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REAL-GAS AEROTHERMODYNAMICS TEST FACILITIES

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1 INTRODUCTION

This chapter provides an overview of the current ground-based aerothermodynamic testing capabilities in Western Europe and the United States. The focus is on facilities capable of producing real-gas effects (dissociation, ionization, and thermochemical nonequilibrium) pertinent to the study of atmospheric flight in the Mach number range of $5 < M < 50$. Perceived mission needs of interest to the Americans and Western Europeans are described where such real-gas flows are important.

The role of Computational Fluid Dynamics (CFD) in modern ground testing is discussed, and the capabilities of selected American and European real-gas facilities are described. An update on the current instrumentation in aerothermodynamic testing is also outlined.

Comments are made regarding the use of new facilities which have been brought on line during the past 3-5 years. Finally, future needs for aerothermodynamic testing, including instrumentation, are discussed and recommendations for implementation are reported.

2 MISSION NEEDS

Figure 1 (adopted from Howe 1990) is an altitude-velocity map of past missions and representative missions of future interest to the aerothermodynamics community in the United States and Western Europe. For reference, the reentry corridor of the Space Shuttle Orbiter is depicted by the line labeled Space Transportation System (STS). The black dot on the curve represents the peak heating point for the STS trajectory as do those for other trajectories to be discussed below. The higher-density ascent flight envelope for the National Aerospace Plane (NASP) type scramjet-propelled vehicle is shown by the shaded bar. Both trajectories asymptote at speeds of about 6.7 km/s to meet their mission objectives of access to Low Earth Orbit (LEO).

The Apollo lunar-return trajectory is also shown for reference in Figure 1 with its entry speed of 11 km/s. This trajectory involved landing and is called a "direct" entry as contrasted to aerocapture type maneuvers to be discussed below. The NASP Program was canceled, and American efforts to replace the Space Shuttle are now embodied in the Reusable Launch Vehicle X-33 and X-34 programs. Missions of interest to the American community involving space beyond Low Earth Orbit include new return missions into the Earth's atmosphere from the Moon, Mars, and comets. In some cases, aerocapture maneuvers will be used to decelerate vehicles by dipping into the atmosphere and exiting the atmosphere into a prescribed, lower energy orbit. The flight envelope for the aerocapture maneuver for return from bodies other than the Moon are not shown. They are similar to the lunar-return case except with higher entry speeds. Also plotted in Figure 1 is the aerocapture trajectory that was planned for the canceled American Aeroassist Flight Experiment which was to have been launched from the Shuttle Orbiter. Future piloted Mars missions must account for abort scenarios with entry speeds of up to 15 km/s as depicted by the Mars abort trajectory with a direct entry to the Earth's surface.

The Magellan spacecraft used aerobraking (Haas and Schmitt 1993) to circularize its orbit about Venus by making many high-altitude "dips" in the atmosphere. While this mission did not exhibit aerobraking where significant aeroheating occurred, it does illustrate the viability of such maneuvers.

Additional missions of interest to the Americans include a 1997 planetary entry into the atmosphere of Mars by the Mars/Pathfinder probe vehicle which is a precursor to the landing of a network of surface stations in a proposed new program called Micromet. A new Discovery class mission called Stardust will collect ejecta at a distance of 60 miles from Comet Wild-2 in 2003 and return to the Earth's atmosphere at 13.5 Km/sec in 2006. Entry technology will be involved in numerous U.S. missions in the coming decade.

Huygens-Cassini, a joint European-American program, will send a sampling probe into the atmosphere of Titan, one of Saturn's moons. This will entail an entry speed of about 6 km/s.

Figure 1 indicates regions of important real-gas flow phenomenon which must be adequately accounted for at increasing speeds. Boundaries are shown in the figure where dissociation of O_2 and N_2 occur and where ionization effects become important. These boundaries are for normal shock/stagnation regions of the flow. At altitudes below about 45 km, flow simulations can be made assuming equilibrium thermochemistry on moderately sized blunt bodies within the Navier-Stokes approximations. On small bodies or sharp leading edges, these approximations may not be valid. At altitudes above about 45 km, finite-rate chemistry must be taken into account. In many instances single-temperature CFD models break down requiring more complicated treatments such as the two-temperature, nonthermochemical models (Park 1990). At higher altitudes, the Navier-Stokes approximations break down, requiring treatments for rarefied flows (Lumpkin and Chapman 1991).

European human mission needs are first focused on the independent access capability to space. The initial mission of the European vehicles is to perform the servicing of manned or man-tended space stations, either in the frame of an international cooperation, or possibly later in full autonomy. The planned duration of the stay in orbit varies from a week to several months, depending on the operational scenario. The vehicles considered have to take into account particular geographic and geopolitical constraints, namely a launch capability from Kourou (French Guyana) and a return capability on European countries. These constraints include a preference for medium-or high-inclination orbits but imply the need for a maneuverable vehicle with significant cross range in case this vehicle is to be operated regularly. The European vehicles, as they are envisaged today, should rely on medium-level technology and yield performance capacities which are unique and complementary to those of the USA and Russia. The aerodynamic shapes under study provide a good lift-to-drag ratio (0.6 to 1) and a positive aerodynamic control in hypersonics but do not allow a conventional landing as an airplane. Other landing modes are under study, ranging from conventional parachutes to guided paragliders.

The fulfillment of these ambitions implies a large effort in Europe in the field of aerothermodynamics in which both computational and experimental design tools are developed. The creation of this new center of competence in Europe is of a nature, through the dialogue between scientists, to speed up scientific progress within NATO in hypersonic aerodynamics.

In the longer term, Europe also considers participating in human space exploration, most probably in the frame of an extended cooperation. However, the present tight-budget restrictions do not allow us to start conceptual studies for the time being. Finally, The Europeans are putting together a technology program to prepare the development of future reusable launchers. For the time being, both single-stage-to-orbit and two-stage-to-orbit concepts are considered, and both airbreathing and advanced rocket options are studied. A technology effort is made in aerothermodynamics, propulsion, and other relevant technologies.

In addition to the European programs described above, some national programs are also underway.

The German Hypersonics Technology programme terminated at the end of 1995. In its last phase (Phase Ic), emphasis was put on the RAM propulsion technology for the lower stage of the two-stage-to-orbit reference concept SÄNGER. The program is continued now in the Future European Space Transportation Investigations program (FESTIP) of ESA. SÄNGER is one of the concepts studied there. The "Sonderforschungsbereiche (centers of excellence)" of the Universities Aachen, München, and Stuttgart on Advanced Space Transportation Technology are still active. Working groups consisting of members of industry and universities are presently formed in order to keep the old contacts alive.

The French PREPHA program (Programme de Recherche et Technologie sur la Propulsion Hypersonique Avancée) is funded jointly by the DGA (Ministry of Defence), CNES, and the Ministry of Research and Technology. The PREPHA program started in 1992 with the following objectives:

- to investigate and perform ground testing of a scramjet and to study a vehicle which could validate such an engine in flight
- to maintain and improve the advance of French research in the field of technologies specific to the hypersonic regime

The program is now in its final plan and many results have been obtained, both from the theoretical (modeling of physical phenomena and improvement of CFD tools) and experimental (scramjet tests performed at Mach 6 in the Aérospatiale, Bourges, Subdray facility) points of view.

The activities performed covered aerodynamics (external and internal) propulsion, materials and structures, as well as system studies of a generic vehicle.

Finally, we note that two recent discoveries that are leading to renewed interest in solar system exploration: 1) meteors thought to have come from Mars contain suggestions of extraterrestrial life, and 2) the Galileo Spacecraft's photographs suggest the possibility of warm, life giving, liquid water on Europa. Since solar system exploration involves hypervelocity, atmospheric flight, this interest is likely to rejuvenate waning interest in hypersonic facilities worldwide.

3. MODERN APPROACH TO AEROTHERMODYNAMICS

The traditional aerodynamic design tools for aeronautics and space projects, i.e., wind tunnels, semiempirical codes, and flight testing, have been supplemented by CFD due to rapid advances in computer power and algorithm developments.

As with all design tools, the CFD codes must be validated; i.e., they must be checked against a range of experiments and experimental conditions spanning the range of involved flow physics and chemistry before they can be considered as serious design tools.

There currently is a dearth of archival quality, benchmark experimental data suitable for CFD code validation and calibration. What defines a benchmark experiment is still open to question and debate. One version of the structure that sorts and classifies the types of testing is shown in Table 1. Included is a statement on the necessary acceptance criteria for the data, the facilities that cover the range of testing required, desirable acceptance criteria and data completeness and accuracy requirements.

TABLE 1

CALIBRATION OF HYPERSONIC CFD CODES

CLASS	PURPOSE	EXAMPLES
1. Phenomenological data	Understand flow physics	Studies of large-scale structures
2. Unit problem data	Assess a model incorporated in a CFD code	Simple shear-layer data
3. Component data	Assess code's ability to analyze comp. of overall flow field	Rotor blade in wind tunnel
4. Performance data data/complete flow field	Assess if code predicts eng. parameters	Nozzle thrust
5. Full or subscale	Assess code ability to analyze specific flow parameters	Blunt body heating

Necessary Acceptance Criteria:

1. Baseline applicability ($M > 3$)
2. Simplicity
3. Specific applicability
4. Well-defined experimental boundary conditions
5. Well-defined experimental error bounds
6. Consistency criterion
7. Adequate documentation of data
8. Adequate spatial resolution of data

Facilities Required:

- conventional hypersonic wind tunnels
- high enthalpy or real-gas facilities
- rarefied facilities
- airbreathing propulsion testing
- materials testing

Table 2 shows another set of criteria for an ideal benchmark experiment for CFD validation as reported from the Antibes meeting on CFD code validation.

TABLE 2

CRITERIA FOR AN IDEAL BENCHMARK EXPERIMENT FOR CFD VALIDATION

1. Appropriate For CFD Validation:
   a. Simple enough for economical CFD treatment.
   b. Universal enough for applicability to numerous real world problems.

2. Model Flow field Adequately Characterized:
   a. Boundary conditions defined: T_w, $\sqrt{\rho c k}$, m, catalytic effects, etc.
   b. Boundary-layer surveys made (laminar or turbulent state defined)
   c. Force, pressure, and heat data available as appropriate.
   d. Shock locations measured (Optically or by other means)
   e. Model attitude accurately measured ($\leq 0.1°$)
   f. Base pressure measured.

3. Model Fidelity Sufficient:
   a. Sharpness/bluntness accurately manufactured, maintained, and defined.
   b. Surface conditions quantified.
   c. Model shape faithful to the defined configuration.

4. External Flow field Characterized:
   a. Mach and Reynolds numbers, T_0, T_∞, T_{vib}, P_∞, and time accurately defined.
   b. Chemical/energy states of test gas defined (frozen, nonequilibrium, etc.)
   c. Gradients of pressure, temperature, velocity defined in terms of the primary variables.
   d. Contamination levels measured; potential effects noted.
   e. Flow angularity measured in the test area.

5. Variables Identified and Controlled:
   a. External flow-field variables varied in an orderly and rational manner.
   b. Model variables varied in an orderly and rational manner.

6. Facility and Instrumentation Adequately Described:
   a. Principles of operation as they affect data.
   b. Limitations.

7. Data Uncertainties Defined For All Measurements:
   a. Defined by standard methods and convention.
   b. Repeatability demonstrated.

CFD clearly can be an essential design tool for hypersonic vehicles because wind tunnels do not provide full simulation at velocities above about 3 km/s (or Mach number above 5 - 10). This raises the question of how to extrapolate wind tunnel results to flight, addressed in chapter V. Semiempirical codes are inadequate and flight tests are exceedingly expensive. Moreover, CFD can directly assist in the improvement of wind-tunnel designs and in the efficient use of wind tunnels. Examples of methods already in use are

1. Hypersonic nozzles are validated and even designed with CFD methods, using available models for transition and turbulence.
2. Decisions on the relative importance of a given flow parameter for vehicle design are made by performing sensitivity studies using CFD.
3. Estimations of the allowable uncertainty for a given measurement technique are made by performing sensitivity studies with CFD.

However, CFD has the potential to play an even more important role in the future. CFD codes could replace the role of empirical design codes when new facility concepts are considered. For example, CFD can be used to predict the time-dependent operation of a new facility concept by simulating the influence of diaphragms, valve ports, shock propagation and reflection, etc. on the overall flow development. In other words, the entire aerodynamic design of a proposed new facility could be carried out with CFD. Optimization of the design may also be envisaged.

This process would permit the construction of a pilot tunnel with enhanced confidence; the pilot tunnel, in turn, would serve to validate the codes whose improved versions could then be employed to design the full-scale facility. A caveat: a pilot tunnel implies different physical scaling which may alter the relative importance of certain physical phenomena!

Clearly, the procedure can be continued to predict the gasdynamic state of the flow in the test section and even the flow nonuniformities. Such knowledge is the first step leading to the elimination of nonuniformities and will improve the credibility of wind-tunnel testing and in the long run will reduce the number of wind-tunnel tests required for a given design effort.

This philosophy is already producing results. In one example, Wilson et al. (1993) have published time-dependent simulations of reflected-shock/boundary-layer interactions in a cold-flow model of the NASA Ames Electric Arc Shock Tube called the E.A.S.T. Facility. The results on the mixing of driver/driven gas are in qualitative agreement with experiments and have shed new light on the effect of this phenomenon on the prediction of reservoir conditions. In another example, Bakos et al. (1996) have used CFD to aid in the design, calibration, and analysis of a tunnel mode of operation of the NASA Hypulse Facility with excellent results. At present, these computations are very CPU intensive, but if past experience is a guide, continued increases in computer power and algorithm efficiency will reduce this limitation.

It can be concluded that the pacing item in the use of CFD for facility design and flow characterization is code validation carried out through the use of building-block experiments. Only when facility designers have confidence in CFD codes will these codes be used for the design of major facilities. The promise to aerothermodynamics held out by this modern approach is so large as to fully justify the important investments that will be necessary to bring it to fruition.

4 EXISTING FACILITIES

This section begins with a broad overview of the simulation capabilities of various types of wind tunnels designed to study hypersonic flows and to simulate aspects of hypersonic flight. Then a brief summary of the characteristics of the major real-gas, high-Reynolds number, and rarefied facilities for aerodynamic research will follow.

Above Mach numbers of 8 to 10, *duplication* of flight conditions cannot be attained in most existing ground-based facilities with models of realistic size, even for entry from LEO; the upper limit is even lower if scramjet vehicles or entry from lunar or higher energy orbits are considered. Thus, *simulation*, which is the duplication of the essential dimensionless parameters characterizing the specific flow problem of interest, is the only methodology generally available at present. (It must be emphasized that if nonequilibrium chemical reactions are present, full simulation cannot be achieved in the most general case. Only full-scale testing at real flight conditions will suffice). As a general rule, blunt bodies require Mach-number, binary-gas-number (ρL), and T_0/T_w simulation; slender bodies require Reynolds-number simulation as well. If surface radiation effects are important, then absolute wall temperature must be duplicated which leads to the so-called "hot-model" technique (Hirschel 1991).

Figure 2 summarizes the Reynolds-Mach capabilities of European and American facilities; an overlay of two mission trajectories is shown for comparison. Figure 2(a) is adopted from Holden et al. (1995) and 2(b) is from Wendt (1992). A clear need for high-Reynolds-number tunnels is evident if the mission is to be a single-stage-to orbit (SSTO), air-breathing vehicle. Figure 3 shows the situation with regard to the simulation of ρL for the European facilities. The newest European and American facilities (F4, HEG, T5, LENS etc.) were designed with this phenomenon in mind.

