7 High-Speed Supersonic and Hypersonic Engines

7.1 INTRODUCTION

In the previous three chapters, four broad categories of aero engines were discussed. The powered vehicles have flight speeds ranging from low subsonic speeds, as in the case of helicopters, and up to moderate supersonic speeds. Supersonic vehicles are mainly military airplanes and the limited number of supersonic transports (SSTs) such as the Concorde and Tu-144 airplanes. Passengers become bored by the long time of trips, say across Atlantic or in general during flights of more than 5–7 hours. Extensive efforts are made these days to satisfy the passenger needs. It is expected in the forthcoming decades that airplanes will satisfy the greatest demand for more comfortable and faster flights. This chapter will concentrate on engines that power such airplanes or airspace planes that will fly at high supersonic and hypersonic flight speeds (Mach 5 and above). The corresponding flight altitudes in such circumstances will be much higher than today's subsonic/transonic civil airliners for some technical reasons, which will be discussed later.

In this chapter, two engines will be analyzed, namely, turboramjet and scramjet. Both were highlighted in Chapter 1. Turboramjet either combines a turbojet/turbofan and a ramjet engine in a single conduit or assembles them in over/under configuration and sometimes identified as turbine-based engine. In Chapters 3 through 5, the ramjet, turbojet, and turbofan engines were analyzed. Here, a brief discussion will be made with a special analysis of their combined layout and combined operation in the dual mode.

Next, scramjet will be analyzed. Similarity with ramjet will be emphasized. Scramjet (sometimes identified as ram-based engine) will be employed in hypersonic vehicles that must be boosted at first by other engines to reach the starting Mach number for its operation.

7.2 SUPERSONIC AIRCRAFT AND PROGRAMS

On October 14, 1947, the Bell X-1 [1] became the first airplane to fly faster than the speed of sound. Piloted by the U.S. Air Force Captain Charles E. Yeager, the X-1 reached a speed of 1127 km (700 mi) per hour, Mach 1.06, at an altitude of 13,000 m (43,000 ft).

It was the first aircraft to have the designation X. The X-1 was air launched at an altitude of approximately 20,000 ft from the Boeing B-29.

With that flight, the supersonic age had started. The second generation of Bell X-1 was modified and redesignated the X-1E (Figure 7.1). It reached a Mach number of 2.44 at an altitude of 90,000 ft. Later on, many aircraft joined the supersonic race. Until now, the fastest manned aircraft is the Blackbird SR-71 (Figure 7.2) long-range, strategic reconnaissance aircraft that can fly at Mach number of 3+ (maximum speed 3.3+ (2200+ mph, 3530+ km/h) at 80,000 ft (24,000 m)). It is powered by 2 Pratt & Whitney J58-1, continuous-bleed afterburning turbojets 32,500 lb force (145 kN) each [2,18]. It was in service from 1964 to 1998.

Continuous R &D programs and testing are performed in the United States, France, Russia, United Kingdom, and Japan. However, the first supersonic programs started by National Advisory Committee for Aeronautics (NACA) first conceptual design of a supersonic commercial transport was designed to carry 10 passengers for a range of about 1500 miles in a cruise speed of 1.5 Mach number.

After that design, there were a lot of other conceptual designs of supersonic commercial aircraft. Very few designs reached completion such as Concorde and Tu-144. Many others were only tested

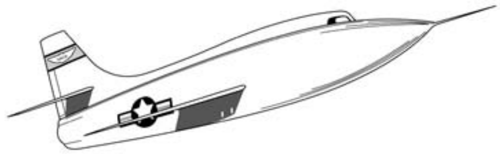

FIGURE 7.1 The first supersonic aircraft X-1E.

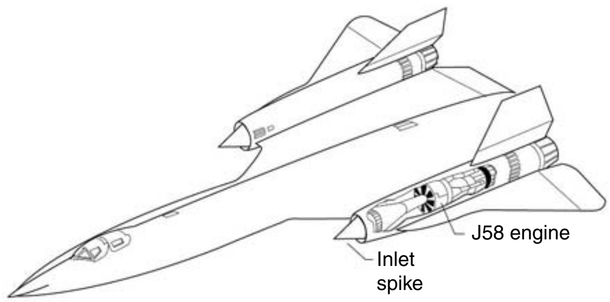

FIGURE 7.2 The fastest manned aircraft SR-71.

in wide tunnels such as Boeing 2707-100, 200, 300, and the Lockheed L-2000. The research on the supersonic commercial air transport started in the United States in 1958 and somewhat earlier in France and Great Britain. The starting date of the Russian project is unknown. A brief description of several programs will be given in the following sections.

7.2.1 Anglo-French Activities

Concorde

The "supersonic transport aircraft committee (STAC)" was established in 1956. STAC conducted a series of design studies, leading to the Bristol Company's "Bristol 198," which was a slim, delta-winged machine with eight turbojet engines designed to cross the Atlantic at Mach 2. This evolved into the somewhat less ambitious "Bristol 223," which had 4 engines and 110 seats.

In the meantime, Sud-Aviation of France designed the "Super Caravelle," which surprisingly was very similar to Bristol 223. On November 29, 1962, the British and French governments signed a collaborative agreement to develop an Anglo-French SST, which became the "Concorde." It was to be built by the *British Aircraft Corporation* (BAC), into which Bristol had been absorbed in the meantime, and *Rolls-Royce in the United Kingdom*; and *Sud-Aviation* and the *Snecma* engine firm in France. In all, a total 20 Concordes (Figure 7.3) were built between 1966 and 1979 [3]. The first two Concordes were prototype models, one built in France and the other in England. Two more preproduction prototypes were built to further refine the design and test out ground breaking systems before the production runs; thus, only 16 aircraft in total, commenced operations in both countries. Concorde is powered by four turbojet engine (Olympus 593) fixed in the wing. The maximum thrust produced during supersonic cruise per engine is 10,000 lb. Maximum weight without fuel (Zero fuel weight) is 203,000 lb (92,080 kg). Maximum operating altitude is 60,000 ft.

BAe-Aerospatiale AST

The European AST was based on the Concorde design although being much larger and more efficient. It had canard foreplanes for extra stability. The new AST design featured the curved Concorde wing unlike all other delta SST designs to date. It was designed as two aircraft: the first would carry 275 passengers and the second would carry 400 passengers. The first version would weigh the same on takeoff as two Concordes. The second version had a slimmer delta shape and a double-delta leading edge wing shape similar to AST designs coming out of the United States.

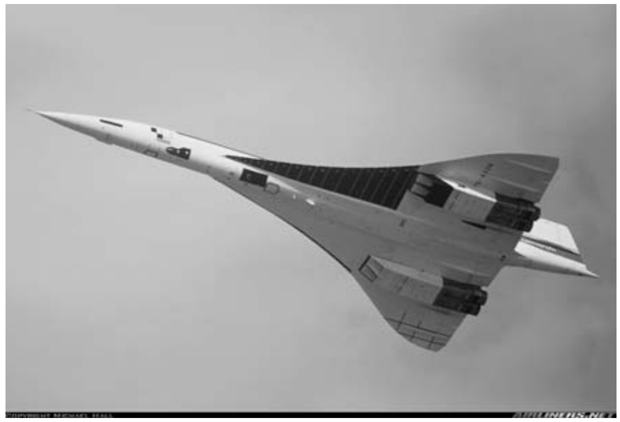

FIGURE 7.3 Concorde supersonic airplane.

FIGURE 7.4 Tupolev TU-144.

7.2.2 Russian Activities

Tupolev TU-144

In the former Soviet Union plans were underway for a trans-Soviet continental SST. The Tupolev Tu-144 "Concord-ski" looked superficially similar to the Anglo-French aircraft (Figure 7.4). It was quite different however. It began life with all four NK144 turbofan engines in a main block underneath the fuselage [4]. This arrangement was changed to a design similar to the Concorde with the four engines in boxes of two each side. The wing design had two delta angles in it. It was withdrawn from service in 1985 after a 10-year line life.

7.2.3 U.S. Activities

In the United States, fears about being left behind in the supersonic airliner race by the Soviet and European SSTs started to take form on paper. Boeing and Lockheed competed in the design of an SST paper plane. The Boeing models were the 2707-100, 200, and 300. Lockheed had also proposed an SST of its own, the Lockheed L-2000.

Boeing 2707-100/200

It was a large aircraft 318 ft long from nose to tail and had a complex swing-wing design with a tail plane behind the main swing wing. This tail place formed the back section of the delta wing when the wings were swung back during supersonic flight and held four large General Electric GE-4 turbojet engines. It would fly at a high Mach 2.7 over 3900 mi with a passenger load of at least 300. This was a truly large aircraft—bigger than a Boeing 747 (Figure 7.5).

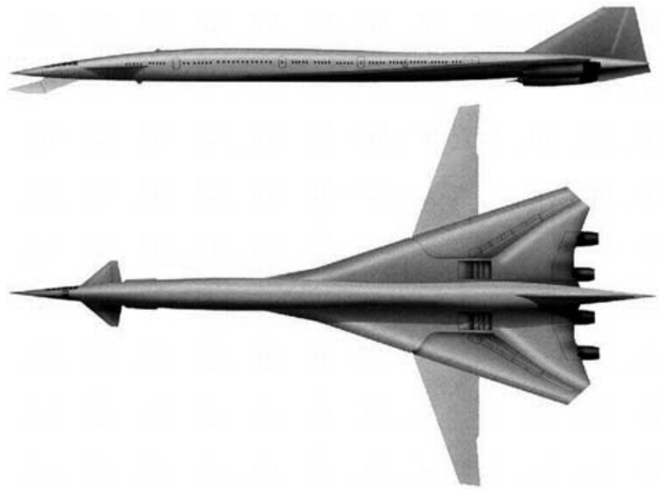

FIGURE 7.5 Boeing 2707-200. (Courtesy Boeing.)

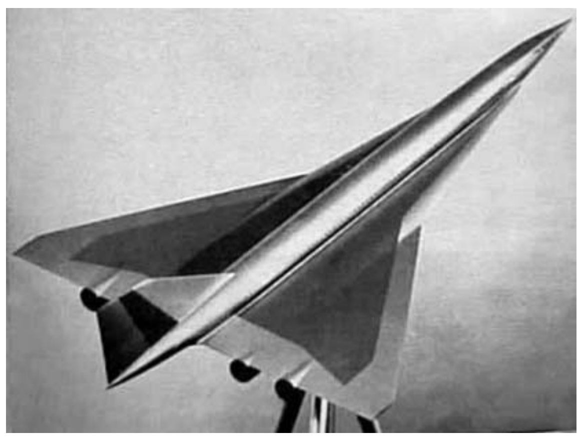

FIGURE 7.6 Lockheed L-2000.

Lockheed L-2000

This jet had a similar range and size to the Boeing jets but was designed as a simpler delta (Figure 7.6). The Lockheed L-2000 was powered by four General Electric GE4/J5P or Pratt & Whitney JTF17A-21L turbojets with augmentation being fixed in the wing. Empty weight was 23,800 lb (10795.5 kg) and maximum takeoff weight is 590,000 lb (276,620 kg).

Boeing 2707-300

Boeing looked again at the dash 200 and replaced it with the Boeing 2707 dash 300 design. This was a more conventional design although it still had a tail plane but with a conventional SST delta wing and the four GE-3 engines mounted separately underneath the wing.

Convair BJ-58

The Convair BJ-58 was designed to fly at Mach 2.5 and cruise at more than 70,000 ft (Figure 7.7). It was powered by four J-58 engines (each of 23,000 lbs) and has a payload capacity of 52 passengers.

AD-1

The Ames-Dryden-1 (AD-1) was a research aircraft designed to investigate the concept of a wing that could be pivoted obliquely from 0° to 60° during flight. AD-1 was 38.8 ft long and 6.75 ft high

FIGURE 7.7 Convair BJ-58.

FIGURE 7.8 The HSCT.

with a wing span of 32.3 ft, unswept and was constructed of plastic reinforced with fiberglass. It weighed 1,450 pounds,empty and was powered by two small turbojet engines, each producing 220 lb of static thrust at sea level. The AD-1 flew a total of 79 times during the research program. It was limited for reasons of safety to a speed of about 170 mph. If flying at speeds up to Mach 1.4, it would have substantially better aerodynamic performance than aircraft with more conventional wings and might achieve twice the fuel economy of an aircraft with conventional wings.

The high-speed civil transport project

A present project is the high-speed civil transport (HSCT), contracted between NASA subdivision high-speed research (HSR), Boeing, General Electric, and Pratt & Whitney (Figure 7.8). This contract was able to solve a lot of the SST problems such as the emissions problem and the sonic boom problem. However, problem of landing noise, takeoff noise, and materials still remain.

7.3 FUTURE OF COMMERCIAL SUPERSONIC TECHNOLOGY

NASA's HSR Program began in 1985 with the objective "to establish the technology foundation by 2002 to support the U.S. transport industry's decision for a 2006 production of an environmentally acceptable, economically viable, 300 passenger, 5000 nautical mile, and Mach 2.4 aircraft" (NRC, 1997). The first flight of a commercial supersonic aircraft was envisioned around 2010, with the first production aircraft to be operational around 2013. NASA expected that program goals would require a large investment.

