATTITUDE CONTROL OF ROCKETS AND MISSILES

Syllabus:

Launch Vehicle Dynamics: Tsiolskovsky's rocket equation, range in the absence of gravity, vertical motion in the earth's gravitational field, inclined motion, flight path at constant pitch angle, motion in the atmosphere, the gravity turn – the culmination altitude, multi staging. Earth launch trajectories – vertical segment, the gravity turn, constant pitch trajectory, orbital injection. Actual launch vehicle trajectories, types. Examples, the Mu 3-S-II, Ariane, Pegasus launchers. Reusable launch vehicles, future launchers, launch assist technologies.

Attitude Control of Rockets and Missiles: Rocket Thrust Vector Control – Methods of Thrusts Vector Control for solid and liquid propulsion systems, thrust magnitude control, thrust termination; stage separation dynamics, separation techniques.

4.1 Tsiolskovsky's Rocket Equation

Tsiolkovsky was faced with the dynamics of a vehicle, the mass of which is decreasing as a jet of matter is projected rearwards. As we shall see later, the force that projects the exhaust is the same force that propels the rocket. It partakes in Newton's third law—'action and reaction are equal and opposite', where 'action' means force.

The rocket equation—more properly called Tsiolkovsky's equation—has a relatively simple derivation. It is based on calculating the acceleration of a rocket vehicle with a mass decreasing continuously due to the expenditure of propellant.

The case we have to consider is that of a rocket vehicle of mass M, expelling combustion products at a rate m, with a constant effective exhaust velocity v_e. The mass of the vehicle is decreasing at the rate m, and, due to the thrust F, developed by the exhaust, the rocket is accelerating. The rocket equation produces the achieved velocity at any time in terms of the initial and current mass of the rocket.

The thrust developed by the exhaust is represented by

$$F = v_e m$$

where

$$m = \frac{dM}{dt}$$

This is a simple application of Newton's third law to the exhaust gases.

The acceleration of the rocket, under the thrust F, is represented by a second application of Newton's law:

$$\frac{dv}{dt} = \frac{F}{M}$$

Substituting for F, from the first equation,

$$\frac{dv}{dt} = -v_e \frac{dM}{dt} \frac{1}{M}$$

because the velocity increases as the mass decreases. Cancelling dt, and rearranging, produces

$$dv = -v_e \frac{dM}{M}$$

Integrating the velocity between limits of zero and V, for a mass change from M_0 to M, produces

$$\int_0^V dv = -v_e \int_{M_0}^{M} \frac{dM}{M}$$

The solution is

$$V = v_e \log_e \left(\frac{M_0}{M} \right)$$

Here M_0 is the mass of the rocket at ignition, and M is the current mass of the rocket. The only other parameter to enter into the formula is v_e, the effective exhaust velocity. This simple formula is the basis of all rocket propulsion. The velocity increases with time as the propellant is burned. It depends on the natural logarithm of the ratio of initial to current mass; that is, on how much of the propellant has been burned. For a fixed amount of propellant burned, it also depends on the exhaust velocity—how fast the mass is being expelled.

This is shown in Figure 1.6, where the rocket velocity is plotted as a function of the *mass ratio*. The mass ratio, often written as R, or Λ, is just the ratio of the initial to the current mass:

$$R = \frac{M_0}{M}$$

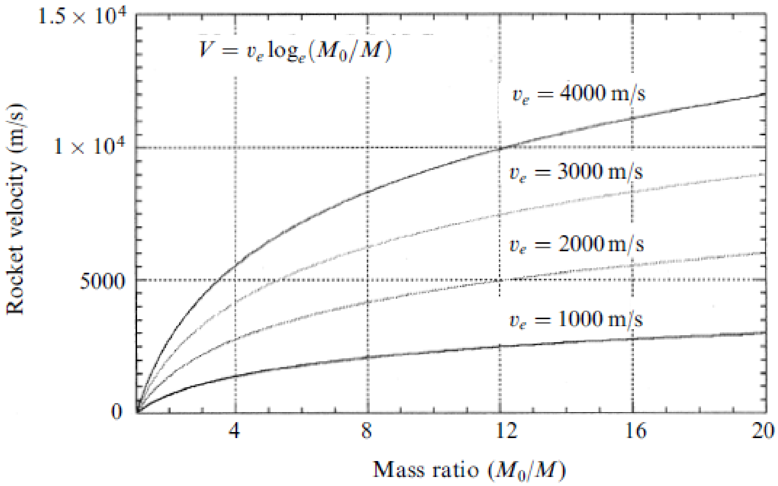


Figure 1.6. Tsiolkovsky's rocket equation.

4.2 Range in the Absence of Gravity

Range is defined here as the distance travelled along the rocket's trajectory during a burn. It is an important parameter in launch dynamics, particularly when gravity has an effect on the acceleration. It is sometimes useful to know the distance travelled during the burn, even in the absence of gravity—such as during an orbital insertion—or for an interplanetary trajectory injection. In the absence of gravity the range is obtained by integrating the rocket equation over time. The rocket equation expresses the velocity in terms of the mass ratio, which translates into an effective integration over mass. This does not require a knowledge of how the mass flow rate dM/dt varies during the burn.

While the velocity achieved is independent of the mass flow rate, the distance the rocket has to travel in order to reach this velocity is not. It can be seen from the rocket equation that in the limit, if all the propellant were to be expended instantaneously then the final velocity would be the same, but the distance travelled in achieving that velocity would be zero (a shell fired from a gun approximates to this situation). The most common case is where the mass flow rate, and hence the thrust, are constant, and we shall make this assumption here.

The integration has to be over time, so we need an expression for time t in terms of mass loss. For constant thrust this is

$$t = \left(\frac{M_0 - M}{m} \right)$$

where the (constant) mass flow rate m is

$$m = dM/dt$$

Dividing through by M_0 produces

$$t = \frac{M_0}{m} \left(1 - \frac{M}{M_0} \right)$$

This expression for the time is conveniently in terms of the mass ratio, M_0/M.

The distance travelled by the rocket is determined by integrating the velocity, which of course varies with time. This is simply integration of the rocket equation. The distance travelled, s, is expressed by

$$s = \int_0^t V(t)\, dt = \int_0^t v_e \log_e \left(\frac{M_0}{M_0 - mt} \right) dt$$

Evaluation of the integral from time zero to time t leads to

$$s = v_e \frac{M_0}{m} \left[\frac{M_0 - mt}{M_0} \left(\log_e \frac{M_0 - mt}{M_0} - 1 \right) \right]$$

Evaluation of the integral from time zero to time t leads to

$$s = v_e \frac{M_0}{m} \left[\frac{M_0 - mt}{M_0} \left(\log_e \frac{M_0 - mt}{M_0} - 1 \right) \right]$$

Substitution of the expression for t, derived above, produces

$$s = v_e \frac{M_0}{m} \left[1 - \frac{M}{M_0} \left(\log_e \frac{M_0}{M} + 1 \right) \right]$$

Therefore, for the distance travelled we have an expression involving the familiar mass ratio M_0/M; but as foreseen, the value of s also depends on the exhaust velocity v_e, and inversely on the mass flow rate m; a high mass flow (a high thrust) leads to a short range. This is intuitive; if all the propellant is exhausted very quickly then the distance travelled will be short. The velocity depends only on v_e and M_0/M, and the range is the distance travelled while reaching this velocity. The above arguments apply only to the range under power; after burn-out the range will continue to increase, but of course the velocity will remain constant in the absence of gravitational effects. Figure 5.2 shows the range as a function of mass ratio for two exhaust velocities. It is important to remember that the functional relationship with mass ratio can apply to any instant during the burn of the rocket, not simply at the moment of burn-out.

The above expression is for zero initial velocity. If the spacecraft already has some velocity—from a previous stage, for instance—then this has to be included in the calculation of the total distance travelled during the burn. The additional distance is just $V_i t$, or expressing t as above:

$$s_i = V_i \frac{M_0}{m} \left(1 - \frac{M}{M_0} \right)$$

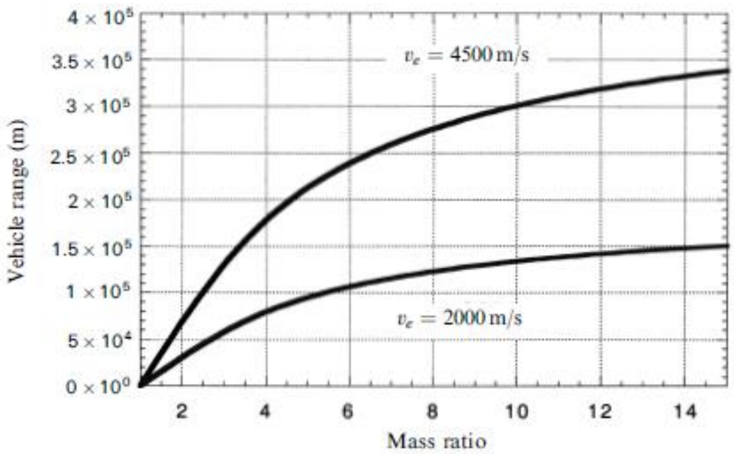


Figure 5.2. Range as a function of mass ratio.

This distance should be added to the distance already calculated.

The above arguments lead to a triad of equations for the velocity, time and range in the absence of gravitational effects:

$$V = v_e \log_e\left(\frac{M_0}{M}\right)$$

$$t = \frac{M_0}{m}\left(1 - \frac{M}{M_0}\right)$$

$$s = v_e \frac{M_0}{m}\left[1 - \frac{M}{M_0}\left(\log_e \frac{M_0}{M} + 1\right)\right] + V_i \frac{M_0}{m}\left(1 - \frac{M}{M_0}\right)$$

These equations can be used for calculations in which gravity plays no part; for example, the injection into orbit of a satellite by the upper stage of a launcher. For orbital injection the thrust and velocity vectors are usually perpendicular to the gravitational field.

4.3 Vertical Motion in The Earth's Gravitational Field

In considering vertical motion, the thrust, velocity and gravitational field vectors are all aligned, and the motion is exclusively one-dimensional. This situation applies in the initial segment of most launches, and is the simplest to treat. As before, we shall derive expressions for the velocity achieved, the distance travelled, and the time, all in terms of the mass ratio.

5.2.1 Vehicle velocity

The above derivation of the rocket equation can easily be adapted to determine the velocity in the presence of gravity. The thrust remains the same, but the acceleration of the rocket is now governed by two forces: the thrust, and the opposing force of gravity.

