CFD in Hypersonic Flight

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ABSTRACT
This is a short review of how CFD contributed to hypersonic flights in the past 50 years. Two unexpected phenomena that occurred in the entry flights of the Apollo and Space Shuttle made us aware of the impact of the high temperature real-gas effects on hypersonic flights: pitching moment anomaly of up to 4 degrees, and radiation overshoot behind a shock wave. The so-called two-temperature nonequilibrium model was introduced to explain these phenomena. CFD techniques were developed to accommodate the two-temperature model. Presently, CFD can predict trim angle of attack to an accuracy of about 1 degree. A concerted effort was made to numerically reproduce the experimentally measured flow-field over a double-cone. As yet, perfect agreement between the experimental data and computation is not achieved. Scramjet technology development is disappointingly slow. The phenomenon of ablation during planetary entries is not yet predicted satisfactorily. In the future, one expects to see more research carried out on planetary entries and space tourism.

Key words: Hypersonic, CFD

Historical Background

The undying human dreams of flying fast, flying high, and flying far are expressed most directly in hypersonic technology. When humans were convinced that supersonic flight is possible, they immediately went on to ask how fast one could fly and how one should use the technology if a hypersonic flight were possible. Space flight was the first achievable goal. Humans successfully managed to fly into space and fly as far as the satellite of Saturn Titan. The by-product of this effort (or the hidden true purpose of the technology according to some) was the long-range ballistic missiles. The second goal was the peaceful civil commercial flight at a hypersonic speed. The calculations made in the 1950s showed that the so-called supersonic combustion jet liner, scramjet for short, should have very high fuel efficiency, efficiency equal to today's Mach 0.8 flights. This second goal has not yet been reached.

The first space flight, achieved by the Russians in Sputnik, stimulated Americans to start the Apollo moon-landing project. From the start, the high temperatures occurring during a hypersonic entry flight drew the most attention: the temperature in the shock layer over the Apollo vehicle during reentry was predicted to be about 12,000 K. The heat transfer rates to the entry vehicles were found to be of the order of several hundred watts per square centimeter. Ablative heat shield was invented.

Pitching Moment Anomaly

The high temperature occurring in the shock layer over a hypersonic flying object produces vibrational excitation, dissociation, ionization, and radiation emission. These phenomena changes the flow field, and thereby changes lift, drag, and moments. Radiative heat transfer becomes a major component of the total heat transfer rate. Most annoyingly, these high-temperature real-gas effects are not in thermodynamic equilibrium condition because the flow does not dwell at a same thermodynamic condition for long. Thus, the so-called nonequilibrium hypersonic aerothermodynamics[1] and the so-called two-temperature model became the central part of this technology.

That these high temperature real-gas effects affect aerodynamics was made painfully aware in the first Apollo return flight. Despite the fact that the pitching moment and the related trim angle of attack were determined through earlier test flights[2], apparently the values obtained were wrong. The trim angle of attack was expected to be 34 degrees, but in the first Apollo return entry, it was only 30 degrees (see Figure 1). As a result, the Apollo vehicle landed in the Pacific Ocean several hundred kilometers away from where the ships were waiting.

Later, when the first Space Shuttle vehicle reentered into the atmosphere, this pitching moment anomaly brought even more perilous impact[3]. The Shuttle was supposed to fly at the angle of attack of 40 degrees. But the vehicle started to pitch up, to about 43 degrees (see Figure 2). The flaps started to deflect to their maximum

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range, to produce the nose-down moment. Despite this flap deflection, the vehicle kept pitching up. 44 degrees was considered the angle of no return: the vehicle would have flipped over to expose its tender back side. By god's help, the pitching angle was stopped at 43 degrees. After landing, the NASA engineers immediately suspected computers. In the second and third flights, the vehicle carried four computers. But the second and third flights repeated the first flight. Then NASA engineers moved the center-of-gravity location forward for the fourth flight to offset the nose-up pitching moment. From then on, Shuttle entry had no problem.

