APPLICATION: IDEAL ROCKET NOZZLE

Flow through an ideal rocket nozzle is everywhere isentropic. Gas comes out of the nozzle with a pressure that matches that of the surroundings. The combustion chamber is taken to be large. Let U denote the combustion chamber, D denote the nozzle exit and T denote the throat. The Mach Number in the combustion chamber is approximately zero and the Mach Number at the throat is unity. The thrust generated by the nozzle is

$$\dot{M} U_D$$

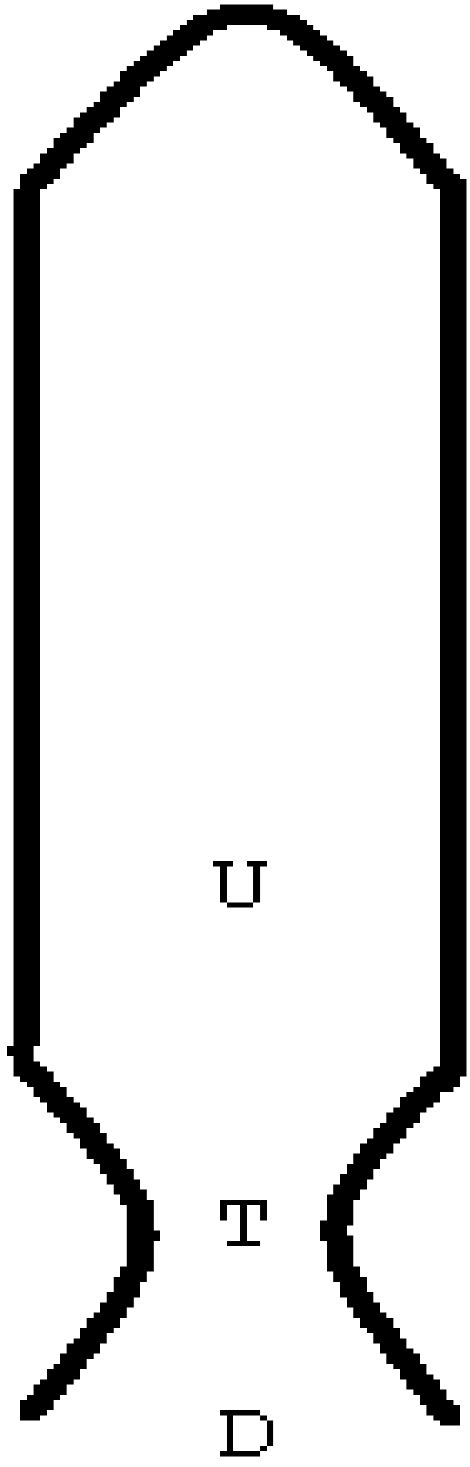
The mass flow rate anywhere in the nozzle is

$$\dot{M} = \rho A U$$

With known pressure and temperature in the combustion chamber, the isentropic ratios can be used to get pressure and temperature at the throat. One can then write