As before, the thrust developed by the exhaust is represented by

$$F = v_e m \quad \text{where } m = \frac{dM}{dt}$$

The acceleration of the rocket, under the thrust F, and the opposing force of gravity, is represented by

$$\frac{dv}{dt} = \frac{F - Mg}{M}$$

where Mg is the current weight of the rocket. Substituting for F,

$$\frac{dv}{dt} = -v_e \frac{dM}{dt} \frac{1}{M} - \frac{Mg}{M}$$

Multiplying through by dt and rearranging produces

$$dv = -v_e \frac{dM}{M} - g\,dt$$

Integration between limits of zero and V, the vehicle velocity, for a mass change from M_0 to M produces

$$\int_0^V dv = -v_e \int_{M_0}^{M} \frac{dM}{M} - g \int_0^t dt$$

The solution is

$$V = v_e \log_e \frac{M_0}{M} - gt$$

Or, using the expression for t in terms of the mass ratio,

$$V = v_e \log_e \frac{M_0}{M} - g \frac{M_0}{m} \left(1 - \frac{M}{M_0}\right)$$

This expression has two terms: the familiar expression from the rocket equation, and a second term involving the acceleration of gravity. The vehicle velocity is equal to that which would have been obtained in the absence of gravity, minus the acceleration of gravity multiplied by the time—a result which could have been arrived at by intuition.

The first part of the expression—the velocity in the absence of gravity—is sometimes referred to as the 'ideal velocity' and the second part as the 'gravity loss'. While the ideal velocity is always independent of the thrust history and the burn time, the gravity loss is not independent. A very short acceleration, with high thrust and high mass-flow rate, leads to a small gravity loss, while a slow acceleration, with low thrust and low mass-flow rate, leads to a high gravity loss. This corresponds with intuition: if the mass flow rate is high, less of the propellant has to be carried to high altitude, or accelerated to high velocity, before it is burned. Since the raising and acceleration of propellant both reduce the amount of energy available to accelerate the payload, exhausting most of the propellant early in the launch is beneficial. Of course high mass-flow rates imply high thrust, and this may be inconvenient,

5.2.2 Range

The range is simply derived, as before, by integration of the velocity expression in the previous section:

$$s = v_e \int_0^t \log_e \frac{M_0}{M}\,dt - g \int_0^t t\,dt$$

The time t is of course unchanged by the presence of gravity. It depends only on the properties of the rocket, and is represented by the expression derived earlier.

With this information, and by analogy with the previous case, we find

$$s = v_e \frac{M_0}{m}\left(1 - \frac{M}{M_0}\left(\log_e \frac{M_0}{M} + 1\right)\right) - \frac{1}{2} g t^2$$

or substituting $t = M_0/m(1 - M/M_0)$

$$s = v_e \frac{M_0}{m}\left(1 - \frac{M}{M_0}\left(\log_e \frac{M_0}{M} + 1\right)\right) - \frac{1}{2} g \left(\frac{M_0}{m}\right)^2 \left(1 - \frac{M}{M_0}\right)^2$$

Comparing this with the non-gravity case, we can again consider the range as the 'ideal range', identical to that in free space, modified by the gravity loss term. As before, the range under power is short when the thrust-to-weight ratio is large.

The equations for velocity, time and range in the presence of gravity are

$$V = v_e \log_e \frac{M_0}{M} - g\frac{M_0}{m}\left(1 - \frac{M}{M_0}\right)$$

$$t = \frac{M_0}{m}\left(1 - \frac{M}{M_0}\right)$$

$$s = v_e \frac{M_0}{m}\left(1 - \frac{M}{M_0}\left(\log_e \frac{M_0}{M} + 1\right)\right) - \frac{1}{2}g\left(\frac{M_0}{m}\right)^2\left(1 - \frac{M}{M_0}\right)^2$$

Note that in all cases, 'range' indicates the distance travelled during acceleration, assuming an initial velocity of zero.

These equations show how the motion of the rocket is altered by gravity when the motion is vertical. This applies to the early stages of most launches, and the effect of gravity can be estimated using these equations. The general effect is that the velocity and the distance travelled are less than would have been predicted by the rocket equation, by the amount of the gravity loss. In launches, the main requirement is to gain horizontal velocity rather than vertical velocity. This is needed to arrive at the necessary orbital velocity. Vertical flight does not contribute to this, and moreover it is very expensive in terms of gravity loss. For this reason, launchers begin to travel horizontally as soon as possible in their flight. In the next section we shall examine the effects of gravity on inclined motion of a rocket vehicle.

4.4 Inclined Motion in a Gravitational Field

It is obvious that if the whole trajectory of the rocket is vertical, then unless escape velocity is reached the payload will ultimately fall back to Earth. To achieve orbit around a planet requires a high *horizontal* velocity. Thus the majority of the flight path of a launch vehicle is inclined to the gravitational field in order to gain velocity in the horizontal direction. Gravity now affects the direction of flight as well as the magnitude of the velocity. As we shall see, the flight path is curved, even if the thrust vector direction is constant. In general, the computation of the flight path is complex and requires numerical solution, although the case for constant pitch angle can be treated, and provides some useful insight into the way launch vehicles behave.

Constant pitch angle

The pitch angle (Figure 5.4) is the angle made by the thrust vector to the horizontal, which in most cases is the same as the angle of inclination of the vehicle axis to the horizontal, since the mean thrust axis coincides with the vehicle axis. This kind of flight is often used during the later stages of the launch because it produces the maximum velocity for a given final injection angle. However, it is not a good flight path for the early parts of the flight, where atmospheric forces are important, and where a pitch angle that varies with time is more desirable.

Since we now have to deal with inclined flight we shall need to consider both vertical and horizontal components of the velocity and distance travelled. The thrust and gravitational force also have to be resolved. The derivation is rather simple because the vertical components are the same as in the previous section, and the horizontal components are unaffected by gravity.

The thrust is independent of the orientation of the rocket:

$$F = v_e m \quad \text{where } m = \frac{dM}{dt}$$

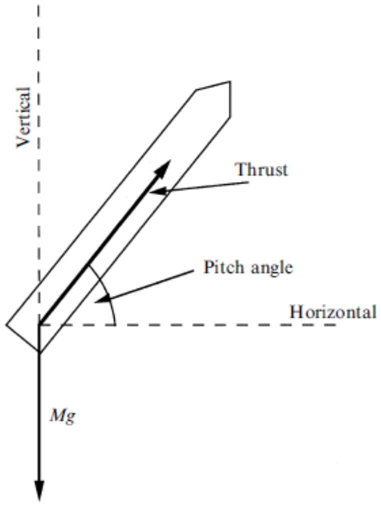


Figure 5.4. Thrust and pitch angle.

The vertical acceleration of the rocket, under the thrust F and the opposing force of gravity, is expressed as

$$\frac{dV_Z}{dt} = \frac{(F \sin\theta - Mg)}{M}$$

where θ is the pitch angle and, as before, Mg is the current weight of the rocket. The thrust is resolved in the vertical direction, and the vertical component of velocity is V_Z. The further steps are identical to those in the previous section, leading to

$$dV_Z = v_e \frac{dM}{M} \sin\theta - g\,dt$$

Integration between limits of zero and V for a mass change from M_0 to M produces

$$\int_0^V dV_Z = v_e \sin\theta \int_M^M \frac{dM}{M} - g \int_0^t dt$$

and the solution is

$$V_Z = v_e \sin\theta \log_e \frac{M_0}{M} - gt$$

To calculate the horizontal component of velocity V_X, the thrust is resolved on to the horizontal axis, and gravity plays no part, leading to

$$\frac{dV_X}{dt} = \frac{F\cos\theta}{M}$$

Integration as before, over the same limits, leads to

$$V_X = v_e \cos\theta \log_e \frac{M_0}{M}$$

The magnitude of the total velocity V—the speed of the rocket along its direction of motion—can be derived from the triangle of velocities, and is represented by the quadratic sum of the components:

$$V = \sqrt{(V_X^2 + V_Z^2}$$

Substitution and simplification leads to

$$V = \sqrt{\left(v_e^2 \log_e^2 \frac{M_0}{M} - 2v_e\, gt \sin\theta \log_e \frac{M_0}{M} + g^2 t^2\right)}$$

This is the total velocity along the current velocity vector. The burning time is independent of inclination, and as before

$$t = \frac{M_0}{m}\left(1 - \frac{M}{M_0}\right)$$

Substitution of M = mass at burn-out, would determine the velocity increment from an individual stage if the pitch angle were constant throughout the burn.

5. The flight path at constant pitch angle

It is important to realise that the flight path angle and the pitch angle are not necessarily identical. The pitch angle is the angle of the thrust vector (and the vehicle axis) to the horizontal and the flight path angle is the angle of the velocity vector to the horizontal.

The flight path angle, γ, can be derived using the above expressions for vertical and horizontal velocity, in a triangle of velocities:

$$\tan\gamma = \frac{V_Z}{V_X} = \frac{v_e \sin\theta \log_e(M_0/M) - gt}{v_e \cos\theta \log_e M_0/M}$$

Cancelling produces

$$\tan\gamma = \tan\theta - \frac{gt}{v_e \cos\theta \log_e M_0/M}$$

The second term is always finite, because $M = M_0 - mt$ and is never equal to zero for a practical rocket. So the flight path angle is *always* different from the pitch angle.

This result is of great practical significance, because it indicates that for a constant pitch angle the rocket is forced to travel with its axis inclined to the direction of motion. If we define the angle between the thrust axis and the velocity vector as the *angle of attack*, then this is always non-zero. This difference between the directions of the thrust and velocity vectors is in accord with intuition. Gravity is pulling the rocket down, so some additional vertical thrust is needed to counteract it, which must result in an upward tilt of the vehicle axis from the flight path. Since atmospheric effects on the rocket depend strongly on the angle of attack, constant pitch angle is not the best approach for low in the atmosphere.

The flight path angle for constant pitch angle varies throughout the flight, being at its greatest offset from the vehicle axis immediately after the vehicle axis departs from the vertical. In the limit when $t = 0$, the above equation can be shown to reduce to

$$\tan\gamma = \tan\theta - \frac{gM_0}{v_e m \cos\theta}$$

The initial angle of attack depends inversely on the thrust-to-weight ratio, and is smallest for high thrust-to-weight. Thereafter the angle of attack decreases as the weight decreases, and the thrust, of course, remains constant. Thus, near burn-out the vehicle axis and the flight path are nearly parallel, but they cannot ever be precisely so because of the residual and payload mass. The flight path angle as a function of time for a number of different pitch angles is show in Figure 5.6. The flight path angle changes instantaneously when the thrust axis changes from vertical, and thereafter converges on the pitch angle.

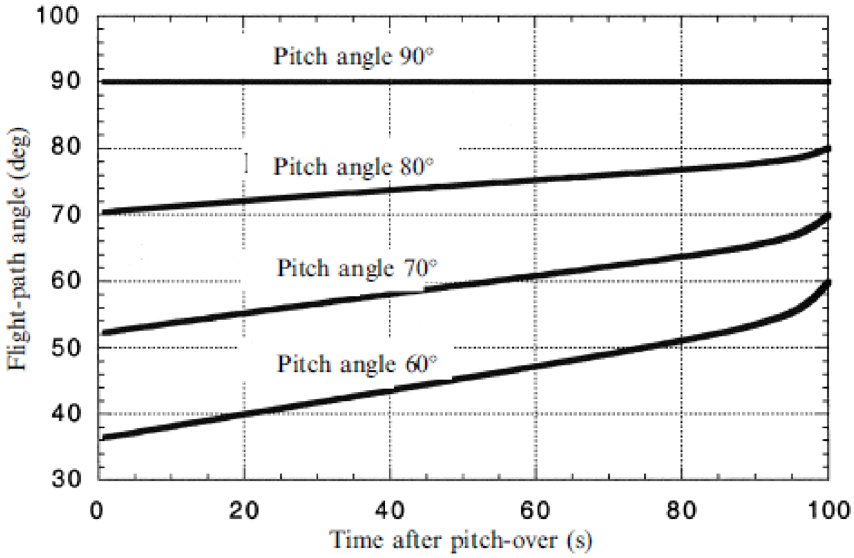


Figure 5.6. Flight path angle as a function of time and pitch angle.