Figure 4 summarizes the capabilities of existing facility types in terms of stagnation temperatures they can produce as a function of run time. The figure shows that shock tubes are useful in simulating gasdynamics and kinetics where the duration of chemical processes are the same or greater than that for which the flow persists (1μs to 10μs). For example, nonequilibrium radiation behind normal shocks. The next realm from 10μs to 10 ms can be studied by shock tubes, shock tunnels, and ballistic ranges. Gasdynamics and kinetics can be studied at the lower range of the time scale. At the upper end of the time scale, the researcher can study aerodynamics and flow field definition in the hypersonic regime where real-gas effects are important. Shock tunnels are generally driven with a free-piston driver or a shock tube. Flow quality and chemical cleanliness are of concern to these facilities. Ballistic ranges generally launch into a quiescent gas and contamination is not a concern. However, current ballistic-range-model scales are quite small. The realm of simulation from 10 ms to 2 s of flow can be explored with hot-shot and long-shot tunnels and high-performance blowdown tunnels. Here, the flow duration is sufficiently long to observe controlled motion of the test article, enabling detailed studies of aerodynamics. Finally, conventional blowdown tunnels can provide long-duration flows of up to minutes, but these facilities cannot usually produce real-gas flows. Most arc-jet facilities can provide long-duration, real-gas flows. An arc jet is basically a wind tunnel in which energy is added to the flow with a high-power arc discharge. Arc jets generally do not have aerodynamic quality flows and are used to study heat shield materials. This chapter will not consider further conventional blowdown tunnels or arc jet facilities.

Thus, a given mission and flight vehicle will define the relative importance of the various dimensionless parameters whose values must be matched in the facility. In addition to the fluid dynamic parameters, it is clear that such elements as cleanliness (freedom from solid particles) and flow uniformity (spatial and temporal, including wind-tunnel-generated pressure fields) must be quantified. It should be noted that some types of measurements may be carried out to an acceptable degree of precision in spite of the presence of noise, dirt, fluctuations, etc.; however, other important phenomena may be totally obscured by some of these effects. (The well-known influence of nozzle-wall turbulent boundary layers on the location and character of boundary-layer transition is a classic example).

An accurate knowledge of facility characteristics is obviously essential if tests of high quality are to be undertaken. High-enthalpy facilities are, however, particularly difficult to characterize due to their short running times, the presence of particles, nonequilibrium conditions, etc. Therefore, the need for sophisticated instrumentation techniques, which is summarized in Section 6 of this chapter, is of paramount importance.

4.1 United States Facilities

4.1.1 NASA Ames 16-Inch Shock Tunnel

The 16-Inch Shock Tunnel is being currently in a standby mode. Until recently, it was used for scramjet propulsion testing in support of the NASP program and the NASA Hypersonics Research Programs. The description below is a contraction of information contained in Cavolowsky et al. (1992).

A schematic of the 16-Inch Shock Tunnel is shown in Figure 5. The driver section consists of a tube 21 m long with an inside diameter of 432 mm. The driven section is 26 m long with an inside diameter of 305 mm. The shock tunnel received its name from the 16-inch naval rifles used to construct its driver section. The shock tunnel is rated at 680 atm maximum driver pressure. The contoured Mach 7 nozzle is 5.8 m long and has an exit diameter of 990 mm. Interchangeable throat sections are used to vary the nozzle area ratio. Results have been obtained for area ratios of 190, and a minimum of 95 as attainable without sacrificing test time or ideal shock tube end wall behavior. The test cabin is a 1.82 m long by 1.37 m square cross-section box located immediately downstream of the nozzle exit. Flow simulations equivalent for Mach 12, 14, and 16 have been achieved in the tunnel. Uncontaminated flow times of 3-5 ms are routinely seen in the 16-Inch Shock Tunnel.

It is important to note that although its recent efforts were directed toward propulsion testing and research studies, the 16-Inch Shock Tunnel is not restricted to this use. It could be valuable to experimental and computational research involving real-gas, blunt-body aerothermodynamics. This includes flight trajectories for spacecraft that will be studied as part of the Mars mission program and NASA's efforts to return to the lunar surface. Future plans will include calibration of test conditions required for these and other flight programs.

4.1.2 Large Energy National Shock Tunnel (LENS)

The LENS facility described by Holden, et al. (1995) is a reflected shock tunnel, and its basic components are shown in Figure 6. The driver/driven configuration consists of a chambered shock tube with an area ratio (driver/driven) of two. The 292 mm internal diameter driver is 7.6 long and is externally heated by a resistance heater to 2270 K. The driven tube has an internal diameter of 203 mm and is 15.2 m long. The test section has a diameter of 2.43 m. Two nozzles are employed to cover the Mach number range from 6 to 18. a contoured nozzle is used for the Mach 6 to 9 range, and its exit plane diameter is 1.06 m. A conical nozzle is used for the Mach 10 to 18 range, and its exit diameter is 1.22 m. The nozzles employ replaceable throat inserts of different diameters so that, with a particular nozzle, the test Mach number can be varied. Both nozzles are calibrated using pitot pressure survey rakes over the Mach number ranges indicated. This facility can produce flow velocities from 0.91 to 4.9 km/sec. Test times vary from 4 milliseconds at the higher velocities to 20 milliseconds at the lower velocities for reservoir pressures up to 2,040 atmospheres. Figure 7 shows the altitude/velocity performance map for the LENS facility and compares its capability to Shuttle entry trajectory and to those of other U.S. facilities.

Aerothermodynamic instrumentation associated with the LENS facility permits surface measurements of heat transfer, pressure, and skin friction. Force and moment instrumentation is available as are schlieren, holographic interferometry, cine and video, total pressure, heat transfer and temperature gages for flow field measurements. Non-intrusive instrumentation includes electron beam, laser diode, PLIF, LIF, spectroscopy, and microwave interferometry. Extensive instrumentation is available for evaluation of the aerothermal and aero-optical performance of hypervelocity missile interceptors.

4.1.3 Boeing Hypersonic Shock Tunnel

The Boeing Hypersonic Shock Tunnel HST was built in the early 1960's and was brought on-line to support the development of the X-20 program. It was deactivated in 1981 and restored in 1987.

The major components of the HST are a 4 m long, 76 mm diameter combustion heated driver, a 7.6 m long, 76 mm diameter driven tube, contoured nozzles (305 mm and 762 mm exit diameter), a test section/dump tank, and a high-speed data recording system. Ignition of the driver gases (hydrogen and oxygen in helium) is initiated by 21 spark plugs placed in a spiral pattern along the length of the driver.

The HST uses a double diaphragm arrangement and operates as a reflected-wave shock tunnel. The Mach 5 to 8 range is covered with the 305 mm diameter nozzle while the 762 mm diameter nozzle is used for the Mach 8 to 20 range. Test times range from 2.5 to 5 ms, depending upon the total temperature being run. Reservoir conditions to 8000 K and 400 atm are available for high-enthalpy simulations.

The primary measurements made in the tunnel are surface pressure and aerodynamic heating rates using either platinum thin-film gauges or coaxial thermocouples.

Free-stream velocity has been measured using an exploding wire and a photo diode array. Instrumentation also includes high-speed cameras, shadowgraph and Planar Laser-Induced Fluorescence (PLIF).

4.1.4 Northrup Grumman Research Detonation Shock Tunnel

The Northrup Grumman Research detonation shock tunnel has been used for over 25 years to simulate high-temperature gas flows for programs including the Shuttle Orbiter and the National Aerospace Plane.

The tunnel consists of a 19.8 m long, 127 mm ID sectional tube assembly exiting through a nozzle into a 1.83 m diameter by 3.7 m long vacuum chamber. The nozzle has a rectangular cross section with a nearly square throat 38 mm in height, varying to a 38 x 114 mm exit plane.

The shock tunnel operates by first rupturing its primary diaphragm, allowing pressurized helium from the 6.1 m long driver to propagate into the driven tube. The driven gas for combustion simulation studies consists of a detonable mixture of nitrogen, oxygen, hydrogen, and argon. The driven section is isolated from the vacuum chamber by a secondary mylar diaphragm at the entry to the facility nozzle. Test times of 2-3 milliseconds are achievable with this facility.

Total enthalpy conditions corresponding to free-stream Mach numbers between 8 and 12 are produced by varying the composition of the driven gas and the pressure of the helium driver gas.

Instrumentation consists of wall pressure, heat flux, and optical systems. The optical systems consist of holographic interferometry; laser extinction and absorption; UV and visible OMA and photometers; and infrared (IR) radiometers, arrays and cameras.

4.1.5 GALCIT T5 Shock Tunnel

The T5 facility is a free-piston, reflected-shock tunnel located at the Graduate Aeronautical Laboratory of California Institute of Technology. The tunnel became operational in December 1990. Hornung (1992) describes the facility and its performance. T5 has been used for graduate research and industrial testing in over 1200 runs as of spring 1996.

The compression tube is 30 m in length and is 300 mm in diameter, while the driven tube is 12 m in length and is 90 mm in diameter. The nozzle has a throat diameter of 31 mm, an exit diameter of 310 mm and a length of 1 m. The driver gas is helium or a helium/argon mixture, and the maximum burst pressure is 1,300 atm. The facility test times range from 0.2 to 5 ms, depending upon operating conditions. The tunnel is equipped with a hydrogen injection system with speeds up to 5 km/s. Instrumentation consists of wall pressure, heat flux, and optical systems: schlieren, interferometry, and differential interferometry.

4.1.6 NASA's HYPULSE Facility at GASL

The HYPULSE Facility is an expansion tube/tunnel which was originally built at NASA Langley in the 1960's, decommissioned in 1983, transferred to GASL, Ronkonkoma, New York, in 1987, refurbished, and recommissioned in 1989. Erdos et al. (1994) have recently described the GASL facility and its expanded operational envelope.

Figure 8 shows a wave diagram describing the basic operation of an expansion tube. This figure assumes a not-yet funded free-piston driver is being used. Typically, about two-thirds of the total enthalpy and total pressure is generated through the unsteady expansion fan and occurs in the test section mainly in the form of velocity. The tunnel mode of operation uses a divergent (throat-less) nozzle at the end of the tube to increase the Mach number and size of the test section.

The HYPULSE Facility components consist of a 2.44 m long, 165 mm ID, driver tube rated for an operating pressure of 1360 atm, a 2.44 m long, 152 mm ID intermediate tube rated for 1000 atm, and a combination of intermediate and acceleration tubes of 152 mm ID, totaling 19.66 m long, rated for 525 atm. The test section/dump tank is 11 m long with a 1.2 m ID. In a typical configuration, the intermediate tube is 7.5 m long, and the acceleration tube is 14.6 m long. The lengths of these two sections are variable. An additional two sections of acceleration tube, each 2.29 m long, are available but not installed, as is a divergent nozzle having an exit diameter of 635 mm. The driver can be operated with either room-temperature helium, helium-hydrogen, or helium-nitrogen mixtures or with detonatively heated hydrogen-oxygen-helium mixtures.

The test section conditions are varied by varying the driver conditions and the fill pressures of the intermediate and acceleration tubes. The facility has been used extensively at a total enthalpy of 15.2 MJ/kg for hypervelocity aerothermodynamic studies. This condition is achieved with room-temperature driver gas and provides a free stream velocity of 5.33 km/s, a static temperature and static pressure of 1200 K and 1.8 kPa, respectively, a unit Reynolds number of 6.6×10^{5} m^{-1}, and a total pressure of 163 MPa. The test time at this condition is 0.3 ms. The facility has also been calibrated and operated at total enthalpies from 7.5 to 17.3 MJ/kg, primarily for supersonic/hypersonic combustion studies. At these conditions, the free-stream velocities vary from 3.5 to 5.7 km/s; the static temperatures vary from 1100 to 2400 k; the static pressures vary from 1.5 to 150 kPa; and the total pressures vary from 60 to 400 MPa. The total test time available for flow establishment and data collection/averaging varies from 0.3 to 0.8 ms. The quoted conditions are all for air as the test gas. The facility has also been calibrated and operated over a similar range of hypervelocity conditions using pure gases, including nitrogen, oxygen, helium, and carbon dioxide as test media.

Instrumentation available at HYPULSE includes pressure and heat flux (up to 160 channels sampled at up to 1 MHz with 12-bit resolution) and various optical devices: laser holographic interferometry, schlieren, shadowgraph, various types of spectroscopy, and time-averaged Mie scattering imagery.

Supporting systems for studies involving gas injection include two Ludweig tubes and a shock tube (installed inside the dump tank) for delivering room-temperature or shock-heated gases to a model in synchronization with start-up of the primary test gas flow.

Instrumentation available for testing include laser holographic interferometry, emission measurements, spontaneous raman spectrascopy (vibrational and rotational temperature measurements), and UV absorption spectroscopy (NO and O concentration measurements).

4.1.7 NASA AMES Electric Arc Shock Tube

NASA Ames's electric arc-driven shock tube facility has been in existence since the 1960's. This facility is currently in standby mode. The operating characteristics of the facility are described in Sharma and Park (1990). A photograph of the facility is shown in Figure 9. The facility consists of one driver system and two parallel driven tubes. One is a 100 mm ID tube 12 m in length, and the other is a 600 mm ID tube 21 m in length. The driver can be operated in two configurations: (1) a 177 mm conical drive configuration with a 101.6 mm exit (driver volume = 0.632 ℓ), and (2) a variable length (340-1370 mm) 100 mm ID cylindrical configuration (driver volume = 2.7 to 10.7 ℓ). The length of the cylindrical drivers can be varied by using a Lexan filler plug.

Energy to the driver is supplied by a 1.24 MJ, 40 kV capacitor energy storage system. By using the two different driven tubes, varying the driver/driven gas combination, driver charge pressure and preset capacitor bank voltages; normal shock velocities in the range of 3.0 - 50.0 km/s, with unshocked test gas pressures at the higher velocities in the range of several tenths to several torr, have been obtained. In order to minimize the level of impurities, contact of the test gas with steel and any material containing carbon or hydrocarbons has been minimized. Past experience shows that the spectra of the test gas, which was in contact with steel wall and carbon (burned mylar diaphragm), were overwhelmed by spurious emission spectral lines of iron and CN-violet (Sharma and Park 1990). For this reason, aluminum diaphragms and an aluminum 100 mm ID driven tube are used. Instrumentation consists of laser holographic interferometry, emission measurements, spontaneous Raman spectroscopy (vibrational and rotational temperature measurements), and UV absorption spectroscopy (NO and O concentration measurements).

4.1.8 NSWC Hypervelocity Tunnel Number 9

The Hypervelocity Wind Tunnel 9, located at the Naval Surface Warfare Center in White Oak, Maryland, is a blow-down facility which currently operates at Mach numbers of 7, 8, 10, 14, and 16.5. Tunnel 9 provides a high Mach-number and Reynolds number testing environment with usable test times up to 15 seconds and a 1.5 meter diameter test section.

A schematic of Tunnel 9 is shown in Figure 10. A vertical heater vessel is used to pressurize and heat a fixed volume of nitrogen to a predetermined operating pressure and temperature. The test section and vacuum sphere are evacuated and separated from the heater by a pair of metal diaphragms. When the nitrogen in the heater reaches the desired temperature and pressure, the diaphragms are ruptured. The gas flows from the top of the heater, expanding through the contoured nozzle into the test section at the desired test conditions. As the hot gas exits the top of the heater, fast-acting valves are opened allowing cooler nitrogen from four pressurized driver vessels to enter the heater base and maintain a constant heater pressure. The cold gas drives the hot gas out of the heater in a "fluid piston" fashion while maintaining constant conditions in the test section during the run.

