178
10

HYPERSONIC TEST FACILITIES:

REQUIREMENTS ANALYSIS AND PRELIMINARY DESIGN

by

Gregory Andrew Drauch

Thesis submitted to the Faculty of the
Virginia Polytechnic Institute and State University
in partial fulfillment of the requirements
for the degree of

MASTER OF SCIENCE

in

Aerospace Engineering

Approved: A. K. Jakubowski

Dr. A.K. Jakubowski, Chairman

Dr. J.A. Schetz

Dr. E. Nikolaidis

April, 1990

Blacksburg, Virginia

HYPERSONIC TEST FACILITIES:
REQUIREMENTS ANALYSIS AND PRELIMINARY DESIGN

by
Gregory Andrew Drauch

Committee Chairman: Dr. A.K. Jakubowski
Aerospace and Ocean Engineering

(Abstract)

There has come about, in recent years, a renewed interest in aerospace vehicles operating in the hypersonic regime. With this interest has come a need to not only reestablish the hypersonic test capability that was available in the 1960s but to enhance this capability to meet the demanding needs of todays proposed vehicles. This will require more capable hypersonic wind tunnels with larger test sections, longer run times and test gases more closely resembling the fluid to be encountered by the vehicle being developed. This document will review the current hypersonic testing capability, examine the operating characteristics of several hypersonic vehicles to develop a set of hypersonic testing requirements, and develop a preliminary design of a required hypersonic facility that addresses the demonstrated requirements. An order of magnitude cost estimate is also presented.

Acknowledgments

I would like to thank my advisor and chairman, Dr. Jakubowski, for all of his help and guidance during the preparation of my thesis. I would also like to thank my wife, Amy, for her patience, understanding, and encouragement in our first year of marriage when faced with the late nights and weekends devoted to this task. She has made this painful task much more tolerable.

<u>Table of Contents</u>

Table of Contents (cont)

Table of Contents (cont)

<u>List of Figures</u>

<u>List of Tables</u>

1.0 Introduction

1.1 Background

In the early 1960s the American aerospace community was very interested in hypersonic vehicles. In response to this interest, a significant hypersonic testing capability was developed. This capability was limited by the technology available but served to advance the communities knowledge of hypersonic flow. There has been a recent rekindling of interest in the hypersonic regime that includes programs such as the National Aerospace Plane (NASP) and hypersonic missile systems. To support the development of these vehicles, the American testing community must not only reestablish the hypersonic test capability that was available in the 1960s but must enhance this capability to meet the demanding needs of todays proposed hypersonic programs.

Hypersonic velocities are defined as velocities exceeding Mach 5 or five times the local speed of sound. There are many interesting events that occur in the hypersonic regime making hypersonic vehicle development and ground testing very difficult. For instance, as an Earth-bound vehicle approaches Mach 10, heating around the vehicle causes dissociation, meaning polyatomic molecules begin breaking apart into single atoms. As the velocity increases further, ionization starts. Ionization is the process whereby a plasma is formed (electrons are stripped from their atoms). At this point the vehicle becomes enveloped in an extremely hot plasma sheet.

Existing hypersonic wind tunnels were built in the 1960s and are characterized by several shortfalls including short run times, small test sections, and exotic test gases. These shortfalls were a result of the limited technology available in that era. In order to meet the current and projected needs in hypersonic testing, the American testing community must decide whether it is most prudent to modernize the existing facilities or to design and build a new, highly capable facility. Several considerations must be analyzed. These include:

- the age of existing facilities, - If the remnants of the test facilities built in the 1960s are still employing vintage or near-vintage equipment then the remaining lifetime of the equipment becomes a serious concern.

- the capabilities of existing facilities, - The existing facilities may not be capable of supporting the testing requirements of proposed hypersonic vehicles.

- and the availability of existing facilities. - Due to the limited capabilities of todays facilities, the most capable facilities are in great demand and typically have a backlog of two to three years.

1.2 Purpose

The purpose of this document is to review the current state of hypersonic test capabilities in the United States, analyze the current and projected needs of the hypersonic testing community, and propose a course of action to meet these needs. This document is intended to 1) stimulate interest in developing the needed hypersonic capabilities, and 2) begin developing the concerns, considerations, and issues associated with advancing the hypersonic capability of the U.S. aerospace community in an effort to maintain our technological edge.

1.3 Scope

This effort includes the assessment of numerous hypersonic test facilities and hypersonic vehicle development programs. It does not claim to be an exhaustive study of all facilities or programs. The test facility information examined represents those facilities that are the most capable, based on information available in an unclassified forum. The programs reviewed represent a sampling of highly documented programs from a variety of missions. In looking at the vehicle development programs, a detailed analysis of the designs was not performed. Instead the operational requirements were reviewed in order to determine the important parameters that must be tested. Higher level information concerning the vehicles anticipated velocity, body forces, or surface heating was examined. Likewise, the course of action proposed was done more as a preliminary analysis as opposed to a detailed, finalized plan.

1.4 Outline

This document is divided into two parts. The first part defines the hypersonic testing capability shortfalls and is organized as follows. Section 2.0 is titled Hypersonic Testing. This section presents a summary of hypersonic testing techniques. It describes several types of wind tunnel facilities. Section 3.0 is titled Hypersonic Programs and presents an outline of the goals, performance requirements, and testing needs of several ongoing or proposed hypersonic vehicle development programs. Section 4.0 is titled Comparison of Capabilities and Requirements and presents the test requirements of the programs outlined in Section 3.0 and their shortfalls based on existing capabilities that are presented in detail in Appendix A.

Part 2 utilizes the information presented in Part 1 and develops a course of action to insure the availability of the needed capabilities. Section 5.0, titled Needed Facility Parameters, presents a proposed facility that corrects most of the current hypersonic facility shortfalls. This proposal is a preliminary design and does not represent a detailed development plan. Section 5.0 also includes an estimate of the cost of the facility proposal. Part 2 concludes with Section 6.0 which summarizes the contents of this document and includes a brief review of the current capabilities, the projected requirements, the needed capabilities, a proposed facility that satisfies most of the needed capabilities, and a cost estimate associated with providing the needed capabilities.