4.5 Motion in the Atmosphere

So far we have described the effects of the gravitational field on the motion of the rocket in some simple circumstances. The atmosphere also has a significant effect on the rocket, which is to be expected, since the velocity quite quickly exceeds the sound speed and the rocket becomes a hypersonic vehicle. Mass ratio arguments require the vehicle to be lightweight, and consequently not well able to withstand the forces so induced. Aerodynamics, especially for hypersonic flight, is a complex subject, but fortunately there are some simple ideas which can be used to estimate the aerodynamic forces on the rocket.

5.4.1 Aerodynamic forces

The motion of the rocket through the atmosphere generates various forces that affect its motion. Discussion of the movements and instabilities which disturb its flight path are beyond the scope of this book, and the effects we shall deal with here are *lift* and *drag*. These both have an impact on the velocity that can be achieved and on the structural integrity of the rocket; they depend strongly on the instantaneous velocity and on the local density of the atmosphere. Figure 5.7 shows how these forces act on the rocket.

The lift is generated by the air flowing over the rocket surface and acts in a direction perpendicular to the flight path of the rocket. The drag is caused by a number of effects, and acts parallel to the flight path and in the opposite direction to the velocity. The transverse force T on the rocket, and the axial retarding force R, are obtained by resolving the lift and drag and adding them, defining the angle of attack as α:

$$T = L\cos\alpha + D\sin\alpha$$

$$R = -L\sin\alpha + D\cos\alpha$$

Here L and D represent the lift and drag respectively, and the minus sign shows that the lift acts as an accelerating force, as shown in the figure. The drag acts as a retarding force, and exists for any angle of attack, including zero. The lift is present only when the angle of attack is non-zero. The magnitude of the lift and drag depend strongly on the velocity, and the form of the dependence is different for velocities below and above the local sound speed. Rockets quickly reach supersonic velocity, in which case the *lift coefficient* is expressed approximately by

$$C_L = 2\alpha$$

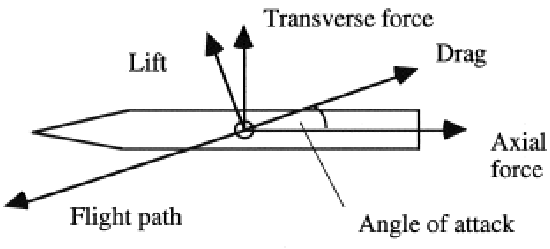


Figure 5.7. The aerodynamic forces acting on a rocket.

For a cylindrical rocket, most of the lift is generated by the nose cone. The drag coefficient is represented by

$$C_D = a + b\mathrm{M}^6$$

for $\mathrm{M} < 1$, and

$$C_D = a + \frac{b}{\mathrm{M}^2}$$

for $\mathrm{M} > 1$. Here a and b are constants which depend on α, and M is the *Mach number*—the ratio of the velocity to the local sound speed. The coefficient peaks around the velocity of sound, and typically takes a value of about 0.2.

5.4.2 Dynamic pressure

The lift and drag forces can be expressed in terms of the above coefficients—the velocity V, the atmospheric density ρ, and a *reference area* A. The latter can be regarded as the frontal area of the rocket projected onto the plane perpendicular to the direction of motion. It increases with the angle of attack.

The lift L is expressed as

$$L = C_L A \frac{\rho V^2}{2}$$

and the drag D is represented by

$$D = C_D A \frac{\rho V^2}{2}$$

The quantity $\rho V^2/2$ has the dimensions of pressure and is known as the *dynamic pressure*, often represented as q. Figure 5.8 shows typical profiles of the dynamic pressure, velocity and altitude as a function of instantaneous mass ratio, or time.

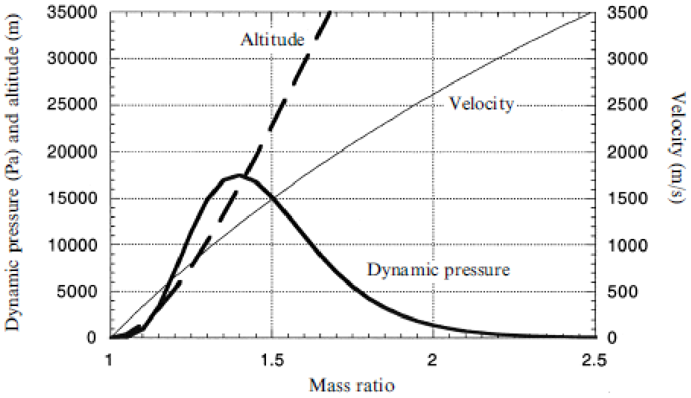


Figure 5.8. Dynamic pressure, velocity and altitude as functions of mass ratio.

Dynamic pressure has two important properties. The V^2 dependence means that the lift and drag, and the disruptive forces on the rocket, increase very rapidly as the rocket accelerates. The effect of drag on first-stage acceleration is quite significant: the acceleration of the vehicle is often almost constant even though the mass is reducing. The dynamic pressure also depends on the atmospheric density, which decreases rapidly as the rocket gains altitude. Thus, with velocity increasing, and density decreasing, with time after launch, every rocket passes through a condition known as maximum dynamic pressure, or 'maximum q'. This is the time when the atmospheric forces are at their maximum, and when the risk to the structural integrity of the rocket is greatest. Significant effort is put into reducing this risk as much as possible. For example, the Space Shuttle throttles back the main engines during this period. The booster thrust profile is also tailored to reduce their thrust. A temporary reduction in thrust will reduce the force on the rocket vehicle, but it will continue to accelerate. This approach is possible because the peak in dynamic pressure is so sharp, as a direct result of the exponential reduction in atmospheric density with altitude and the V^2 dependence of the dynamic pressure on velocity.

Reduction in thrust is not always possible, and in any case it does not affect the aerodynamic forces, only the thrust force. It is also advisable to attempt to reduce the aerodynamic forces. A common approach is to hold the angle of attack at zero during the early part of the flight. This not only reduces stress due to lift and drag, but also minimises the drag losses on the rocket when low in the atmosphere, maximising the velocity gain where drag dominates. This means that the rocket velocity vector is axial, so that the area presented to the atmosphere is just the cross-section. Because the angle of attack is zero, the lift is also zero. The result is a significant reduction in the aerodynamic forces. This technique results in the rocket following a curved path with the nose gradually dropping, as opposed to the upwardly curved path arising in constant pitch angle flight, and is known as a *gravity turn*.

4.6 The Gravity Turn

In constant pitch angle flight the downward force of gravity is counteracted by an upward thrust component due to the upward tilt of the thrust vector with respect to the velocity vector. If the angle of attack is zero, then the thrust and velocity vectors coincide, and there is no additional upward thrust. The flight path is therefore curved downwards by the influence of gravity—a 'gravity turn'. The gravity turn should be distinguished from the downwardly curved flight path of a ballistic projectile such as a shell, where there is no thrust, and the path depends only on the initial velocity and angle of projection. In the gravity turn a rocket moves under the combined forces of thrust and gravity, and follows a path that is differently curved.

The flight path for a gravity turn has to be computed numerically, but some insight into the motion can be gained through a simplified analysis. The differential equations for the vertical and horizontal motion are the same as for the case of constant pitch:

$$\frac{dV_Z}{dt} = (F - Mg)\frac{\sin\theta}{M}$$

$$\frac{dV_X}{dt} = F\frac{\cos\theta}{M}$$

However, θ is no longer a constant. It is set equal to γ, the flight path angle, which varies and is itself defined from the triangle of velocities by

$$\tan\gamma = \frac{V_Z}{V_X}$$

We must therefore manipulate the differential equations before integrating in order to define expressions for the total velocity V, and for the flight path angle, as a function of time.

Substituting for $\sin\gamma$ and $\cos\gamma$ from the triangle of velocities, the above equations can be written as

$$\frac{dV_Z}{dt} = \frac{F}{M}\frac{V_Z}{V} - g$$

$$\frac{dV_X}{dt} = \frac{F}{M}\frac{V_X}{V}$$

Multiplying the first by V_Z, the second by V_X, and then adding the equations leads, after some manipulation, to an expression for dV/dt, where $V = \sqrt{V_X^2 + V_Z^2}$ is the total velocity along the direction of motion:

$$\frac{dV}{dt} = \frac{F}{M} - g\sin\gamma$$

Multiplying the first by V_X, the second by V_Z, and then subtracting leads in a similar way to an expression for $d\gamma/dt$, where γ is the flight path angle:

$$\frac{d\gamma}{dt} = -\frac{g}{V}\cos\gamma$$

As mentioned above, these equations can only be integrated numerically for the general case. This can be conveniently carried out with a spreadsheet programme. An example calculation of V and γ as functions of time is illustrated in Figure 5.9.

In this particular case the initial flight path angle is 60°, and a gravity turn is followed by setting the flight path and pitch angles equal. Initially the flight path angle changes quickly, but as the velocity increases and the mass decreases the rate of change becomes smaller, and the path stabilises at around 53°.

In the specific case in which $d\gamma/dt$ is a constant (the pitch angle changes uniformly with time) there is an analytical solution. Writing $d\gamma/dt = c$ for the constant pitch rate, the solution is

$$V = V_0 + \frac{g^2}{2} v_e \log_e \frac{M_0}{M}$$

$$\cos\gamma = \cos\gamma_0 + c\frac{g}{2} v_e \log_e \frac{M_0}{M}$$

where V_0 and γ_0 are the initial values of velocity and pitch angle. It can be seen that the pitch angle increases with time, giving the downwardly curved path, and the velocity increases with time, as expected.

The total velocity is increasing with time quite rapidly, but because of the downward curve of the flight path, altitude is not gained so quickly. This penalty is justified in the reduced risk to the rocket structure of the zero angle of attack trajectory. Once the maximum dynamic pressure region is passed, a more efficient pitch programme can be followed. Modern launchers use computer controlled pitch programmes to optimise the velocity and altitude achieved during the gravity turn. This reduces the gravity losses while still minimising the effects of dynamic pressure on the rocket.