The governments of France, Japan, Russia, and the United Kingdom are also sponsoring development of supersonic technology with commercial applications, although none has embarked on a formal program to produce a new commercial supersonic aircraft. The development of a commercial SST that can meet international environment standards and compete successfully with subsonic transports may be a larger effort than the industry of any single nation, including the United States, might wish to undertake. As with many such innovations, the first manufacturer to market will have the potential to dominate the market worldwide. A small supersonic jet could be developed by a single aircraft manufacturer and might lead to important technological innovation.

7.4 TECHNOLOGY CHALLENGES OF THE FUTURE FLIGHT

The key technology challenges that are derived from the customer requirements and vehicle characteristics are related to economics, environment, or certification [5].

1. Environment
   - Benign effect on climate and atmospheric ozone
   - Low landing and takeoff noise
   - Low sonic boom
2. Economics—range, payload, fuel burn, and so on
   - Low weight and low empty weight fraction
   - Improved aerodynamic performance
   - Highly integrated airframe/propulsion systems
   - Low thrust-specific fuel consumption (TSFC)
   - Long life
3. Certification for commercial operations
   - Acceptable handling and ride qualities
   - Passenger and crew safety at high altitudes
   - Reliability of advanced technologies, including synthetic vision
   - Technical justification for revising regulations to allow supersonic operations over land

7.5 HIGH-SPEED SUPERSONIC AND HYPERSONIC PROPULSION

7.5.1 Introduction

For an aircraft cruising at Mach 6, the aircraft must be fitted with an engine that can give sufficient thrust for takeoff, climb and cruise, and again deceleration, descent and landing [6]. There are two suggestions for such a vehicle and its propulsion.

1. Multistage vehicle

In this method, the aircraft is to be composed of two stages (Figure 7.9). The first stage has turbine engines (either a turbojet or turbofan) to power the aircraft during takeoff, climb, and acceleration to supersonic speed. At such speed the first stage separates and return back to the ground while the engines of the second stage start working, which are either ramjet or scramjet engines.

2. Hybrid cycle engine

The hybrid or combined engine is an engine that works on two or more cycles (Figure 7.10). The first cycle is a cycle of a normal turbojet or turbofan engine. The engine stays working with that cycle till the aircraft reaches a specified speed at which the engine changes its cycle and works on a ramjet, scramjet, or even a rocket cycle depending on the aircraft mission and concept.

For sure, the second concept will be used because in the first concept there will be a lot of problems concerning connecting/disconnecting the stages, how the first stage will be back to airport, and many others.

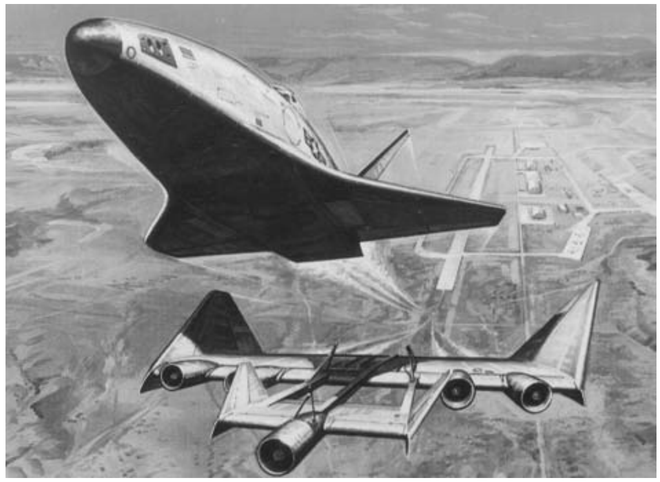

FIGURE 7.9 Multistage aircraft concept.

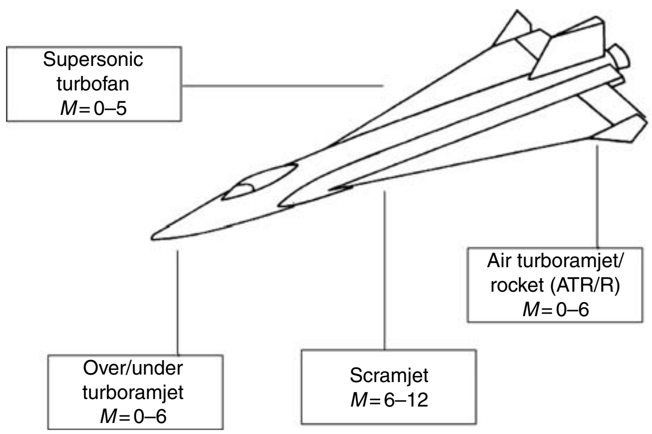

FIGURE 7.10 Hybrid (combined) cycle engine.

7.5.2 Hybrid Cycle Engine

The question arising here is why a hybrid (combined) cycle is needed.

The main engine for aircraft is to be a turbine engine (turbojet or turbofan) but from the performance analysis of those engines (Chapters 4 and 5) it was found that the engine thrust will start decreasing after a certain Mach number (about 2–3) depending on the overall pressure ratio of the engine and the bypass ratio for turbofans. On the contrary, ramjet still have a reasonable thrust for Mach numbers up to 5 or 6 and scramjet engine can provide thrust theoretically up to Mach number of 20. Ramjet and scramjet engines, however, cannot generate thrust at static conditions or low Mach numbers. Ramjet can start generating thrust at Mach numbers close to unity while scramjet develops thrust at Mach numbers greater than 2.5. Therefore, the hypersonic aircraft need both the turbine engine and the ram/scram engine. The above argument is displayed in Figure 7.11 illustrating the specific impulse versus Mach number.

The hybrid engine works on the combined cycle where in the first flight segment the engine works as a turbojet or a turbofan and then switches to a ramjet or a scramjet. It may also have a poster rocket to help the engine in the transition region or drive the vehicle alone if it is to fly in space (the aerospace plane).

In the succeeding sections, both turboramjet and scramjet engines will be treated in detail.

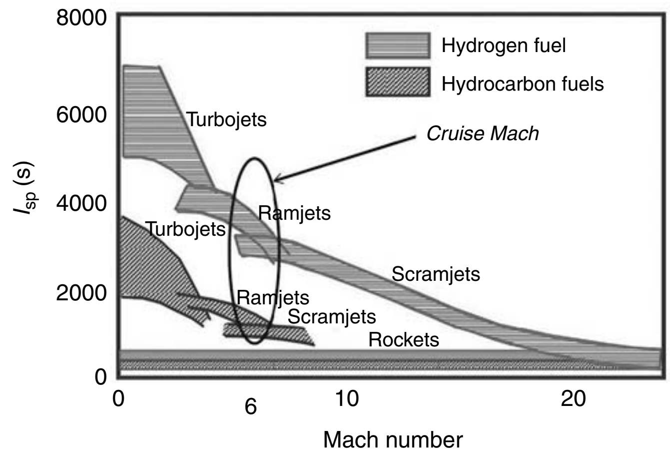

FIGURE 7.11 Specific impulse versus Mach number.

7.6 TURBORAMJET ENGINE

As defined earlier, turboramjet engine is a hybrid engine composed of a ramjet engine in conjunction with either a turbojet or turbofan engines. The turboramjet can be run in turbojet mode at takeoff and during low-speed flight but then switched to ramjet mode to accelerate to high Mach numbers. It is constructed in either of the following forms:

1. Wraparound turboramjet
2. Over/under turboramjet

The differences between them are

1. The position of the ram with respect to turbojet
2. The position of the afterburner of the turbojet with respect to the ramjet.

7.7 WRAPAROUND TURBORAMJET

In that configuration the turbojet is mounted inside a ramjet (Figure 7.12). The turbojet core is mounted inside a duct that contains a combustion chamber downstream of the turbojet nozzle. The operation of the engine is controlled using bypass flaps located just downstream of the diffuser. During low speed flight, these controllable flaps close the bypass duct and force air directly into the compressor section of the turbojet. During high-speed flight, the flaps block the flow into the turbojet, and the engine operates like a ramjet using the aft combustion chamber to produce thrust.

The wraparound turboramjet layout is found in Convair BJ-58 and SR-71 aircraft.

The engine cycle is plotted on the T–S diagram in Figure 7.13. It has two modes of operation: either working as a simple turbojet or as a ramjet. A plot for the cycle on the T–S plane is illustrated in Figure 7.13.

7.7.1 Operation as a Turbojet Engine

In the turbojet mode, a chain description of the different processes through the different modules is described in Table 7.1.

A brief description of the different processes is given subsequently.

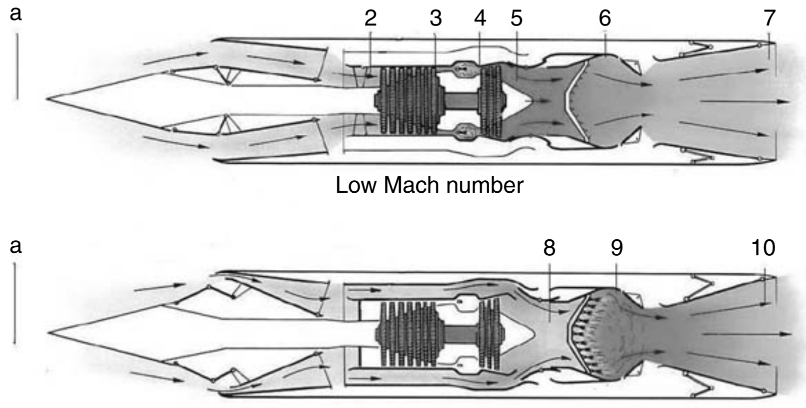

FIGURE 7.12 Wraparound turboramjet.

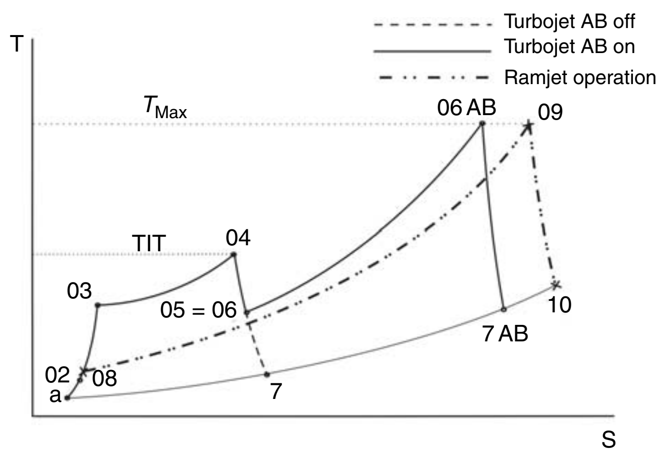

FIGURE 7.13 T–S diagram of the wraparound turboramjet engine.

TABLE 7.1
Description of Different Processes in a Turbojet Engine

Part	States	Processes
Intake	a–2	Compression process with isentropic efficiency η_d
Compressor	2–3	Compression process with isentropic efficiency η_C
Combustion chamber	3–4	Heat addition at constant pressure or with a pressure drop of $\Delta P_{C.C.}$
Turbine	4–5	Expansion process with isentropic efficiency η_t
Afterburner	5–6AB	Heat addition at constant pressure or with a pressure drop ΔP_{AB}
Nozzle	6(or 6AB)–7	Expansion process with isentropic efficiency η_N

Intake

As described previously, the inlet conditions are the ambient ones. For a flight Mach number of (M), the outlet conditions will be

$$T_{02} = T_a\left(1 + \frac{\gamma_c - 1}{2}M^2\right). \tag{7.1}$$

$$P_{02} = P_a \left(1 + \eta_d \frac{\gamma_c - 1}{2} M^2\right)^{\frac{\gamma_c}{\gamma_c - 1}} \tag{7.2}$$

Compressor

With a pressure ratio of π_C the outlet conditions will be

$$P_{03} = \pi_C \times P_{02} \tag{7.3}$$

$$T_{03} = T_{02} \left(1 + \frac{\pi_C^{(\gamma_c - 1)\gamma_c} - 1}{\eta_c}\right) \tag{7.4}$$

Combustion chamber

Owing to friction, pressure losses ΔP_{CC} will be encountered. The output of the combustion chamber is to be controlled by the turbine inlet temperature (TIT) that depends on the turbine material. The outlet pressure and fuel-to-air ratio are then given by the following equations:

$$P_{04} = P_{03} \left(1 - \Delta P_{CC}\right) \tag{7.5}$$

$$f = \frac{C_{Ph} T_{04} - C_{Pc} T_{03}}{\eta_{c.c} Q_{HV} - C_{Ph} T_{04}} \tag{7.6}$$

Turbine

As usual it is the main source of power in aircraft so the power of turbine must be equal to the power needed to drive the compressor plus power needed by all systems in aircraft. For a preliminary design, it is assumed that the power of both turbine and compressor are equal. Thus power balance results in the following:

$$T_{05} = T_{04} - \frac{C_{Pc} \left(T_{03} - T_{02}\right)}{\eta_{mech} C_{Ph} \left(1 + f\right)} \tag{7.7}$$

$$P_{05} = P_{04} \left(1 - \frac{T_{04} - T_{05}}{\eta_t T_{04}}\right)^{\frac{\gamma_h}{\gamma_h - 1}} \tag{7.8}$$

Afterburner

Afterburner will be lit if more thrust is needed from the engine. When no extra thrust is needed the afterburner is to be turned off and treated as a pipe. Combustion in the afterburner is associated with pressure loss ΔP_{AB}. The fuel burned in the afterburner and the outlet pressure are calculated:

$$f_{AB} = C_{Ph} \left(1 + f\right) \frac{T_{06AB} - T_{05}}{\eta_{AB} Q_{HV} - C_{Ph} T_{06}} \tag{7.9}$$

$$P_{06} = P_{05} \left(1 - \Delta P_{AB}\right) \tag{7.10}$$

Nozzle

The gases expand in the nozzle to high velocities to generate the required thrust. The outlet conditions of the gases and nozzle exit area are

$$T_7 = T_{06} - \eta_N T_{06} \left[1 - \left(\frac{P_a}{P_{06}}\right)^{\frac{\gamma_h - 1}{\gamma_h}}\right] \tag{7.11}$$

$$\mathrm{V}_7 = \sqrt{2 C_{Ph} \left(T_{06} - T_7\right)} \tag{7.12}$$

$$\rho_7 = \frac{P_{\text{a}}}{RT_7} \tag{7.13}$$

$$\frac{\text{A}_{\text{ex}}}{m_{\text{a}}} = \frac{1 + f + f_{\text{AB}}}{\rho_7 \text{V}_7}. \tag{7.14}$$

In Equations 7.9, 7.11 and 7.12, the symbol T_{06} is equally used for T_{06} and $T_{06\text{AB}}$.