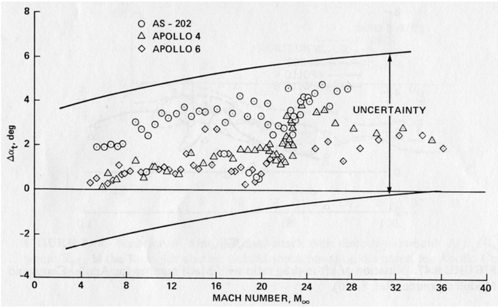

Figure 1 Trim angle uncertainties in Apollo.

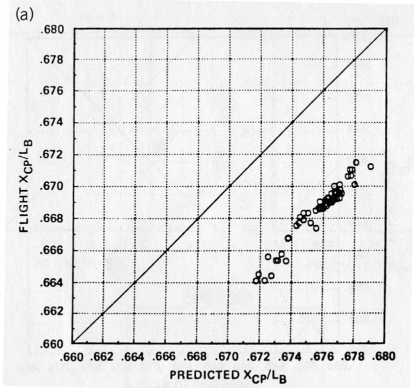

Figure 2 Center-of-pressure uncertainties for Space Shuttle.

Then the earnest search for the cause of the nose-up pitching moment began. There were three theories. The first was Mach number effect. All aerodynamic coefficients for the Space Shuttle were obtained experimentally in a cold-flow hypersonic wind tunnel at Mach 7. The actual entry flight occurred at Mach 25. The second theory was the Reynolds number effect. The wind tunnel tests were conducted at Reynolds numbers much lower than in flight. The third theory was high temperature real-gas effect[4]. After a long discussion, people agreed that the high temperature real-gas effect was indeed mostly responsible for the pitching moment anomaly[5].

With the advent of computational fluid dynamics (CFD), efforts began to computationally explain this pitching moment anomaly. In order to correctly predict the pitching moment, one must first correctly predict the shock stand-off distance for a sphere. Shock stand-off distances for spheres and cylinders have been measured previously in shock tubes and shock tunnels. But these were not trustworthy because of the possible imperfections in the test flows such freezing of dissociation in the nozzle. The only reliable test method was considered to be a ballistic range. A spherical model was flown in a ballistic range in Tohoku University to measure the shock stand-off distance in the 1990s (Ref. 6). Because the sphere had a diameter of 15 mm, the shock stand-off distance was about 1 millimeter. Measuring this shock stand-off distance accurately at a flight speed of 3500 m/s was very difficult, but was done. Then the CFD effort began in Japan to numerically reproduce it. In Figure 3, the measured shock stand-off distance and the calculated values are compared.

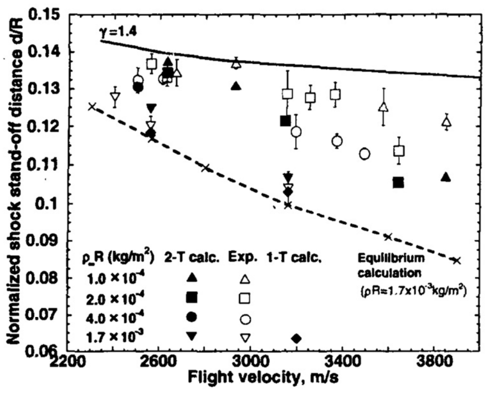

Figure 3 The shock stand-off distance for a sphere measured in a ballistic range and CFD calculations[6].

The CFD calculation shown in Figure 3 was made using the standard two-temperature model. As the figure shows, the calculated shock stand-off distances are slightly smaller than the measured values. This implies that the calculated pitching moment will be smaller than the true value for the Space Shuttle. Ref. 6 hypothesizes that this is caused by the inaccuracy in the two-temperature model and that a three-temperature model accounting for rotational temperature nonequilibrium is needed to correctly predict the shock stand-off distance and pitching moment.

Radiation Overshoot

The need for and the credibility of the two-temperature model were established by comparing the radiation emitted by the hot gas behind a normal shock wave. Before the flight of the first Apollo vehicle, shock tube experiments were conducted at speeds of up to 11,000 m/s in order to observe radiation from the hot gas. In Figure 4, a raw data from such an experiment[7] is shown. As the figure shows, radiation intensity increases and decreases, i.e, radiation overshoot occurs, behind the shock wave. This phenomenon was explained by the use of the two-temperature model[8]. In addition to predicting the intensity of radiation at its peak point, the two-temperature model was able to predict the time to the peak of radiation, as shown in Figure 5.