Tunnel performance characteristics are given below:

Contoured Nozzle [Mach]	Supply Pressure [MPa]	Supply Temperature [K]	Reynolds Number [10^6 m^{-1}	Usable Test [sec]
7	13 - 90	1920	6 - 55	1 - 6
8	13 - 83	920	28 - 183	0.2 - 0.75
10	3.5 - 97	1005	3 - 72	0.2 - 15
14	0.7 - 131	1755	02. - 20	0.7 - 15
16.5	131 - 145	1810	9 - 11	3

Tunnel 9 Capabilities are under continuous development. In 1995, a new facility leg with full-flight duplication at Mach 7 for providing a thermal/structural test environment was brought on line. This capability matches the true temperature and pressure at flight altitudes as low as 15 km with run times up to 6 seconds. Future developments will allow this test leg to duplicate flight altitudes down to 10 km. Ragsdale et al. (1993) and Lafferty et al. (1996) have discussed the capabilities of NSWC Tunnel 9.

4.1.9 Ballistic Ranges

With its clean test-gas environment, the ballistic range provides correct thermochemistry at true-flight enthalpy. The enthalpy comes from the use of light-gas guns which can launch projectiles at speeds in the 3 to 9 km/s range. However, the model scale is currently very small and there is concern that the test capability is inadequate. A very detailed AGARD Report (AGARDograph 138), primarily discussing the NASA Ames ballistic ranges of the 1970 time frame, details the methodology of the ballistic range, including the use of a shock tunnel to provide a counter flow, enabling the simulation of lunar return (11 km/s) entry of the Apollo vehicles. With the advent of CFD, the role of the ballistic range has changed to become more of a validation tool.

Below is a synopsis of ballistic ranges in the U. S. as in the 1996 time frame, based on a more detailed review by Chapman (1992).

NASA Ames Research Center

Ames has four two-stage light gas launchers ranging in size from 7.1 mm to 38.1 mm diameter. These launchers were designed to provide low-acceleration (soft) launching. Ames has two facilities which use these launchers: the Hypervelocity Free-Flight Aerodynamic and Radiation Facilities. The aerodynamic facility uses the 16-Inch Shock Tunnel discussed above as its counterflow source, but the facility has not been operated in this mode for over 20 years. The facility has a 25 m long test section with 16 orthogonal shadowgraph stations. The test section was sized to the capability of the shock tunnel to provide a slug of moving test gas of this length. It is capable of conducting aerodynamic testing at hypervelocities and can yield good quality flow visualization and aerodynamic coefficients for simply-shaped vehicles. The aerodynamic range is currently operated as an impact facility, while the 16-Inch Shock Tunnel is in standby status. The radiation facility is configured solely for gun development and impact testing.

Arnold Engineering and Development Center: AEDC has several launchers. The largest is 62.5 mm in diameter, and there is one being designed and built which is 82 mm in diameter. All of these launchers have been optimized with operational experience for low acceleration launch loads. The new launcher has been optimized from the design stage. There are two long variable pressure ranges, the longest being 300 m in length. This range is also designed for either free-flight launches or launching onto a rail. The rail launch system also allows for recovery of models. Besides the conventional range instrumentation, this range is currently instrumented with spectrometers for wake-flow diagnostics. There is also ongoing work to develop planer laser induced fluorescence (PLIF). This facility has the best set of diagnostics equipment of any in the United States at the present time.

University of Alabama, Huntsville: This range was previously located at the Delco facility in Santa Barbara, CA. It has a 62.5 mm diameter launcher that has been extensively optimized to minimize launch accelerations. The range is over 300 m long with variable pressure capability. This facility has the radiometric instrumentation that was at Delco. This instrumentation needs to be updated if it is to be useful for detailed flow-field studies and CFD code validation.

Wright Laboratory Armament Directorate at Eglin Air Force Base: This is a sea-level atmospheric pressure range of over 200 meters in length. The launcher room is small, and hence the light-gas launcher is small and not optimized for soft launches. It also has an optical system that cannot reject optical radiation from the model and hence is limited to 3 to 4 km/s. However, in this speed range it has the best developed aerodynamic determination system in operation at the present time. There are plans to replace all of the conventional cameras with electronic cameras and to fully automate the aerodynamic data-reduction procedure to provide rapid determination of aerodynamic parameters.

Lawrence Livermore National Laboratory: LLNL has developed a 100 mm two-stage light-gas launcher for potential application as a space launcher. This launcher has two novel features. First, the driver tube (1st stage) and the launch tube (2nd stage) are at right angles; hence the launch tube can be elevated for firing without altering the driver. This could be a useful feature for saving space in a constrained area. It also presents an opportunity for using the pump tube (first stage) as a free-piston driver for a shock tunnel without significant interference with the ballistic-range portion of the facility. The second feature is that it uses methane-oxygen combustion as the driver for the first stage piston rather than the conventionally used gunpowder. This launcher with the extensive advanced instrumentation base that exists at LLNL could be the nucleus of an aerothermodynamic testing capability. What is missing is a variable pressure test range. A total of about

30 shots have been fired as of fall 1995. Some were in support of scramjet testing. These shots were with 5 kg models with launch speeds of about 3 km/sec. Whether this facility contributes to the aerothermodynamic testing capability remains to be seen.

4.1.10 UC-Berkeley Rarefied Gas Wind Tunnel

The configuration and instrumentation of the University of California at Berkeley rarefied-gas wind tunnel have changed greatly from their state in the 1960's and 1970's (see Figure 11 for schematic of present facility Gochberg (1993). Currently, the electron-beam fluorescence technique is used as the primary experimental diagnostic tool, measuring density and rotational and vibrational temperatures in the hypersonic, low-density flows generated using free-jet expansions. A ceramic resistance heater with a maximum operating temperature of 2000 K functions as the flow reservoir and can be operated with virtually any gas, including oxygen. The facility is capable of producing shock Mach numbers for nitrogen in excess of 20 at the highest stagnation temperature available. The shock barrel is 65 mm long with a Mach disk diameter of 55 mm for this condition.

4.1.11 AEDC Free-Piston Shock Tunnel

A new concept for a very high pressure free-piston shock tunnel being developed at AEDC has been described in Maus et al. (1992). Figure 12 depicts the evolution of this concept from the conventional free-piston, light-gas gun. In the light-gas gun, the disposable piston is driven by gunpowder, compressing hydrogen to a high pressure. The diaphragm bursts at a prescribed pressure and the projectile is accelerated through the launch tube into free flight.

In the disposable free-piston shock tunnel, the piston propellant is moderate-to-high-pressure air, and the compression gas is helium. The diaphragm bursts and drives the rest of the facility as a conventional shock tunnel. Conventional free-piston shock tunnels are limited to about 2000 atm stagnation pressures to avoid damage to their reusable pistons and gas leakage. In the light-gas-gun operation, the deformable piston seals the gases by extruding the piston into a tapered section. Pressures in excess of 10^4 atm are routinely achieved in this manner. Maus et al. (1992) state that this concept for a disposable free-piston shock tunnel has the potential of attaining stagnation pressures as high as 10^4 atm with enthalpies over 20 MJ/kg. Their paper discusses pilot experiments for this concept and makes comparisons against theory. Blanks (1996) reported that the facility has been constructed and low pressure (up to 650 bar) calibration data have been obtained for enthalpy of 12.5 MJ/Kg. The facility has a 27.5 m compression tube whose ID is 203 mm. The length of the shock tube is 12.2 m and its diameter is 7.62 cm. The conical nozzle has throat diameter that vary from 9.5 - 19 mm. The 8 degree conical nozzle exit diameter is 45.7 cm. Flow times are about 1 -2 milliseconds at the aforementioned low pressure test condition. It is also reported that good comparison with CFD modeling of the facility has been obtained. It was reported that a copper liner was effective in preventing reservoir erosion. It was noted by Blanks that the attainment of nozzle stagnation pressures in excess of 2000 bar will require solution of the erosion problems.

4.2 European Facilities

4.2.1 HEG Shock Tunnel (Germany)

The HEG Göttingen facility is a free-piston-driven shock tunnel (Fig. 13). The tunnel is 60m long with an internal diameter of 20 cm and has a test section diameter of 1.2 m. The maximum reservoir pressure achievable is 1000 bar which allows a maximum binary scaling parameter, Rho-L (or pL), of 1/1000 kg/m^2 (Fig. 14). This parameter represents the number of molecular collisions. The total temperature can reach 10, 000 K. The speed reached in the test section varies from 4.5 to 7 km/s. The core of parallel flow has been estimated to be 0.55 m with a static temperature of 1000-2000 K. The flow is frozen non-equilibrium in the undisturbed free stream. This nonequilibrium occurs because of the low density which is reached in the expansion through the nozzle. The nozzle exit Mach number is approximately equal to 10. The testing time is currently about 2 ms. Forces and balances are not currently worked on. The instrumentation consists of pitot tubes and static pressure transducers to measure the pressure and Mach number. Heat transfer measurements are also performed with coax thermocouples and laser-induced fluorescence to measure density and possibly temperature to get the flow velocity. For flow visualization, a 2-D holographic interferometer is used, from which the density can also be computed. A laser-schlieren flow visualization setup is also used in the facility.

4.2.2 ONERA F4 WIND TUNNEL

The F4 depicted in Figure 15 is the ONERA's high-enthalpy hypersonic testing facility. It is an intermittent blowdown (impulse) "hot-shot" type of wind tunnel. It has three different steel and fiberglass contoured nozzles with different exit diameters, the largest measuring 0.67 m with a length of 3.9 m. The electric power needed to operate it is 150 MW. This arc-heated facility can attain a stagnation pressure of 2000 atm. The flow velocity can reach 5.5 km/s, and the binary scaling parameter ρL goes from $10^{-3}\,\mathrm{kg/m^2}$ at a velocity of 5.5 km/s to $10^{-2}\,\mathrm{kg/m^2}$ at a velocity of about 3 km/s (see Figure 14). The testing time is between 50 and 150 ms. The Mach number range of the facility is 7 to 18. The main area of interest is at a Mach number of about 16 and a unit Reynolds number of about $3\times10^6\,\mathrm{m^{-1}}$. A typical model size is about 0.3 m in length. With the relatively long testing time, accurate force and moment measurements can be performed. The instrumentation includes balances, heat transfer gauges, and pressure transducers.

4.2.3 RWTH Aachen Shock Tunnel (Germany)

The RWTH Aachen facility TH2 is a high-enthalpy shock tunnel (Fig. 16) driven by a resistance-heated helium driver. In the reflected mode, the shock tunnel has a driver section of 6 m, a driven section of 16 m, and conical nozzles with exit diameter of 0.57 m, 1 m and 2 m. A contoured nozzle with an exit diameter of 0.57 m is also available. The inner diameter of the driver, as well as driven section, amounts to 140 mm. The tunnel can simulate Mach (6 to 15) and Reynolds (12 million/m) numbers, duplicate the flight velocity up to 4km/s, and simulate real-gas effects. Measurements include pressure and heat transfer. To meet the requirements of the shock tunnel operation, a special 6-component balance has been developed which allows force and moment measurements for flow duration of at least 2ms. Flow visualization is achieved by color schlieren and shadow optics, and interferometry. The maximum total pressure is 1500 atm, and the maximum total temperature is 5000 K; the maximum testing time is 10 ms. Currently the shock tunnel is calibrated for 11 different flow conditions with a Mach number ranging from 6.1 to 12.1 and for total temperatures ranging from 1500K to 4700K. To improve the performance of the tunnel, a detonation driver has been built which is currently in the testing phase. With this new driver in the tailored interface mode, the total maximum pressure will be 2800 atm and the maximum total temperature 7500K.

4.2.4 LRBA C2 Reflected Shock Tunnel (France)

The LRBA C2 is a classic shock tunnel with stagnation conditions to 2400 K and 350 bar; test times are 10 to 20 ms. The main feature of this tunnel is the large nozzle exit diameter of 1.2 m. Mach numbers can be varied from 8 to 16 using conical nozzles; however, a contoured nozzle for Mach 16 is generally employed.

4.2.5 VKI Longshot (Belgium)

The von Karman Institute's Longshot is a free-piston tunnel (Fig. 17). It has one contoured nozzle with a 0.43 m exit diameter and a 6 degree conical nozzle with a 0.355 m exit diameter. The total pressure can reach 4000 bar, and the total temperature about 2500K. For the calibrated flow conditions with the contoured and conical nozzles, the Mach number range is 11 to 15; the Reynolds number ranges between 4 and 14 million/m and the useful running time is about 10 to 20 ms. With the contoured nozzle, the tunnel is operated with nitrogen; with the conical nozzle it can also be operated with carbon dioxide. Four operating points (two for N2 and two for CO2) are calibrated with the conical nozzle so that the effect of variation of the specific heat ratio γ at constant viscous interaction parameter can be studied. The models are mounted on a high precision incidence mechanism for pitch, yaw, and roll. Instrumentation includes a 6 component strain gauge balance with accelerometers to account for impulse forces; infrared photography, thin-film gauges, and coaxial thermocouples for heat transfer measurements; piezo-electric pressure gauges; a 64 channel acquisition system with integrated amplifiers and filters and a schlieren system for flow visualization. The research conducted includes support to the development and validation of the physical modeling used in the numerical codes and investigations of the aerothermodynamics of reentry vehicles.

4.2.6 CNRS SR-3 Low-Density Tunnel (France)

The SR-3 wind tunnel (Figure 18) of the National Center of Scientific Research can achieve Mach numbers from 2 to 20. Its flow is of low density, and the maximum Reynolds number obtainable is 7.3×10^4 at Mach 30. The gas used in the tests is nitrogen. The nozzle exit diameter goes from 0.15 m to 0.40 m. The tunnel flow is continuous and the flow regimes can be from near-free-molecular to continuum. The instrumentation includes electron-beam probes for low-density measurements, pressure transducers, devices for heat-flow measurements, i.e., thin-wall technique and infrared thermography, hot-wire probes, and aerodynamic balances. Flow visualization is obtained by sweeping the electron beam by glow discharge. The research conducted includes plume interaction studies (launcher stage separation, spacecraft control) and low-density aerothermodynamics.

4.2.7 VG Low-Reynolds-Number Tunnels (Germany)

The V1G and V2G facilities at the DLR Göttingen are resistance-heated continuous tunnels which were designed for hypersonic low-Reynolds number (low-density) flow research. V1G and V2G have nozzle exit diameters of 0.25 m and 0.4 m, respectively. Because they are low Reynolds-number facilities, the useful cores are much smaller than the geometric cores: from 0.05 m to 0.3 m, depending on selected conditions. Reservoir temperatures can reach 1500 K. Force balances, electron-beams, thin-wall heat transfer techniques, and flow visualization by glow discharge are just some of the instrumentation methods which have been developed over many years.

4.2.8 Facilities in Development ISL-RAMAC (France)

Ram-accelerator research has been under way at ISL (French-German Research Institute of Saint Louis) since 1988. The largest facility now operational is RAMAC 90, consisting of a ram accelerator tube of 90 mm in diameter and a conventional powder gun as pre-accelerator. The length of the accelerator tube is at present 16.2 m (180 calibers). Extension to about a 30 m length is planned within the next years. The facility is located in a 120 m long ballistic range already existing in ISL since 1958.

The first ram acceleration was attained in March 1992. A recent result has been increasing the velocity of a 1.340 kg body from 1335 m/s to about 2000 m/s within the tube of 16.2 m length. The main future objective of this facility is the acceleration of important masses to velocities up to 3 km/s.

A smaller facility is RAMAC 30, which consists of a ram accelerator tube of 30 mm in diameter of up to 12 m using a conventional powder gun as the pre-accelerator. Stepwise extension up to about 40 m is planned for the next years.