7.7.2 Operation as a Ramjet Engine

In the ramjet mode the flow is passing through the processes and states described in Table 7.2.

Analysis of the processes is briefly described as follows.

Intake

The delivery pressure and temperature will be evaluated in terms of the total pressure ratio (r_{d}), which is to be calculated from Equation 7.15

$$r_{\text{d}} \equiv \frac{P_{08}}{P_{0\text{a}}} = \left\{ \begin{array}{cc} 1 & M \le 1 \\ 1 - 0.75\,(M-1)^{1.35} & 1 \le M \le 5 \\ \dfrac{800}{M^4 + 938} & M \ge 5 \end{array} \right\} \tag{7.15}$$

The outlet temperature and pressure are

$$T_{08} = T_{\text{a}} \left(1 + \frac{\gamma_{\text{c}} - 1}{2} M^2\right)^{\frac{\gamma_{\text{c}}}{\gamma_{\text{c}}-1}} \tag{7.16}$$

$$P_{08} = r_{\text{d}} P_{0\text{a}} = P_{\text{a}} r_{\text{d}} \left(1 + \frac{\gamma_{\text{c}} - 1}{2} M^2\right)^{\frac{\gamma_{\text{c}}}{\gamma_{\text{c}}-1}}. \tag{7.17}$$

The combustion chamber is treated in the same way as the afterburner of turbojet.

$$P_{09} = P_{08}\,(1 - \Delta P_{\text{CCR}}) \tag{7.18}$$

$$f_{\text{R}} = \frac{C_{\text{Ph}} T_{09} - C_{\text{Pc}} T_{08}}{\eta_{\text{C.C.R}} Q_{\text{HV}} - C_{\text{Ph}} T_{09}} \tag{7.19}$$

TABLE 7.2
Description of Different Processes in a Ramjet Engine

Part	States	Processes
Intake	a–8	Compression process with total pressure ratio r_{d}
Combustion chamber	8–9	Heat addition at constant pressure or with a pressure drop of $\Delta P_{\text{C.C.}}$
Nozzle	9–10	Expansion with isentropic efficiency (η_{N}).

Nozzle
It will be also handled in the same way as in the turbojet engine:

$$T_{10} = T_{09} - \eta_N T_{09}\left[1 - \left(\frac{P_a}{P_{09}}\right)^{\frac{\gamma_h - 1}{\gamma_h}}\right] \tag{7.20}$$

$$V_{10} = \sqrt{2C_{Ph}\,(T_{09} - T_{10})} \tag{7.21}$$

$$\rho_{10} = \frac{P_a}{RT_{10}} \tag{7.22}$$

$$\frac{A_{ex}}{m_a} = \frac{1 + f_R}{\rho_{10} V_{10}} \tag{7.23}$$

7.8 OVER/UNDER TURBORAMJET

As it is clear from the name over/under, the configuration here has separated turbojet and ramjet engines, but may have the same intake (inlet) and nozzle (outlet) as in the wraparound configuration. However, in some cases, each engine has its separate intake and nozzle. In such configuration, the turbojet engine operates at takeoff and low subsonic flight where the movable inlet ramp is deployed to allow for the maximum air flow rate into its intake. At higher Mach number, the engine operates at a dual mode where both turbojet and ramjet engines are operative for a few seconds until the Mach number reaches 2.5 or 3.0. Next, the turbojet engine is shut down and only ramjet becomes operative. An example for such layout is found in the four over/under airbreathing turboramjet engines powering the Mach 5 wave rider [7]. The complexity of both aerodynamic and mechanical design of two variable throat nozzles and necessary flaps led to another design including only one nozzle [8]. The suggested nozzle uses a single-expansion ramp nozzle (SERN) instead of a conventional, two-dimensional, convergent–divergent nozzle.

Figures 7.14 and 7.15 illustrate a layout and T–S diagram of the over/under configuration of the turboramjet engine.

For the over/under configuration of turboramjet engine, the engine height is large compared to the wraparound configuration. The reason is clear as the height in this case is the sum of the heights of turbojet and ramjet engines. Usually, a part of the engine is to be buried inside the fuselage or inside the wing. Since the engine will operate at high Mach number, it needs a long intake. Thus, a part of the aircraft forebody is used as a part of the intake. Concerning installation, the engine is either installed under the fuselage or the wing. For fuselage installation, the turbojet is buried

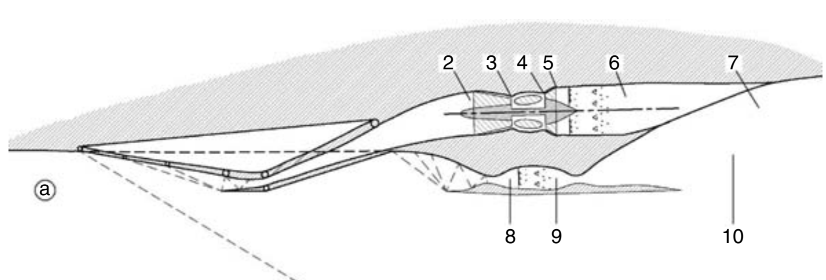

FIGURE 7.14 Over/under layout of turboramjet engine.

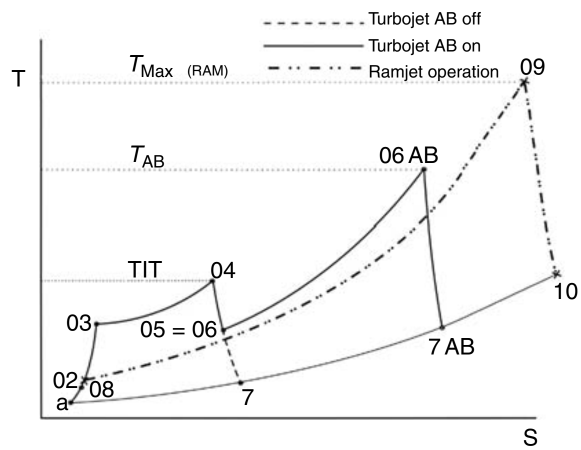

FIGURE 7.15 T–S diagram of the over/under layout of turboramjet engine.

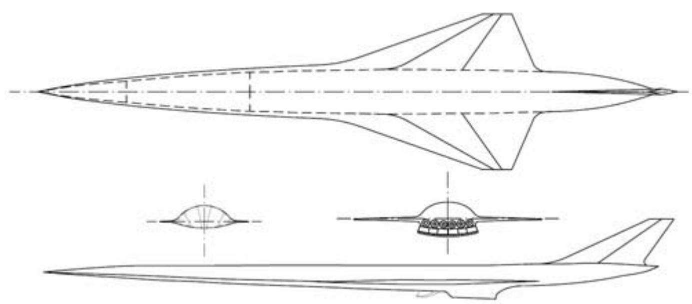

FIGURE 7.16 Hypersonic aircraft (HYCAT-1) [9].

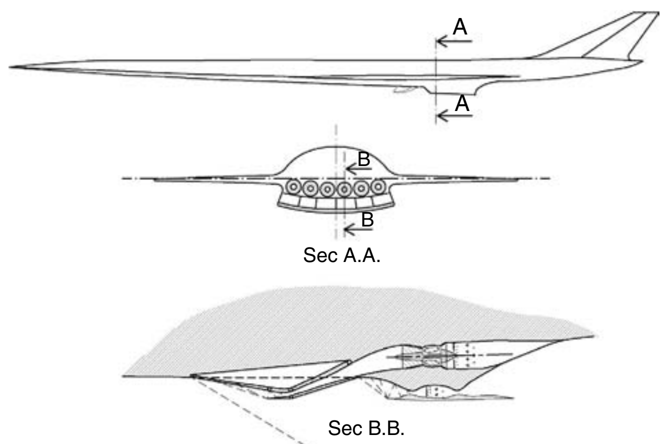

FIGURE 7.17 Turboramjet engine installed under the fuselage.

inside the fuselage and the ramjet is located under the fuselage as in Figures 7.16 and 7.17. For wing installation, the turbojet is situated above the wing and the ramjet is under the wing.

Concerning the thermodynamic cycle of an over/under configuration, there are three modes of operation; namely, a turbojet, a ramjet, or a dual mode. Dual mode represents the combined or transition mode in which both the turbojet and ramjet are operating simultaneously.

7.8.1 Turbojet Mode

In that region, the engine is working as a simple turbojet engine and develops all the thrust needed by the aircraft. The states and governing equations are the same as the turbojet in the wraparound configuration. Cold air passes through the ramjet in this mode.

7.8.2 Dual Mode

It is the mode, in which both the turbojet and ramjet are operating simultaneously. Turbojet starts declining and its developed thrust will be intentionally decreased by reducing the inlet air mass flow rate via its variable geometrical inlet. The ramjet starts working by adding fuel and starting ignition in its combustion chamber. The thrust generated in the ramjet is increased by also increasing the air mass flow rate through its variable area inlet. The generated thrust force will then be the sum of both thrusts of turbojet and ramjet engines.

7.8.3 Ramjet Mode

In this mode, turbojet stops working and its intake is completely closed. All the air mass flow is passing through the ramjet intake. With Mach number increase, the forebody acts as a part of the intake with the foremost oblique shock wave located close to the aircraft nose, as shown in Figure 7.18.

7.9 TURBORAMJET PERFORMANCE

Performance parameters: specific thrust, thrust-specific fuel consumption (TSFC), total thrust, and propulsive, thermal, and overall efficiencies are defined here. The following equations are valid for both types of turboramjet engines.

7.9.1 Turbojet Mode

The specific thrust is

$$\left(\frac{T}{\dot{m}_{\mathrm{a}}}\right)_{\mathrm{TJ}} = (1 + f + f_{\mathrm{AB}})\, V_7 - V \tag{7.24}$$

The TSFC is

$$(\mathrm{TSFC})_{\mathrm{TJ}} = \frac{(f + f_{\mathrm{AB}})}{T/\dot{m}_{\mathrm{a}}} \tag{7.25}$$

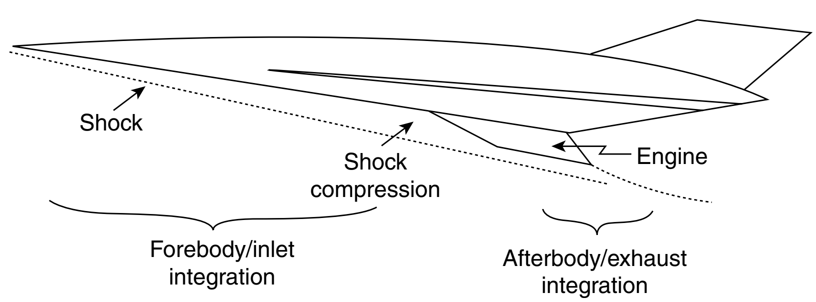

FIGURE 7.18 Intake of the ramjet of the turboramjet engine.

The engine propulsive, thermal, and overall efficiencies are

$$\eta_p = \frac{(T/\dot{m}_a)\, V}{(T/\dot{m}_a)\, V + \left((V_7 - V)^2/2\right)(1 + f + f_{AB})} \tag{7.26}$$

$$\eta_{th} = \frac{(T/\dot{m}_a)\, V + \left((V_7 - V)^2/2\right)(1 + f + f_{AB})}{Q_{HV}\,(f + f_{AB})} \tag{7.27}$$

$$\eta_o = \eta_p \times \eta_{th} \tag{7.28}$$

7.9.2 Ramjet Mode

For the ramjet engine, the performance parameters will be calculated in the same way.