2.0 Hypersonic Testing

2.1 Introduction

There are various ways to generate hypersonic test data. These methods will be discussed in this section. The topics that will be covered include:

- Wind Tunnel Testing
- Computer Simulation
- A Unique Concept in Hypersonic Testing

Another viable method of generating test data is through flight experiments. Specific examples of these types of programs are discussed in Section 3.0. While each of these methods have unique characteristics and merit, wind tunnel testing will be emphasized because of the primary topic of this document.

The impetus for implementing a test program is cost. A vehicle development program would be far too costly, and dangerous, without a comprehensive test program that plays an integral part of the design process. Flying an expensive, untested vehicle could risk millions of dollars and possibly human lives. Clearly, a very capable, validated testing capability is necessary to build safe, reliable, trustworthy flight vehicles. To this end, hypersonic test methods must provide the ability to advance the development of many critical technologies. Figure 2.1-1 presents these needs and divides them into two categories; aerothermodynamic and aeromechanic. This section reviews many of the current hypersonic facility technologies available in the United States.

2.2 Wind Tunnel Testing

Undoubtedly the most common method of aerospace testing is wind tunnel testing. While low-speed wind tunnels are fairly inexpensive and easy to build, hypersonic wind tunnels are very costly and complex. The main driver of this cost and complexity is the fact that a hypersonic wind tunnel, by definition, requires very high-speed flow. The need to generate a high speed flow results in the need to use a great deal of energy which is primarily needed to power compressors and heaters. While hypersonic tunnels do not follow a common low speed tunnel rule of thumb that states the power required varies as the cube of the tunnel speed, however "the implication of rapidly increasing power requirements with increasing test speed is correct." [1]

Aerothermodynamic Advances Needed in:

Gas Dynamics – Thermodynamics and transport properties, turbulent heat transfer validation, radiative heat transfer validation

Aerothermochemistry – Surface – gas phase, thermal penetration

Thermal Protection Materials – Evaluation and development of new composites, evaluation of contemporary materials and reuse capability

Aeromechanical Advances Needed in:

Configuration Refinement – Nose bend and aft slice trades, flap geometry and sizing, enhanced yaw stability

Six – DOF Aerodynamic Characteristics – Force and moment coefficients and derivatives, asymmetric vehicle effects, ablative mass addition phase lag effects, flap shape change, skewed laminar/ turbulent transition front effect, ablator roughness effects, gas composition effects

Six – DOF Trajectory and Control Simulations

Figure 2.1 – 1 Needed Critical Technology Advancements

The concept behind wind tunnel testing is simple. Namely, if a certain set of flow conditions can be created in a controlled environment, tests can be performed on a vehicle in that environment. Wind tunnel tests are used to generate data that will demonstrate the performance of a vehicle in actual flight conditions. This allows the design to be adjusted or "fine tuned." By using a small scale model, the development and construction of costly, large scale vehicles whose untested designs will most likely be altered is avoided. Unfortunately, complications arise in hypersonic flows that make wind tunnel testing at hypersonic speeds difficult.

> During the hypersonic flight of a vehicle through the atmosphere of the the Earth, the gas that passes through the bow shock experiences what is called real-gas effects. These effects are the result of excitation of vibration, dissociation, and ionization energy modes of the atmospheric gas. The excitation to higher energy modes causes changes in the shock-layer flow, such as reduction in the static temperature as energy is absorbed from the flow, a small increase in static pressure, and a large increase in density. [2]

Simulating these effects is very difficult. Some other problems arise including scaling problems and inaccuracies that result from the inability, at present, to simultaneously recreate all of the parameters associated with hypersonic flight in the atmosphere. A result of the deficiencies in existing facilities is the need for hypersonic test programs to utilize several different wind tunnels that have different abilities. Several tests must be performed in an attempt to test all or most of the important flow parameters. Some degree of uncertainty is introduced because the tests do not incorporate all necessary parameters, possibly missing a significant interaction.

2.2.1 Types of Hypersonic Tunnels

There are various wind tunnel configurations suitable for hypersonic testing. The paragraphs that follow will present a description of each of these types. They include:

- Intermittent Blowdown Tunnels
- Continuous Blowdown Tunnels
- Pressure-Vacuum Tunnels
- Hotshot Tunnels
- Plasma-Arc Tunnels
- Shock Tunnels, and
- Gas Gun Tunnels

<u>Intermittent Blowdown Wind Tunnel</u> - A typical intermittent blowdown tunnel configuration that vents to the atmosphere is shown in Figure 2.2-1. (A closed cycle configuration blowdown tunnel would include some method of collecting the test gas after passing through the