The facility is built for basic research mainly in the superdetonative flight mode. The objectives are to achieve velocities beyond 4 km/s and to identify and overcome possible limiting factors such as aerodynamic heating which may lead to ablation and unstart phenomena; i.e., a detonation wave moves in front of the projectile and ends the acceleration phase.

Two versions of ram tubes have been used: (1) a rail tube and (2) a circular bore tube. With the rail tube, cylindrical aluminum projectiles of 130 g could be accelerated with up to 90,000 m/s^2. In this case, ablation problems were not as dominant as in the circular bore experiments using fin guided projectiles. Here heating and ablation caused a strong erosion especially on the projectile's fins, resulting in a projectile canting followed by an unstart.

4.3 Russian Facilities

The first version of this chapter edited by Saric, et al. (1996) included a section by W. Calarese, based on his visits to Russia. A publication by Czajkowski (1994) contains excellent descriptions and photographs of the Russian facilities. This publication can now take the place of the section in our previous AGARDOGRAPH.

5 ASSESSMENT OF FACILITIES

As shown in Figures 2, 3, and 4, no facility can reproduce all the conditions required for complete reentry or ascent simulation. Consequently, each class of facilities has aimed at reproducing some of the required conditions, and all classes can be seen as complementary to one another. Depending on the problem to be investigated, certain characteristics of a given class may range from very undesirable to acceptable; e.g., contamination may strongly influence combustion processes but have negligible influence on force measurements.

All the facilities discussed above which are more than five years old can be said to be useful contributors to our experimental database within their specific limitations and advantages.

We also comment here on the progress being made with the newer facilities: LENS, T5, HEG, and F4 which now have one to several years of regular calibration, operation, and practical testing. They all represent very valuable additions to our testing capabilities and, in particular, to our code validation capabilities.

5.1 LENS

The LENS facility has become an important asset for the measurement of aero-optic and aerothermodynamic effects as well as scramjet propulsion testing. The calibration of the facility is described by Holden et al. (1995). LENS is capable of producing critical design data in turbulent, non-equilibrium hypervelocity gas flows over large, well-instrumented models. Recent major use of the facility has been to evaluate the aerothermal and aero-optical performance of full-scale interceptor configurations, including seeker head geometries. Importantly, the actual aerothermal environment encountered in flight and its effect on optics can be demonstrated. Tests of large-scale scramjet engines in the facility have demonstrated its use for ground testing and development of engines in the range of velocities from 1.8 to 4.6 km/sec. The long test times and clean airflow generated in the LENS tunnel under conditions where nonequilibrium real-gas effects are important enable study of vehicles of interest to Earth re-entry and planetary entry as well. Fundamental studies in high-enthalpy flows also provide an excellent opportunity for code validation and the models of turbulence, vibrational relaxation and dissociation, which are employed in Navier-Stokes and Direct-Simulation Monte Carlo computational schemes. Initial results of these studies are described by Holden et al. (1995) and by Holden, Chadwick, Gallis, and Harvey (1995).

5.2 GALCIT T5 Facility

As mentioned in section 4.1.5, the GALCIT T5 facility at the California Institute of Technology has been used in over 1200 runs in the time frame from December 1990 through the spring of 1996. Clearly, this facility is serving as an important focal point for the advancement of understanding real-gas, hypervelocity flows. This advancement is occurring because faculty, students, and researchers from a wide range of institutions (academia, industry, and government) with both computational and experimental interests are focusing their attention on the capabilities of T5. Part of the focus is on the prediction of tunnel mechanisms, operations, and flow quality, e.g., the behavior of free pistons using CFD as described by Belanger and Hornung (1994); improvements in diaphragm manufacture and nozzle throat materials; and in corroboration of the predicted free-stream flow conditions of the tunnel. As described by Candler, Dimotakis, Hornung, Leonard, Meiron, McKoy, Pullin, and Sturtevant (1995), excellent progress is being made by an interdisciplinary effort by computational fluid dynamists, experimentalists, and computational chemists in understanding the interactions of chemistry, turbulence, and shock waves in hypervelocity flows. Results of this work to date include a clear understanding of two important parameters that define hypervelocity flow over spheres. This reference also discusses detailed experimental and theoretical studies which show that real-gas effects do not further enhance heat flux in type IV shock-shock interactions as compared to ideal-gas flows. These shock-shock interaction studies resulted in the first high-resolution interferograms of such flows, and these also established a good measure of the flow quality in T5. Candler et al. (1995) also report on an important computational discovery of a flow field that is sensitive to vibration-dissociation coupling, which suggests an important shock tunnel measurement that could lead to improved real-gas CFD modeling. Finally, this reference shows how computational chemists are providing reliable information on real-gas properties via first-principles quantum mechanical calculations on collision cross sections of electronic excitations of OH, NO, and CO2, important in weakly ionized flows.

5.3 HEG Facility

The HEG (High-Enthalpy Göttingen) facility has demonstrated its ability to produce hypervelocity flows characterized by nondimensional binary scaling parameters identical to those experienced in the high-velocity regime of reentry from Earth orbit. At present (late 1996), 350 shots, of which the first (50) were part of a commissioning process, have been made. The facility was commissioned in July 1993. Since then, it has been used mainly for European Space Agency programs which the major efforts have been devoted to the calibration and understanding of the flow. Since the number of measured parameters is limited by the experimental capabilities, all calibration efforts are performed concurrently with numerical calculations (e.g., Hannerman et al., 1995).

As a result, a realization grew that the HEG contoured nozzle does not always produce sufficient flow quality, due to centre line perturbations/focusing. Therefore a new conical nozzle was installed, providing a more uniform central core flow. However, for many experiments, the flow produced by the contoured nozzle was of sufficient quality for studying real gas effects. These effects are measurable with conventional measurement techniques adapted to the HEG flow charcteristic (Eitelberg 1996). The studies of on-going interest are

- Flows over blunt or blunted objects. These can be spheres (Eitelberg et al., 1996), missile-shaped objects (ELECTRE), or blunt coned (70° case, ref. Ch. 4). In all cases, shock shapes and shock stand-off-distances provide significant fluid dynamics/real gas effect data for code validation. In all these blunt cases the existence of a non-equilibrium (neither frozen nor equilibrium) shock layer has been shown to be present. The influence of nonequilibrium flow on the configuration of flow features is demonstrated unequivocally. The determination of its influence on surface quantities (pressure and heat transfer) is complicated, and this stage is not always unequivocal.
- Shock/shock interaction serves as a test case for code validation. Here again the geometry dependence on flow-field shape upon high-enthalpy is easy to demonstrate. The high-enthalpy effects lead to changes in the characteristics of peak heat loads.
- Capsules and flight configurations (Halis, HERMES). Here data have been provided in the framework of ESA projects.

In order to obtain good quality data, a large effort has been dedicated to instrumentation development. In particular, for the study of real gases, spectroscopic methods have been developed. The LIF technology has been applied for visualization of changes in species concentrations and thereby the flow-field shape (Beck et al., 1996) and temperature profiles (Rosenhauer 1994). Work towards fully quantitative measurements (T, C_i) is continuing. Also under development is a laser disk absorption technique to determine free-stream temperatures and velocities. Flow visualisation with schlieren and interferometry is mature and reliable (Kastell, Eitelberg 1995). There is good experience with pressure and heat transfer measurements.

It is important to note that all current experimental programs have been accompanied by numerical analyses (e.g., Hanneman 1995). Further numerical analysis into the shock tube behaviour and nozzle flow starting process is ongoing.

5.4 F4 ONERA

The ONERA F4 wind tunnel (Figure 15) was built in the early 1990s to simulate the atmospheric re-entry of hypersonic vehicles. It is a hot-shot type, meaning that the settling conditions are obtained by heating the test gas with an electric arc in an arc chamber. The energy is delivered by an impulse generator, at a power of up to 150 MW for several tens of milliseconds. The settling-chamber pressure can be as high as 500 bar and the reduced total enthalpy H/RT_0 can be as high as 250 (about 20 MJ/kg). After the arc-chamber conditions reach the desired levels, the arc is stopped and the nozzle throat is opened by igniting a pyrotechnic plug to initiate the nozzle flow. The blowdown is interrupted by firing a pyrotechnic valve in the arc chamber, quickly evacuating the remaining gas into a dump tank. Run duration of up to 400 ms can be achieved, but with reservoir conditions decreasing with time. Reservoir pressure and enthalpy decays are slow enough (1%/ms) to allow force measurements to be performed. The useful run period is established after a perturbed period of 30 ms because of the throat-plug-expelling phase. Synthetic air and pure nitrogen are used as test gases. F4 can be equipped with four different contoured nozzles with area ratios varying from 1850 to 32,000. Nozzle 2 with an area ratio of 4490, a length of 3.4 m, and an exit diameter of 0.7 m, was used for most of the aerothermal testing. Figure 14 shows the range of the binary scaling factor versus velocity.

Measurement techniques comprise forces and moments with six-component balances and inertia compensation, model pressure and heat transfer, infrared thermography either with the scanning line technique (2500 Hz) or with the 2D high rate ONERA camera (400 Hz), emission spectroscopy, absorption spectroscopy and velocity (Doppler effect) with Diode Laser infrared Absorption Spectroscopy (DLAS) on NO, H_2O, and spectroscopy with Electron Beam Fluorescence (EBF) on NO, N_2. Recently, free stream velocity has been measured with a time-of-flight technique using a pseudospark electron gun.

6 ASSESSMENT OF INSTRUMENTATION

6.1 Requirements

Experimental testing in hypervelocity flows must consider four basic parameters (Seibert et al. 1992):

(1) The total temperature or enthalpy determines the maximum velocity attainable.
(2) The total pressure determines the test pressure and, therefore, the altitude to be simulated, and it has a profound effect on the nature of the test gas.
(3) The size of the facility determines the largest model scale that can be used.
(4) Test duration determines the type of instrumentation that can be used, the ability to "soak" structures in the hot flow, and the relationship of the chemical relaxation times.

The following fluid properties need to be measured: pressure, P, density, ρ, temperature, T, the components of velocity, u, v, w, the stream or global velocity, U, and the sound speed in the flowing medium, a. Flow profiles of chemical species, X_i, ionization, and the transport properties: viscosity, μ, thermal conductivity, k, and diffusion, D_{ij}, must be measured or determined.

In order to provide the physical interpretation of the experiment, the following topics must be characterized.

<u>Flow Patterns</u>: Shock Shape, shock locations, boundary-layer transition locations, reattachment, boundary-layer thickness and profiles, and vortex patterns all significantly affect the determination of flight performance and ultimately the design of the flight vehicle.

Turbulence: Fluctuations in pressure, P', density, ρ', temperature, T', and velocities u', v', w', as well as frequency spectra and power spectral densities, must be measured or determined. Since the determination of these parameters depends heavily on statistical analyses, extreme care must be taken in their measurement, estimation, or calculation.

Thermodynamic States: Spectroscopic techniques must be employed to measure the state of the gases in conditions where physical probing is not feasible or would so adversely affect the measurement that the data could not be adequately corrected. Measurements include, but are not limited to, rotational lines, vibrational bands, luminescence, induced fluorescence, Rayleigh scattering, exited electronic states (electronic excitation), net charge, currents, and electron beam fields.

These measurements may be made in conjunction with or separately from forces and moments and heat transfer rates. In addition, combustion requirements and effects must be considered if propulsive studies are to be made. All this must include measuring the parameters that describe performance in flight and those that describe deviations or departure from the flight conditions.

One of the driving forces behind the research and development of diagnostic techniques for hypersonic flows is the need to validate CFD codes (Marvin 1988). This matter is discussed in some detail in Section 3.

6.2 Status - January, 1994

AGARD-CD-514, Theoretical and Experimental Methods in Hypersonic Flows, published in April 1993, extensively categorized the state-of-the-art in hypersonic diagnostics. A hypersonics mini-symposium, held at Wright-Patterson AFB in May 1993 was the forum for update discussions on some of the technologies presented and discussed one year earlier at the AGARD meeting. This report, therefore, will concentrate on updating the activities that have occurred in the U.S. since the AGARD meeting.

Diagnostic developments in hypersonic flow measurements in the United States, since that symposium, have reached, essentially, an evolutionary phase in their progress. Direct measurement of skin friction and heat transfer in rough and smooth surfaces was made in the Mach 6, high-Reynolds-number facility at WPAFB Wagner (1993). The direct use of skin-friction and heat-flow sensors gave performance levels of ±6.0% in the conventional heat-transfer coefficient and ±0.2% full scale (nonlinearity and hysteresis) in shear-stress measurement.

Also at Wright Laboratory, LDV measurements were accomplished at M = 6 and M = 12 in the cold-gas facilities in the Flight Dynamics Directorate (Maurice 1993; Schmisseur and Maurice 1994) 2-LDV measurements were made in a Mach 6 flow over and through a generic hypersonic inlet model where the flow had been calculated using a Navier-Stokes code. Particle response through oblique shocks was corrected for particle lag, and comparisons were made with the CFD solutions showing good agreement where there was no shock-wave/boundary-layer interaction, but a significant discrepancy existed internally in the nozzle where the shock-wave pattern was complex. Pressure distributions on the cowl and ramp matched the CFD solution, indicating the danger of just matching pressure distributions with the CFD solution.

Additional measurements in the 20-inch Mach 12 tunnel were made in the shear layer at the nozzle exit in the free stream and behind the shock on a cylinder model injected into the flow. CFD solutions of the nozzle flow field were made using a full N-S solution in the throat region, a PNS code in the expansion section, and an Euler scheme in the free-jet portion of the test section. A notable result of the work showed that the alumina seed, regardless of size, never reached the theoretical free-stream velocity, even though it had three meters of nozzle length within which to equilibrate.

Other particle techniques that have emerged include an LV system proposed by Smeets which is being developed at NSWC for their hypervelocity facilities. This technique allows the use of submicron size particles, and since it incorporates a spectrometer with a one-microsecond response, the measurement of highly turbulent flow is possible.

A technique proposed for propulsion testing in HYPULSE at GASL involves seeding the H_2 plenum chamber with Silane and some O_2, which then spontaneously burns, creating SiO_2 as a by-product in the submicron size range. A long-pulsed, flash-lamp, pumped-dye laser with a 50 μs pulse width, giving 150 mm of flow passing through the sheet, is used to track the flow using Mie scattering. This process gives the opportunity to observe mixing and also relate concentrations to intensity to get time averaged measurements of the H_2 mass fraction. CFD calculations are also being made. Standard video cameras are being used to prove the concept before going to higher resolution optics. A practical problem is the vaporization of the SiO_2 at very high temperatures, causing data dropout.

Efforts at AEDC include PLIF imaging in the Mach 8-14 impulse facility, imaging NO to determine concentration and temperature and also doing nonabsorptive Rayleigh scattering to determine He arrival. Dual pulse LIF is planned for the next fiscal year. Work will also be done in Tunnel B at Mach 8 measuring jet-interaction phenomena using LIF of NO looking at parts/trillion.

Later this year, a dual line LIF system using O_2 in the H2 facility measuring temperature and density and LIF velocity measurement using atomic copper also in H2 will be attempted. Plans also included the use of a pulsed e-beam in H2 to measure densities. AEDC has also looked at flow fields in shuttle engines using naturally occurring sodium in the hydrogen, where the sodium is vaporized in the hot hydrogen and imaged. One can scan the laser and then measure temperature, density, and pressure.

Boeing is continuing to pursue PLIF measurements in their shock tunnel by looking at large NO concentrations (1% or more) to enhance signal levels. Quantitative measurements of temperature and density are being sought. Coaxial thermocouple heat transfer gauges are being used instead of thin-film gauges due to reliability. Iron-constantan gauges are used in steel models, or plugs are used in aluminum models.