The thrust per unit air mass flow $(T/\dot{m}_a)$ is

$$\left(\frac{T}{\dot{m}_a}\right)_{RJ} = (1 + f_R)\, V_{10} - V \tag{7.29}$$

The TSFC is

$$(\text{TSFC})_{RJ} = \frac{f_R}{((T/\dot{m}_a))} \tag{7.30}$$

The engine propulsive, thermal, and overall efficiencies are

$$\eta_p = \frac{(T/\dot{m}_a)_{RJ} V}{(T/\dot{m}_a)_{RJ} V + \left((V_{10} - V)^2/2\right)(1 + f_R)} \tag{7.31}$$

$$\eta_{th} = \frac{(T/\dot{m}_a)_{RJ} V + \left((V_{10} - V)^2/2\right)(1 + f_R)}{Q_{HV} f_R} \tag{7.32}$$

$$\eta_o = \eta_p \times \eta_{th} \tag{7.33}$$

7.9.3 Dual Mode

Both turbojet and ramjet are operating simultaneously. The total thrust will be

$$T = (T)_{TJ} + (T)_{RJ} \tag{7.34}$$

$$T = (\dot{m}_a)_{TJ}\,[(1 + f + f_{AB})\, V_7 - V] + (\dot{m}_a)_{R.J}\,[(1 + f_R)\, V_{10} - V] \tag{7.35a}$$

$$T = (\dot{m}_a)_{TJ}\,(1 + f + f_{AB})\, V_7 + (\dot{m}_a)_{RJ}\,(1 + f_R)\, V_{10} - (\dot{m}_a)_{total}\, V \tag{7.35b}$$

The specific thrust is

$$\frac{T}{\dot{m}_a} = \frac{T_{total}}{(\dot{m}_a)_{total}} = \frac{(T)_{TJ} + (T)_{RJ}}{(\dot{m}_a)_{TJ} + (\dot{m}_a)_{RJ}} \tag{7.36}$$

The total fuel mass flow rate is

$$\dot{m}_f = (\dot{m}_a)_{TJ}\,(f + f_{AB}) + (\dot{m}_a)_{RJ} f_R \tag{7.37}$$

The specific thrust fuel consumption is

$$\frac{(\dot{m}_{\mathrm{f}})_{\mathrm{total}}}{(T)_{\mathrm{total}}} = \frac{(\dot{m}_{\mathrm{f}})_{\mathrm{TJ}} + (\dot{m}_{\mathrm{f}})_{\mathrm{RJ}}}{(T)_{\mathrm{TJ}} + (T)_{\mathrm{RJ}}} \tag{7.38}$$

7.10 CASE STUDY

A simplified design procedure for a turboramjet engine will be given here. The procedure is the same as followed in any airliner design. It starts by selecting a proper engine for a certain aircraft. It is not an easy task such as selecting a tie for a suit, but it is a very complicated process. The following steps, among many others, are to be followed. The function of the air vehicle is first defined and then its mission is to be specified. During each point on the flight envelop the drag and lift forces are calculated. On the basis of this lift force, the lifting surfaces are designed and a variation of the lift coefficient against Mach number is determined. The corresponding drag force is also calculated and a graph for drag coefficient versus Mach number is also plotted. Now from this last curve, the appropriate power plant is selected. The case of one engine type as described in the previous four chapters is a straightforward process. The case of a hybrid engine is rather a complicated design procedure. The difficulty arises from the proper selection of switching points, or in other words, when one engine is being stopped and another one takes over. If more than two engines are available, the process gets more and more complicated. In addition even for two engines, there is also the combined or dual modes. Such a transition process must be carefully analyzed to keep safe margins of lift and thrust forces above the weight and drag forces. A sophisticated control system controls the air and fuel mass flow rates into different engines is needed. Variable intakes or inlet doors, nozzles, and possibly stator vanes in compressors are activated during such a transition mode.

The following steps summarize the procedure for high supersonic/hypersonic cruise aircraft propulsion integration:

1. Mission or flight envelop has to be selected. The cruise altitude is important from fuel economy point of view, while takeoff, climb, and acceleration are critical from the surplus thrust that must be available over the aircraft drag force.
2. Calculate the drag force based on drag coefficient variations with Mach number.
3. The needed thrust force must be greater than the drag value all over its Mach number operation. A margin of 10% is selected here.
4. Determination of the number of engines based on the maximum thrust is suggested for the hybrid engine at its different modes.
5. The performance of each constituent of the hybrid engine is separately determined. For example, if the hybrid engine includes a ramjet, turbojet, and scramjet, each module is examined separately. Thus, the performance of turbojet engine running alone is considered first. The specific thrust and TSFC are determined for the cases of operative and inoperative afterburner. Next, the performance of the ramjet operating alone is also considered. Since the ramjet operates up to Mach number of 6, liquid hydrogen is considered as its fuel. If a scramjet is available, then its performance is also determined separately.
6. Optimization of the hybrid engine represents the next difficult step. From the specific thrust and TSFC, the switching points are defined. Thus in this step, the designer selects at which Mach number the turbojet operates or stops the afterburner and at which Mach number the turbojet must be completely stopped and replaced by the ramjet. If a scramjet is also available, another decision has to be made concerning the switching Mach number from ramjet to scramjet, the switching process duration, and the accompanying procedures regarding the air and fuel flow rates and inlet doors actuation.

TABLE 7.3
Mission of HYCAT-1-A

Passengers	**200**
Range	9260 km
Cruise Mach number	6
Field length	3200 m
Fuel	JA-7 For Turbojet LH2 for ramjet
Aircraft life	Commensurate with current aircraft
Fuel reserves	5% of block fuel
Subsonic flight to alternate airport	482 km
Descent Mach at which turbojet turn on	0.8 Mach
Cruise altitude	90,000 ft
Fuselage length	118.26 m
Wing reference area	816.8 m^2
Horizontal Tail Total Area	177.8 m^2
Vertical Tail Area	90.2 m^2
Gross weight	350,953 kg
Payload	19,051 kg
Empty weight	198,729 kg

Since the plan of the twenty-first century is to have hypersonic/high supersonic civil transports, a modified version of the HYCAT-1 aircraft [9] is selected for this analysis as its aerodynamic characteristics up to Mach 6 are available. This selected aircraft may be identified as HYCAT-1-A, which includes a completely buried turbojet engine in the fuselage. Other studies for supersonic aircraft having test results up to Mach 6 are also available [10]. Test results for supersonic aircraft of Mach 2.96, 3.96, and 4.63 are found in Reference 11, while results for a supersonic aircraft flying at Mach number of 4 are discussed in [12]. Some data of the aircraft are given in Table 7.3.

Now after selecting the aircraft, illustrated in Figures 7.16 and 7.17, the drag force is calculated for different Mach numbers. Some important data concerning the turbojet engine are compressor pressure ratio of 10, TIT of 1750 K, maximum temperature in the afterburner of 3000 K, and heating value of hydrocarbon fuel of 45,000 kJ/kg. The efficiencies of the diffuser, compressor, turbine, and nozzle are 0.9, 0.91, 0.92, and 0.98, respectively. The efficiency of the combustion chamber and afterburner are 0.99, and the pressure drops in the combustion chamber and afterburner are 2%. The mechanical efficiency of the compressor turbine shaft is 0.99. Concerning the ramjet, the maximum temperature is 3000 K and heating value of liquid hydrogen is 119,000 kJ/kg. Efficiencies of diffuser, combustion chamber, and nozzle are 0.9, 0.99, and 0.98, respectively. The pressure drop in the combustion chamber is 2%.

The specific thrust and specific fuel consumption for individual turbojet engine and ramjet engine are calculated and plotted in Figures 7.19 and 7.20.

From Figures 7.19 and 7.20, the following points may be selected as the switching points; refer to Figure 7.21:

1. The ramjet starts burning fuel (R1)
2. The turbojet starts its gradual shut down (T1)
3. The ramjet works with its maximum burner temperature (R2)
4. The turbojet completely shuts down (T2)

Details of selections and identification of these points are given in Table 7.4.

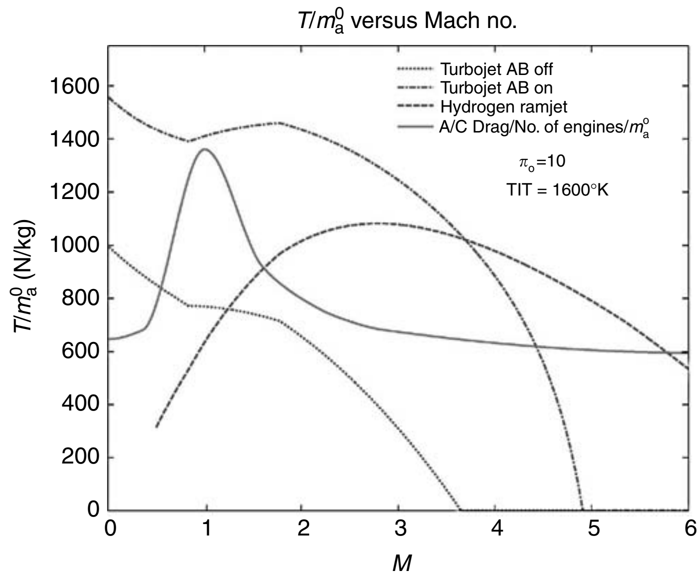

FIGURE 7.19 Specific thrust for individual turbojet and ramjet engines.

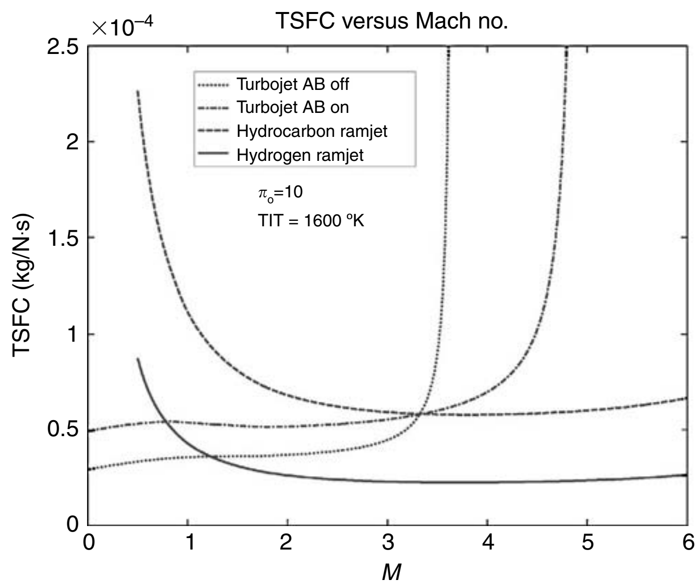

FIGURE 7.20 Thrust-specific fuel consumption for individual turbojet and ramjet engines.

The following two assumptions are identified here concerning both engines:

(a) *Concerning the turbojet engine:* The turbojet has three regions of operation, namely,

1. Constant mass flow rate with afterburner off (from $M = 0$ to $M = 0.5$)
2. Constant mass flow rate with afterburner on (from $M = 0.5$ to $M = 1.5$)
3. Decreasing mass flow rate with afterburner off (from $M = 1.5$ to $M = 2.5$)

The air flow in this last region decreases gradually and linearly from points T1 to T2.

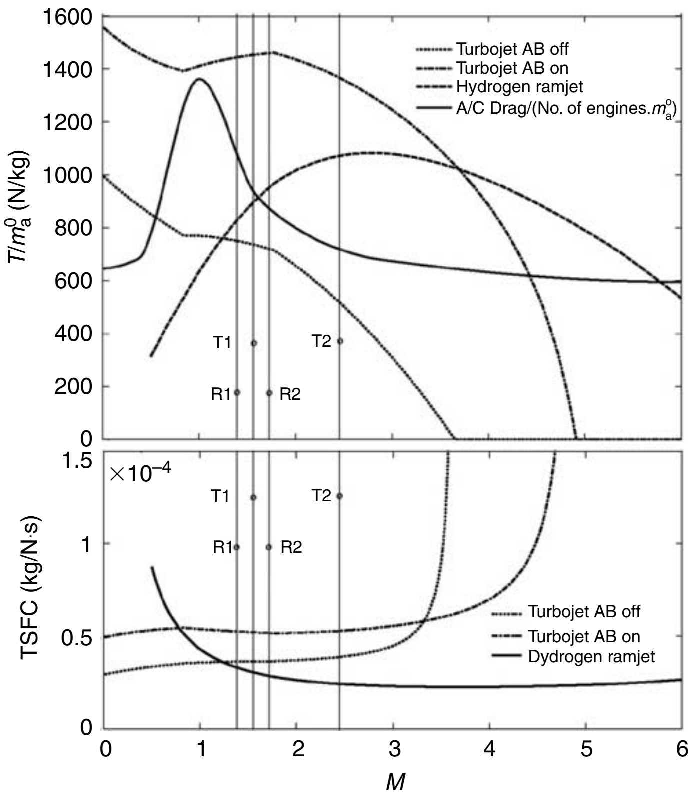

FIGURE 7.21 Regions boundaries.