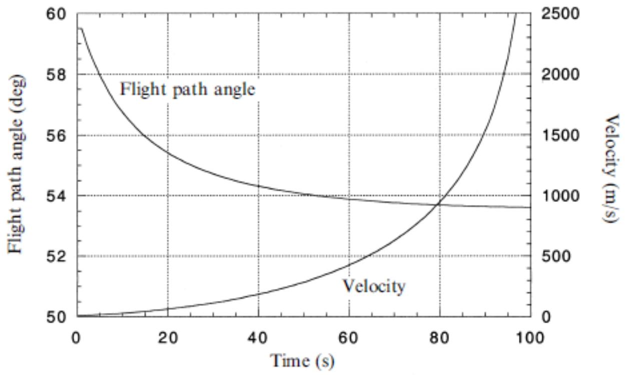


Figure 5.9. Flight path angles and velocity as functions of time for a gravity turn.

4.7 Earth Launch Trajectories

Launches from the surface of the Earth are considerably more complicated than those from the surface of the Moon. The gravitational potential well is much deeper, and a multi-stage rocket is required to reach orbital altitude and velocity. The atmosphere causes drag and lift which reduce the acceleration, and are potentially destructive forces. The mass ratio of each stage of the launcher needs to be as high as possible, and this requires the lightest structure. It is considerably easier to design a lightweight structure for a vertical launch than for a horizontal launch. For all these reasons, terrestrial launches begin vertically. It is clear from earlier discussion that ultimately the launcher has to travel horizontally, and so there has to be a transition somewhere in the trajectory.

The lower part of the flight is in the atmosphere. Precautions have to be taken to minimise drag and lift, which define the length of the vertical segment and the nature of the transition from vertical to horizontal. The nature of the trajectory is also different because of the much higher velocity increment needed for Earth orbit. This means a higher mass ratio and therefore a longer burn time for reasonable acceleration. These factors take the trajectory even further from the ideal described above, and greatly increases the gravity loss. Another aspect which may play a part is the capability of stage engines to be restarted. If this is not possible, then the rocket has to be powered continuously into orbit, with further gravity loss. The design of launchers has to take into account these issues, and optimise the trajectory and the launcher itself to maximise the payload capability. This is, of necessity, a very complex process.

The vertical segment of the trajectory

The length of the vertical section varies from vehicle to vehicle. Indeed, the more modern launchers have very short vertical segments; for example, the Space Shuttle just clears the launch tower. A long vertical section reduces stress on the rocket structure by placing the vehicle quickly, and at relatively low velocity, above the densest part of the atmosphere; and, of course, the angle of attack is always zero for vertical flight. The vertical segment, however, does not contribute to the eventual orbital velocity, and the gravity losses associated with transporting most of the propellant, tanks and so on to high altitude are large for vertical flight. The choice is then one of protecting the rocket or losing efficiency. As technology has developed, the length of the vertical segment has tended to decrease, and with modern guidance, engineers can protect the rocket from aerodynamic stress more readily, and take advantage of the increased efficiency of a short vertical segment to launch larger payloads. In the vertical segment, drag rises with the square of the velocity, and limits the acceleration. It does not normally reach its peak value (maximum dynamic pressure) in the vertical segment. This is important in that the densest part of the atmosphere is traversed at a relatively low velocity. The vertical segment can be regarded simply as transportation of the rocket to a region of lower atmospheric density, where the real business of gaining velocity can be carried out under more benign conditions.

The gravity turn or transition trajectory

The gravity turn—or its more modern analogue, the controlled transition trajectory—begins as soon as the rocket departs from its vertical flight, which, with some modern launchers, can be very early. Whatever the precise nature of the trajectory, it minimises the stress on the rocket by keeping the angle of attack close to zero, while the vehicle accelerates through the maximum dynamic pressure zone. The reduction in gravity loss from an inclined trajectory, and the continuing reduction in mass, means that the velocity gain is more rapid. Again, for modern launchers the use of advanced guidance and thrust vector control maximises the efficiency while limiting the stress on the rocket. The use of a controllable thrust engine, as on the Space Shuttle, can be included in this programme to lower the thrust as maximum dynamic pressure is encountered. These advanced guidance programmes are not strictly gravity turns, but the same principles apply. This part of the trajectory is truly a transition region, because the velocity is increasing rapidly, and the atmospheric density is still significant. The objective is to gain as much velocity as possible while at the same time minimising the risk to the structural integrity of the spacecraft by active control of the angle of attack and the thrust. The flight path must not be too flat, because it is important to rise above the region of tangible atmosphere during this portion of the flight.

At this stage, the alternative to a curved path would be either to continue the vertical ascent until the atmospheric density is negligible, or to attempt to gain velocity and altitude quickly by using constant pitch flight. The former would produce a very high gravity loss, while the latter would exert potentially destructive transverse loads on the rocket. Gravity loss should not be underrated: the loss during a typical launch can be as much as 20% of the total velocity.

Constant pitch or the vacuum trajectory

As soon as the atmosphere is sufficiently tenuous to allow the angle of attack to increase, the trajectory can be freely chosen to take full advantage of the rocket performance. Everything is then concentrated on maximising the velocity and altitude. The only restraint is on the acceleration, which should not be too great for the payload or for the astronauts. The nature of the trajectory here can be, at its simplest, constant pitch flight, the objective in this case being to burn the propellant as quickly as possible, with the intention being then to coast to the final orbital altitude. This is somewhat similar to the lunar launch situation. There are, however, other factors to be considered. One of these is the argument of perigee for the orbit: in simple terms, the longitude of the injection point. This may require a more complicated trajectory, which we shall consider after a brief review of orbital injection.

Orbital injection

Once the orbital injection point is reached, the final segment comprises horizontal acceleration to orbital velocity. This is in many cases the simplest segment, since there are no gravitational or aerodynamic disturbing forces: the objective is simply to reach the necessary horizontal velocity. A variety of orbits can be created, depending on the velocity specified (see Chapter 1). If the velocity is equal to $\sqrt{GM_{\oplus}/r_{\oplus}+h}$, then the orbit is circular. If the velocity is higher than this, then a variety of elliptical orbits, with the injection point as perigee, can be generated—even a parabolic or escape orbit if the velocity is as high as $\sqrt{2GM_{\oplus}/r_{\oplus}+h}$. If the velocity is lower than the circular velocity, then elliptical orbits with the injection point as apogee are possible. Usually the injection point is chosen to be the lowest point of the orbit and to be just high enough to prevent atmospheric drag from causing rapid decay. Orbits with injection point as apogee are therefore usually short-lived and the result of an injection error or motor failure.

In many cases this injection is not final, and the orbit is called a 'parking orbit', which is to allow the final orbit to have different properties. The most common use is to create an orbit with a different argument of perigee; that is, the location of the lowest and highest points of the orbit. Communication satellites, for example, need to be placed in a circular orbit with a 24-hour period, above a particular longitude on the Earth's surface. It is usual to use a parking orbit to allow the spacecraft to reach the correct perigee location before inducing another acceleration to create an elliptical orbit with its apogee at the final position of the satellite. A further burn at apogee then circularises the orbit with the correct period and phase. Similar considerations apply to interplanetary missions, and to missions which require a change of orbital plane.

The vacuum trajectory segment can be varied from the simple constant pitch burn and coast which is suitable for injection into low circular orbits, or for a low perigee parking orbit. To place a satellite efficiently into a transfer orbit, either the upper stage has to be capable of a restart, or a more complex vacuum trajectory has to be followed. The combination of parking orbit and restart is the most flexible, but it increases the risk and complicates the upper stage. If no restart is used, then the upper stage has to reach the correct injection point for the transfer orbit while still under power. In fact, the trajectory has to be tailored so that it culminates in the transfer orbit injection. A long continuous burn has the potential to generate a large gravity loss, and a slow acceleration to the injection point is therefore not desirable. Alternatively, a rapid acceleration would bring the stage to injection velocity in the wrong place. The solution adopted is to use a rapid and roughly constant pitch acceleration towards an apogee *higher* than the desired perigee of injection. This burns propellant quickly, and stores its energy in the initial form of kinetic energy, and then of potential energy, at the temporary apogee. The subsequent direction of thrust is chosen so that the rocket accelerates downwards, under power, towards the correct injection point, gaining velocity both from the thrust and from its decreasing potential energy. The whole trajectory is designed to bring the vehicle flight path angle and the thrust vector horizontal at the injection point, and to be at the correct altitude. This so-called vertical 'dog-leg' manoeuvre minimises gravity loss, and at the same time allows freedom in the injection point of the spacecraft.

The orbital inclination is defined by the latitude of the launch site and the azimuth of the launch direction. It is easy to see that spacecraft can be launched into orbits with inclination higher than, or equal to, the latitude of the site. For example, a launch site at 30° latitude can provide inclinations of 30° and greater. To inject a spacecraft into an orbit of lower inclination than the launch site requires a large additional velocity increment, because the plane of the natural orbit has to be rotated through the necessary angle, which requires a significant change of orbital angular momentum compared with the original angular momentum. This can be accomplished from a parking orbit by an out-of-plane burn, but is more usually carried out by a horizontal dog-leg manoeuvre. The critical point is the latitude of

injection. To achieve, say, an equatorial orbit from a non-equatorial launch site, the injection point has to be vertically over the equator. The vehicle begins by being launched into a simple coplanar trajectory, as if it were intended to enter an orbit with an inclination the same as that of the launch site latitude. At some point in the trajectory it is given lateral acceleration, so that the final injection is over, and parallel to, the equator. Projected on the surface of the Earth, this trajectory shows at least one change in direction. Consideration of the triangle of velocities shows that this change in direction requires considerable transverse velocity increment, and that it is best carried out early in the flight when the vehicle velocity is smaller, but it must be done in a vacuum so that there is no stress on the rocket. This results in its being part of the vacuum trajectory. The most common case is the launch of a geostationary satellite, which must be in an orbit coplanar with the equator.

4.8 Actual Launch Vehicle Trajectories, Types

There is a variety of launch vehicle trajectories ranging from simple launches with solid-fuelled rockets to those of highly sophisticated vehicles with advanced guidance programmes. The trajectory can also vary for the same vehicle, depending on the purpose. Here we shall consider only a few examples to illustrate the considerations elaborated earlier.

The Mu-3-S-II launcher

This is a simple solid fuelled Japanese launch vehicle, which was used to place satellites of about 0.5 tonne into a circular 550-km orbit, from the launch range near Uchinoura, in southern Kyushu. It was developed by the Institute of Space and Astronautical Science, to launch scientific satellites, and is now superseded by the M5 rocket (see Appendix 2). The inclination of the orbit is 33°, given by the latitude of the launch site; the right ascension of the ascending node follows from the time of injection. This launcher, now obsolete, is considered here because it illustrates some of the basic trajectory concepts described above.

The Mu-3-S-II launcher is remarkable in that there is no vertical segment; instead, the launcher is guided by a rail on the launch tower into a flight path with an initial 71° inclination. This breaks all the dynamical rules, but is necessary due to the densely populated nature of the Japanese coast. The fishing village of Uchinoura is only a few kilometres away, and the inclination of the flight path is chosen to ensure that even the most dramatic failure will not result in parts of the rocket falling on the land. The rocket has to be robust in order to survive the aerodynamic forces. The Mu is a very conservative design, and is highly reliable, having launched some 20 satellites successfully, with no failures.