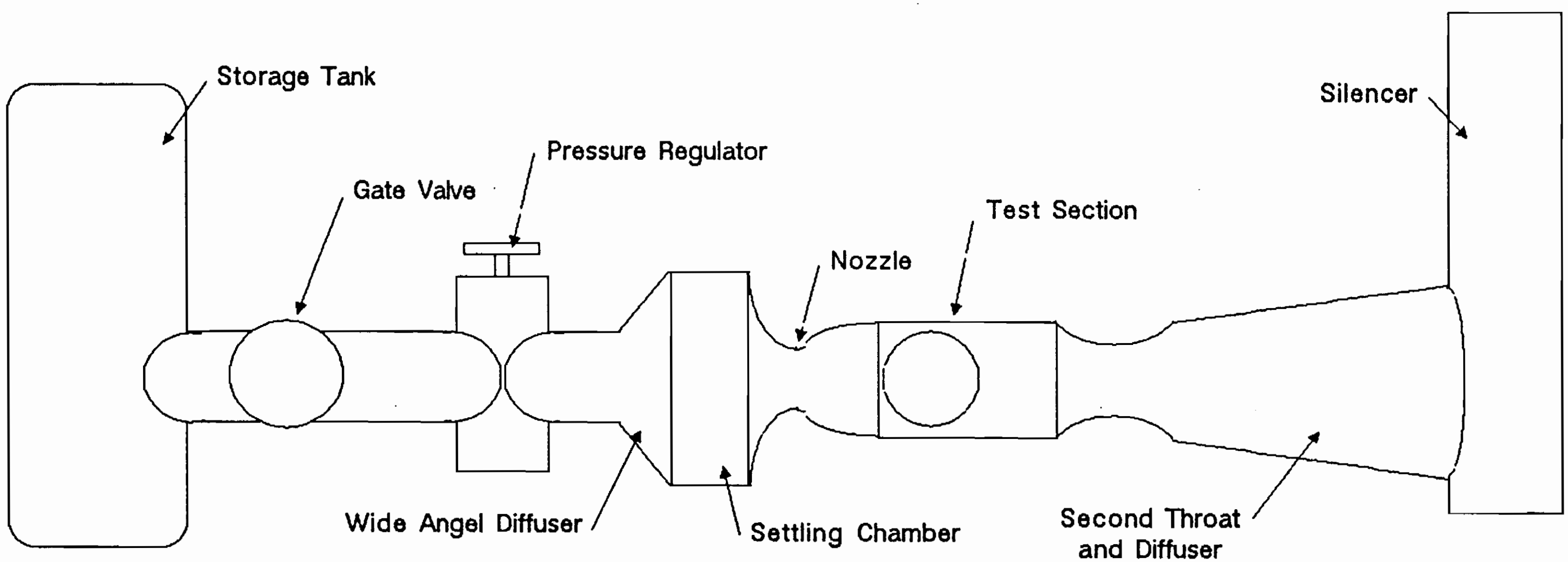


Figure 2.2 – 1 Intermittent Blowdown Wind Tunnel

diffuser.) The intermittent blowdown tunnel utilizes a source of a pressurized test gas to create the desired flow for test runs 10s of seconds long. These flows are typically in the lower range of the hypersonic regime. Once the storage tank is sufficiently pressurized, the tunnel is ready to be started.

The starting process involves a complex series of events and careful consideration of these events is necessary when designing a tunnel. Ideally, once the flow in a blowdown tunnel has been initiated, the nozzle throat velocity increases until it reaches Mach 1 and a shock forms in the throat. As the tunnel velocity increases, the shock moves through the diverging portion of the nozzle, the test section, past the model, through the second throat and into the diffuser. These events occur if the tunnel has been properly designed.

There are two crucial points in this process. The first occurs in the test section where the shock moves past the model. The tunnel must be designed to provide a sufficient pressure ratio to push the shock past the model. This is a crucial point because of the reduced cross sectional area in the test section due to the model.

Once again, the pressure ratio must be great enough to overcome the adverse pressure gradient caused by the second throat. In fact, some wind tunnels have a variable area second throat to reduce the pressure ratio requirement. This approach however, does not save money and adds to the tunnel complexity. Therefore, most tunnel designers simply design a sufficient pressure ratio capability into the facility to overcome a constant area second throat.

In order to establish a uniform flow through the test section, a settling chamber must be incorporated into the design. These chambers typically have flow settling screens and are serviced by a wide angle diffuser that disperses the flow from a comparatively small duct into the large settling chamber. The flow must be evenly distributed throughout the chamber in order to have uniform flow in the nozzle. A typical diffuser utilizes a conical device with holes positioned to evenly disperse the flow.

Figure 2.2-1 includes those components that are necessary to run the tunnel and perform tests (except for the data acquisition system). There are a few components that are not shown. They include the compressors, aftercoolers, oil filters, and dryers.

The compressor must be sized to accommodate the desired facility duty cycle. There are basically two scenarios. One uses a storage tank large enough to accommodate one eight hour shift of testing. Generally in this case, a slower, smaller compressor can be used and is usually left running all night. By morning, the storage tank is fully pressurized and ready for an eight hour shift of operation. It should be noted that this does not mean that the tunnel will operate for eight hours. This means that one shift's worth of testing may be performed. This usually corresponds to four to ten runs per day. The other scenario uses a smaller storage tank and a faster compressor. Here, the tank is refilled between each run; a process that generally takes 20 to 40 minutes or about the time it takes to adjust the model for another test. The choice of scenarios depends on the desired duty cycle of the wind tunnel. The costs of each are comparable and is not a driving issue in the decision.

The aftercooler, oil filter and dryer are used to process and purify the test gas after the compression stage. The aftercooler is required because most oil filters do not operate at the higher temperatures encountered in the compression process. The oil filter removes impurities that are introduced in the compression process. These impurities must be removed since accelerating them to hypersonic speeds could damage the model or some of the wind tunnel components. The dryer is used to remove moisture from the air so that it does not condense, damage facility equipment, and make the flow unsuitable for testing.

The last piece of equipment not appearing in Figure 2.2-1 is a heater. Generally a heater is necessary to increase the stagnation temperature of the test gas so that the static temperature of the gas after expansion remains sufficiently high to avoid liquefaction. This is also the only method of increasing the stagnation temperature on the model when performing aerothermal tests. These heaters are available in many configurations and may be used before the gas enters the storage tank or after.

<u>Continuous Wind Tunnel</u> - Figure 2.2-2 presents a typical continuous tunnel configuration. It is very similar to the intermittent blowdown tunnel and in fact can be used like one. The difference lies in the compressor capacity. A continuous tunnel incorporates a series of compressors that are capable of matching the mass flow rate of the wind tunnel. Therefore, once an equilibrium condition has been established, the tunnel can run for hours.