(b) *Concerning the ramjet engine:* The ramjet has three regions of operation, namely,

1. The cold flow region (burner is off) in which the ramjet does not generate any thrust but generates drag (from $M = 0$ to point R1).
2. The starting region in which the burner temperature increases gradually and linearly. At the end of this region, the temperature reaches its maximum value (from points R1 to R2).
3. The working region in which the burner temperature reaches its maximum value and keeps working (from point R2 to the maximum Mach number).

On the basis of the above explanation, the operation of the hybrid engine is divided into four regions that are identified in Table 7.4 and in Figures 7.21 through 7.25.

Region I is a typical operation of a turbojet engine with an inoperative afterburner.

Region II is also a typical operation of a turbojet engine with an operative afterburner.

Region III (dual region) is important as in this transition region the ramjet engine is starting to build up thrust while the turbojet engine is declining out. A linear variation of the air mass flow rate in the turbojet is assumed during its shut down (region III), and then at any Mach number the thrust generated by the turbojet engine is known. The thrust needed from the ramjet is the difference between the aircraft required thrust and the turbojet thrust. Since the specific thrust of the ramjet is known at this Mach number, the air mass flow rate through the ramjet can be calculated from the following relations:

$$(T)_{\mathrm{RJ}} = (T_{\mathrm{R}})_{\mathrm{A/C}} - (T)_{\mathrm{TJ}} \tag{7.39a}$$

$$(T)_{\mathrm{TJ}} = (T/\dot{m}_{\mathrm{a}})_{\mathrm{TJ}}\,(\dot{m}_{\mathrm{a}})_{\mathrm{TJ}} \tag{7.39b}$$

TABLE 7.4
Regions of Turboramjet Operation

Region	I	II	III	IV
Mach Number	0–0.5	0.5–1.5	1.5–2.5	2.5–6
Ramjet	Off (cold flow)	Off (cold flow)	Starting (combustion starts and $(\dot{m}_a)_{RJ}$ increases to required value)	On (working alone and stable)
Turbojet	On $(\dot{m}_a)_{TJ} = 160$ kg/s (alone and stable)	On $(\dot{m}_a)_{TJ} = 160$ kg/s (alone and stable)	On $[(\dot{m}_a)_{TJ}$ starts to decrease to reach zero at $M = 2.5]$	Off $(\dot{m}_a)_{TJ} = 0$
Gas generator				
After burner	Off	On $T_{max} = 3000$ K	Off	Off

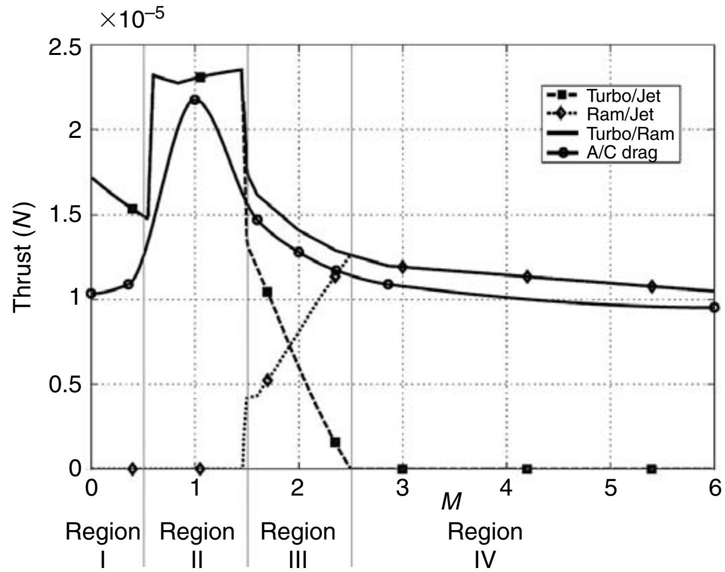

FIGURE 7.22 Thrust versus drag of hybrid engine.

where $(T_R)_{A/C}$ is the thrust required by airplane, say 110% of aircraft drag

$$(\dot{m}_a)_{RJ} = \frac{(T)_{RJ}}{(T/\dot{m}_a)_{RJ}} \tag{7.40}$$

In region IV, the ramjet is working alone so the desired thrust from ramjet is equal to the aircraft required thrust. In the same way, the ramjet air mass flow rate can be calculated from Equations 7.41 and 7.42:

$$(T)_{RJ} = (T_R)_{A/C} \tag{7.41}$$

$$(\dot{m}_a)_{RJ} = \frac{(T)_{RJ}}{(T/\dot{m}_a)_{RJ}} \tag{7.42}$$

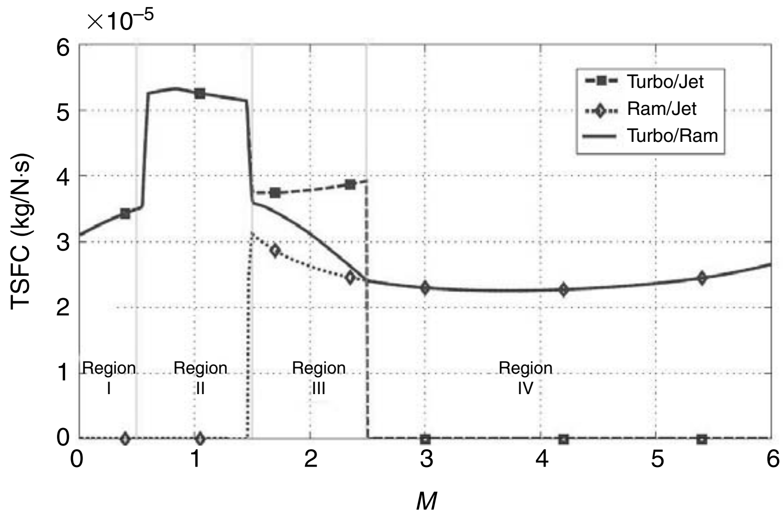

FIGURE 7.23 Thrust-specific fuel consumption (TSFC) of hybrid engine.

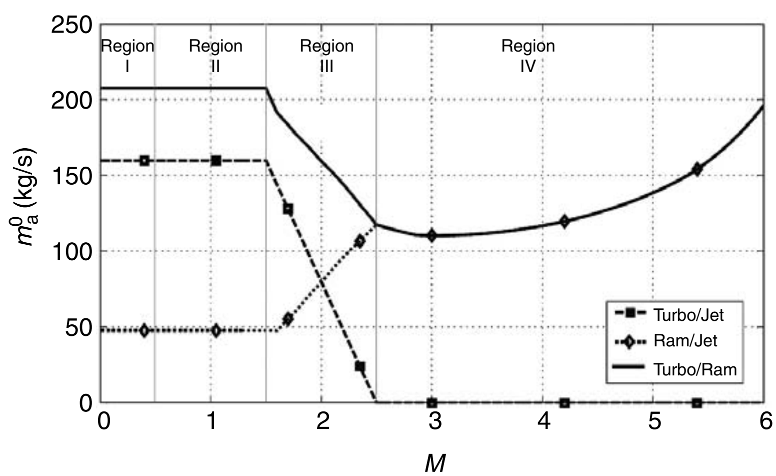

FIGURE 7.24 Air mass flow rate of hybrid engine.

The mass flow rate of the ramjet engine is known all over the engine working interval. Thus, the different performance parameters of the turboramjet engine can be calculated and plotted as illustrated in Figures 7.22 through 7.26.

7.11 EXAMPLES FOR TURBORAMJET ENGINES

Turboramjet engines are not as common as the pure ramjet engine. Two famous known engines may be identified. Most of the others are either secret or only in experimental phases.

1. Pratt & Whitney J58-1, continuous-bleed afterburning turbo (145 kN), two of which are powering the SR-71 strategic reconnaissance aircraft (1964–1998). This enables it to takeoff under turbojet power and switch to ramjets at higher supersonic speeds. Cruise speed is between $M = 3$ and $M = 3.5$.
2. The ATREX turboramjet engine [13]: This engine is developed in Japan. As shown in Figure 7.27 the air flow is pressurized inside the engine by an air fan, while the

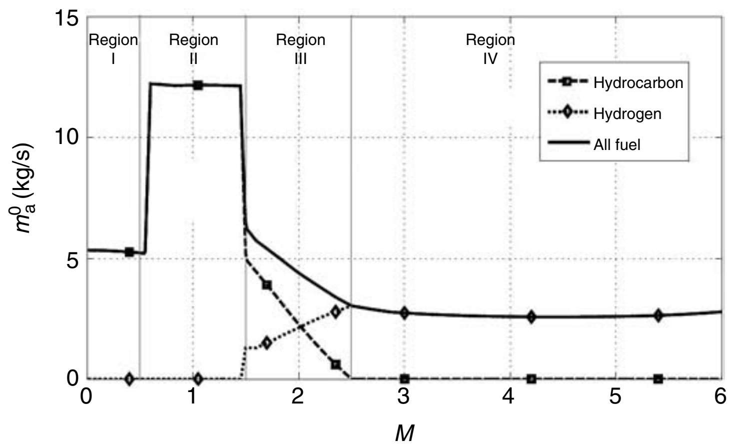

FIGURE 7.25 Fuel mass flow rate of hybrid engine.

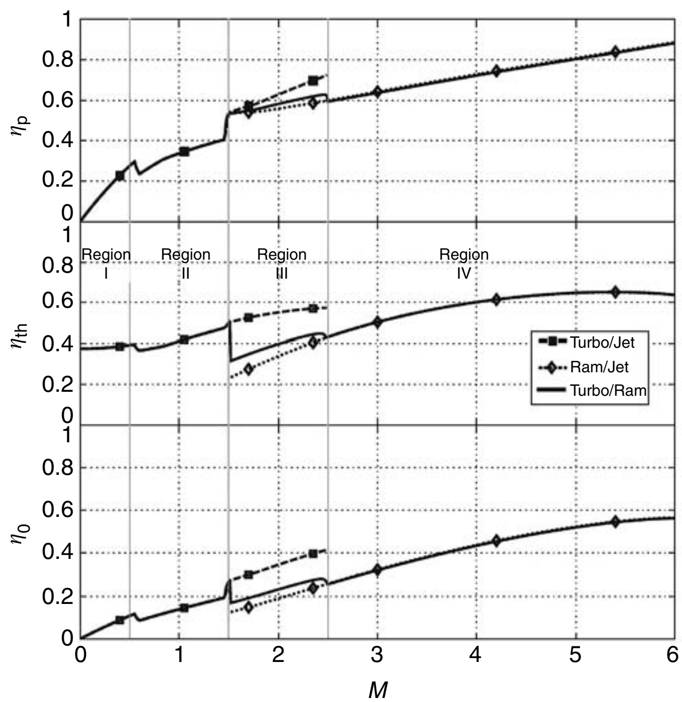

FIGURE 7.26 Propulsive, thermal, and overall efficiencies.

liquid hydrogen (fuel used) is pressurized by a turbopump. Highlights for both flows are given here.

- *Air Flow:* The air intake that reduces velocity of air incoming at high flight Mach number. Next, this air is cooled down by the precooler. The cooled air is pressurized by a fan and injected into the combustion chamber through the mixer together with the hydrogen discharged out of the tip turbine. The plug nozzle provides the effective nozzle expansion over the wide range of flight environment.

- *Hydrogen (H_2) Flow:* The hydrogen supplied from storage tank is pressurized by turbopump and heated regeneretively in the precooler.

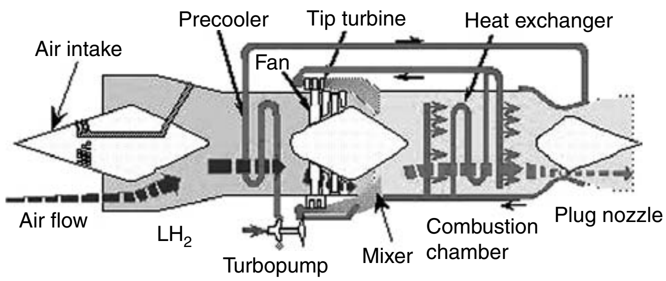

FIGURE 7.27 ATREX turboramjet engine.

7.12 HYPERSONIC FLIGHT

Hypersonic flight is identified as a flight with a Mach number exceeding 5 [14]. The main problem of that speed range is the very large wave drag. To minimize the wave drag, both the aircraft and engine must be completely reconfigured. Until now apart from research work, no civil aircraft can fly at hypersonic speeds. Only rockets can.

7.12.1 History of Hypersonic Vehicles

Some important events will be highlighted here.

WAC Corporal

The first vehicle ever to achieved hypersonic speeds was the WAC Corporal, a U.S. Army second stage rocket mounted atop a captured German V-2 rocket. The origin of the WAC label is a bit unclear, but some sources state it stood for "Without Attitude Control," referring to the fact that the simple rocket had no stabilization and guidance system. On February 24, 1949, the WAC Corporal separated from the V-2 at 3500 mph before its own engine powered it to an altitude of 244 miles and a top speed of 5150 mph upon atmospheric reentry.