In Figure 5.10, the launch sequence is shown in the form of curves for the velocity, acceleration and altitude, as functions of time. The Mu is a simple three-stage solid propellant rocket with two strap-on boosters. The boosters and the main motors all have cog-shaped charge voids to produce roughly constant thrust.

First stage

The main motor and the solid boosters both ignite at time zero, and the boosters burn for 30 seconds, and separate at zero + 40 seconds. The effect can be seen in Figure 5.10 as a dip in the first-stage acceleration curve at about 35 seconds. The dynamic pressure and pitch angle are shown in Figure 5.11. Maximum dynamic pressure occurs at 30 seconds. Note how the pitch angle is raised while this peak is passed, in order to reduce the angle of attack, while the thrust decreases as the boosters burn out. The rocket then pitches over under thrust vector control until the burn-out of the main motor of stage 1; the pitch angle is then 58 degrees. Note that the acceleration remains more or less constant during the first stage flight. This is the result of drag (proportional to V^2) in the lower part of the atmosphere, cancelling the effect of the reducing mass of the rocket.

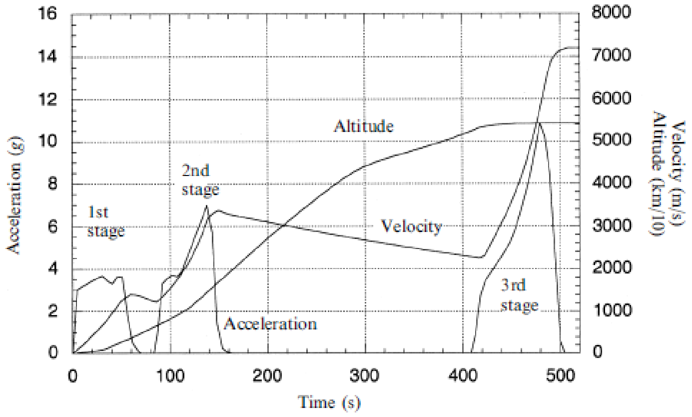


Figure 5.10. Velocity, acceleration and altitude as functions of time.

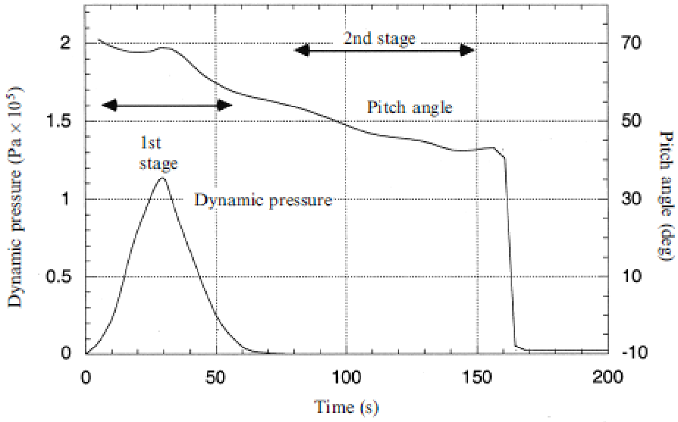


Figure 5.11. Dynamic pressure and pitch angle as functions of time.

Second stage

After first-stage separation there is a short coast phase, during which the velocity drops. Following second-stage ignition, the acceleration can be seen to increase rapidly as the mass of the rocket decreases; atmospheric drag is now small. The pitch angle is soon guided to about 45°, and then remains relatively constant. The objective here is to gain maximum velocity and altitude using the second stage. After burn-out the second stage uses small guidance rockets to pitch over until the axis is about 9° *below* the horizontal, which is the correct angle for the final injection into orbit by the third stage, some 1,100 km down-range, allowing for the curvature of the orbit. The second stage then spins up to 60 rpm to stabilise the third stage in this orientation prior to separation.

There is then a long gap in the time sequence before the third stage is ignited. This coast phase is occupied by the rocket in reaching the correct orbital altitude via a ballistic trajectory. Note that the orientation of the rocket is irrelevant during the coast phase. The kinetic energy imparted by the second stage is converted partially into potential energy, as can be seen in the decreasing velocity. This is a good example of the early and efficient burning of the fuel, to avoid the gravity loss associated with carrying unburned fuel to high altitude.

Third stage

At the peak of the ballistic path, where the residual velocity is purely horizontal, the third stage ignites and produces the increase in horizontal velocity necessary to secure the orbit. In another example of efficiency, this stage is unguided and has no thrust vector control; all the guidance is carried out by the second stage before separation. This maximises the mass ratio of the third stage by reducing its dry weight, and enables it to generate the necessary velocity efficiently, maximising the payload mass. The third stage ignites after separation. Its orientation has been set by the second stage, and is maintained by the stabilising effect of the 65 rpm spin. Note the very rapid acceleration; there is no gravity loss in this orientation, and the thrust-to-weight ratio is also high.

Following burn-out, the spacecraft is separated from the third stage, despins, and assumes autonomous existence for the life of the mission. The simple nature of the final injection means that there is some spread in the possible final velocity and direction, mostly from variability in the total velocity increment of the third-stage solid motor. This has to be allowed for in the specification of the orbit.

Ariane 4

The Mu is a very simple and robust design, intended to obtain the best performance from small solid-fuel rocket motors which have low exhaust velocity. This places a corresponding burden on the designer, to maximise efficiency, for small payloads into low Earth orbit (LEO). It serves well to illustrate the basic launcher dynamics discussed above. The Ariane rocket uses powerful liquid-fuelled motors to place large spacecraft in geostationary orbit. The guidance is more sophisticated, and the parallels with the simple dynamics we have discussed are not exact. Figure 5.12 shows the Ariane 4 dynamic parameters. While the Mu weighs about 92 tonnes and launches a payload of 0.5 tonnes into LEO, the Ariane weighs about 470 tonnes and launches a payload of 4.5 tonnes into geostationary transfer orbit (GTO).

The Ariane 44L—the largest in the Ariane 4 series—has four strap-on liquid-fuelled boosters, a first and second stage using storable liquid propellants, and a third stage with liquid hydrogen and liquid oxygen. The exhaust velocities are much higher than those of the Mu. The Mu launch lasts about 500 seconds and includes a coast phase, while for Ariane the time to injection is about 900 seconds. The Mu has burned both first and second stages during the time required for first-stage burn-out on Ariane, and the acceleration of Ariane is much less that of the Mu. 60 seconds after launch, when the Mu first stage has burned out and the velocity is 1.4 km/s, the Ariane velocity is only 200 m/s. By the time the first stage of Ariane has burnt out, the Mu velocity is 3.4 km/s, while Ariane has reached around 1.2 km/s.

First stage

The first stage burns for 204 seconds, and the boosters for 135 seconds. There is a short vertical segment lasting about 60 seconds, followed by a rapid guided gravity turn during which the pitch angle drops to 20° – much flatter than the Mu. The altitude gain is small because of this, and indeed peak altitude during injection is only 210 km. This results in a considerable saving in gravity loss.

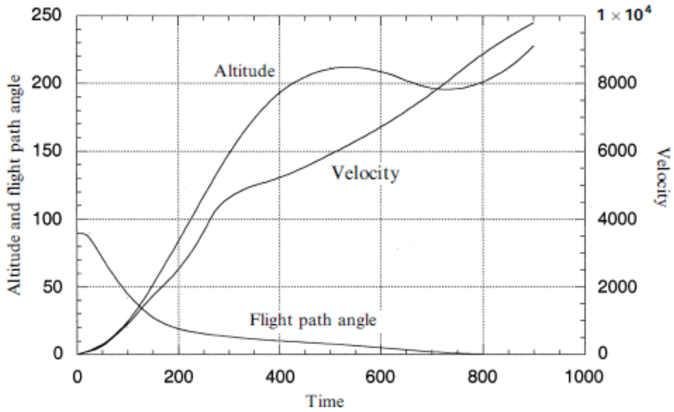


Figure 5.12. Ariane 4 dynamic parameters.

Second stage

The second stage burns for 124 seconds, the pitch angle decreases further while the velocity rises rapidly to 4.7 km/s, and the rocket climbs to 150 km. The fairing is jettisoned at a height of about 90 km.

Third stage

The cryogenic third stage has the task of increasing the vehicle velocity to more than 10 km/s—in order to place the payload in geostationary transfer orbit—an ellipse with apogee at 36,000 km. The acceleration is very slow, and never exceeds 1.75 g. The objective is to build up sufficient velocity to enter GTO. The rocket passes its maximum altitude at about 500 seconds and enters a 'dog-leg' manoeuvre during which some of the potential energy gained by early burning of fuel is converted to kinetic energy. The injection point is at a lower altitude of around 190 km. During the final stages the vehicle is accelerated both by the high exhaust velocity cryogenic engine and by gravity. This whole trajectory is designed to minimise gravity loss and to give maximum kinetic energy during the long burn of the third stage. Since the final orbit will be at geostationary altitude, there is no need for the initial injection point to be high. This approach is closer to the lunar launch strategy outlined above, with a low injection point to minimise gravity loss and potential energy requirements, and with no requirement of restart capability.

Pegasus

A contrasting flight path is adopted by the Pegasus small launcher, which is carried to significant altitude by an aircraft and launched horizontally. This uses the lift of the aircraft wings to gain the initial altitude and so reduce the expenditure of rocket propellant; being above the densest part of the atmosphere reduces drag and dynamic pressure. A further advantage is the ability to launch from anywhere, provided there is a suitable airfield nearby. The Pegasus parameters are illustrated in Figure 5.13.

First stage

The three-stage Pegasus is slung below a Lockheed L1011 aircraft and carried to an altitude of 11.6 km. It is then dropped from the aircraft, and glides for 5 seconds before first-stage ignition. For the whole of the first-stage burn it gains altitude and velocity as a hypersonic aircraft, passing through the maximum dynamic pressure after about 40 seconds. At this altitude the atmospheric pressure is of course much lower than at sea level, and the dynamic pressure at peak is therefore only 48.8 kPa. The flight path angle is initially a few degrees, and rises, after maximum dynamic pressure, to reach 33° by the end of the first-stage burn. The Pegasus gains velocity rapidly, reaching Mach 7.9 (2,300 m/s) by the time the first stage burns out after 76 seconds.

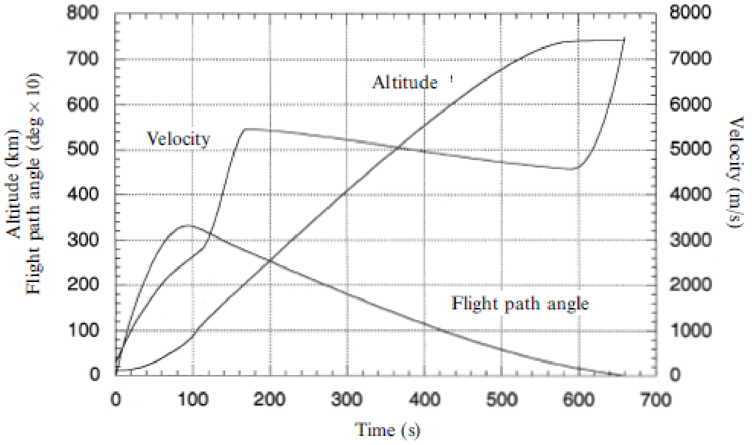


Figure 5.13. Pegasus dynamic parameters.