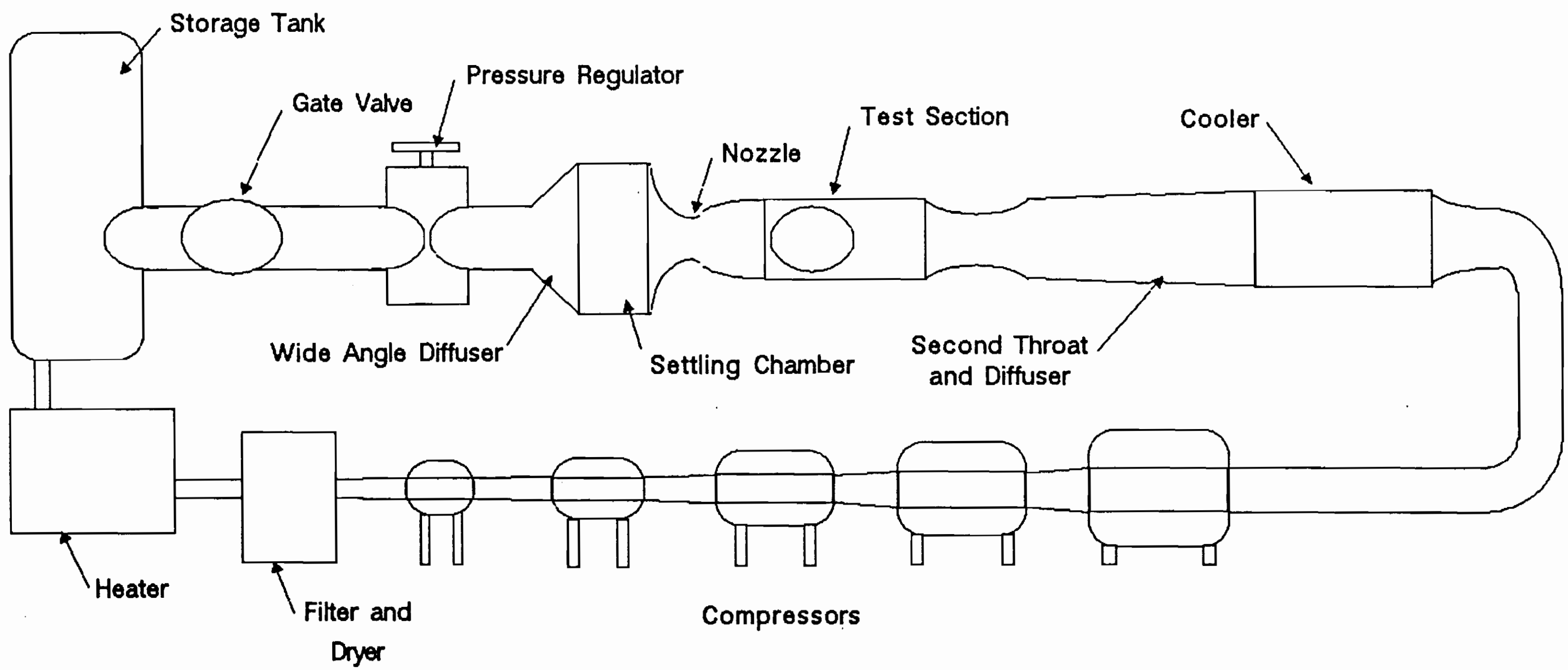


Figure 2.2 – 2 Continuous Wind Tunnel

Continuous tunnels have the same difficulties and design considerations as intermittent tunnels. The added components, namely the cooler and the increased compressor capacity, are present to allow the continuous operation of the tunnel. The cooler reduces the temperature of the gas thereby increasing its density. This somewhat eases the compressor requirements. Again, the extra compressor capacity must match the mass flow of the tunnel to allow continuous operation.

The primary difficulty with operating hypersonic continuous facilities is the tremendous amount of power required to run the compressors throughout the test run. This power requirement drastically increases the cost of each test and affects the operation of other tunnels collocated with a continuous hypersonic tunnel.

<u>Pressure-Vacuum Wind Tunnel</u> - The pressure-vacuum wind tunnel is one of the most common types of hypersonic wind tunnels currently operated. It incorporates features of a blowdown tunnel and an indraft tunnel. A typical configuration is shown in Figure 2.2-3. There are some obvious differences between the pressure-vacuum tunnel and the blowdown tunnel. However, many of the components of a pressure-vacuum tunnel are also used in the blowdown