The LENS facility at CALSPAN is using an advanced version of pulsed e-beam and LIF technologies to probe the flow field for measurements of temperature and density.

NASA Ames is continuing development of optical techniques in their 16-Inch Shock Tunnel, including an optical-probe layout in a scramjet model looking at the Raman scatter of nitrogen (Cavolowsky et al. 1993). NASA Ames is also looking at two classes of laser systems that are being developed and applied to absorption measurements of the critical species O_2, OH, and H_2O in hypersonic reacting flow. An Argon-ion pumped tunable ring-dye UV laser system at 306 nm probing OH has been tested.

Also under development are two laser-diode systems: one for the measurement of O_2 in the near IR at about 760 nm and the other for the measurement of H_2O, also in the IR at about 1385 nm. These systems have great potential for flight-vehicle application since they are small and rugged. The systems have all been validated in shock tube experiments simulating pressures, temperatures, and velocities applicable to hypersonic simulations. OH mole and temperature were measured in an expanding nozzle flow in the 16-Inch Shock Tunnel at a simulated flight Mach number of 14.

Flow-visualization techniques included double-pulsed laser holographic interferometry in the Ballistic Range Facility at a Mach number of 14.4 (Tam et al. 1991). "Synthetic" infinite fringe interferograms are also calculated to examine the intensity pattern of the experimental finite fringe interferogram. These results show flow features in the wake region not found on the experimental interferograms.

Work is ongoing for the development of Resonant Holographic Interferometry Spectroscopy Tomography (RHIST) flow diagnostics of hypersonic flows and combustion. RHIST will be used to quantitatively measure OH concentration in combusting flows.

NASA Langley is continuing to pursue the CARS technique for the measurement of temperature and species in scramjet flow.

NASA Langley is also testing the use of modulation absorption spectroscopy for their 8-foot High-Temperature Hypersonic Tunnel to perform scramjet thrust tests, measuring gaseous concentration and temperatures. Both amplitude modulation spectroscopy and wavelength modulation are being tested. Infrared absorption spectra of constituent gases by using diode lasers is measured. Test-cell results measure changes in oxygen concentration of 0.1% using the $X^3\Sigma g^- \rightarrow b^1\Sigma g^+$ transition. Raleigh imaging is being used at Langley to look at condensate fog in the M = 6 realm where velocity can be measured.

In Europe, work continues in the application of spectroscopic diagnostic techniques to studies on HEG in Göttingen (Beck et al. 1993). LIF measurements have been carried out in the vacuum wind tunnel V2G in Göttingen, the arc-heated tunnel LBK in Cologne, and on the shock tunnel TH2 in Aachen in preparation for measurements in HEG. Emission spectra over the range of 200-850 nm from the hot gases behind a model bow shock has been carried out. NO excitation spectra were measured in V2G and LBK with temperature and NO concentration being measured in the free stream in LBK.

Free-stream temperature and NO concentration were also measured in TH2 using a single-shot, two-line measurement. Early emission spectra in HEG shots with nitrogen reevaluated that a major limiting species was atomic iron (Fe) requiring the introduction of a copper liner in HEG to prevent the ablating wall effect.

Other work in Göttingen using LIF with an ArF eximer laser (192.8-193.8 nm) in V2G using 90% He + 10% NO and 90% N_2 + 10% NO revealed the rotational temperatures of NO and O_2 at low densities (Grundlach and Hirai 1993). Rotational-temperature measurements were made near a hot copper model of the reflected NO molecules coming back from the surface at nearly free-molecular conditions. Results show a significant deviation from total accommodation at surface temperatures $T_w > 600$ K where the results indicate that the gas surfaces can be studied by LIF spectroscopy.

A study has been conducted at the von Kármán Institute looking at the application of Particle Image Velocimetry (PIV) in hypersonic facilities (Moraitis and Simeonides 1993). Mie scattering calculations indicate that very small particles, with a diameter of 50 nm or smaller can be detected with readily available lasers and films. Problems from flow contamination by foreign particles would have to be investigated.

ONERA has conducted tests on a heat-flux measurement technique based on a luminescence coating in their R3CL Hypersonic Blowdown Wind tunnel at Mach 10 (LeSant and Edy 1993). Tests were performed at a stagnation pressure of 12.5 MPa and a stagnation temperature of 1050 K. Work was done using a model made of insulating material since the coating has insulating thermal properties. The technique must be proven on metal models to allow for standard temperature measurements.

Work at Caltech in the shock tubes/shock tunnels is being done with PLIF techniques using multiple lines in the same shot to look at more complex flows. Measurements are being made to study chemical effects on boundary-layer stability, boundary-layer transition, and on nonequilibrium flows. Resonant Holography is being used to take holograms of flows that are resonating in some species like O_2 or NO and capturing shock structures.

Rayleigh imaging is being considered by several researchers, but whether it works in the low densities of hypersonic facilities is questioned. One approach is to use multiple-reflecting-mirror sets where one passes the laser beam time and again through the flow and then rasters down to take an image. This way one increases laser power by 10 or 20 to compensate for low-signal levels. This approach, combined with filtered Rayleigh scattering, can eliminate the background scattering from windows and walls.

Another technique getting another look is sodium laser induced fluorescence, previously called Resonant Doppler Velocimetry. In heated facilities, which have sodium and copper, one can look at laser induced fluorescence from these species and get good images of the flow structure. One can look at flows with sodium in the parts per billion range.

Many other diagnostic concepts and variations of existing techniques are under development or testing in various centers around the world. The evolutionary process for diagnostic development continues in all speed regimes; whether it continues to expand in hypersonic facilities will be dictated by rapidly changing events.

6.2 Status - January, 1997

Since the last report, activities in hypersonic studies in the U.S. have moderated and are being compromised by continuing budget cuts and restrictions.

Work at Wright Laboratory has been on hold since the hypersonic facilities have been put on a standby status. Work is being completed on tests made in the Mach 6 facility using Rayleigh Scattering to measure density profiles, with mixed results due to scattering effects from condensing water vapor and carbon dioxide, which form nuclei around which oxygen and nitrogen can condense at low degrees of supersaturation. Holographic Interferometry detection of "rope-like" structures in the boundary layer of a cone at Mach 6 are being analyzed to determine stability characteristics and to examine the validity of linear stability theory for these flows. All other activity is on hold with no new testing scheduled for the near term and no new programs being considered.

Progress is continuing on the Radiatively Driven Hypersonic Wind Tunnel Program (RDHWT) which as a facility is discussed elsewhere in this report. Instrumentation and diagnostics development to measure the relevant flow parameters involved with adding megawatts of energy to air at pressure of 700 to 1000 atm is ongoing. The current focus is to implement CO2 Enhanced Filtered Rayleigh Scattering flow visualization and sodium Laser induced Fluorescence for the measurement of injection and mixing in the Mach 8 tunnel at the Princeton Gasdynamic Laboratory. The latter technique has the potential for direct application to mixing and combustion tests that would be conducted in the RDHWT. In addition, sodium seeded flows have been suggested as a medium for studying radiative energy addition processes to support development of the facility. Additional, related work includes the continued development of a pulse burst Laser system which includes a 1 Megahertz pulsed Nd:Yag laser coupled to a 1 Megahertz framing rate camera which allows for 30 frames to be captured at 1 microsec intervals, synchronized with the pulsed laser. Sequential images of a Mach 2.5 wind tunnel boundary layer have been taken, including a shock wave interaction, qualitatively capturing the boundary layer growth and development in the flow and the shock wave fluctuation previously not seen.

Since the Phase I report, Purdue University researchers have made fully operational a Mach 4 Ludweig Tube and are currently developing new flow diagnostics to measure high-speed laminar to turbulent transition in very quiet flow. Currently they have developed a laser perturber, a glow-perturber, hot-film, hot-wire, and differential-interferometer techniques to measure the transition mechanisms using elliptic cross-section cones. These techniques have been sufficiently developed and could be implemented into larger, higher Mach number facilities.

In the NASA Ames 20MW Arcjet facility, where the test gas was a mixture of argon and air, emission measurements within a blunt shock layer were made using a CCD camera attached to a spectrograph. Spatially resolved emission spectra over a 200-890 nm wavelength were obtained to determine line-of-sight averaged thermodynamic properties, including rotational temperature of the free stream and rotational, vibrational, electronic temperatures and species number density in the shock layer. T_r, T_v and T_e measured in the shock layer agreed within their uncertainties for two positions closest to the model. T_r was measured in the free stream using five different NO bands, and the value from each band system was in the estimated error bound from the other band systems.

NASA Ames is also using Laser Induced Fluorescence (LIF) in its arc-jet facility to measure velocity, temperature and species concentration in the very high temperature arc jet flow. NASA Ames is looking at two photon excitation of N2 and single photon excitation of NO.

The Lens Facility at CALSPAN continues to develop advanced electron beam technology to measure the rotational temperature of N2, to obtain the N2 number density and vibrational populations, as well as to detect unknown gases and atomic species in the flow. USC has developed a Pulsed Electron Beam (PEB) which can operate at potentials of order 40 KeV with currents of 500 - 2000 A with a 10 nSec rise time. This technique delays serious complications of collisional quenching to higher densities up to $10^{18}/cm^3$. Also, due to high currents, a high density of excited states are produced, providing a strong signal relative to the high background levels. The plan is to use this device in the LENS Facility following the success of the continuous wave electron beam. Also in LENS, a tunable semiconductor diode laser, pioneered by Hanson at Stanford, was used to measure temperature, velocity and water partial pressure. This measurement was accomplished by using a hardened probe installed directly into the flow field. Spectrographic techniques are also being employed to determine the species concentrations. Advances in thin film and coaxial gauge instrumentation continue to be accomplished in LENS, where DSMC and Navier-Stokes codes are used to compare experiment and computation. Results indicate good agreement using DSMC in the low-density regions around a planetary probe model, although the computed rates did not exhibit similar characteristics as the measurements did.

The HYPULSE Facility at GASL is conducting fuel plume imaging measurements using monodisperse, one micron SiO2 particles which are illuminated using a Laser light sheet from an Alexandrite solid-state Laser with a 100 micro-sec pulse width. From a time-averaged image of the density of the particles, one can track the H2 accurately and back out the H2 concentration. This method replaces the silane technique which was not successful due to the unknown size of the particles from agglomeration during a run. Also, a water vapor measuring technique, similar to the one being used at CALSPAN, using a tunable diode laser where a beam splitter sends three parallel beams across the combustion field to detectors in a heterodyning fashion, is being employed. However the technique does not give concentration or temperature, so a variation using two water bands is being developed.

In T5 at the California Institute of Technology, to supplement the diagnostic systems currently in place and reported elsewhere, consideration is being given to the use of Particle-Image Velocimetry to measure profiles of transitional and turbulent boundary layers in the presence of dissociation and weak ionization. Recently in T5, a simple duct device was used to detect driver gas arrival at the test section. The technique measures the pressure rise in the duct detector which is affected by the arrival of the driver gas for very carefully designed detectors. Initial results showed good consistency over a wide range of enthalpies and suggest the duct detector will be a useful tool to determine test time for these type facilities.

At AEDC in their Impulse Facility, a dual PLIF system has been installed to measure the temperature and density of NO. in the flow, both in the free stream and inside a shock layer around a conical model. CFD calculations were compared with the PLIF measurements showing temperatures lower by 400-500 °K than the theoretical values. The PLIF data are complicated by laser absorption and spectral hole burning in the free stream. Ablation contamination of the flow is considered to be the leading candidate as the source of the temperature discrepancy.

In Europe, spectroscopic work at DLR in Göttingen, Germany in the HEG Facility is being used to measure quantitative, absolute temperature. LIF tests were conducted in HEG at enthalpies of 21 MJ/Kg and pressures of 39 Mpa. Two counterpropagating ArF eximerlaser beams at 193 nm are tuned to two different transitions of NO. and excite fluorescence from the free stream, behind shocks and in wakes. The short tunnel test times (1-4 msec) require a single-shot technique, making the quantification of the flow parameters difficult. Results to date indicate that the technique must be refined to reduce the uncertainty in the data. The tunable diode laser absorption technique has also been tried in HEG, attempting to detect absorption from an excited state of 0 so a kinetic temperature (from the absorption bandwidth) and a flow velocity (from the Doppler shift) could be measured. First attempts could not detect the 0 atom absorption and the sensitivity of the technique is being examined.

Onera, in France at their various research facilities, has, for several years, been pursuing advanced diagnostic techniques for hypersonic and hypervelocity flows. Their preferred methods for high enthalpy facilities include diode laser absorption spectroscopy for free-stream measurement of velocity and static temperature; CARS for point measurement of rotational and vibrational temperatures, and N2 density with velocity a possibility and Electron Beam Fluorescence and LIF for qualitative imaging and visualization. Work is planned for their R2Ch wind tunnel at Chalais-Maudon using CARS for measuring temperature. They hope to expand the capability for use in their high enthalpy, short duration tunnel, F4.

7 FUTURE FACILITY NEEDS AND FACILITIES IN DEVELOPMENT

7.1 Results of American Studies

Potential future missions planned by the Americans involving hypersonic flight (air-breathing access to orbit/hypersonic cruise/planetary and earth entry/aeroassist) will require ground-test capabilities for aerothermodynamics testing which cannot be met with simple modifications to existing facilities. Further, some of the required technologies and methodologies for new facilities which can meet the requirements are not in place. This shortfall has led to advocacy within the United States for substantial investments in "facilities research" which will ensure the efficient design/construction/operation of the next generation of U.S. hypersonic facilities.

The following is a synopsis from the aerothermodynamics section of the December 1992 United States Department of Defense/NASA Hypersonics Test Investment Plan (HTIP). This plan is for U.S. Government use only, but the following synopsis has been approved for inclusion in this AGARD document by the HTIP Co-Chairs K. Richey (USAF) and Wayne McKinney (NASA). According to the HTIP report, facilities filling the anticipated needs for aerothermodynamics in the near, mid, and far term are specified below:

Near Term: Research should be done to enable the implementation of a large-scale, advanced expansion-tube/shock-tunnel to be used for study in the true-enthalpy flight regime of Mach 16 - 20+. Real-gas effects could be studied in air and planetary atmospheres. The facility would employ the double-diaphragm, shock-tube approach where energy is added to the moving stream as depicted in Figure 8. Arbitrary test gases can be used and, ideally, low dissociation will be experienced in the free stream because the flow is not stagnated. In full scale, this facility would use a 610 mm diameter free-piston driver, a 1.52 m diameter test section, and would have test times of approximately 2 ms. Early research for this facility would include analysis of driver options as well as issues of losses/disturbances and unwanted dissociation effects associated with the secondary diaphragm. The research will include CFD and experimental studies in existing small expansion tubes such as the NASA Langley GASL Facility.

Mid term: Studies are advocated to proceed preconstruction of a large facility with several second flow duration at Mach 16+ equivalent enthalpies in a 1.52 m test section. These flows would be driven by a reservoir with temperatures and pressures of 8000 K and 14000 atm, respectively. Facility research would focus early on exploring an extension of the Russian approach to increasing flow time in impulse facilities by using the type of driver depicted in Figure 19. This driver, called a Piston Gasdynamics Unit (PGU), is operational at the TSNIMASH research center (Anfimov 1992; Anfimov and Kislykh 1990) and uses special valves between a piston and the stagnation chamber to subject the test gas to multiple shock passage/heating cycles. Early research on this facility would include analysis of nozzles which minimize reservoir dissociation products (O and NO) in the test section as well as studies of materials which can withstand formidable heat transfer to the accumulators, valves, stagnation chamber, and nozzle throats.