Bell aircraft corporation X-2

First flight was on June 27, 1952. The fastest flight recorded a Mach of 3.169. The last flight was on September 27, 1956.

Lockheed Missiles & Space Co (X-7)

First flight was on April 24, 1951. The fastest flight was at a Mach number of 4.31. The last flight was on July 20, 1960.

North American Aviation X-10

First flight on October 14, 1953 and the fastest flight speed was at Mach 2.05. Last flight was on January 29, 1959.

North American Aviation X-15

Robert White, an American flew the X-15 rocket-powered research aircraft at Mach 5.3, thereby becoming the first pilot to reach hypersonic velocities (Figure 7.28). White's mark was later topped by his X-15 co-pilot Pete Knight who flew the craft (denoted X15A-2) to a maximum speed of Mach 6.72 in 1967. The record still stands today as the highest velocity ever reached by an aircraft.

FIGURE 7.28 X-15.

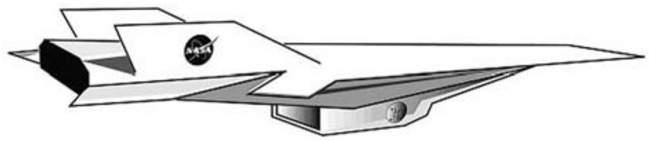

FIGURE 7.29 X-43.

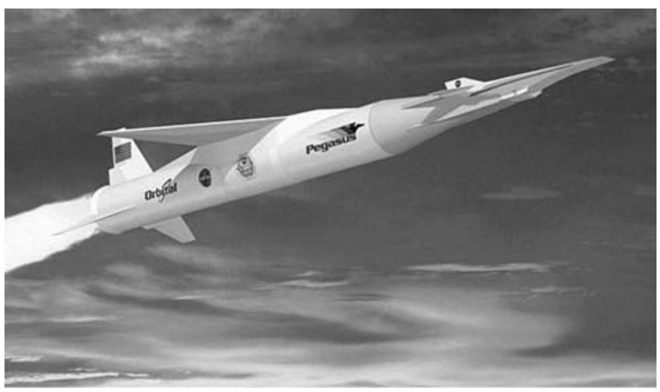

FIGURE 7.30 Booster rocket and X-43.

National Aerospace Plane (NASP); X-30

During the 1980s, NASA began considering a hypersonic single-stage-to-orbit (SSTO) vehicle to replace the Space Shuttle. The proposed national aerospace plane (NASP) would takeoff from a standard runway using some kind of low-speed jet engine. Once the aircraft had reached sufficient speed, airbreathing ramjet or scramjet engines would power the aircraft to hypersonic velocities (Mach 20 or more) and to the edge of the atmosphere. A small rocket system would provide the final push into orbit, but the attractiveness of the concept was using the atmosphere to provide most of the fuel needed to get into space. NASP eventually matured into the X-30 research vehicle, which used an integrated scramjet propulsion system.

X-43 Hyper-X

X-43 is a part of the NASA's Hyper-X project (Figure 7.29). A winged booster rocket with the X-43 itself at the tip, called a "stack," is launched from a carrier plane (B-52). After the booster rocket (a modified first stage of the Pegasus rocket) brings the stack to the target speed and altitude, it is discarded, and the X-43 flies free using its own scramjet engine (Figure 7.30). The first X-43 test flight, conducted in June 2001, ended in failure after the Pegasus booster rocket became unstable and went out of control. The X-43A's successful second flight made it the fastest free flying airbreathing aircraft in the world, though it was preceded by an Australian Hyshot as the first operating scramjet

FIGURE 7.31 The German sänger space transportation systems.

engine flight. The third flight of a Boeing X-43A set a new speed record of 7546 mph (12,144 km/h), or Mach 9.8, on November 16, 2004. It was boosted by a modified Pegasus rocket, which was launched from a Boeing B-52 at 13,157 m (43,166 ft). After a free flight where the scramjet operated for about 10 s, the craft made a planned crash into the Pacific Ocean off the coast of southern California.

LoFlyte

To further the research in this area, the Air Force and Navy sponsored the LoFlyte test vehicle to study the flight characteristics of a wave rider at low-speed conditions, such as takeoff and landing. LoFlyte is another small, unmanned vehicle that can be reconfigured with different types of control surfaces and flight control algorithms to determine the most effective combination. Though designed using a Mach 5.5 conical flowfield, the LoFlyte vehicle is not actually capable of flying at hypersonic speeds. It has instead been tested for basic handling characteristics at low speeds.

The German sänger space transportation systems

The German hypersonic technology program was initiated in 1988, the reference configuration is known as sänger space transportation systems and it is reusable in two stages (Figure 7.31). The piloted first stage is also known as the European hypersonic transport vehicle (EHTV); the EHTV is propelled up to the stage separation Mach number of 6.8 by a hydrogen-fueled turboramjet propulsion system. There are two identical second stages similar to the space shuttle orbiter and powered by hydrogen–oxygen rocket.

7.12.2 Hypersonic Commercial Transport

Hypersonic vehicles in general and wave riders in particular have long been touted as potential high-speed commercial transports to replace the Concorde. Some aerospace companies, airlines, and government officials have proposed vehicles cruising at Mach 7–12 capable of carrying passengers from New York to Tokyo in less than 2 h. Perhaps one of the most well-known concepts along these lines was the Orient Express, envisioned as a commercial derivative of the NASP project.

The expected flight path for the commercial flight is something unbelievable. Hypersonic vehicles escapes heat buildup on the airframe by skipping along the edge of Earth's atmosphere, much like a rock skipped across water. Hypersonic vehicles would ascend via its power outside the Earth's atmosphere, and then turn off its engines and coast back to the surface of the atmosphere. There, it would again fire its airbreathing engines and skip back into space. The craft would repeat this process until it reached its destination. A flight from the United States to Japan will take 25 such skips (Figure 7.33). The skips will be angled at only 5°. The passengers will feel a force of 1.5 Gs, the same as what one would experience on a child's swing. The plane will power up to 39 km, from where it will coast to double that altitude, before it starts to descend. Each skip will be 450 km long.

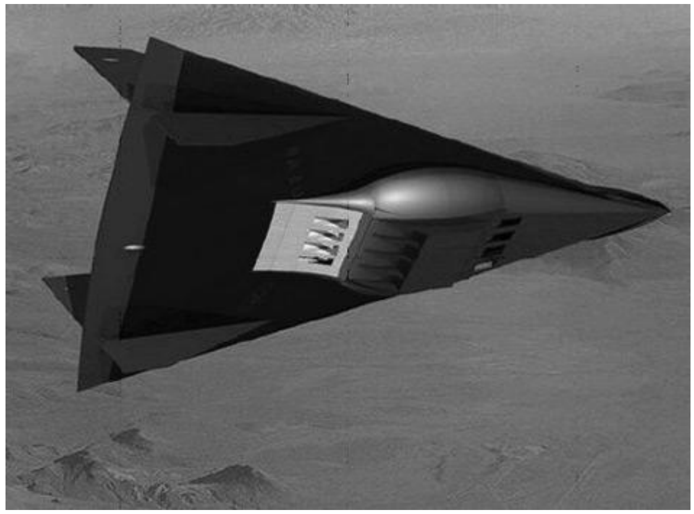

FIGURE 7.32 Artist's concept of a hypersonic aircraft (Aurora).

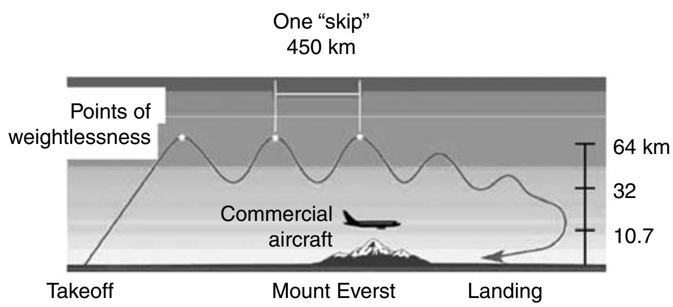

FIGURE 7.33 The expected flight path for hypersonic civil transports.

7.12.3 Military Applications

Trends of the 1950s and 1960s indicated that military aircraft had to fly faster and higher to survive, so concepts for high-altitude fighters and bombers cruising at Mach 4 or more were not uncommon. Although the trends soon fizzled and military planners looked to maneuverability and stealth for survival, the military has recently shown renewed interest in hypersonic flight. For example, many have conjectured about the existence of a Mach 5 spy plane, the Aurora (Figures 1.72 and 7.32), may be either under development or perhaps already flying. If so, the Aurora may be scramjet-powered.

7.13 SCRAMJET ENGINES

7.13.1 Introduction

Scramjet engines find their applications in many recent hypersonic speed vehicles such as rockets, commercial transports in the twenty-first century, as well as SSTO launchers. Scramjet is an acronym for supersonic combustion ramjet.

From a thermodynamic point of view a scramjet engine is similar to a ramjet (Figure 7.34) engine as both consist of an intake, a combustion chamber, and a nozzle. A dominant characteristic of airbreathing engines for hypersonic flight is that the engine is mostly inlet and nozzle (Figure 7.35), and the engine occupies the entire lower surface of the vehicle body.

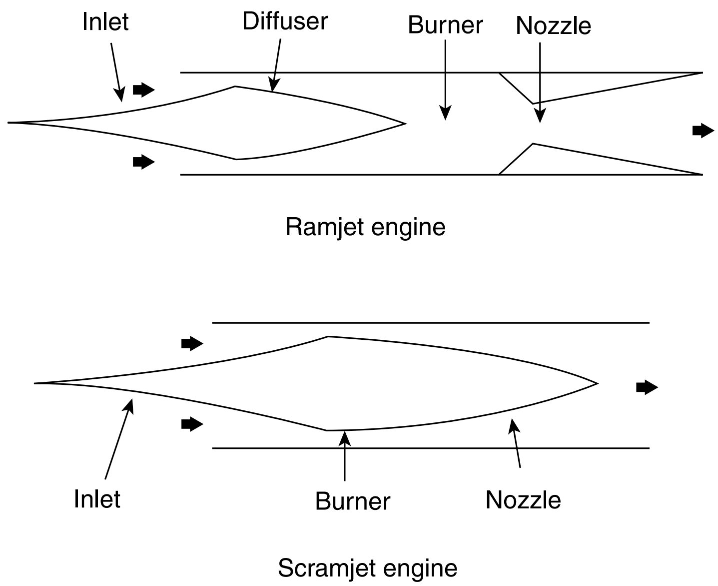

FIGURE 7.34 Ramjet and scramjet engines.

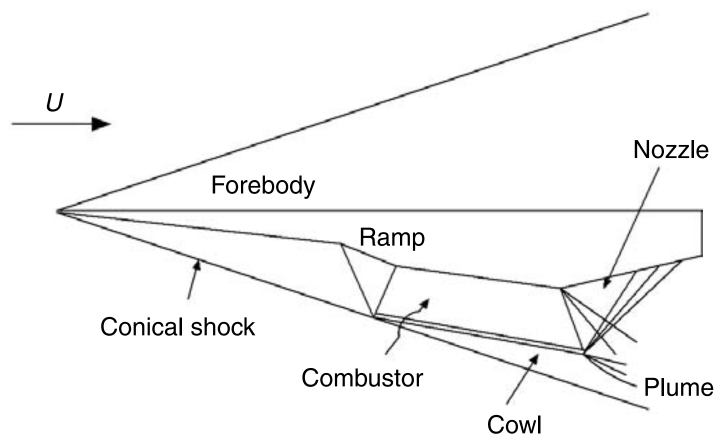

FIGURE 7.35 Schematic representation of a hypersonic scramjet engine powering a hypersonic aircraft.

In both engines, the cycle pressure rise is achieved by purely ram compression [15]. The main difference between ramjet and scramjet relies upon the speed of combustion, which is subsonic in a ramjet engine and supersonic in a scramjet engine. A conventional hydrogen-fueled ramjet with a subsonic combustor is capable of operating up to around Mach 5–6 at which the limiting effect of dissociation reduces the effective heat addition to the airflow resulting in a rapid loss in net thrust.