Second and third stages

Second-stage ignition takes place after 95 seconds, and both velocity and altitude increase until burn-out at 166 seconds. The vehicle then coasts until 594 seconds. During this coast phase, kinetic energy obtained by early burning of fuel is converted into potential energy. As the altitude increases to the required orbital insertion value (in this case, 740 km), the velocity drops from 5,469 m/s to 4,564 m/s. The third stage then ignites to generate the required insertion velocity of 7,487 m/s.

The Pegasus is an example of new launchers exploiting modern technology and filling a niche in the market, in this case for small satellites. The guidance throughout uses sophisticated control algorithms which constantly monitor the velocity and flight path and readjust the pitch angle to ensure optimum insertion. The high-altitude launch reduces dynamic pressure, and the low pitch angle throughout maximises horizontal velocity. Burning the fuel early and coasting to orbital altitude is a further efficiency gain.

Reviewing the other launchers, it is obvious that the first stages are very heavy, and in general burn out quite low in the atmosphere; boosters typically separate at about 30 km. The velocity is not very high because the flight is near vertical and because of atmospheric drag, and for these reasons it is inefficient to use a rocket to reach this altitude. The use of an aircraft is better, because lift is a very much more fuel-efficient way of gaining altitude than is thrust, and to launch Pegasus in this way is therefore a very sensible approach. Neither the velocity nor the altitude at launch are as large as can be achieved with boosters, but the saving is still considerable. Aircraft are, of course, not large enough to launch very big rockets, and can therefore only be used with small launchers.

4.9 Reusable Launch Vehicle

What is RLV?

- Conventional launch vehicle such as PSLV and GSLV can be launched once and then discarded because it burns out on re-entering the atmosphere.
- RLV is a system capable of launching payloads into space more than one time.
- RLV can launch spacecraft, including satellites, into space and re-enter the earth's atmosphere withstanding extreme pressure and heat conditions and land in the target spot.

Why RLV?

- The cost of access to space is the major deterrent in space exploration and space utilization mainly because of the launch vehicle and fuel costs.
- Therefore, RLV is considered to be the solution to achieve low cost, reliable and on-demand space access as it can be reused.

How does it work?

- The first stage or the special booster is powered using a solid fuel that carries the payload to about 70 km into the atmosphere and releases it. Then the descent begins.
- During the descent phase, small thrusters would help the vehicle navigate itself into the landing area. For the safe descent, RLV is steered by its navigation, guidance and control system.
- RLV is designed with special materials like special alloys, composites and insulation materials so that it can withstand extreme pressure and heat conditions while re-entering the atmosphere.
- The configuration of RLV is similar to that of an aircraft and combines the functionalities of both launch vehicles and aircraft for successful launch and descent.

What are the advantages of RLV and How India would benefit?

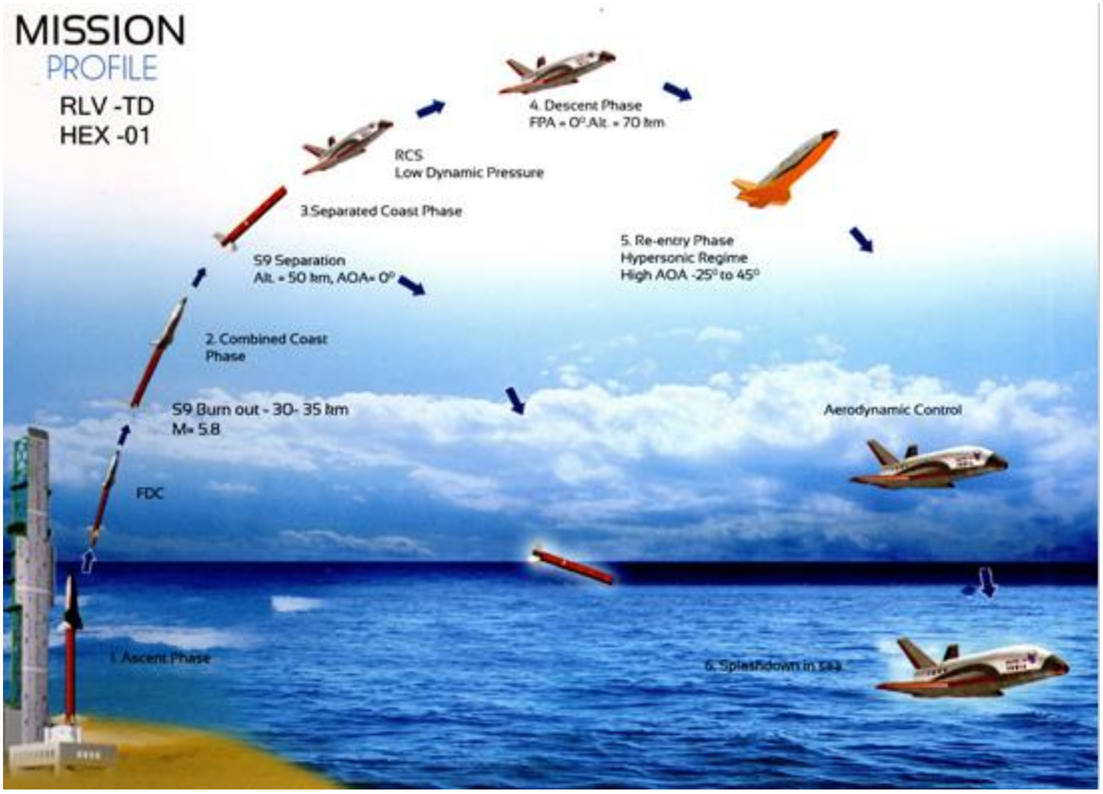
- Reusability of RLV will eliminate the cost to create a new launch vehicle every time.
- The technology used in RLV can also be used in other spacecraft, be it the manned mission to Moon or Mars. Thus it will help economize time and cost.
- Due to cost-effectiveness and reduced operation cost, India will attract more foreign business to launch their satellites.
- As it can be reused again, it decreases the growing space debris (remember normal LVs burns out on re-entering the atmosphere = leaving some debris). Therefore, RLV is seen as preferred clean space

technology at the international level, thus boosting Indian's space sector further.

- It will boost ISRO's credentials further, and motivate other space agencies to work together with ISRO.
- With RLV, India will join a select group of nations having their own space flights.
  - USA – Columbia, Challenger, Discovery, Endeavour, and Atlantis
  - Russia – Soyuz
  - China – Shenzhou
- India can use the RLV to launch the satellites of smaller neighbours with cheap cost. Thus India would get a geo-strategic advantage with RLV.

What are the disadvantages of RLVs?

- They cost higher than expendable launch vehicles because getting back to Earth safely requires more propellants and more hardware.
- Fuel cannot be reused in the rocket as it is released into the atmosphere at an exhaust rate of 300 pounds/second.
- They pollute the atmosphere more since they emit CO2 at stratosphere and mesosphere layer.
- Due to cheaper launch cost, there will be more launches at frequent intervals. This leads to more exhaust and more harm to the atmosphere.



4.10 Rocket Thrust Vector Control

In addition to providing a propulsive force to a flying vehicle, a rocket propulsion system can provide moments to rotate the flying vehicle and thus provide control of the vehicle's attitude and flight path. By controlling the direction of the thrust vectors through the mechanisms described later in the chapter, it is possible to control a vehicle's pitch, yaw, and roll motions. All chemical propulsion systems can be provided with one of several types of thrust vector control (TVC) mechanisms. Some of these apply either to solid, hybrid, or to liquid propellant rocket propulsion systems, but most are specific to only one of these propulsion categories. We will describe two types of thrust vector control concept:

- For an engine or a motor with a single nozzle; and
- For those that have two or more nozzles.

Thrust vector control is effective only while the propulsion system is operating and creating an exhaust jet. For the flight period, when a rocket propulsion system is not firing and therefore its TVC is inoperative, a separate mechanism needs to be provided to the flying vehicle for achieving control over its attitude or flight path.

Aerodynamic fins (fixed and movable) continue to be very effective for controlling vehicle flight within the earth's atmosphere, and almost all weather rockets, antiaircraft missiles, and air-to-surface missiles use them. Even though aerodynamic control surfaces provide some additional drag, their effectiveness in terms of vehicle weight, turning moment, and actuating power consumption is difficult to surpass with any other flight control method. Vehicle flight control can also be achieved by a separate attitude control propulsion system. Here six or more small liquid propellant thrusters (with a separate feed system and a separate control) provide small moments to the vehicle in flight during, before, or after the operation of the main rocket propulsion system.

The reasons for TVC are:

- To willfully change a flight path or trajectory (e.g., changing the direction of the flight path of a target-seeking missile);
- To rotate the vehicle or change its attitude during powered flight;
- To correct for deviation from the intended trajectory or the attitude during powered flight; or
- To correct for thrust misalignment of a fixed nozzle in the main propulsion system during its operation, when the main thrust vector misses the vehicle's center of gravity.

Pitch moments are those that raise or lower the nose of a vehicle; yaw moments turn the nose sideways; and roll moments are applied about the main axis of the flying vehicle (Fig. 16-1). Usually, the thrust vector of the main rocket nozzle is in the direction of the vehicle axis and goes through the vehicle's center of gravity. Thus it is possible to obtain pitch and yaw control moments by the simple deflection of the main rocket thrust vector; however, roll control usually requires the use of two or more rotary vanes or two or more separately hinged propulsion system nozzles. Figure 16-2 explains the pitch moment obtained by a hinged thrust chamber or nozzle. The side force and the pitch moment vary as the sine of the effective angle of thrust vector deflection.

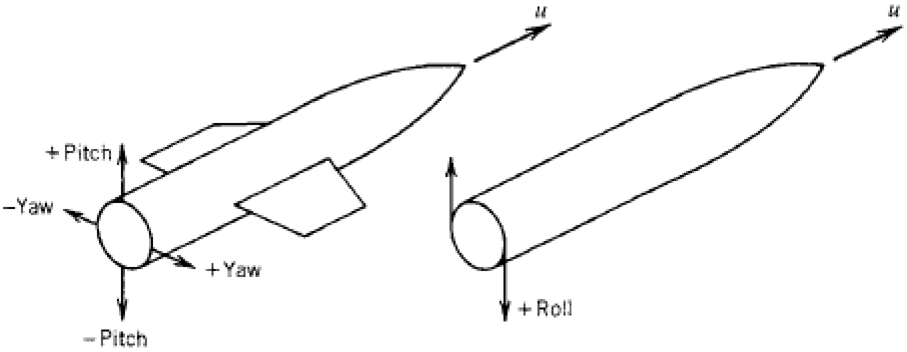


FIGURE 16–1. Moments applied to a flying vehicle.