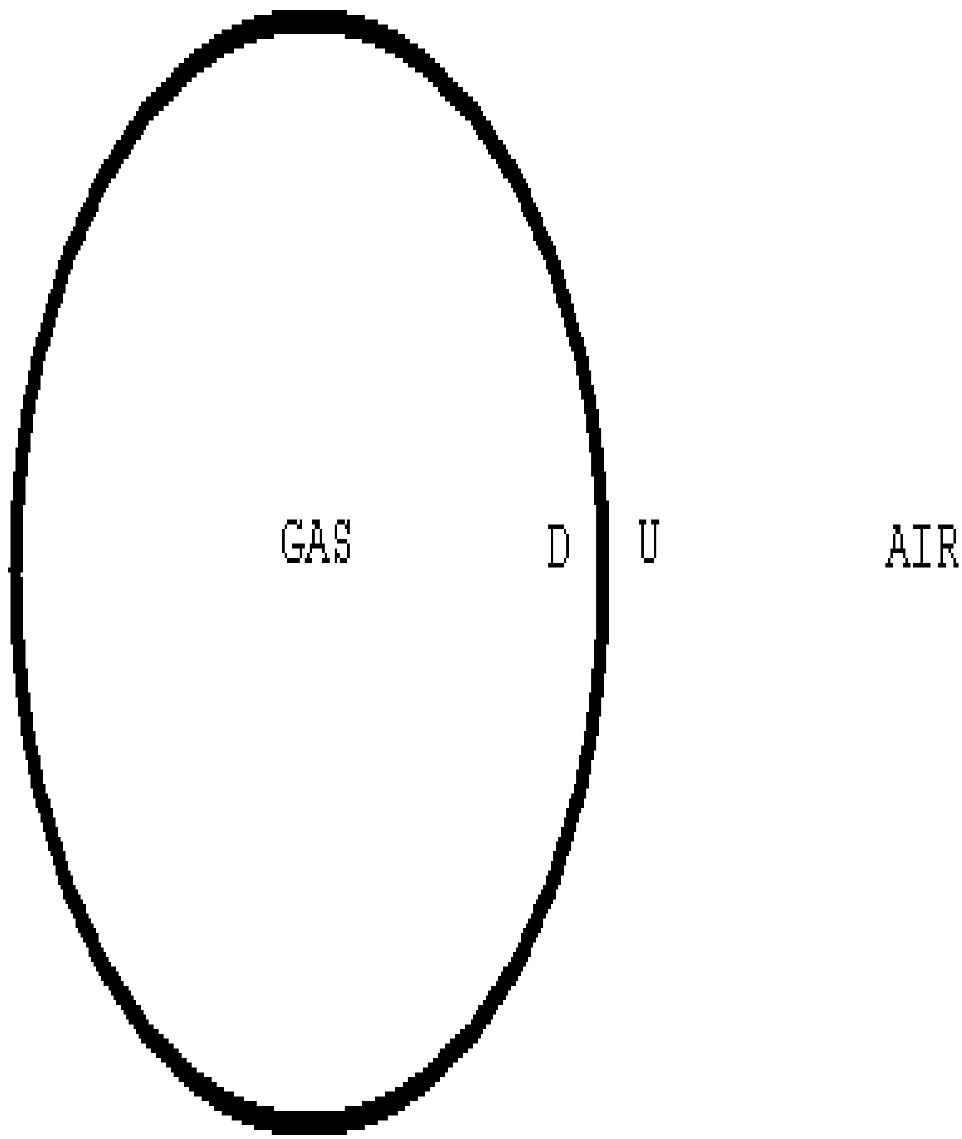
$$\dot{M} = P_T/[RT_T]\ A_T\ \sqrt{[kRT_T]}$$

With the combustion chamber and nozzle exit pressures known, the pressure ratio equation gives the Mach Number M_D at the exit. The temperature ratio equation then gives the temperature T_D at the exit. This allows us to calculate the sound speed C_D at the exit. With known Mach Number and sound speed, one can calculate the gas speed U_D at the exit.