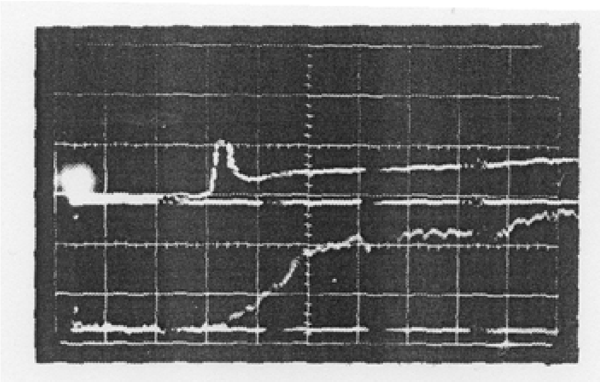

Figure 4 Radiation intensity behind the normal shock wave observed in a shock tube[7]. The top trace is at 500 nm. The bottom trace is at 6,1 μm. Shock speed = 9500 m/s.

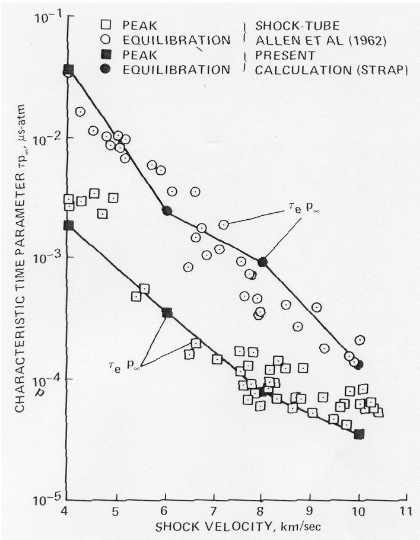

Figure 5 The time to the peak of radiation measured in a shock tube and its prediction using the two-temperature model[8].

Double-Cone Flow

In recent years, there was a concerted effort to compare the CFD calculations with experimental data. For this purpose, a double-cone, shown in Figure 6, was chosen. The distribution of static pressure and heat transfer rate over the surface were measured in a shock tunnel. CFDers were invited to compute this flow. Initially, agreement was poor between the calculations and the experimental data. After the experimental values were given, CFD calculations were able to reproduce them crudely[9] using the standard two-temperature model, as shown in Figure 7. Improvement is needed in the two-temperature model for a better agreement.

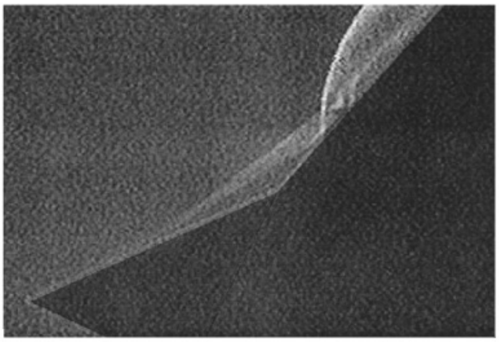

Figure 6 Double-cone flow observed in a shock tunnel.

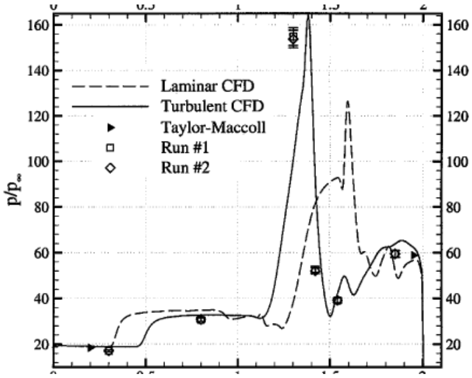

Figure 7 Comparison between the measured and calculated pressure distribution over a double-cone[9].