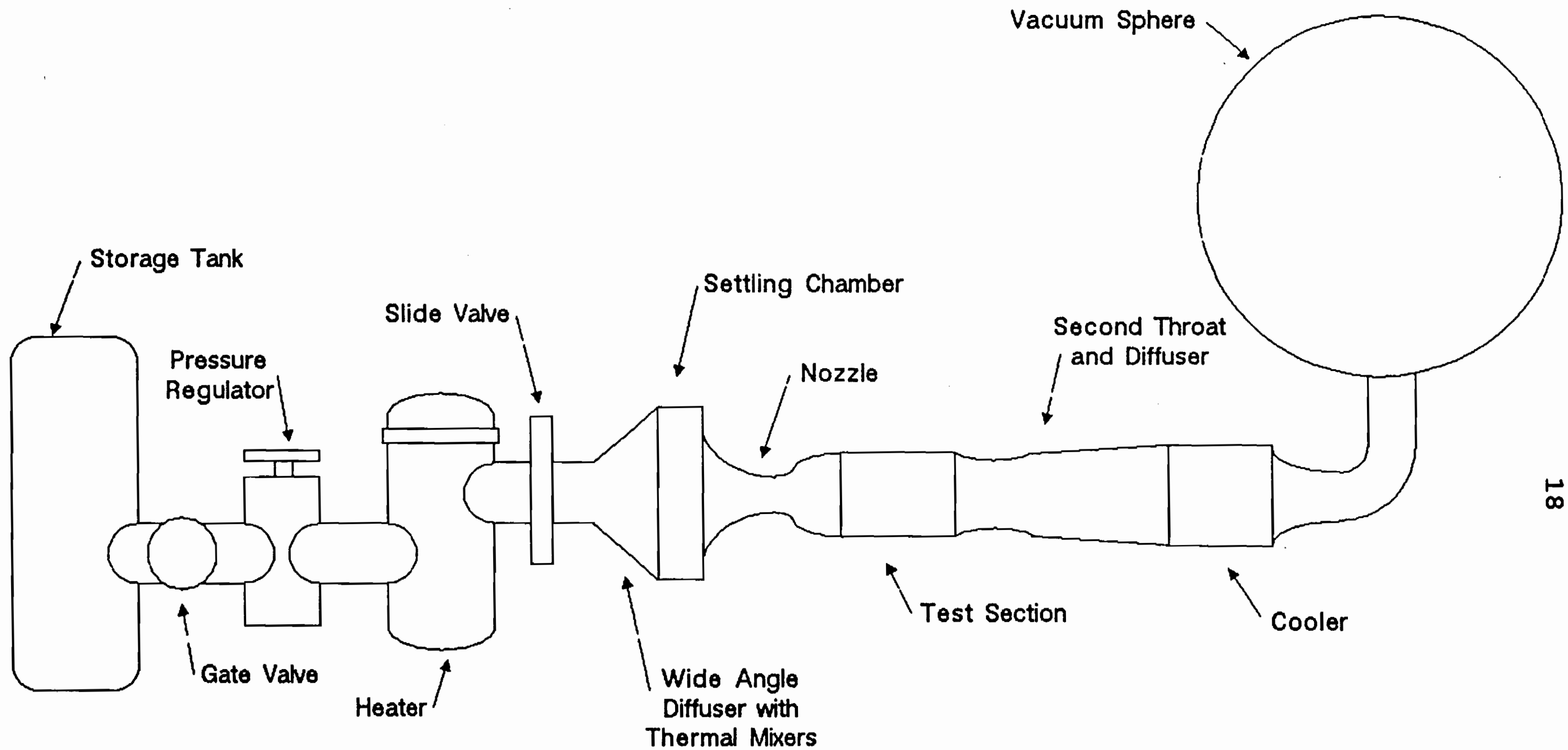


Figure 2.2 – 3 Pressure – Vacuum Wind Tunnel

tunnel. The following paragraphs focus on the major differences between the pressure-vacuum and blowdown tunnels.

One of the first differences is the presence of a heater downstream of the pressure regulator valve. These are generally of the heat storage type and are charged over an extended period of time.

Another difference is the addition of a thermal mixer at the inlet to the settling chamber. This is necessary because of the extreme thermal stratification that can occur in the heater. If this gradient were allowed to continue through the the nozzle and to the test section, the flow through the test section would not be uniform. The use of thermal mixers is not without problems. At higher temperatures, material problems reduce or eliminate the ability to use mixers. In addition, thermal mixers tend to reduce the pressure of the flow thereby reducing the pressure ratio, they cause a temperature loss, and they cause a great deal of turbulence. The turbulence is difficult to remove and in many cases introduces another thermal gradient. These problems make the use of thermal mixers at high temperatures very difficult.

Because of the higher pressure ratios and temperatures, some of the tunnel circuit may have to be cooled. Candidate regions include the nozzle throat and the pressure regulator. Some areas will require active cooling measures while others will only require some passive cooling method, for example a heat sink.

The last significant difference stems from the addition of a test gas cooler downstream of the diffuser and before the vacuum sphere. A gas cooler is necessary for operation of a pressure-vacuum tunnel. Without it, "the rate of pressure rise in the [vacuum] tanks will be relatively high because of the high temperature and the relatively large volume of air." [1] Therefore, by cooling the gas, the pressure ratio change is slowed and the run time is extended.

An important variation of the pressure-vacuum tunnel is known as a low-density tunnel. The primary difference is the absolute pressures at which these tunnels operate. Low-density tunnels operate at lower pressures and mass flows than normal pressure-vacuum tunnels and therefore have subtle differences with respect to normal pressure-vacuum tunnels.

First, in low density tunnels the test section is generally a vacuum chamber that includes several low temperature plates that liquefy the incoming gas thereby reducing the gas pressure. The liquid is then frozen or pumped out of the vacuum chamber. This process increases the run time and maintains the required low pressure. The pressurized test gas is usually kept at one atmosphere or less. The mass flow is kept low by using a very small nozzle. These types of facilities are used to simulate hypersonic flight in the extreme upper atmosphere. There are three regimes in this rarefied flow region that these altitudes encompass including the "slip flow" regime, the "transition flow" regime, and the "free molecule" regime. These types of tunnels are only useful for simulation in these regimes. (See Figure 2.4-1 for definition of these regimes)

<u>Hotshot Tunnels</u> - Hotshot tunnels are very short duration, high temperature, high pressure test facilities that can generate very high Mach numbers. Figure 2.2-4 shows a typical hotshot tunnel configuration. The high velocity flow is established by discharging a large electrical charge in a small volume of pressurized air called an arc chamber. The arc chamber is separated from the rest of the tunnel, which is maintained at a low pressure, by a thin metal or plastic diaphragm. The discharged electricity quickly heats the air and increases

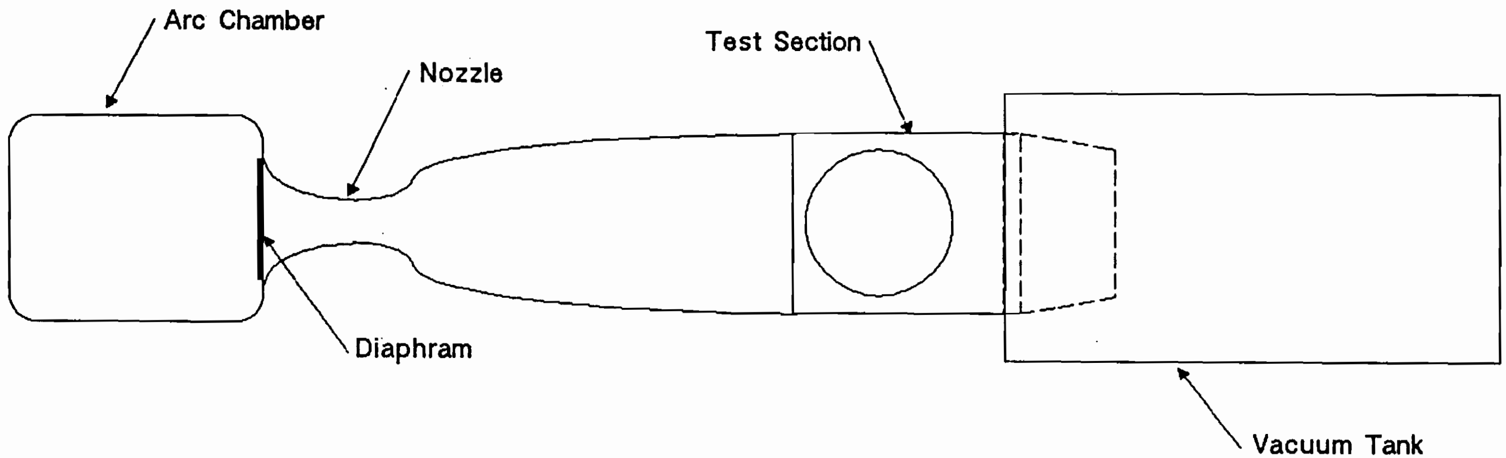


Figure 2.2 – 4 Hotshot Wind Tunnel

the chamber pressure. This increase in pressure and temperature causes the diaphragm to rupture allowing the high pressure, high temperature air to expand through a nozzle and into the test section. The run stops when the shock wave that is created by the sudden expansion reflects off the back wall of the tunnel and reaches the model. Run times on the order of 10 to 100 microseconds are achieved. However, because of the decay of the pressure and temperature with time, the velocity varies continuously during the run and must be considered when analyzing the data. [1]