The idea behind the scramjet engine is to avoid dissociation limit by partially slowing the air stream through the intake system (thus reducing the static temperature rise) and hence permitting greater useful heat addition in the now supersonic combustor. Consequently, scramjet engines offer the tantalizing prospect of achieving a high specific impulse up to very high Mach number (Figure 7.11). Scramjet engines have a relatively low specific thrust due to the moderate combustor temperature rise and pressure ratio, and therefore a very large air mass flow rate is required to give adequate vehicle thrust/weight ratio. As outlined in [15], the captured air mass flow reduces for a given intake area as speed rises above Mach 1. Consequently, the entire vehicle frontal area is needed to serve as an intake at scramjet speeds and similarly the exhaust flow has to be re-expanded back into the original stream tube in order to achieve a reasonable exhaust speed. Employing the vehicle forebody and aftbody as part of the propulsion system has the following disadvantages:

1. The forebody boundary layer reaches nearly 40% of the intake flow that upsets the intake flow stability. The conventional solution of bleeding the boundary layer off would be unacceptable due to the prohibitive momentum drag penalty.
2. The vehicle surface must be flat in order to provide the intake with a uniform flowfield. This flattened vehicle cross section is poorly suitable for pressurized tankage and has a higher surface area/volume than a circular cross section.
3. Since the engine and airframe are physically inseparable, little freedom is available to the designer to control the vehicle pitch balance. The single-sided intake and nozzle generate both lift and pitching moments and this adds difficulties to achieve both vehicle pitch balance over the entire Mach number range and center of gravity (CG) movement to trim the vehicle.
4. Normally, scramjet engines are clustered into a compact package underneath the vehicle. This results in interdependent flowfield and thus the failure in one engine with a consequent loss of internal flow is likely to unstart the entire engine installation precipitating a violent change in vehicle pitching moment.
5. To focus the intake shock system and generate the correct duct flow areas over the whole Mach number, variable geometry intake/combustor and nozzle surfaces are required. The large variation in flow passage shape forces the adoption of a rectangular engine cross section with flat moving ramps thereby incurring a severe penalty in the pressure vessel mass.
6. To maximize the installed engine performance requires a high dynamic pressure trajectory with the high Mach number imposing severe heating rates on the airframe. Active cooling of significant portions of the airframe will be necessary with further penalties in mass and complexity.
7. The net thrust of a scramjet is very sensitive to the intake, combustion, and nozzle efficiencies due to the exceptionally poor work ratio of the cycle. Since the exhaust velocity is only slightly greater than the incoming free stream velocity, a small reduction in pressure recovery is likely to convert a small net thrust into a small net drag.

7.13.2 Thermodynamics

A typical layout of a scramjet engine is shown in Figure 7.36. Three processes only are found, namely, compression, combustion, and expansion. Both compression and expansion take place internally and externally. The first part of compression takes place external to the engine while the last part of expansion occurs outside the engine. The same governing equations described in Section 7.7.2 are applied here. Thus, Equations 7.15 through 7.23 are also applicable here. The five performance parameters, namely, specific thrust, TSFC, and the propulsive, thermal, and overall efficiencies, are also calculated from equations similar to Equations 7.29 through 7.33.

The previously described pressure recovery factors in Chapter 3 for ramjet engine may be also employed here. Thus, Equations 3.23 through 3.29 may also be applied.

The different modules will be briefly described in the following sections.

7.14 INTAKE OF A SCRAMJET ENGINE

The design of an air intake (inlet) of any airbreathing engine is crucial in achieving optimal engine performance. The flow pattern for supersonic flow is more complicated due to shock waves/turbulent boundary layer interaction in an adverse pressure gradient. It is necessary to design such intake for a maximum pressure recovery $r_{\mathrm{d}} = P_{02}/P_{0\mathrm{a}}$.

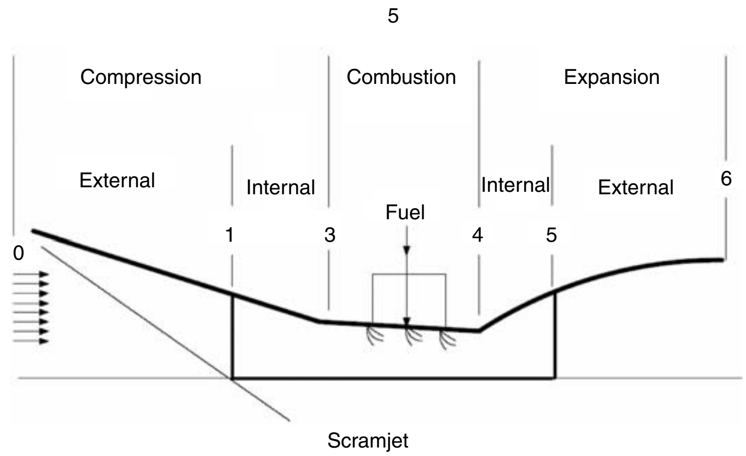

FIGURE 7.36 Layout of a scramjet engine.

Supersonic intakes are classified into three basic types, characterized by the location of the supersonic compression wave system, namely, internal compression, external compression, and mixed compression. The first type (*internal compression*) achieves compression through a series of internal oblique shock waves. The second (*external compression*) achieves compression through either one or a series of external oblique shock waves. The last (*mixed compression*) type makes oblique shock waves in various external and internal blends in order to achieve compression and flow turn. Apart from scramjet engine, in all other types of airbreathing engines the oblique shock waves described previously are followed by a normal shock to convert the flow into a subsonic one. The intake of a scramjet engine is unique by itself. It is of the supersonic–supersonic type. Both inlet and outlet flow are supersonic. The flow is decelerated through several shock waves in the intake, but all of them are of the oblique type. The flow is to be decelerated to the supersonic speed suitable for the succeeding supersonic combustion chamber. The outlet Mach number could be three or more [16].

The pressure recovery of the intake may be calculated from Equation 7.15 based on the flight Mach number or calculated from the product of the pressure ratio across different shock waves.

The details of intake design are given in Chapter 9 together with the governing equations. The pressure recovery for scramjet engines attains very small values. It depends on both the flight Mach number and the number of shocks in the intake. It reaches small values for large Mach numbers. It is in the range of 0.1–0.2 for Mach numbers close to 10 and 0.01–0.02 for Mach numbers close to 20 [16].

Case study

The intake of a scramjet engine is designed in two cases, 4 and 8 external shock waves. The free stream Mach number is 6. The intake shape together with the shock train is shown in Figure 7.37. The intake aspect ratio (height/length) is 1:5.0 and 1:10.65, respectively. The efficiency is 76.82% and 86.45% for the 4 and 8 shock intakes, respectively. The pressure recovery factor (r_d) is 0.451 and 0.642 for the four and eight shocks respectively. The intake length increases continuously with the increase of the number of shocks.

7.15 COMBUSTION CHAMBER

Combustion process here represents a heat addition process at high-speed flow (supersonic flow). Generally, the Mach number at the inlet to combustion chamber is nearly one-third of the flight Mach

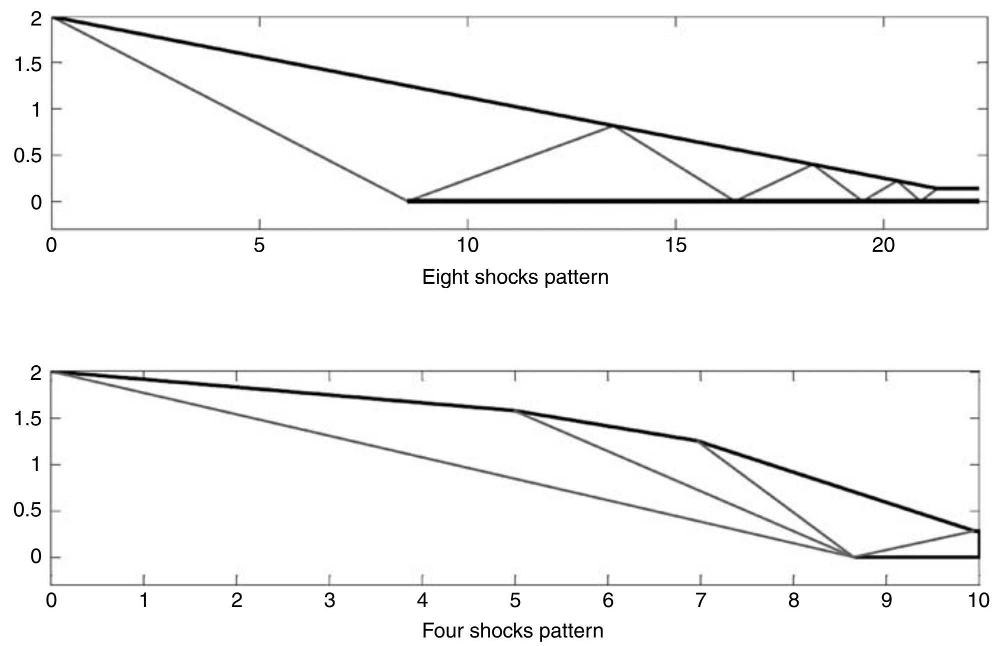

FIGURE 7.37 Eight versus four external shock wave intakes.

number. The cross-sectional area of the combustion chamber is assumed constant for simplicity. The flow may be treated as a Rayleigh flow as a first approximation. In such a simplified model, both the specific heat at constant pressure and the specific heat ratio are assumed constant. Supersonic combustion will be treated in detail in Chapter 11. Here some highlights will be given. First, the practical problems of employing supersonic combustion may be stated as follows:

1. Capture a stream tube of supersonic air.
2. Inject fuel into the air stream.
3. Achieve a fairly uniform mixture of fuel and air.
4. Carry out the combustion process within a reasonable length without causing a normal shock within the engine.

For a hydrocarbon fuel C_xH_y, the maximum combustion temperature occurs when hydrocarbon fuel molecules are mixed with just enough air so that all of the hydrogen atoms form water vapor and all of the carbon atoms form carbon dioxide. The stoichiometric fuel-to-air ratio is

$$f_{st} = \frac{36x + 3y}{103\,(4x + y)} \text{ kg fuel/kg air}$$

Fuel mixtures could be rich or lean depending on the equivalence ratio, the ratio of the actual fuel-to-air ratio to the stoichiometric fuel-to-air ratio, or

$$\phi = \frac{f}{f_{st}}$$

Combustion includes successive processes including fuel injection, mixing, and ignition.

The fuel injector for supersonic combustion should be capable of generating vertical motions, which cause the fuel–air interface area to increase rapidly and enhance the micromixing of molecular

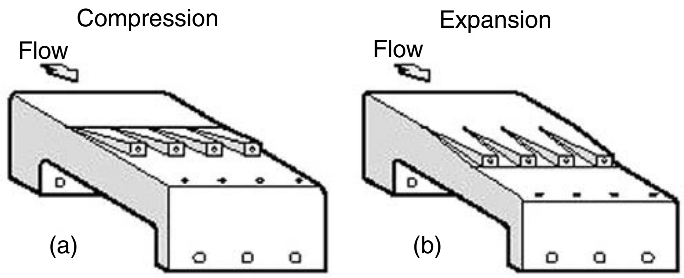

FIGURE 7.38 (a) Compression ramps. (b) Expansion ramps. (From W.H. Heiser and D.T. Pratt, *Hypersonic Airbreathing Propulsion*, AIAA Education Series, 1994, p. 278. With permission.)

level, hence leading to efficient combustion. Injection and mixing are correlated. There are two types of injection that exist either in parallel or normal streams.
(a) *Fuel mixing in parallel stream*

Ramp Injectors

Ramp injectors are passive mixing devices that produce counterrotating streamwise vortices to enhance the mixing of fuel and air (Figure 7.38).

Compression ramps, configuration (a), is associated with an oblique shock standing at the base of the ramp while the compressed air above the ramp spills over the sides into the lower pressure, forming a counterrotating pair of axial vortices. Concerning expansion ramps, configuration (b), the wall is turned away from the flow while the top surface of the ramp remains in the plane of the upstream wall, thus leading to a Prandtl–Meyer expansion so the pressure difference between the flow on the upper ramp surface and the expanded flow is produced. The location of the shock was found to affect the mixing performance, and the expansion design produces higher combustion efficiencies despite having lower overall mixing performance. This was attributed to the improved local, small-scale mixing near the fuel injection port.

(b) *Fuel mixing in normal stream*

Performance of high-speed combustor systems requires fuel and air mixing at the molecular level in the near field of the fuel injection. One of the simplest approaches is the transverse (normal) injection of fuel from a wall orifice. As the fuel jet, sonic at the exit, interacts with the supersonic cross flow, an interesting but rather complicated flowfield is generated.

Fuel–air mixing is so important and numerous studies for such process are available. Two types of mixing are examined, namely, laminar and turbulent mixing. Upon ignition and flame generation, it is necessary to have a flame holder to reduce the ignition delay time and to provide a continuous source of radicals for the chemical reaction to be established in the shortest distance possible.

Analysis of the combustion chamber may be simplified and treated as one-dimensional flow. If the flow is supersonic, adding heat lowers the Mach number and vice versa. A case study examined in [16] outlined that if the inlet Mach number to the combustion chamber is 3.0 and the heat added due to fuel burning increases the total temperature by 30%, then the outlet Mach number will be 1.7 and the total pressure ratio between the outlet and inlet states will be 0.37 [16]. Finally, two points may be added here:

1. Hydrogen fuel is mostly used in scramjet engines, which generally has heating value per unit mass some 2.3 times the conventional hydrocarbon or jet fuel.
2. Computational fluid dynamics (CFD) are extensively used in the analysis of combustion chambers to verify and overcome any difficulties in performing a complete experimental analysis.

7.16 NOZZLE

The nozzle is very similar to the intake. It has an internal part and an external part formed by the vehicle body. The internal part is a typical two-dimensional nozzle while the aftbody is the external part, which is a critical part of the nozzle component.