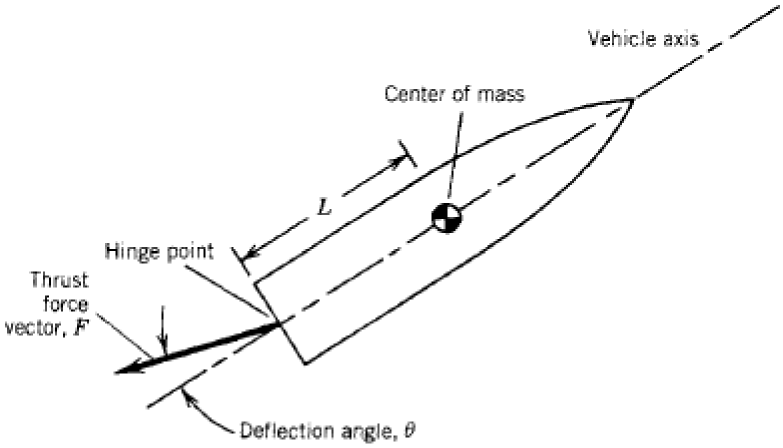


FIGURE 16–2. The pitch moment applied to the vehicle is $FL \sin\theta$.

4.11 TVC Mechanisms with a Single Nozzle

Many different mechanisms have been used successfully. ***They can be classified into four categories:***

- Mechanical deflection of the nozzle or thrust chamber.
- Insertion of heat-resistant movable bodies into the exhaust jet; these experience aerodynamic forces and cause a deflection of a part of the exhaust gas flow.
- Injection of fluid into the side of the diverging nozzle section, causing an asymmetrical distortion of the supersonic exhaust flow.
- Separate thrust-producing devices that are not part of the main flow through the nozzle.

Each ***category is described briefly below and in Table 16-1,*** where the four categories are separated by horizontal lines. ***Figure 16-3 illustrates several TVC mechanisms.*** All of the TVC schemes shown here have been used in production vehicles.

TABLE 16–1. Thrust Vector Control Mechanisms

Type	L/S[a]	Advantages	Disadvantages
Gimbal or hinge	L	Simple, proven technology; low torques, low power; ±12° duration limited only by propellant supply; very small thrust loss	Requires flexible piping; high inertia; large actuators for high slew rate
Movable nozzle (flexible bearing)	S	Proven technology; no sliding, moving seals; predictable actuation power; up to ±12°	High actuation forces; high torque at low temperatures; variable actuation force
Movable nozzle (rotary ball with gas seal)	S	Proven technology; no thrust loss if entire nozzle is moved; ±20° possible	Sliding, moving hot gas spherical seal; highly variable actuation power; limited duration; needs continuous load to maintain seal
Jet vanes	L/S	Proven technology; low actuation power; high slew rate; roll control with single nozzle; ±9°	Thrust loss of 0.5 to 3%; erosion of jet vanes; limited duration; extends missile length
Jet tabs	S	Proven technology; high slew rate; low actuation power; compact package	Erosion of tabs; thrust loss, but only when tab is in the jet; limited duration
Jetavator	S	Proven on Polaris missile; low actuation power; can be lightweight	Erosion and thrust loss; induces vehicle base hot gas recirculation; limited duration
Liquid-side injection	S/L	Proven technology; specific impulse of injectant nearly offsets weight penalty; high slew rate; easy to adapt to various motors; can check out before flight; components are reusable; duration limited by liquid supply; ±6°	Toxic liquids are needed for high performance; often difficult packaging for tanks and feed system; sometimes requires excessive maintenance; potential spills and toxic fumes with some propellants; limited to low vector angle applications
Hot-gas-side injection	S/L	Lightweight; low actuation power; high slew rate; low volume/compact; low performance loss	Multiple hot sliding contacts and seals in hot gas valve; hot piping expansion; limited duration; requires special hot gas valves; technology is not yet proven
Hinged auxiliary thrust chambers for high thrust engine	L	Proven technology; feed from main turbopump; low performance loss; compact; low actuation power; no hot moving surfaces; unlimited duration	Additional components and complexity; moments applied to vehicle are small; not used for 15 years in USA
Turbine exhaust gas swivel for large engine	L	Swivel joint is at low pressure; low performance loss; lightweight; proven technology	Limited side forces; moderately hot swivel joint; used for roll control only

[a]L, used with liquid propellant engines; S, used with solid propellant motors.

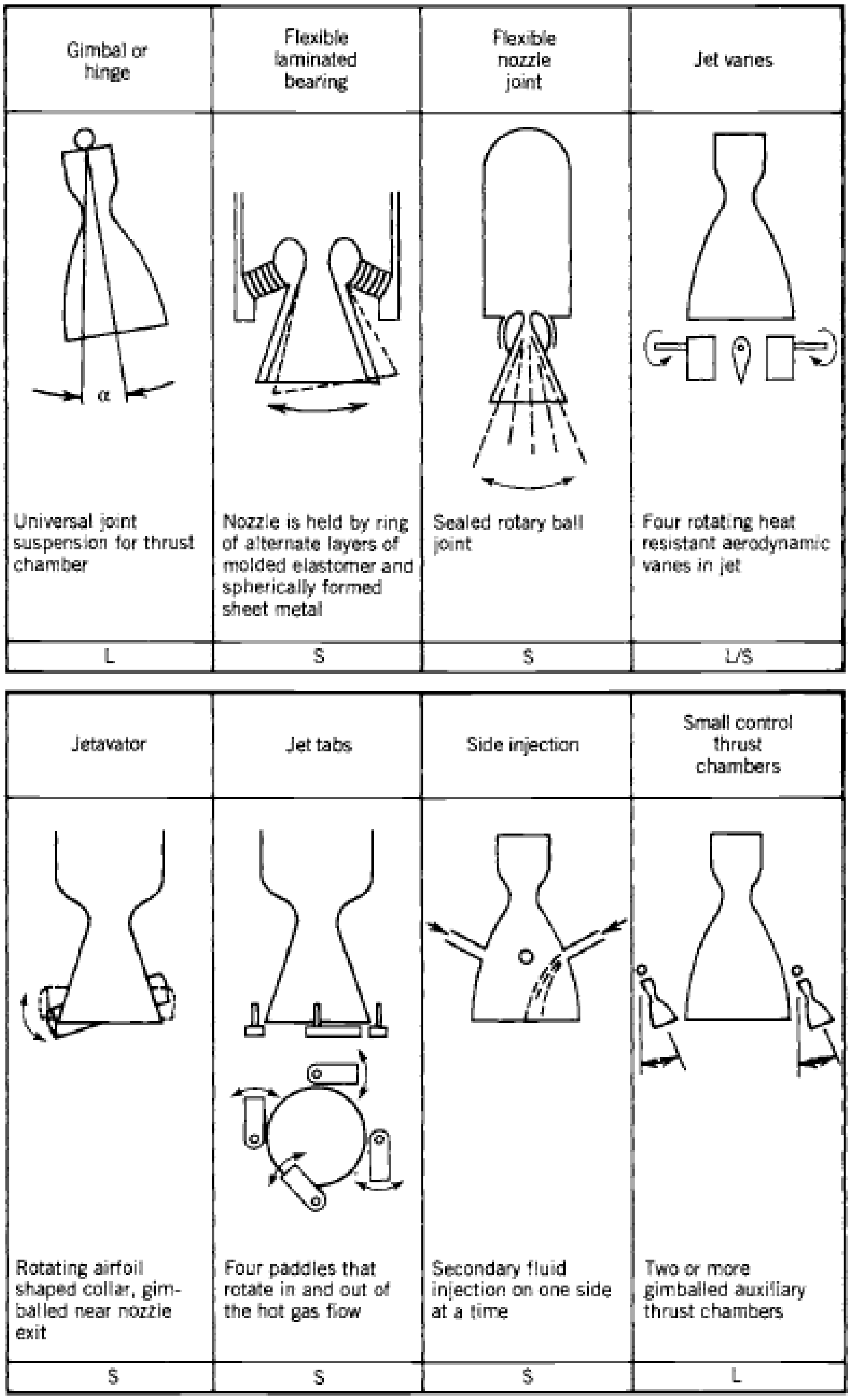


FIGURE 16–3. Simple schematic diagrams of eight different TVC mechanisms. Actuators and structural details are not shown. The letter L means it is used with liquid propellant rocket engines and S means it is used with solid propellant motors.

In the ***hinge or gimbal*** scheme (a hinge permits rotation about one axis only, whereas a gimbal is essentially a universal joint), the whole engine is pivoted on a bearing and thus the thrust vector is rotated. For small angles this scheme has negligible losses in specific impulse and is used in many vehicles. It requires a flexible set of propellant piping (bellows) to allow the propellant to flow from the tanks of the vehicle to the movable engine.

Jet vanes are pairs of heat-resistant, aerodynamic wing-shaped surfaces submerged in the exhaust jet of a fixed rocket nozzle. They were first used about 55 years ago. They cause extra drag (2 to 5% less Is; drag increases with larger vane deflections) and erosion of the vane material. Graphite jet vanes were used in the German V-2 missile in World War II and in the Scud missiles fired by Iraq in 1991. The advantage of having roll control with a single nozzle often outweighs the performance penalties.

Small auxiliary thrust chambers were used in the Thor and early version of Atlas missiles. They provide roll control while the principal rocket engine operates. They are fed from the same feed system as the main rocket engine. This scheme is still used on some Russian booster rocket vehicles.

The ***injection of secondary fluid*** through the wall of the nozzle into the main gas stream has the effect of forming oblique shocks in the nozzle diverging section, thus causing an unsymmetrical distribution of the main gas flow, which produces a side force. The secondary fluid can be stored liquid or gas from a separate hot gas generator (the gas would then still be sufficiently cool to be piped), a direct bleed from the chamber, or the injection of a catalyzed monopropellant. When the deflections are small, this is a low-loss scheme, but for large moments (large side forces) the amount of secondary fluid becomes excessive. This scheme has found application in a few large solid propellant rockets, such as Titan IIIC and one version of Minuteman.

The ***jet tab*** TVC system has low torque, and is simple for flight vehicles with low-area-ratio nozzles. Its thrust loss is high when tabs are rotated at full angle into the jet, but is zero when the tabs are in their neutral position outside of the jet. On most flights the time-averaged position of the tab is a very small angle and the average thrust loss is small. Jet tabs can form a very compact mechanism and have been used successfully on tactical missiles.

The ***jetavator*** was used on submarine-launched missiles. The thrust loss is roughly proportional to the vector angle.

4.12 TVC with Multiple Thrust Chambers or Nozzles

Several concepts have been developed and flown that use two or more rocket engines or a single engine or motor with two or more actuated nozzles. Two fully gimballed thrust chambers or motor nozzles can provide roll control with very slight differential angular deflections. For pitch and yaw control, the deflection would be larger, be of the same angle and direction for both nozzles, and the deflection magnitude would be the same for both nozzles. This can also be achieved with four hinged or gimbaled nozzles.