Typical chamber pressures and temperatures are $2.07x10^8$ N/m^2 and 4000 K, respectively. Temperatures and pressures higher than these are possible but problems with erosion tend to contaminate the flow and significantly alter the results. For example, an early hotshot tunnel was designed to operate at $6.90x10^8$ N/m^2 and 10,000 K. Results from initial test runs in this tunnel showed that the mass flow due to erosion contaminants was several times the mass flow of the test gas. In order to reduce or eliminate the erosion problem hotshot tunnels must operate at lower pressures and temperatures. One method of increasing the operating pressure and temperature in a hotshot tunnel is to use nitrogen as a test gas. While this does not allow pressures as high as $6.90x10^8$ N/m^2

or temperatures as high as 10,000 K, it does allow slightly higher pressures and temperatures when compared to air. The major drawback of using nitrogen, or of operating at reduced pressures and temperatures with air, is the loss of real gas effects in the vicinity of the model. Dissociation and ionization do not occur in nitrogen at the same temperature as air and they do not occur at lower temperatures. Mach numbers in excess of 20 can be generated in existing facilities. [1]

Plasma-Arc Tunnels - Plasma-arc tunnels are very high temperature tunnels generally used for thermal load experiments. A schematic drawing of a typical plasma-arc tunnel is shown in Figure 2.2-5. Temperatures as high as 13,900 K are possible at Mach numbers spanning the supersonic regime and extending into the lower portion of the hypersonic regime. Low stagnation pressures are generally used and test gases other than air are common. Argon is a popular gas because of the higher temperatures achievable. [1]

As shown in Figure 2.2-5, plasma-arc tunnels consist of an arc chamber, a nozzle, and an evacuated test chamber. A cold stream of test gas enters the arc chamber where the gas is heated beyond the ionization point and it becomes a plasma. The plasma is accelerated by the nozzle

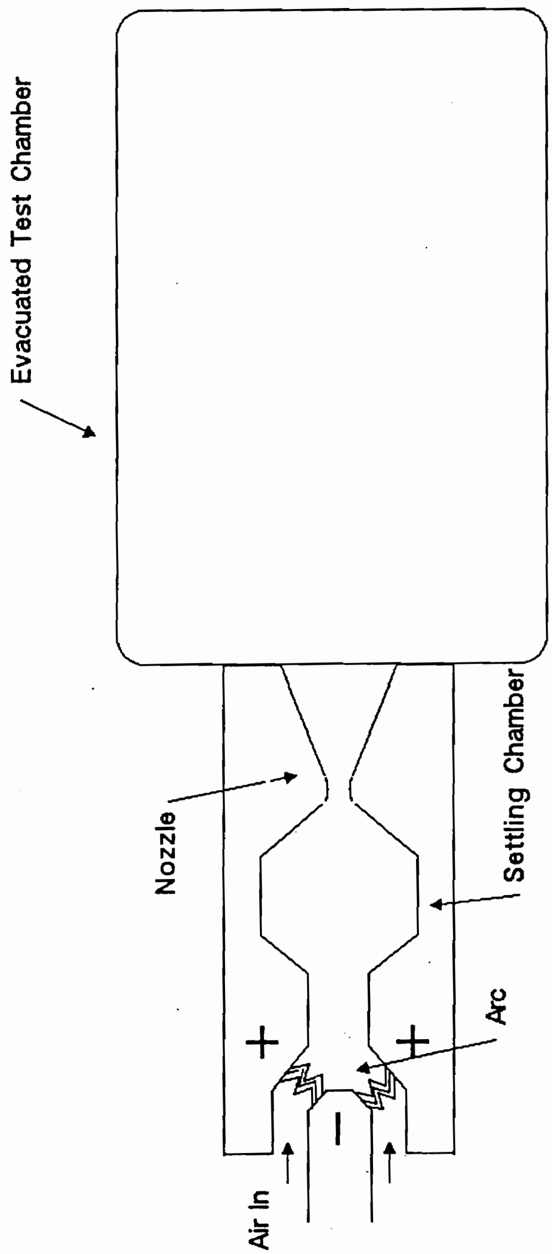


Figure 2.2 – 5 Plasma – Arc Tunnel

and enters the test chamber. A major problem with plasma-arc tunnels is the variation of temperature across the plasma stream. [1]

Shock Tunnels - Shock tunnels are very high velocity, short run time wind tunnels that utilize the heating and compression effects of a shock tube. Mach numbers exceeding 25 are possible. However, run times are limited to a few milliseconds. A shock tunnel configuration is presented in Figure 2.2-6. [1]

The shock tunnel flow is driven by a shock tube. The tube is fired and the high-pressure driver gas sets up a shock wave in the low pressure driven gas. The resulting shock travels toward the nozzle. When the shock reaches the nozzle the nozzle diaphragm is ruptured and the heated, compressed air behind the shock is accelerated through the nozzle and into the test section. The run is terminated when the shock, which was reflected at the nozzle, reflects off the driver-driven gas interface and reaches the nozzle again. The run time is generally on the order of 5 msec. Figure 2.2.7 shows a shock tunnel wave diagram. Run times can be extended by tailoring the driver-driven gas interface so that the pressure rise from the driven gas to the driver gas is the same as the pressure ratio across the reflected shock. Equalizing the pressure ratio in this way