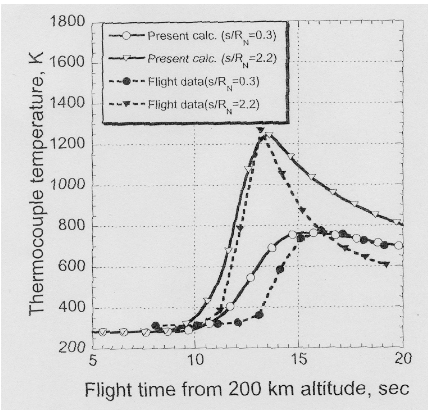

Figure 8 Comparison between the measured and calculated temperature variation over the Day Probe in Pioneer-Venus mission[10].

Planetary Entries

The Earth entry in the Apollo mission could be considered to be a planetary entry, because Earth is a planet. Over the years, entry flights were made into Venus, Mars, Jupiter, and Titan. In a Venus and the Jupiter mission, temporal changes in temperature or the recession of the heat shield surface were measured. There were four vehicles in the Pioneer-Venus mission with thermocouples[10]. For two of these, accelerometers failed to function, and therefore, entry conditions could not be determined. For the two vehicles in which the accelerometer functioned, concerted efforts were made to numerically reconstruct the heating history. As seen in Figure

8, agreement is fairly good for this case. This was still the best case: for the second vehicle, agreement was poorer.

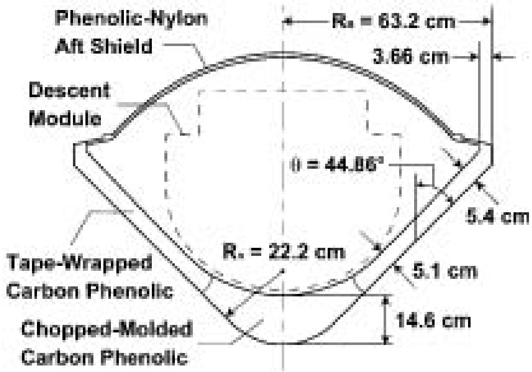

Figure 9 Sketch of the Galileo Probe which entered Jupiter's atmosphere.

The entry vehicle for Jupiter, named Galileo Probe, is sketched in Figure 9. The mass of the heat shield for this vehicle was nearly half of the total vehicle weight. The recession of the heat shield surface measured in the flight is compared with the pre-flight (Moss and Simmonds) and pos-flight (Matsuyama) calculations[10] in Figure 10. As seen, the post-flight calculation still does not fully agree with the flight data. Work is still continuing on this subject.

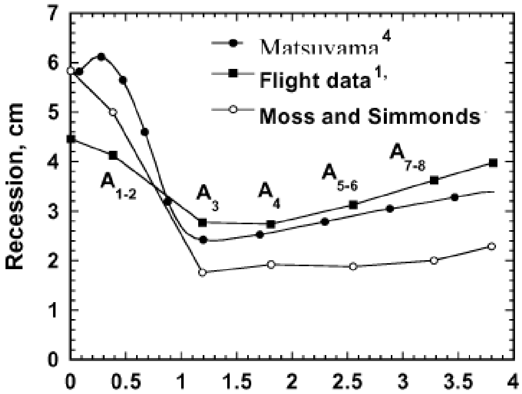

Figure 10 Comparison between the flight-measured and calculated recession of the heat shield for Galileo Probe[11].

Scramjet

The progress in scramjet technology has been one big disappointment in hypersonics. The argument for a scramjet can be made as follows: The lift-to-drag ratio of an airplane falls sharply as the flight Mach number exceeds 1 as shown in Figure 11(a). But engine efficiency should increase at hypersonic speeds because scramjet is not a heat engine but a mechanical energy converter, as shown in Figure 11(b). The net fuel efficiency should reach a high value at Mach 7, a value close to that at Mach 0.8, as shown in Figure 11(c). In the late 1980s, a very comprehensive scramjet research program named National Aerospace Plane (NASP) was conducted in the U.S. Its results are still kept secret. The main conclusion of the program seems to be that scramjet propulsion is possible with liquid hydrogen. No effort was made with hydrocarbon fuel. A simple calculation shows that only hydrogen can propel a vehicle to a Mach number in excess of 6 and only hydrogen can produce an equilvalent specific impulse substantially greater than a rocket engine.