Far term: Research includes work on a large ballistic range which would employ large (up to 300 mm diameter) models up to 15 km/s with advanced onboard and nonintrusive instrumentation A major research issue here is a model launcher. Possible solutions are the University of

Washington's "Ram accelerator" and the Russian TSNIMASH approach with an evacuated tube and timed explosives on walls to accelerate the model (Figure 20 from D. Wilson at the University of Texas at Austin). Advantages of ballistic ranges for aerothermodynamics are well documented in Witcofski et al. (1991).

Finally, the general feeling in the U.S., documented in the HTIP report, is that new, innovative ideas for hypersonic facilities should be nurtured. An example is the high-pressure, cryogenic arc concept as described by Rizkalla et al. (1992).

The United States Air Force Scientific Advisory Board SAB (May 1989) document also is available only to U. S. Government Agencies. The recommendations listed therein are consistent with those discussed in the HTIP report outlined above. The SAB report recommended that research on large arc jet wind tunnels be conducted at the NASA Ames Research Center and at the Arnold Engineering and Development Center, and this is being done, albeit at low levels.

As of August 1996, it is noted that no monies have been made available for using a piston driver for expansion tubes within the United States. However, innovative work at GASL Bakos, et al. 1996) is showing that new test capabilities may be obtained in expansion tubes by drivers employing shock-induced detonation waves. Further, the only research potentially taking advantage of the Russian piston gasdynamic units in the U.S. appears to be within the new studies of a radiatively driven wind tunnel discussed in the next section.

7.1.1 Radiatively Driven Hypersonic Wind Tunnel RDHWT

A novel concept is currently being studied (Miles et al., 1994 and Macheret et al., 1995) in the US to circumvent the well-known limitations of providing long-duration, ground aerothermodynamic test flows with true flight simulation for Mach number in excess of about 10. The limitation is that conventional isentropic expansion wind tunnels require that high-temperature air must be contained in a plenum and then flow though a small throat before expansion in the nozzle. This limitation limits one to the lower Mach number regime for long-duration flows or to short (milliseconds) test section flows, possibly contaminated throat material and/or NO created by the high temperatures in the plenum.

Work on the RDHWT has been ongoing for about two years. The approach is to take advantage of Russian Technology discussed above for piston gasdynamic units (PGU's) to provide low-temperature, ultra-high pressure air in a plenum. The concept is that the PGU provides cold, high-density flow through the throat, possibly circumventing both the production of NO in the plenum and the throat-erosion problem. After an initial supersonic expansion, additional energy is added by coupling to either optical or microwave sources. Figure 21 from Miles et al. (1994) shows a schematic comparison of the two approaches.

As discussed by Miles, the ultra-high pressure in the plenum is such that the air cannot be treated as an ideal gas. They believe that the real-gas effects can lead to much higher kinetic energy passing through the throat at Mach 1 than would be possible for an ideal gas.

Considerable effort and thought is being focused on this activity, including study of energy addition and thermalization of the flow as discussed by Marcheret et al., 1995.

7.2 Results of European Studies

An ESA study was initiated in the early 1980's to assess the level of European competence in hypersonic facilities and computational tools required for the design of specific two-stage launchers to LEO. Recommendations were made which concentrated on bringing back into useful operation a series of tunnels that had been constructed in the 1960's. Many of the recommendations were put into effect as the Hermes program developed. However, to the author's European knowledge (J. F. W.), no more recent ESA study looking toward an entire series of missions involving hypersonic flight has been commissioned.

Except for two thermal protection testing facilities called Sirocco and Plasmatron, which are under construction, no new facility is currently planned in Europe for high-enthalpy aerodynamic studies. The efforts will be devoted in the short term to get fully in line the HEG and F4 wind tunnels and to develop appropriate flow diagnostic methods.

The time is not well chosen to consider the possible development in the longer term of new facilities. There is certainly no money available in the foreseeable future for new developments of large size, and there is not even enough activity to keep the present facilities busy. As far as facilities are open for industrial testing, any new duplication should certainly be avoided within NATO in order not to decrease further the workload of each facility.

However, one should keep in mind, when looking at the technical needs, that the development of such facilities takes 5 years if on a national basis, and up to 15 years if in the frame of an international agreement. Therefore, it is still time for AGARD to think about technical needs for future hypersonic aerodynamics facilities, as far as really new needs are identified or new opportunities appear to fulfill unsatisfied needs. Selected authors have expressed their personal views (Muylaert et al. 1992; Wendt 1992; Kuczera and Weingartner 1993; Hirschel 1993).

The long-term objective should be seen as full-flight simulation with clean equilibrium flows to Mach 16-18 and constant conditions for at least some tens of ms. It is clear that much effort must be expended in the interim on such areas as

1. specially designed code validation tunnels (e.g., the iodine vapor facility of Pham-Van-Diep et al. 1992)
2. the "hot-model" technique for radiation dominated flows
3. transition triggering mechanisms, so that a rational decision concerning the need for a "quiet" hypersonic tunnel can be made
4. instrumentation to reliably measure all appropriate temperatures and constituent concentrations with sub-ms response times in the harsh environment of real facilities

5. techniques to add energy to a flowing gas, e.g. by lasers, to avoid the need of stagnating the flow
6. large-scale ballistic ranges and associated instrumentation; e.g., the ram accelerator method

7.2.1 Plasmatron

A Plasmatron is an induction-heated wind tunnel in which a jet of air (or other gases), heated at temperatures from 6000 K to 10,000 K to a plasma state, is directed onto a target, primarily for the purpose of testing the resistance of thermal protection systems. Figure 22 shows the working elements of the facility.

In 1992, facilities of this kind were found to exist in Russia, where they have been extensively used for the testing and optimization of the ceramic composites used as thermal protection tiles for space reentry objects such as the soviet capsules and the shuttle Buran.

The European Space Agency, recognizing the advantages offered by such facilities in terms of chemical purity of plasma, compared to the arc-jet facilities traditionally used in the western world for the same purpose, decided to sponsor together with the Belgian Federal Office for Scientific, Technical and Cultural Affairs, the construction of a 1.2 MW Plasmatron at the von Karman Institute.

Two interchangeable torches of 80 mm and 160 mm diameter will allow the generation of subsonic and supersonic plasma flows with stagnation pressures ranging from 5 to 175 mbar, producing (catalytic cold wall) stagnation heat fluxes of 350 to 1200 kW/m^2. The facility will feature segmented water-cooled cold cages in the torches and a solid-state thyristor rectifier and MOS-inverter oscillator. The control system will allow fully automatic operation of the facility from warm-up to complete stop, with varying test parameters to simulate re-entry trajectories.

Intrusive and nonintrusive measurement techniques will be used, including emission spectroscopy and LIF.

The planned completion of the Plasmatron is scheduled for October 1997.

7.2.2 SCIROCCO

During studies on the HERMES Spaceplane Programme, the European Space Agency (ESA) identified the need to have a large, high-enthalpy plasma wind tunnel to test and qualify real-scale parts of the spacecraft. SCIROCCO is a 70-MW arc-heated free-jet wind tunnel. This facility would be called an arcjet in the U.S. It is now (late 1996) in its realization phase and will be fully operational at CIRA in Capua by early 1999. This facility is cofunded by ESA and the Italian Ministry of University, Scientific and Technological Research (MURST).

8 RECOMMENDATIONS ON POSSIBLE AREAS OF COOPERATION AND POTENTIAL BENEFITS

The benefits of collaboration will be more pronounced if a specific joint project(s) can be defined; at the present time, only Huygens/Cassini is firm, and collaboration on the aerothermodynamic issues is not part of the accord.

However, members of the AGARD WG 18 hope that the future will bring one or more joint projects involving Earth or planetary entry. Obvious candidates are the robotic and human exploration of Mars and exploration of one of Jupiter's moons, Europa. These possibilities arise because of the very recent discoveries of possible evidence of life in Mars meteorites reported by U.S. and British scientists and images from the Galileo spacecraft showing apparent ice covers on Europa. These images suggest the possibility of warm water beneath the ice where life could exist. The use of aerocapture (the use of atmospheric drag forces rather than retropropulsion) for orbit capture and high-speed entry to planetary surfaces from interplanetary trajectories will be important for the transportation systems for these missions. Cost sharing for the aerothermodynamic testing and development of such space transportation systems could hasten the day when humankind will know the answers to these very profound questions. Prior to this time, a certain number of actions will be very useful as precursors to an eventual joint project.

Based on the perspective of the space transportation and space vehicle programs considered by Europe in the near and far future, ESA should initiate a comprehensive study on Europe's needs for new facilities; the study should take into account opportunities offered by CFD to supplement experimental tools. It should also take into account the facilities existing worldwide and the effective possibility to use them for development purposes. The results of this study should be confronted within AGARD in a manner similar to the studies performed in the U.S. The role of flight testing in design tool validation should be analyzed together with the AGARD community. The result of these studies should bring near-term, mid-term, and long-term recommendations and may bring forward a basis for a possible collaborative effort on an international scale.

Within the limited travel funds available, AGARD-FDP-sponsored Symposia, workshops, professional visits, etc. should focus on real-gas hypersonics. Stress should be put on experiences with testing techniques in the recently developed real-gas facilities such as T5, LENS, HEG, F4, etc.

Exchanges of experience with Russian hypersonic facilities to understand their potential should be undertaken in cooperation among Russia, Western Europe, and America. This cooperative effort will serve to ensure that critical decisions at a later date concerning the possible construction of new facilities in the West versus the use of existing Russian facilities can be made with full background knowledge and confidence.

9 SUMMARY

Missions of current and future interest to the United States and Europe which involve hypersonic flight within or entry into the atmosphere of the Earth or other planets have been summarized. Aerothermodynamic issues such as heating and chemical reaction rates which arise because of the high-flight velocities have been discussed.

The problems posed for the simulation of these effects in ground-based facilities can be summarized as follows. It should be clear that presently a wide variety of facility types is required to simulate, even partially, the expected effects. As a result, Computational Fluid Dynamics is an essential tool in the regime, assuming that the codes can be fully validated by means of appropriate experiments on the ground and in flight.

A number of new facilities have been developed in recent years for the express purpose of addressing the crucial aerothermochemistry problems posed by hypervelocity flight. These facilities are now in the "production" phase, as are the nonintrusive instrumentation techniques which have been introduced. Together, they are providing a clearer understanding of hypersonic flows.

Recommendations for the near and mid-term are

Collaborative and cost-effective efforts on projects of benefit to humankind should be encouraged, e.g., robotic and human exploration of the solar system.

Various agencies in the United States have conducted individual or joint studies on future facility needs; a similar study should be undertaken by the European Space Agency.

AGARD symposia, workshops, lecture series, etc. will also serve as a mechanism to bring researchers interested in hypersonic flight together for an exchange of ideas and experiences. Members of the AGARD WG 18 will be an important part of this activity.

A continuing effort should be made to understand Russian Facilities and their methodology of testing and design. This activity should be a cooperative one between Western Europeans, Americans, and the Russians.

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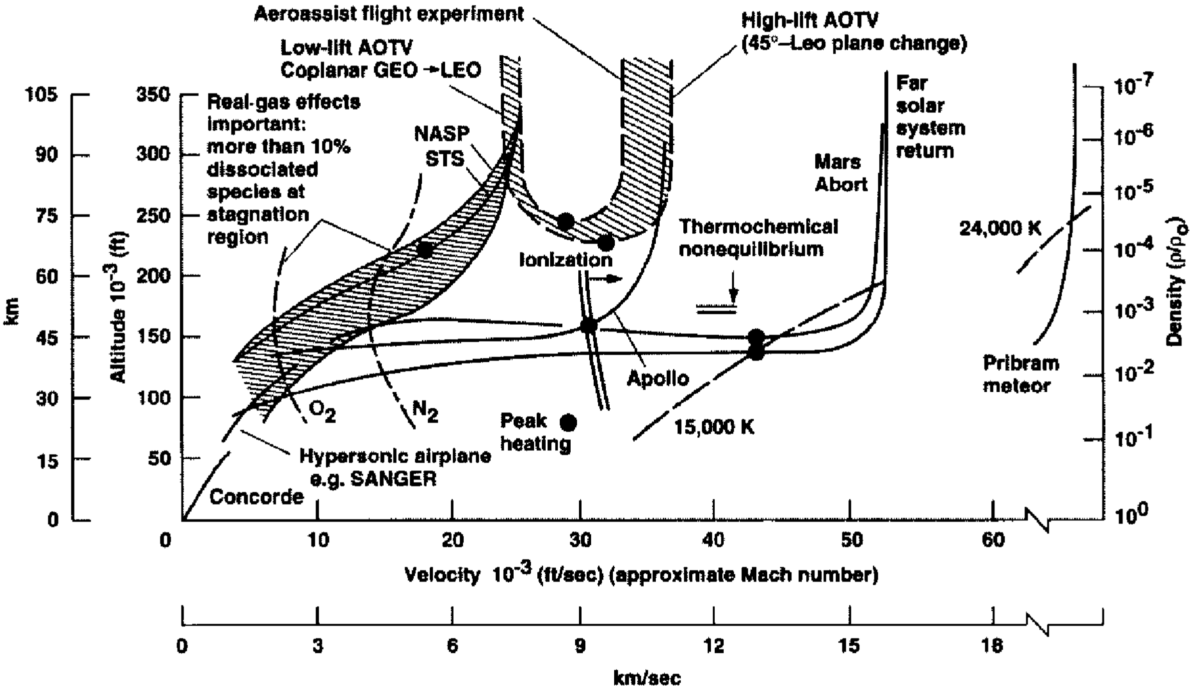

Figure 1. Comparison of vehicle flight regimes in the Earth's atmosphere of interest to the United States and Western European aerothermodynamics community.

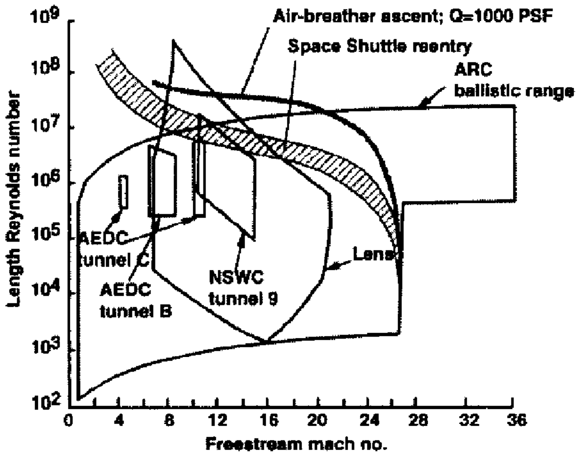

Figure 2(a). Reynolds number versus Mach number plot for American facilities with representative vehicle trajectories.

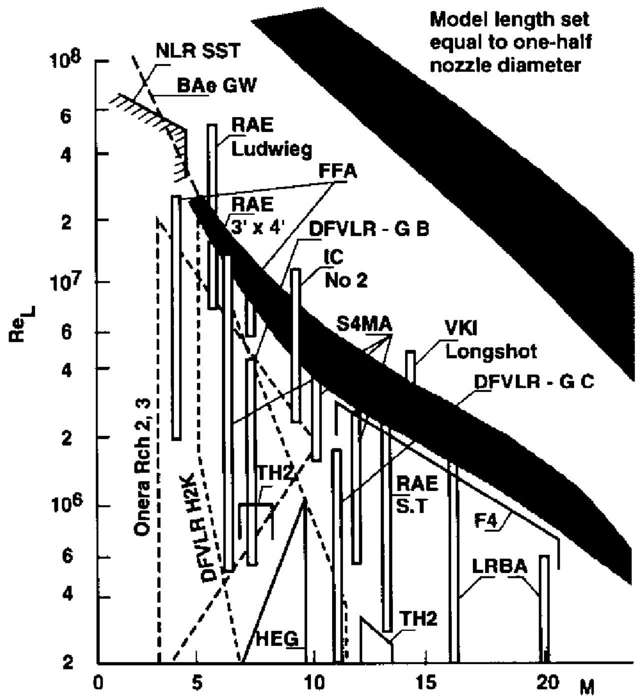

Figure 2(b). Reynolds number versus Mach number plot for European facilities with representative vehicle trajectories.