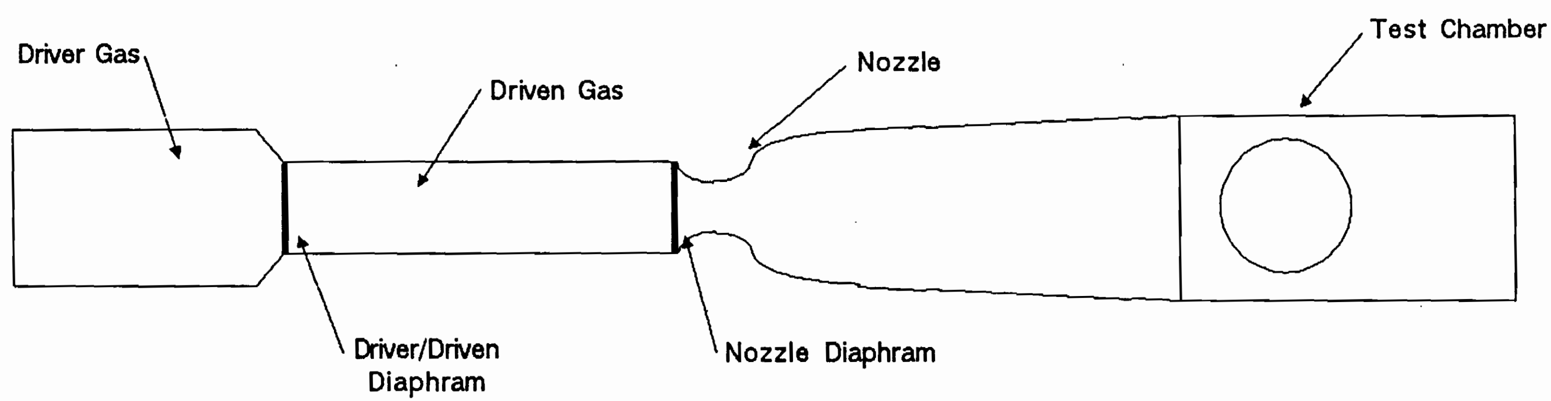


Figure 2.2 – 6 Shock Tunnel

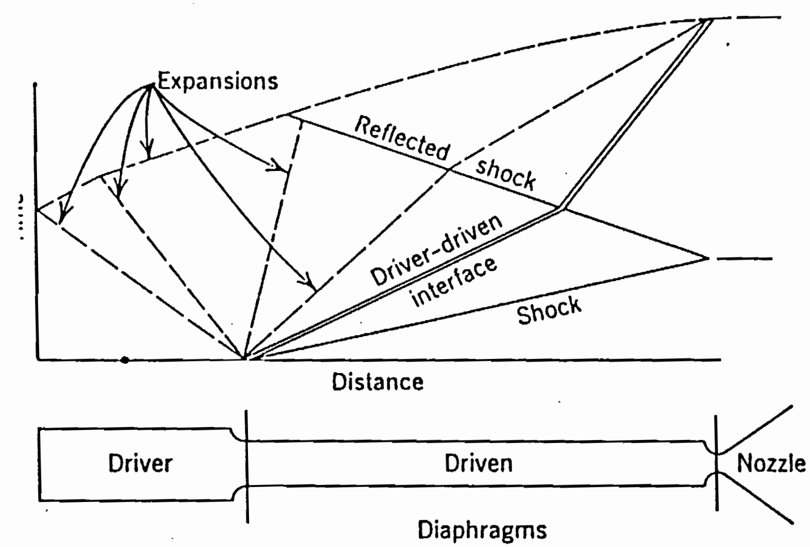


Figure 2.2 – 7 Shock Tunnel Wave Diagram

causes the reflected shock to pass through the interface without interfering with the flow, thus extending the run. Pressures in excess of 2000 atmospheres and temperatures greater than 7,800 K are possible. [1]

Gas Gun Tunnels - Gas gun tunnels, depicted in Figure 2.2-8, are very similar to shock tunnels. The difference is the presence of a piston in the driven gas at the driver-driven gas interface diaphragm. This piston is accelerated and drives the flow. As in the shock tunnel, a shock tube initiates the flow. The shock ruptures the driver-driven gas interface diaphragm and accelerates the piston. The piston in turn compresses the driven gas and a shock from the acceleration of the piston heats the driven gas. As the pressure and temperature of the driven gas increases, the diaphragm directly upstream of the nozzle is ruptured allowing the driven gas to be accelerated through the nozzle. [1]

The piston is the critical technology of a gun tunnel and its durability limits the pressures and temperatures achievable. Typically the piston weighs on the order of 10 to 15 grams. It must be able to withstand high temperatures, high accelerations and non-uniform loading. Because of these limitations, gun tunnels operate at lower pressures and temperatures than shock tunnels.

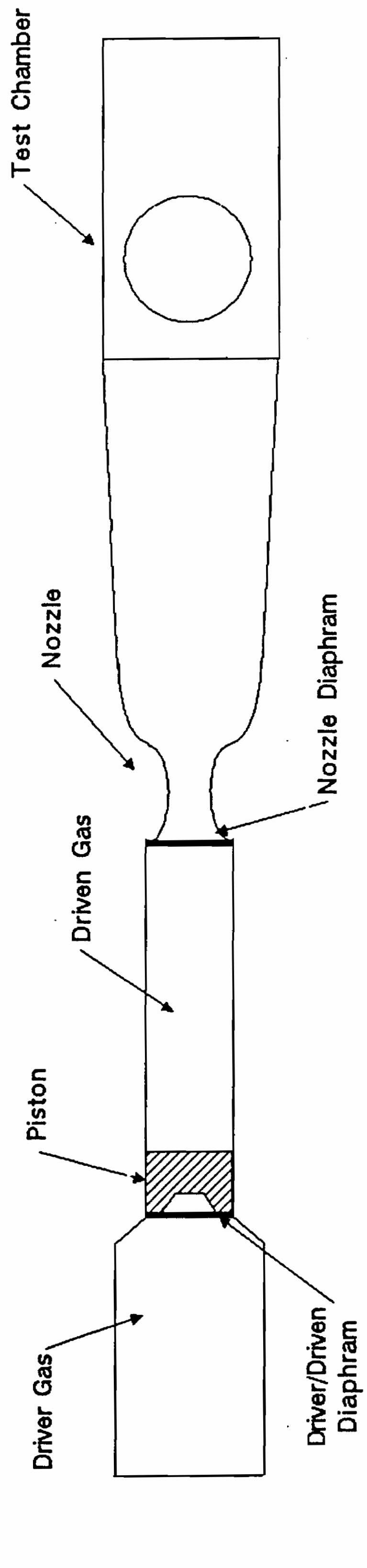


Figure 2.2 – 8 Gas Gun Tunnel

However, the run time is increased by an order of magnitude and the pressure in the test section remains nearly constant because of the piston.