U
T
D

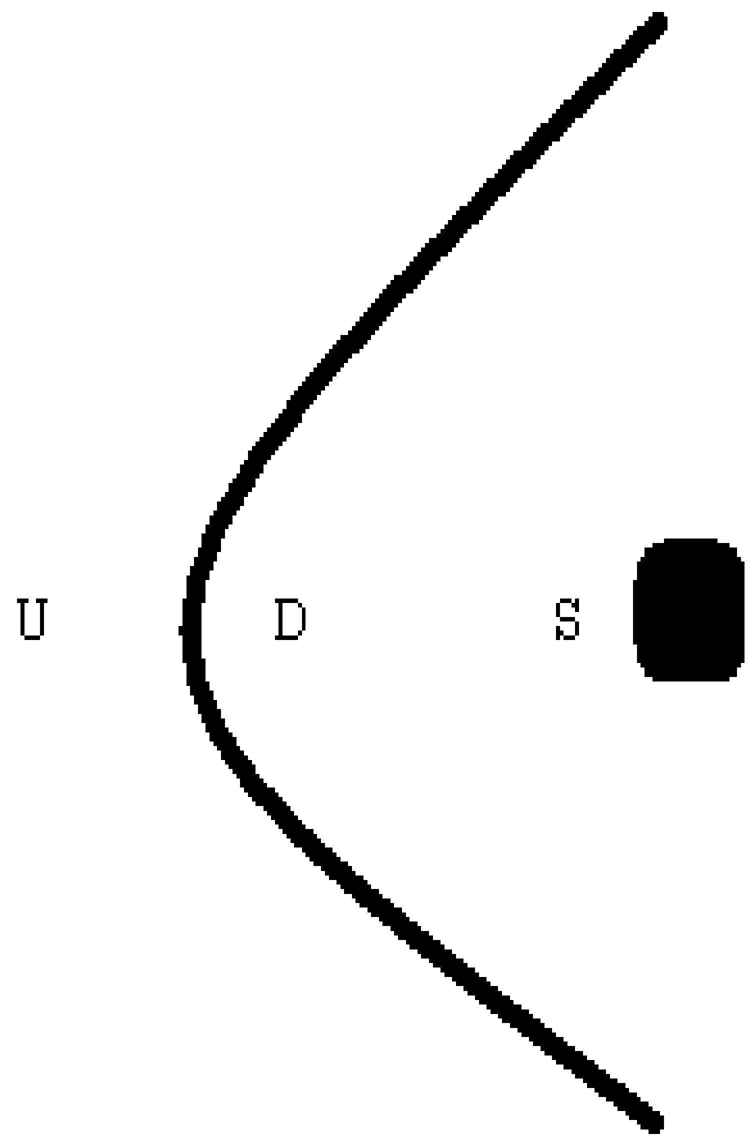
APPLICATION: SHOCKS DUE TO EXPLOSIONS

Consider an explosion in still air with known pressure and temperature. Let the gas ball pressure be known. The explosion generates a spherical shock wave. Let U indicate just upstream of the shock and D indicate just downstream of the shock. In a reference frame moving with the shock, the still air seems to approach at a supersonic speed. With known air and gas pressures, the normal shock equation gives the Mach Number M_U of the shock. With the still air temperature T_U known, one can calculate the sound speed front of the shock. This allows us to calculate the shock speed U_U. The Mach Number connection gives the Mach Number M_D downstream relative to the shock. The temperature ratio equation gives the temperature T_D downstream of the shock. This allows us to calculate the sound speed C_D there. With known Mach Number and sound speed, one can calculate the flow speed U_D relative to the shock. The absolute flow speed U_A downstream is equal to the shock speed U_U minus the flow speed U_D downstream relative to the shock. This is known as the drift speed. It is a supersonic speed.