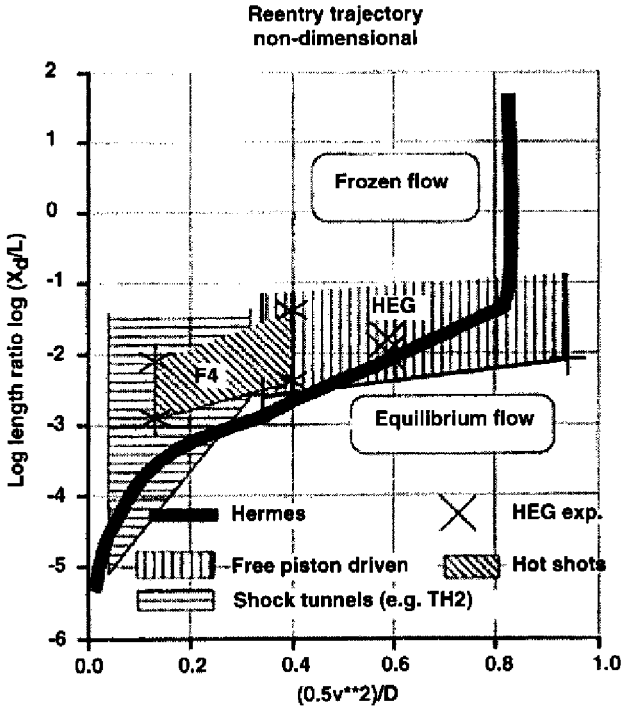

Figure 3. Simulation capabilities with respect to gas dissociation.
Here, X_D is the dissociation length for a strong shock;
D is the dissociation energy of nitrogen;
L is the length of the reentry vehicle;
v is the free-stream velocity

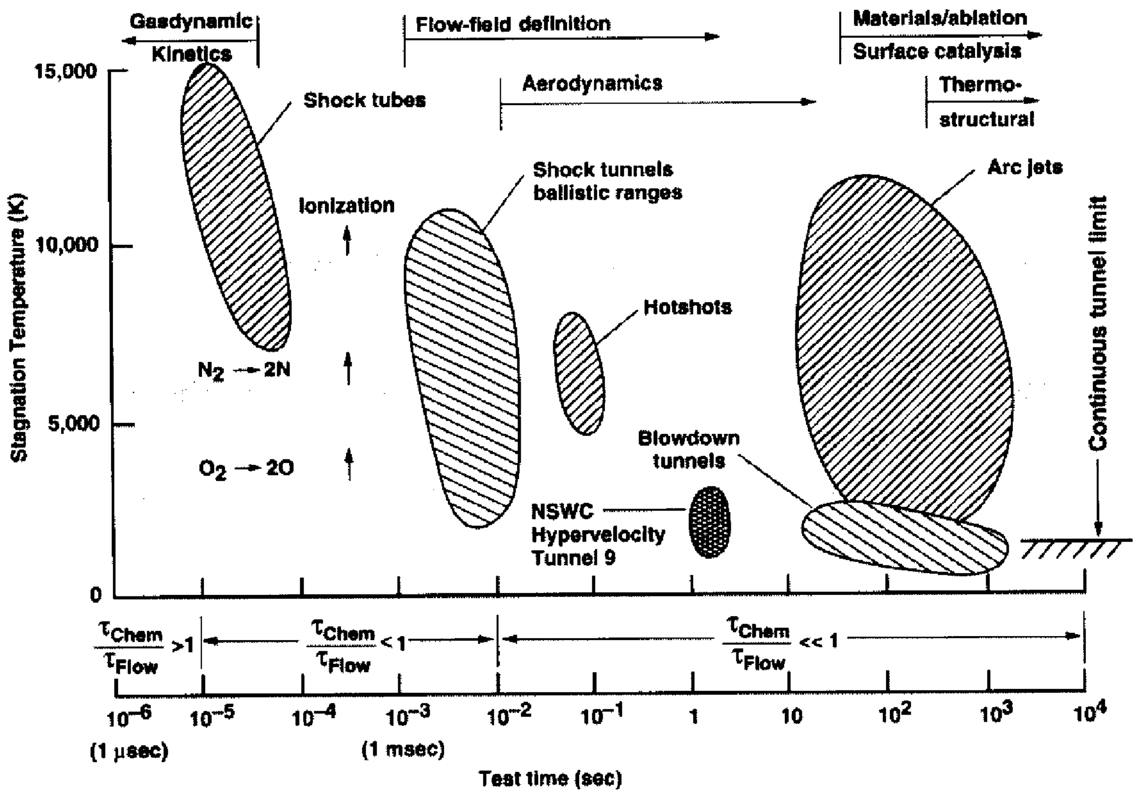

Figure 4. Stagnation point temperatures and flow duration domains for hypersonic simulation facilities.

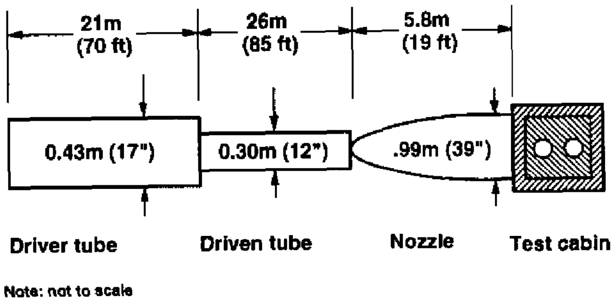

Figure 5. Schematic of NASA Ames 16-Inch Shock Tunnel.

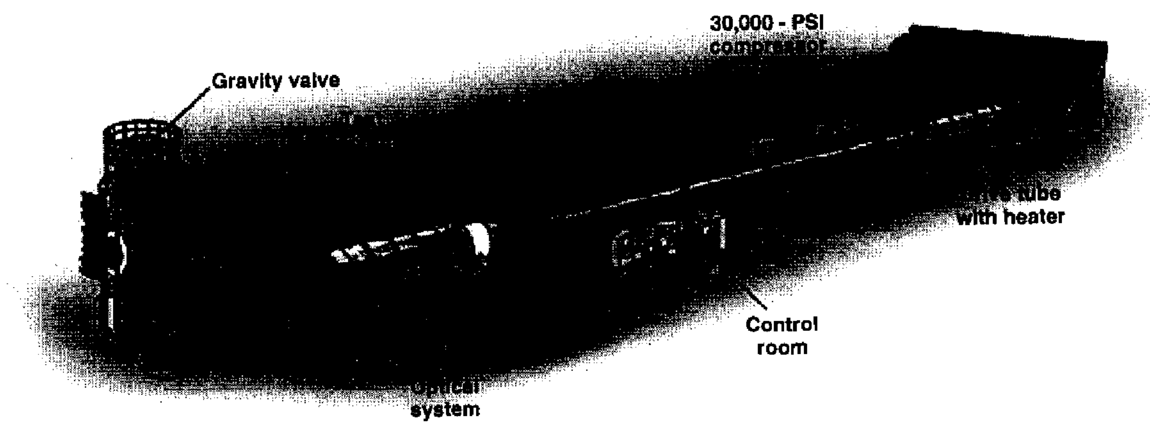

Figure 6. Calspan Large Enthalpy Shock Tunnel LENS.

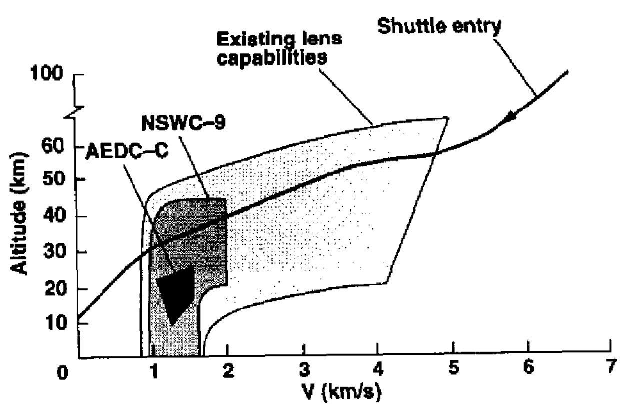

Figure 7. Performance range of Calspan LENS Facility.

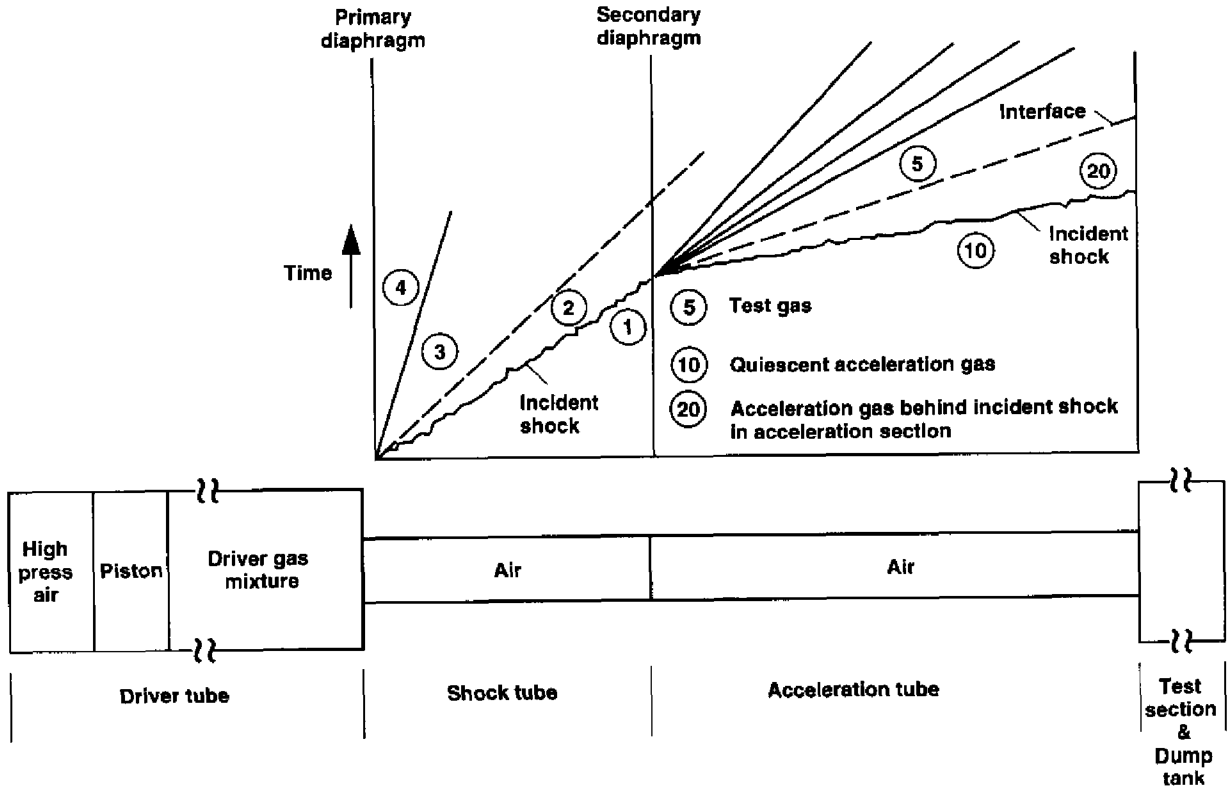

Figure 8. Operation of Expansion Tunnel with free piston driver.

Figure 9. NASA Ames Electric Arc Shock Tube.

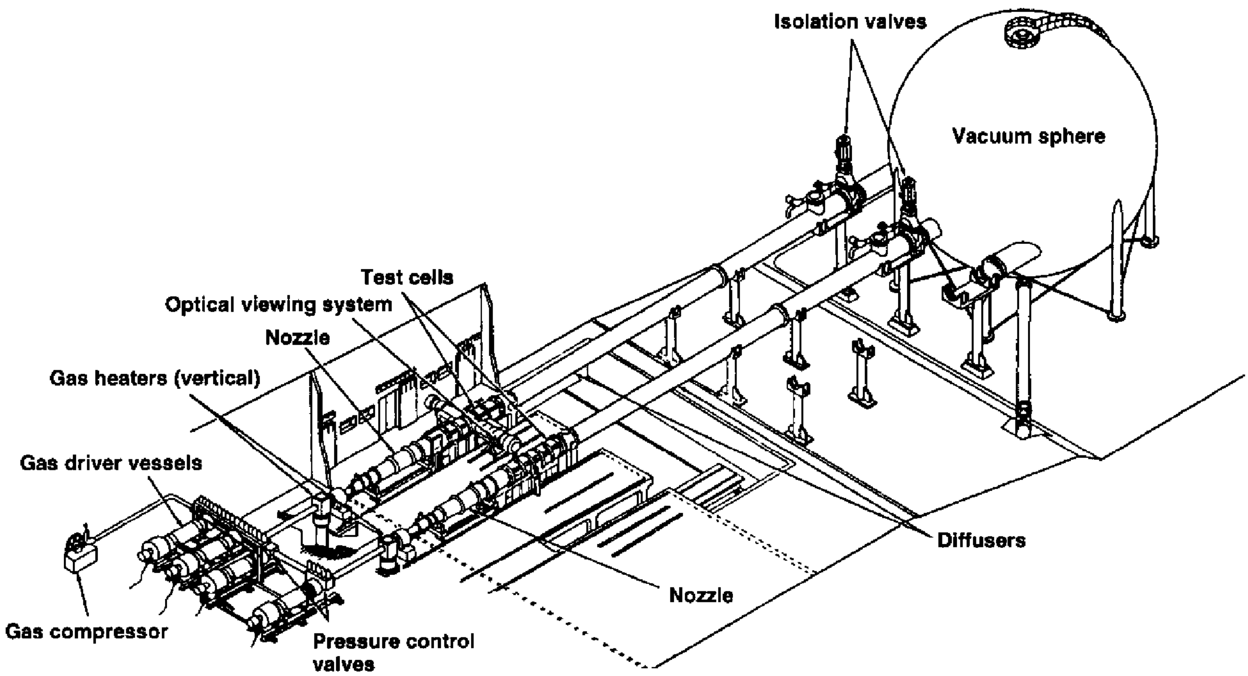

Figure 10. Naval Surface Warfare Center Hypervelocity Tunnel Number 9.

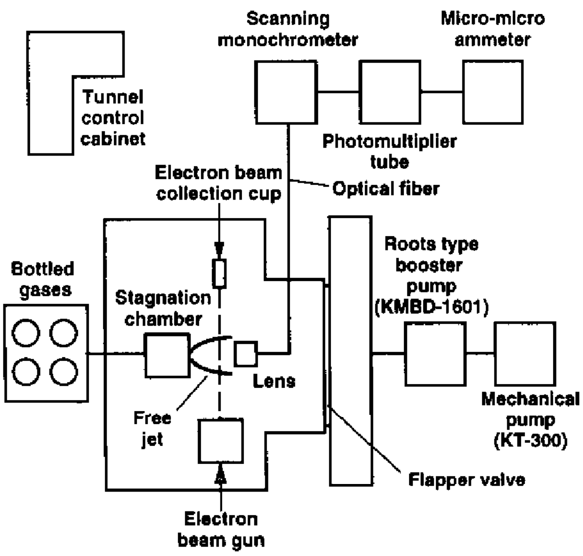

Figure 11. Schematic for University of California at Berkeley Low Density Wind Tunnel.