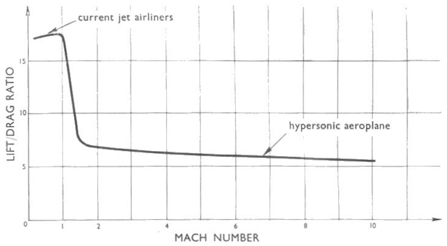

Figure 11 Ideal characteristics of a scramjet. a) L/D.

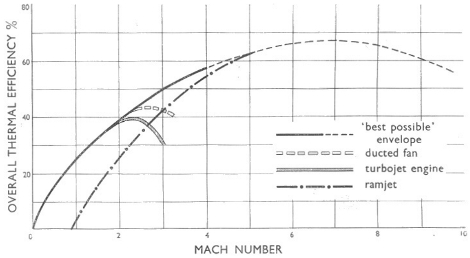

Figure 11 b) Engine thermal efficiency.

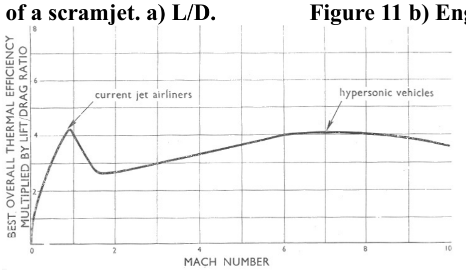

Figure 11 c) Overall fuel efficiency.

But the tank for the liquid hydrogen fuel is bulky because of the low density of liquid hydrogen. This leads to a large pressure drag. This fuel volume problem can be overcome only by making the vehicle large: the scale effect makes the relative volume of the tank insignificant when the vehicle is large. In order to bring in this scale effect, the vehicle length should be at least 50 m. Country after country, usefulness of scramjet vehicle was examined and discarded as being unconvincing. Scramjet technology is being pursued presently only as an academic subject.

The main roadblock seems to be that the high hypersonic engine efficiency envisioned in Figure 11 (b) cannot be realized because fuel and air do not mix so easily in the supersonic combustor. The exact value of supersonic mixing efficiency is still secret. But the rumor has it that it is about 2/3 at Mach 1.5 and about 1/2 at Mach 2. Such low mixing efficiency rules out any possibility of making an efficient hypersonic cruiser using hydrogen.

If hydrocarbon fuel is used, the most optimum cruising Mach number becomes about 4. As Figure 11(b) shows, the ideal engine efficiency at such a Mach number is so low that, even if it were possible to produce a hydrocarbon-burning scramjet, it would not be worthy of much interest.

Features of CFD for Hypersonic Flows

The main strength of the modern CFD stems from the fact that it can track pressure variation in an inviscid flow efficiency through the use of the so-called upwind schemes. This is achieved in turn by being able to make use of the fact that pressure waves propagate with the speed of sound. For the perfect gas flow and for a thermochemically-equilibrium flow, the speed of sound is calculable accurately. Unfortunately, speed of sound is not unique in a nonequillibrium flow: it is a function of the frequency of the pressure disturbance. As a result, upwinding implicit schemes converge not much faster than an explicit scheme in a chemically nonequilibrium flow. Therefore, several of the CFD schemes used for hypersonic flows are explicit, at least for the part that solves flow motions.

On the other hand, the chemical species production rate, i.e., the source terms, can be mathematically very stiff, which means that the associated eigenvalues are large and negative. The Adams integrating scheme can handle eigenvalues of up to -2. The Runge-Kutta integrating scheme can handle up to -3. The solution becomes unstable and blows up beyond these eigenvalues. In a practical hypersonic flow problem, eigenvalues can be easily as large as minus ten thousand. Only a fully implicit scheme can integrate such a flow. Thus, these two conflicting requirements, explicit scheme for the flow part and implicit scheme for chemistry part, led researchers to invent ingenious hybrid algorithms for hypersonic flows.