The ***differential throttling concept*** shown in Fig. 16-10 has no gimbal and does not use any of the methods used with single nozzles as described in Fig. 16-3. It has four fixed thrust chambers and their axes are almost parallel to and set off from the vehicle's centerline. Two of the four thrust chambers are selectively throttled (typically the thrust is reduced by only 2 to 15 %). The four nozzles may be supplied from the same feed system or they may belong to four separate but identical rocket engines. This differential throttling system is used on the Aerospike rocket engine and on a Russian launch vehicle.

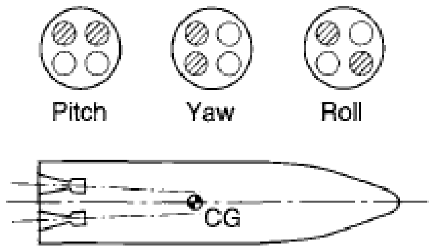


FIGURE 16–10. Differential throttling with four fixed-position thrust chambers can provide flight maneuvers. In this simple diagram the shaded nozzle exits indicate a throttled condition or reduced thrust. The larger forces from the unthrottled engines impose turning moments on the vehicle. For roll control the nozzles are slightly inclined and their individual thrust vectors do not go through the center of gravity of the vehicle.

4.13 Testing of TVC

Testing of thrust vector control systems often includes actuation of the system when assembled on the propulsion system and the vehicle. For example, the Space Shuttle main engine can be put through some gimbal motions (without rocket firing) prior to a flight. A typical acceptance test series of the TVC system (prior to the delivery to an engine manufacturer) may include the determination of input power, accuracy of deflected positions, angular speeds or accelerations, signal response characteristics, or validation of over travel stops. The ability to operate under extreme thermal environment, operation under various vehicle or propulsion system generated vibrations, temperature cycling, and ignition shock (high momentary acceleration) would probably be a part of the qualification tests.

Side forces and roll torques are usually relatively small compared to the main thrust and the pitch or yaw torques. Their accurate static test measurement can be difficult, particularly at low vector angles. Elaborate, multicomponent test stands employing multiple load cells and isolation flexures are needed to assure valid measurements.

4.14 TVC Integration with Vehicle

The actuations or movements of the TVC system are directed by the vehicle's guidance and control system. This system measures the three dimensional position, velocity vectors, and rotational rates of the vehicle and compares them with the desired position, velocity, and rates. The error signals between these two sets of parameters are transformed by computers in TVC controllers into control commands for actuating the TVC system until the error signals are reduced to zero. The vehicle's computer control system determines the timing of the actuation, the direction, and magnitude of the deflection. With servomechanisms, power supplies, monitoring/failure detection devices, actuators with their controllers, and kinetic compensation, the systems tend to become complex.

The criteria governing the selection and design of a TVC system stem from vehicle needs and include the steering-force moments, force rates of change, flight accelerations, duration, performance losses, dimensional and weight limitations, available vehicle power, reliability, delivery schedules, and cost. For the TVC designer these translate into such factors as duty cycle, deflection angle, angle slew rate, power requirement, kinematic position errors, and many vehicle-TVC and motor-TVC interface details, besides the program aspects of costs and delivery schedules.

Interface details include electrical connections to and from the vehicle flight controller, the power supply, mechanical attachment with fasteners for actuators, and sensors to measure the position of the thrust axis or the actuators. Design features to facilitate the testing of the TVC system, easy access for checkout or repair, or to facilitate resistance to a high-vibration environment, are usually included. The TVC subsystem is usually physically connected to the vehicle and mounted to the rocket's nozzle. The designs of these components must be coordinated and integrated.

The actuators can be hydraulic, pneumatic, or electromechemical (lead screw), and usually include a position sensor to allow feedback to the controller. The proven power supplies include high-pressure cold stored gas, batteries, warm gas from a gas generator, hydraulic fluid pressurized by cold gas or a warm gas generator, electric or hydraulic power from the vehicle's power supply, and electric or hydraulic power from a separate turbogenerator (in turn driven by a gas generator). The last type is used for relatively long-duration high-power applications, such as the power package used in the Space Shuttle solid rocket booster TVC. The selection of the actuation scheme and its power supply depends on the minimum weight, minimum performance loss, simple controls, ruggedness, reliability, ease of integration, linearity between actuating force and vehicle moments, cost, and other factors. The required frequency response is higher if the vehicle is small, such as with small tactical missiles. Sometimes the TVC system is integrated with a movable aerodynamic fin system.

4.15 Thrust Termination

The engine thrust must be cut off, the instant that proper velocity is achieved to conserve fuel and or the rocket obtained a desired orbit. The thrust termination in liquid propellant rocket engines is easily accomplished by closing the fuel valve.

But in solid propellant motors the problem is more difficult. One method to terminate the thrust in solid motor is the rupture disks, which vent the combustion chamber, reducing the thrust to zero.

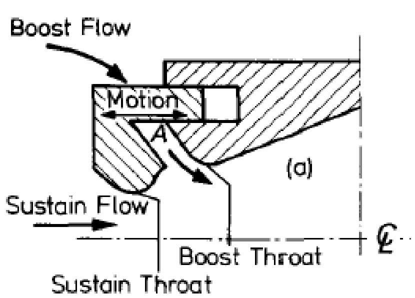
Recent development has indicated that thrust termination can be affected by nozzle blow out or by blowing out the forward section of the combustion chamber. By this means the combustion chamber pressure can be reduced below that required for sustained burning and hence terminate the thrust.

Careful design must be made in order that random re ignition does not occur once burning is stopped. The accomplishment of thrust termination on TVC paves the way for application of this type of motor to ballistic missiles, which require thrust cut off for different range missions. It has been an error in the burn out velocity that has a large detrimental effect of the accuracy of the ballistic missiles. Hence it is important that the thrust termination of the propellant unit can be accurately accomplished and with a good degree of repeatability for one motor to another.

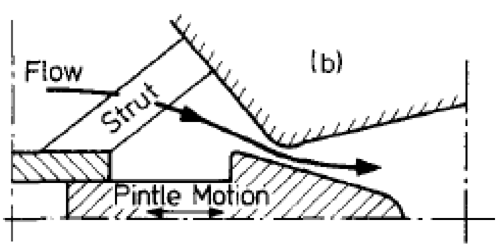
4.16 Thrust Magnitude Control

Thrust Magnitude Control (TMC) allows for large thrust variations usually with small variations in chamber pressure.

TRANSLATING INLET NOZZLE



PINTLE NOZZLE



In some solid propellant rocket motors, TMC has been used without varying the throat area by reducing the mass flow into the chamber. As a result of the reduced mass flow, the chamber pressure decreases too. This may cause irregular combustion, or even extinguishments. Apart from this, the exhaust velocity is also lowered. Two possible systems without these adverse effects are the translating inlet nozzle and the pintle nozzle. Both systems vary the throat to modulate the thrust. The translating nozzle is primarily designed for two different thrust magnitudes. In the figure port A is either closed or fully opened. If the port A is closed, the sustain throat is the only way through which propulsive gases can leave the rocket engine. If port A is opened, an extra boost flow can leave the combustion chamber and the boost throat acts as a nozzle throat.

The pintle nozzle employs a centre body that can move in an axial direction; thereby continuously vary in throat area. The central body, which holds the movable pintle, is mounted on the nozzle inlet. It is of course, possible to combine TVC and TMC to obtain real thrust vector control, i.e both magnitude and direction of the thrust can be varied.

Another TMC device that should be mentioned in this section is the extendable exit cone. If during powered flight under expansion losses become unacceptably large, one can increase the thrust by lengthening the exit cone. This may be done by moving aft an extension to the divergent part of the nozzle. This concept was planned for the space shuttle engine but has be abandoned to keep the mechanism simple.

4.17 Stage Separation Dynamics/Techniques

In multistage launch vehicles the stage separation process is broadly classified into two categories. They are,

4.17.1 Separation Occurring Within the Atmosphere

Separation within the atmosphere is otherwise known as booster separation/lower stage separation/strapton separation.

The burn out of the first stage generally occurs within the upper regions of the atmosphere (i.e) 45km to 60km, to minimize the energy lost due to the aerodynamic forces. The ignition of the second stage must be done as quickly as possible after the first stage burnout. There are two techniques of separation are avail within the atmosphere. They are,

Firing In the Hole Technique:

Firing in the hole staging is also known as vented inter stage separation or hot separation. This technique involves the firing of the upper stage motor before the thrust level of the lower stage motor has decayed to zero (i.e. before the actual separation takes place). Drawbacks:

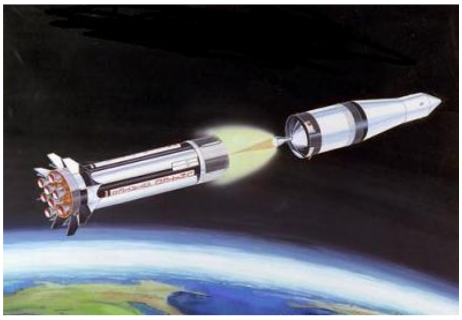
- Care must be taken
- Adequate ventilation holes are provided in the structure of the lower stage separation bay to prevent an excessive buildup of pressure from the jet efflux which might cause the rupture of lower stage tanks.
- In practice even though burnout conditions have been reached, the tanks still usually consists of unusable propellant, which may cause hazard.
- There is a risk of tank rupture by direct jet impingement. So the upper surfaces of the tank should be stronger and hence heavier which imposes additional weight penalty.

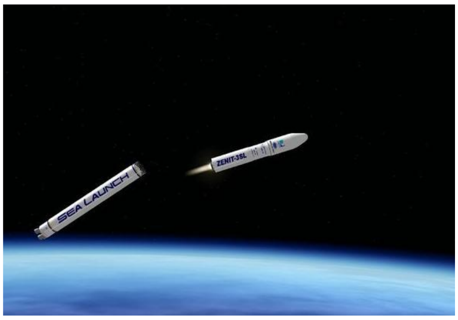
Ullage Rocket Technique:

This technique involves the use of short duration solid propellant rockets which are called ullage rockets, to bridge the gap caused by the decay of lower stage thrust and subsequent buildup of lower stage thrust. The nominal thrust level of the upper stage motor is not reached until there is an appreciable separation distance between the two stages. Drawback:

- Heavy weight.

4.17.2 Separation Out of the Atmosphere (In Space)





Separation occurring out of atmosphere is also known as vacuum/space/upper stage separation. The separation of subsequent stages takes place either at extreme high altitudes in space. The problem of separation is relatively simplified when occurs in space because of absence of aerodynamic forces but it does not mean as soon as burn out of lower stage occurs the ignition of upper stage is initiated. This separation technique involves two methods:

Helical Compression Spring Technology:

In this technology, separation may be obtained by a single compression spring centrally located but in practice it was a large number of small springs located symmetrically around the periphery. This is done in case of accommodation and secondly to minimize the possibility of separation aborting through spring failure.

Advantages:

- No separate command is needed for actuation
- Highly reliable

Draw Back:

- Much heavier when compared to other jettisoning system.