A two-dimensional (or planar rather than axisymmetric or circular) nozzle is used in scramjet engines planned for powering hypersonic aircraft; Mach number is greater than 5, because of the following reasons:

First, circular nozzles are comparatively heavy and do not lend themselves easily to variable geometry, and the need for tight integration.

Second, the flow at the exhaust nozzle entry is supersonic, rather than the sonic or choked throat condition of convergent–divergent nozzles.

Third, the design produces uniform and parallel flow at the desired exit Mach number because that maximizes the resulting thrust.

Fourth, the design is a minimum length exhaust nozzle. This is achieved by placing sharp corners that generate centered simple or Prandtl–Meyer expansion fans at the nozzle entry.

Case Study

An axisymmetric nozzle of a scramjet has the following data:

Inlet Mach number = 1.56

Outlet Mach number = 3.65

Inlet radius 0.366 m

An analytical procedure is followed for drawing the nozzle contour and the Mach number within the nozzle. The procedure is described in full details in Section 11.3.2.1 of the nozzle chapter. Here, only the final results are given in Table 7.5. The nozzle contours are plotted in Figure 7.39 and the Mach number within the nozzle is also given in Figure 7.40.

7.17 PERFORMANCE PARAMETERS

The generated thrust force is expressed by the following relation: as the nozzle is unchoked

$$\frac{T}{\dot{m}_a} = [(1+f)\,u_e - u_a]$$

TABLE 7.5
Nozzle Characteristics

	1	2	3	4	5	6	7	8	9	10
M	1.56	1.79	2.02	2.25	2.49	2.72	2.95	3.18	3.42	3.65
μ	39.94	33.96	29.64	26.33	23.71	21.58	19.81	18.34	17.02	15.91
θ_o	47.21	47.21	47.21	47.21	47.21	47.21	47.21	47.21	47.21	47.21
θ	47.21	40.38	33.83	27.69	22.01	16.79	12.01	7.65	3.65	0
r_c	0.366									
r	0.366	0.429	0.517	0.632	0.782	0.974	1.215	1.515	1.886	2.340
x	0	0.146	0.394	0.789	1.391	2.272	3.527	5.270	7.638	10.798
y	0.404	0.542	0.728	0.961	1.237	1.544	1.863	2.16	2.389	2.484

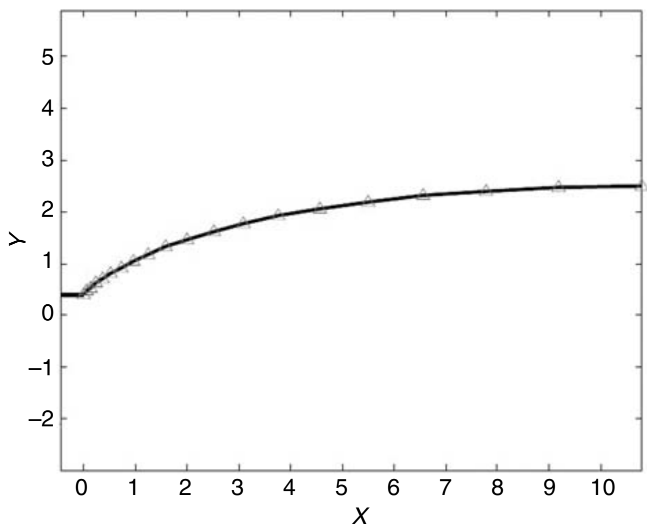

FIGURE 7.39 Nozzle geometry for a scramjet engine.

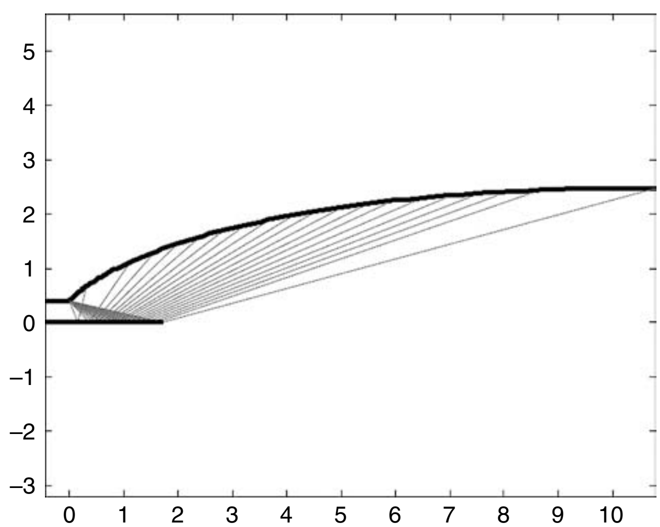

FIGURE 7.40 Mach number distribution within the nozzle.

The fuel-to-air ratio can be given by

$$f = \frac{\left(C_{p_4}T_{04}/C_{p_2}T_{0a}\right) - 1}{\left(\eta_b Q_R/C_{p_2}T_{0a}\right) - \left(C_{p_4}T_{04}/C_{p_2}T_{0a}\right)}$$

The propulsive, thermal, and overall efficiencies are calculated using the appropriate relations.

Example 1 Figure 7.41 illustrates a scramjet engine powering an airplane flying at Mach number equal to 5.0 at an altitude of 55,000 ft where $T_a = 216.67$ K and $P_a = 9.122$ kPa. Two oblique shock waves are formed in the intake before entering the combustion chamber at supersonic speed and having a deflection angle $\delta = 10°$. Hydrogen fuel is burned that gives rise a maximum temperature of 2000 K. The fuel-to-air ratio is 0.025. The nozzle has an expansion ratio $A_5/A_4 = 5$. The inlet

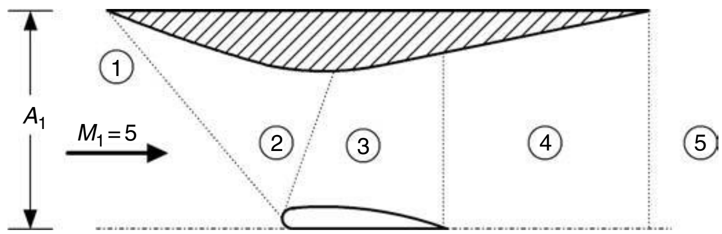

FIGURE 7.41 Scramjet layout.

and exit areas of the engine are equal, $A_1 = A_5 = 0.2\,\text{m}^2$, and the hydrogen fuel heating value is 120,900 kJ/kg. It is required to

1. Calculate the inlet Mach number to the combustion chamber.
2. Calculate exhaust jet velocity.
3. Calculate the overall efficiency.

$(C_\text{p})_\text{cc} = 1.51$ kJ/kg K, $\gamma_\text{n} = 1.238$, and burner efficiency is 0.8.

Solution:
Diffuser section
Flight speed $V_\text{f} = M_1\sqrt{\gamma R T_\text{a}} = 5\sqrt{1.4 \times 287 \times 216.7} = 1475.4$ m/s

$$T_{01} = T_\text{a}\left(1 + \frac{\gamma-1}{2}M^2\right) = 1300\text{ K}$$

An oblique shock wave separates region (1) and (2) with $M_1 = 5$ and $\delta = 10°$, then either from the governing equations listed in Chapter 3 or from oblique chock wave table/figure, the shock wave inclination angle is $\sigma_1 = 19.38°$. The normal component of Mach number ahead of the shock is

$$M_{1\text{n}} = M_1 \sin(\sigma - \delta) = 1.659$$

The normal component of Mach number behind (downstream) the shock wave is $M_{2\text{n}} = 0.65119$. The corresponding Mach number is then

$$M_2 = \frac{M_{2\text{n}}}{\sin(\sigma - \delta)} = 4.0$$

The static temperature ratio across the shock is $T_2/T_1 = 1.429$.

A second oblique shock wave is developed between regions (2) and (3). With inlet Mach number and wedge angle having the values $M_2 = 4.0$ and $\delta = 10°$, then the shock angle is $\sigma_2 = 22.23°$ and the normal component of the inlet Mach number is $M_{2\text{n}} = M_2 \sin\sigma = 1.513$.

From oblique shock tables, the normal component of the outlet Mach number is obtained from tables or normal shock relations as

$$M_{3\text{n}} = 0.698$$

$$M_3 = \frac{M_{3\text{n}}}{\sin(\sigma - \delta)} = \frac{0.698}{\sin 12.23} = 3.295$$

$$\frac{T_3}{T_2} = 1.328$$

The inlet Mach number to combustion chamber is $M_3 = 3.295$.

Combustion chamber (Rayleigh flow)

Within the combustion chamber (from 3 to 4), heat is added. The flow is a Rayleigh flow with $M_3 = 3.316$, from Rayleigh flow tables

$$\frac{T_{03}}{T_0^*} = 0.6282, \quad \frac{P_{03}}{P_0^*} = 4.53$$

Since $T_{01} = T_{02} = T_{03} = 1300$ K and $T_{04} = 2000$ K, then

$$\frac{T_{04}}{T_0^*} = \frac{T_{04}}{T_{03}}\frac{T_{03}}{T_0^*} = \frac{2000}{1300} \times 0.6282 = 0.966$$

From Rayleigh flow tables

$$M_4 = 1.26 \quad \text{and} \quad \frac{P_{04}}{P_0^*} = 1.0328$$

The outlet Mach number from combustion chamber is $M_4 = 1.26$. This confirms the fact that adding heat to a supersonic flow reduces its speed.

The pressure ratio in the combustion chamber (P_{04}/P_{03}) is calculated as

$$\frac{P_{04}}{P_{03}} = \frac{P_{04}}{P_0^*}\frac{P_0^*}{P_{03}} = \frac{1.0328}{4.53} = 0.2279$$

Note that this value is rather a small value, as normally $(P_{04}/P_{03}) \approx 0.3 - 0.4$.

The fuel- to-air ratio

$$f = \frac{C_p T_{04} - C_p T_{03}}{\eta_b Q_R - C_p T_{04}} = \frac{1.51\,(2000 - 1300)}{0.8 \times 120{,}900 - 1.51 \times 2000} = 0.01128$$

sometimes the following simple formula is used for (f):

$$f = \frac{C_p\,(T_{04} - T_{03})}{\eta_b Q_R} = \frac{1.51\,(2000 - 1300)}{0.8 \times 120,900} = 0.011$$

Nozzle

The flow in the nozzle is isentropic. From isentropic flow tables with inlet Mach number $M_4 = 1.26$, then $A_4/A^* = 1.05$ and $T_{04}/T_4 = 1.317$.

$$\frac{A_5}{A^*} = \frac{A_5}{A_4}\frac{A_4}{A^*} = 5 \times 1.05 = 5.25$$

The outlet Mach number and temperature ratio are

$$M_5 = 3.23 \quad \text{and} \quad \frac{T_{05}}{T_5} = 3.11$$

$$T_5 = \frac{T_5}{T_{05}}\frac{T_{05}}{T_{04}}\frac{T_{04}}{T_4}\frac{T_4}{T_3}\frac{T_3}{T_2}\frac{T_2}{T_1}T_1$$

with $T_{05} = T_{04}$, then

$$T_5 = \frac{1}{3.11} \times 1.317 \times 3.744 \times 1.333 \times 1.429 \times 216.7 = 654.5\,\text{K}$$

The exhaust speed is $V_5 = M_5\sqrt{\gamma_h R T_5} = 1558$ m/s.

Note also that the difference between flight and exhaust speeds are very small as identified previously, which is one of the drawbacks of scramjet engines.

The air mass flow rate is

$$\dot{m} = \rho_1 V_1 A_1 = \frac{P_1}{RT_1} V_1 A_1 = \frac{9122}{287 \times 216.67} \times 1475.4 \times 0.2 = 43.3\,\text{kg/s}$$

Thrust Force

For unchoked nozzle as usual in scramjet engines, the thrust force is

$$T = \dot{m}\,[(1+f)\,V_5 - V]$$
$$T = (43.3)\,[1.012 \times 1558 - 1475.4] = 4386\,\text{N}$$

This value is small and it is appropriate for the high-altitude flights as the drag force on air vehicles is also small.

Overall efficiency

$$\eta_0 = \frac{TV_F}{\dot{m}_f Q_{HV}} = \frac{TV_F}{\dot{m}_a Q_{AV}} = \frac{4386 \times 1475}{0.012 \times 43.3 \times 120.9 \times 10^6} = 10.29\%$$

A summary for Mach numbers at different stations in the engine is given as

$$M_1 = 5, \quad M_2 = 4, \quad M_3 = 3.295, \quad M_4 = 1.26, \quad M_5 = 3.23$$

PROBLEMS

7.1 The drag of a hypersonic future businesses jet is to be simplified by the following equations:

Flight Mach Number	**Drag (N)**
M = 0–0.75	$D = 10^5 \cdot \left(4.1M^2 - 1.1M + 2.1\right)$
M = 0.75–1.25	$D = 10^5 \cdot \left(-9.7M^2 + 20M - 5.9\right)$
M = 1.25–6	$D = 10^5 \cdot \left(\frac{21}{M^4} - \frac{37}{M^3} + \frac{26}{M^2} - \frac{5.9}{M} + 2.3\right)$