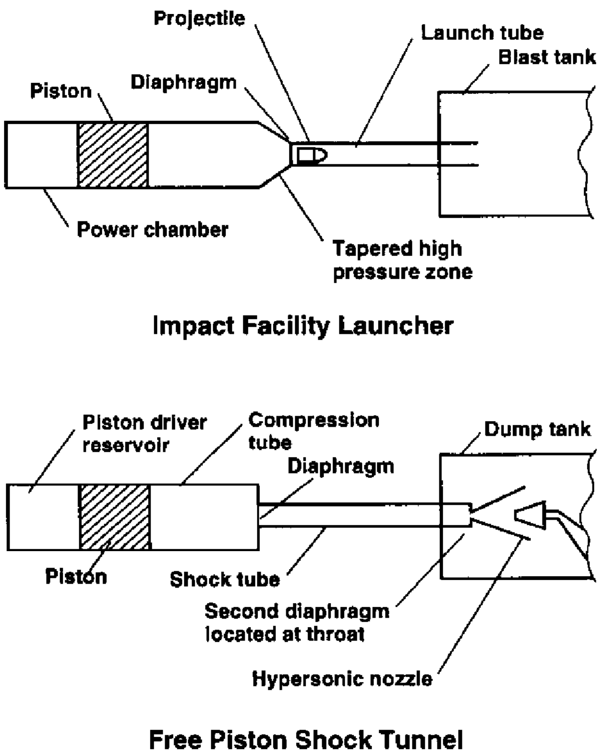

Figure 12. Comparison of light-gas gun and AEDC Concept for disposable-piston, High Performance Shock Tunnel.

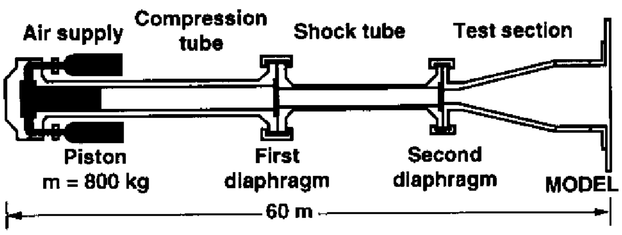

Figure 13. Sketch of the HEG DLR Free Piston Shock Tunnel in Göttingen.

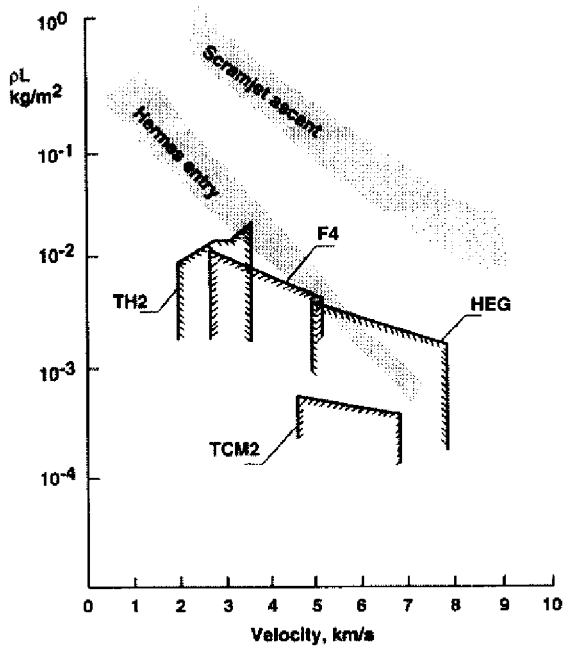

Figure 14. HEG performance and HERMES reentry trajectory.

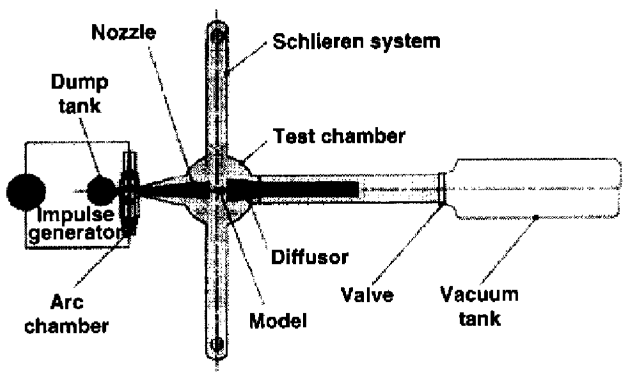

Figure 15. The ONERA High Enthalpy Wind Tunnel F4.

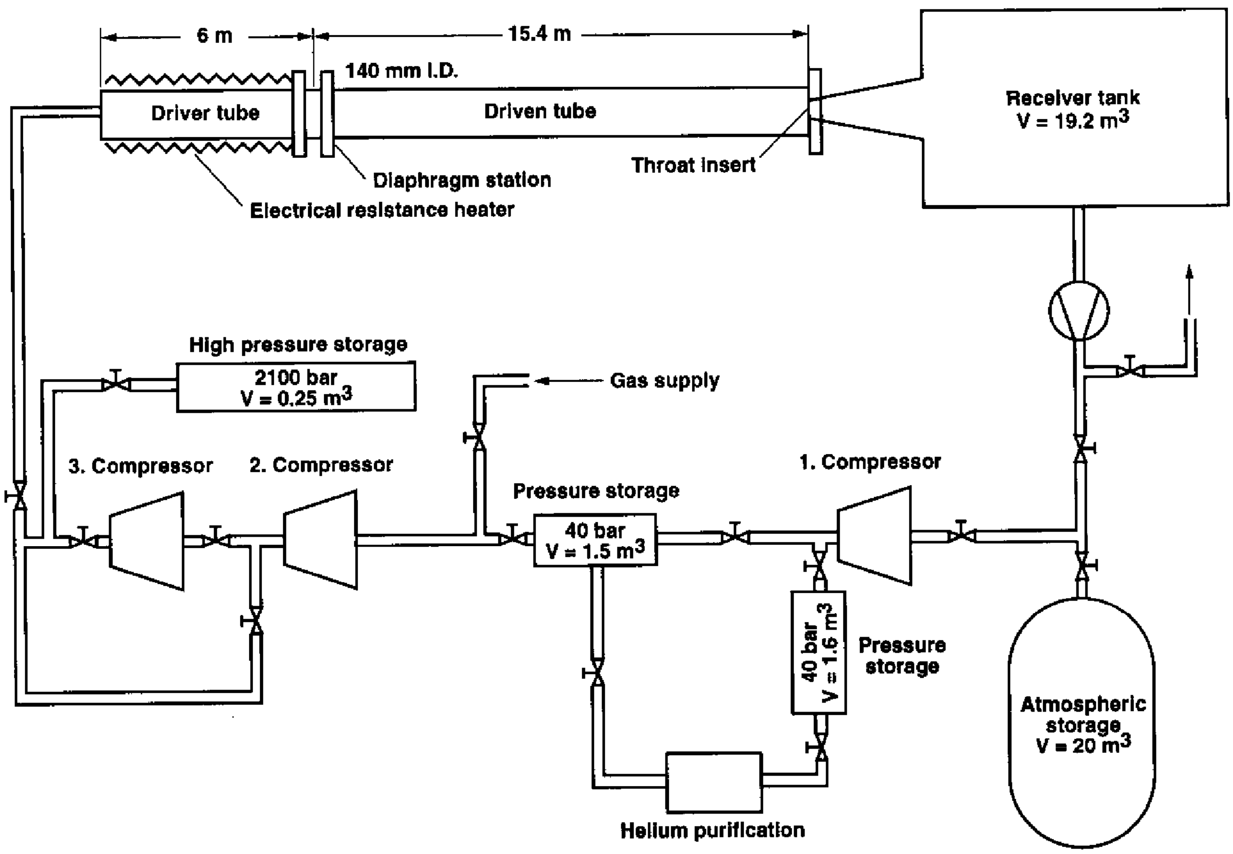

Figure 16. Basic components of the Aachen Shock Tunnel TH2.

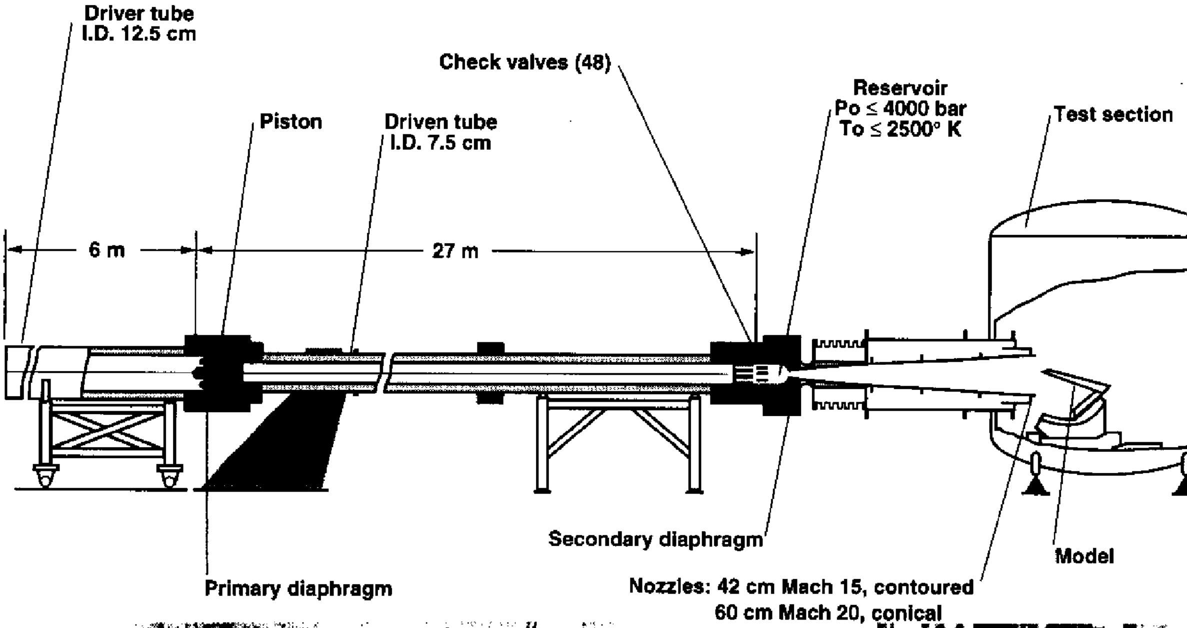

Figure 17. The VKI Longshot Shock Tunnel.

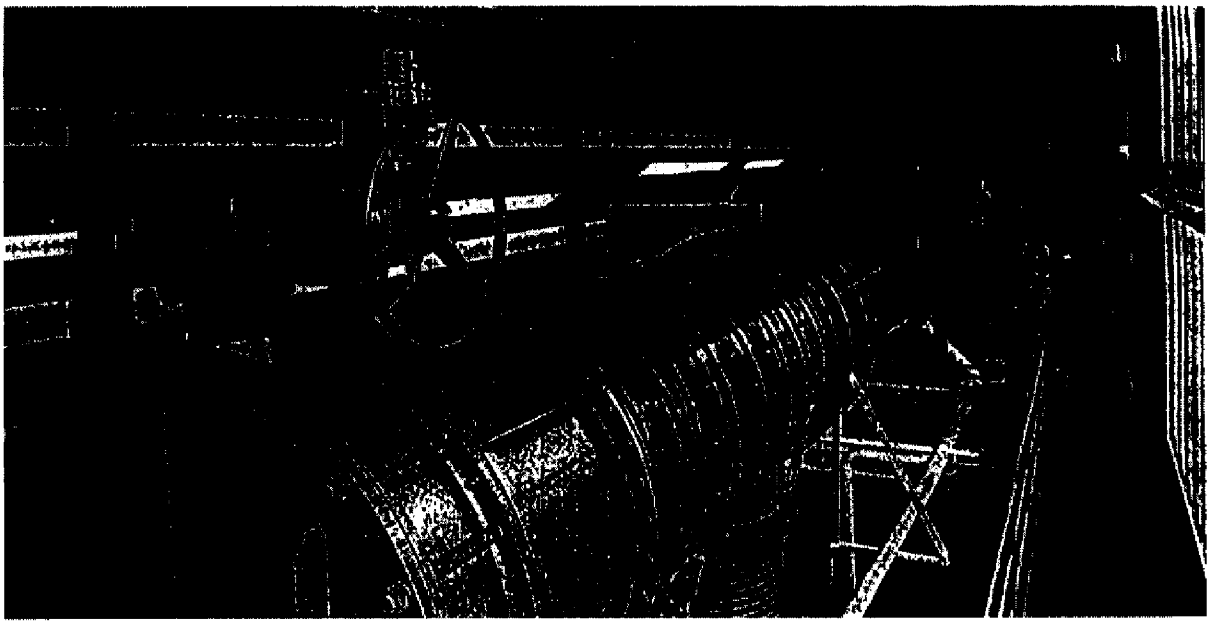

Figure 18. Photograph of the SR-3 Wind Tunnel at the CNRS (France).

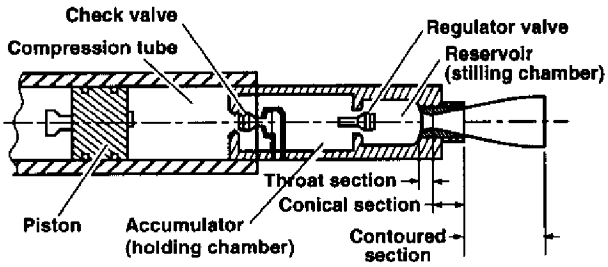

Figure 19. General Arrangement of PGU Compression Cascade.

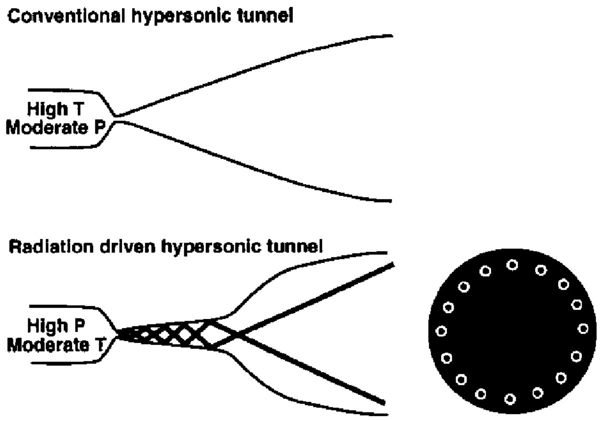

Figure 21. Conventional and radiation-driven wind tunnel philosophies.

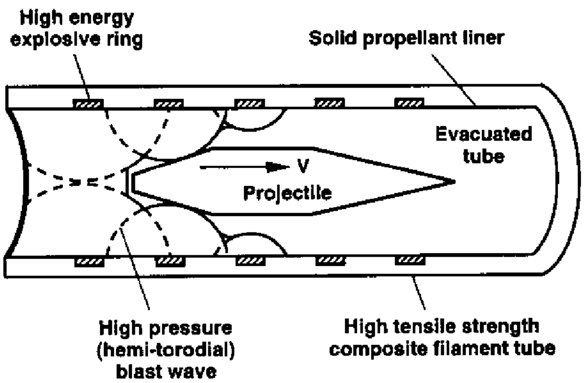

Figure 20. Oblique detonation wave driver.

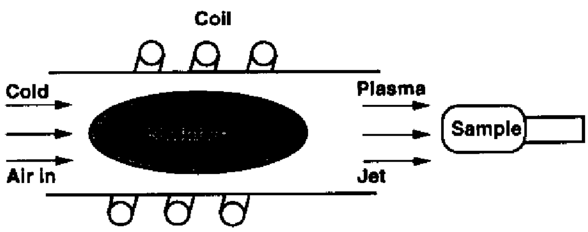

Figure 22. Schematic of VKI Plasmatron.