GAS
D
U
AIR

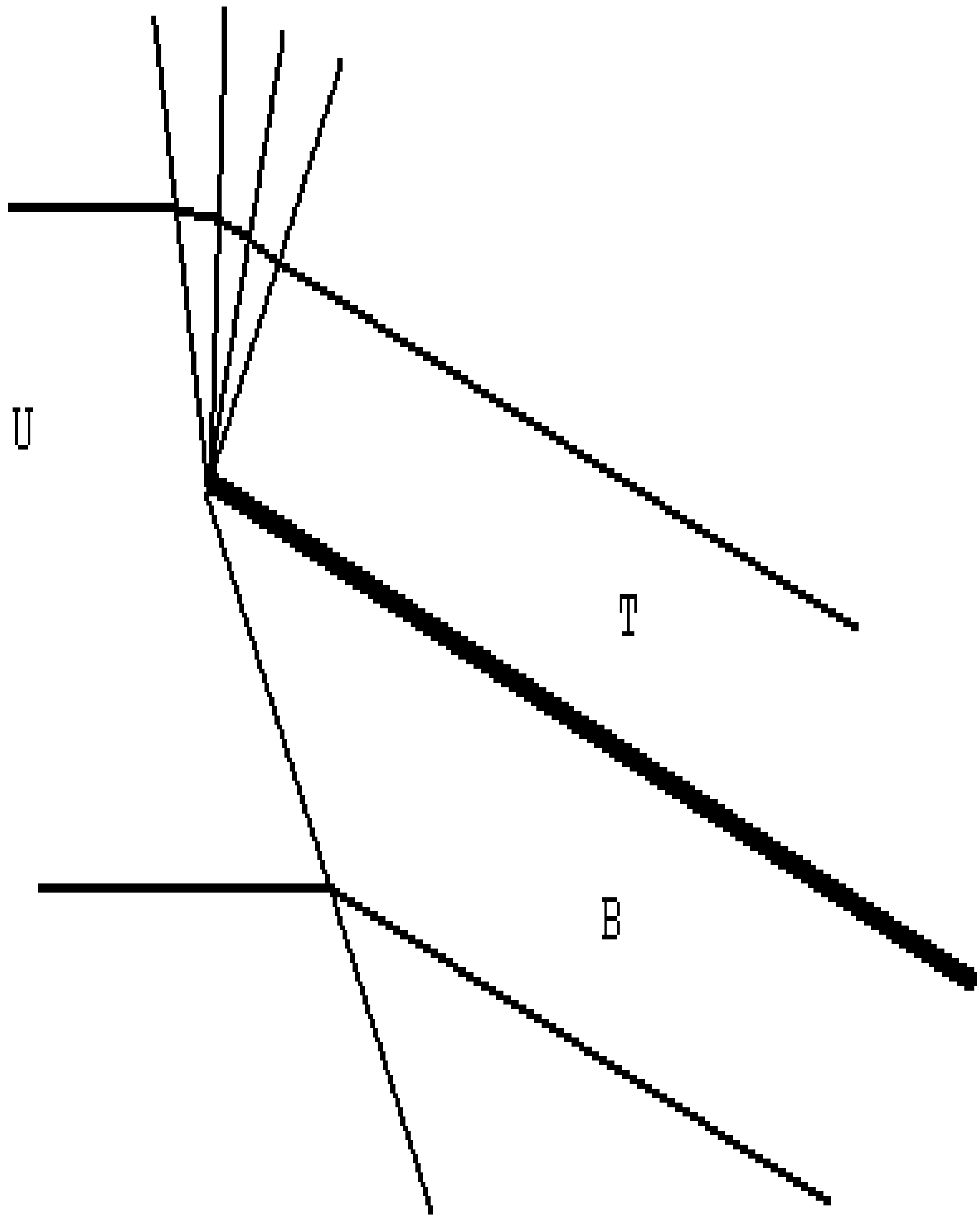
APPLICATION: BLUNT OBJECT IN SUPERSONIC FLOW

Consider a blunt object in a supersonic flow. A bow shock wave forms upstream of the object. Directly in front of the object this shock wave is a normal shock wave. Let U indicate just upstream of the shock and D indicate just downstream of the shock. Let S indicate the stagnation point on the object. Assume that the pressure P_U and temperature T_U upstream of the shock are known and that the Mach Number M_U of the shock is also known. One can use the pressure ratio equation for a normal shock wave to find the pressure P_D just downstream of the shock. One can then use the pressure ratio equation for isentropic flow to get the pressure P_S at the stagnation point. One can use the temperature ratio equation from upstream to the stagnation point to get the temperature T_S at the stagnation point.


U
D
S

APPLICATION: SUPERSONIC FLAT PLATE FOIL

Consider a supersonic flat plate foil moving at a known Mach Number through still air with known pressure and temperature. When the foil has a moderate angle of attack, an oblique shock wave forms below it and an expansion wave forms above it. These turn the flow parallel to the plate. Let U indicate conditions upstream of the foil. Let T indicate conditions top of the foil and B indicate conditions bottom of the foil. One can use the expansion wave plot to find the Mach Number M_T on top of the foil. With known Mach Number there and upstream of the foil, the isentropic pressure ratio equation gives the pressure P_T there. One can use the oblique shock plot to find the shock angle β. One can use this to get the Mach Number N_U of the normal component of the flow. Substitution into the pressure ratio equation for oblique shock waves then gives the pressure P_B below the foil. With pressure top and bottom of the foil known, one can then calculate the lift.


U
T
B