Another important feature of hypersonic CFD is the need for incorporating radiation and ablation. Determination of intensities of radiation emission and absorption became an inseparable part of hypersonic CFD. Radiation can transfer energy from the downstream to the upstream regions. Therefore, the flow field becomes mathematically elliptic. As yet, only iteration between flow solution and radiation solution is practiced.

In order to account for ablation, gas-surface interaction must be handled correctly at least. The mathematical relationship at a chemically-reacting wall produces a system of simultaneous nonlinear algebraic equations[12]. Hypersonic CFD community must yet learn to handle this correctly.

The Future of Hypersonics

The planetary exploration will continue and grow. Higher and higher accuracy will be demanded of the CFD predictions of the flow field over a vehicle entering a planetary atmosphere. The atmospheres of Mars and Venus consist of a mixture of carbon dioxide and nitrogen. The thermo-chemistry of this mixture has been studied in depth, and a great deal is known. But the outer planets' atmosphere consist of hydrogen, helium, and methane. Titan's atmosphere consists most of nitrogen and methane. Nonequilibrium thermo-chemistry and radiation properties of these gas mixtures are not unknown well. Much work is anticipated for these.

One frequently asked question is whether military application of hypersonic technology should be considered seriously in South Korea. Military application of any technology is inevitable. In the case of hypersonics, military application will be a technological fallout from planetary entry technology. Planetary exploration is pursued on its own merit, not for the sake of military application.

Another subject of interest is emergence of space tourism. Virgin Atlantic Airways of Britain, headed by Sir Richard Branson, is nearly completing its first system for carrying a layman to a sub-orbital space flight. An objective calculation reveals that, in the future, the cost for one passenger in a sub-orbital space-plane is only about 20,000 U.S. Dollars. For a fare of 50,000 U.S. Dollars a head, such a service should have enough customers. This space tourism business is bound to grow. NASA, European Space Agency, and Japan Aerospace Exploration Agency all examined space tourism as its subject of pursuit and discarded it. But, when one examine sin depth why they discarded this technology, one finds that the fundamental reason is that a government cannot spend tax payer's money to serve only the rich who can afford to pay 50,000 dollar fare.

Sir Richard has enough counsel of able aerospace engineers at present. However, the technology used by his company is quite primitive: much improvement could be made in the design of his space-plane. Figure 12 shows one such improved design of the sub-orbital space plane designed for use in South Korea[13]. No CFD work has yet been made for such space tourism applications.

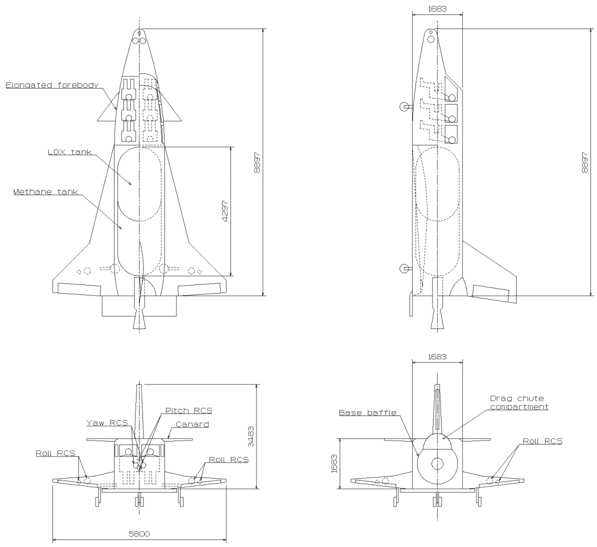

Figure 12 A sub-orbital space-plane for space tourism for use in South Korea[13].

Concluding Remarks

In the above, a very brief review is made of how CFD contributed to hypersonic technology and what is expected to happen in the future. CFD made substantial contributions in understanding the pitching moment anomaly phenomena, radiation, and ablation. Presently, trim angle of attack in hypersonic flight regime can be predicted to within an accuracy of one degree. To improve further, improvement is needed in the two-temperature model. The CFD computation of a hypersonic flow is slow because of undetermined sound speed and stiffness of chemical reactions. Radiation and ablation phenomena occurring during planetary entry cannot be predicted with any reliability still. Scramjet technology is a disappointment. In the future, planetary exploration and space tourism will grow.

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