The piston generates the requirement to limit the ratio of driver to driven gas pressure which combined with the desirability to have a high operating pressure, results in a higher pressure and an increased mass in the driven tube. The higher pressure and increased mass in the driven tube increases mass available for acceleration through the nozzle translating into longer run times. Run times in excess of 25 milliseconds are possible. [1]

2.2.2 Data Acquisition, Recording, and Processing

The data acquisition techniques used for hypersonic testing are similar to those used for supersonic testing. They include methods such as Pitot tubes, force and moment balances, Schlieren photography, interferometry, shadowgraph visualization, laser doppler velocimetry, oil flow visualization, thermocouples, pressure transducers, etc. These techniques are fairly common and will not be discussed in detail in this document. Data recording systems are generally multiple input devices capable of recording up to 256 inputs per cycle with each cycle lasting less than a second. Typically, the data is recorded on magnetic tape. The data processing systems commonly found are capable of some level of real-time or near real-time data reduction. This capability is particularly useful as it allows experimenters to make sure they have reasonable and sufficient data as well as allowing some flexibility in a test program. For instance, if one run results in exceptionally interesting data, the experimenter can tailor a subsequent run to further investigate some phenomena. Similarly, a run resulting in questionable data can be redone. [3]

The above capabilities are generally found at most hypersonic tunnels. A primary concern in designing a wind tunnel to insure adequate data acquisition, recording and processing equipment. Without adequate equipment a wind tunnel's utility is greatly reduced. [1]

2.3 Computer Simulation

A great deal of interest in developing hypersonic flow computer simulations has been generated by the increased cost of constructing capable hypersonic facilities and by the increased availability, and reduced cost, of extremely capable computers. There are numerous computational fluid dynamic (CFD) codes applicable to hypersonic flow, however a lack of data to verify these codes causes uncertainties in code capabilities. [4]

The problem with code verification is exemplified in the work being done with the National Aerospace Plane (NASP - see section 3.2).

> ... There are no wind tunnels which can test the airflow around complete models at speeds above Mach 8. So [NASP] has to use computers to determine how hypersonic airflows will affect the [vehicle]. [5]

NASP scale models have been tested to Mach 8. These results have been compared to the aerodynamic codes being used so that the codes can be refined to match the scale tests. However, the lack of full scale test data introduces uncertainty in the vehicle development program. [5]

Computational solutions must take numerous factors into account. These include:

- radiative cooling
- massive blowing of ablation gases
- diffusion
- radiative transport
- flowfield turbulence
- shock shape [6]

However, the Navier-Stokes equations that describe the flow are too complex to solve and simplifications must be made. There are numerous simplifications possible including making various assumptions about the flow. [6] These assumptions include using inviscid flow, linearization of the Navier-Stokes equations, and a host of boundary layer assumptions.

Complex equations have also hampered CFD work in the design and analysis of hypersonic propulsion systems. For these systems, the equations involve hypersonic flow as well as a variety of chemical and thermodynamic reactions. In addition, mixing rates and reaction rates are complicated by velocities on the order of 3,500 meters per second. [5] All of these difficulties combine to form a very complex problem. Models do exist but they are hard to verify because of the extreme conditions. [6] Testing is difficult because the flow conditions are hard to simulate in a ground test facility.

A factor that further complicates the modeling of airflow around a complex vehicle like NASP is the coupling effects that arise from a complex, integrated vehicle design. The NASP vehicle, for example, uses an integrated engine design with body components that blend into one another (see section 3.2). This results in complex flowfields where the flow over one component is coupled with the flow over another. [5] (This coupling effect would be present in any complex, integrated vehicle design.)

There are basically two types of computer simulations. One is very detailed and attempts to accurately model as many parameters as possible. These are costly to develop, difficult to verify, and very expensive to run. CPU times on the order of 100 minutes are not uncommon. This type of detailed effort requires extensive resources, both computer and human. [6]

The other type of model is the engineering code. These tend to be more simplified but reliable approximations that generate solutions that are generally within 10% of the detailed model solutions. This type of model tends to be inexpensive to develop and cost effective to run. [6] A prudent design team would likely make use of both types of models. The engineering model would be used to quickly narrow the design to something close to optimum. This would then be refined using the detailed model. This scheme minimizes computer time and design time.

The most important activity necessary for the future of computer modeling of hypersonic vehicles is the development of more complete experimental data bases from which the models may be verified. Many processes that are currently being modeled remain to be verified. For instance, "... increased confidence in [computer] solutions is possible if the modeling of transition to turbulent flow and turbulence for such flows can be substantiated with experimental results." [6] The lack of experimental data that can be used with confidence, hampers the verification and validation of existing models.

2.4 A Unique Concept in Hypersonic Testing

This section discusses a unique concept in hypersonic testing. The concept involves using a tether connected to an orbiting Space Shuttle to tow an instrumented model through the upper atmosphere. This concept would provide a continuous "wind tunnel" capable of simultaneously "simulating" all of the necessary flow parameters for long periods of time. [7]

The concept, as described in the NASA "Tethers in Space Handbook", would provide for a tether capable of being lowered from the Space Shuttle to a minimum altitude ranging from 125 to 150 kilometers (Km), depending on the altitude of the Shuttle orbit. This allows for the opportunity to make unique measurements due to the low Reynolds number and high Mach number regimes available. Figure 2.4-1 outlines the concept and the flow regimes available. Based on a minimum altitude of 125 Km, this

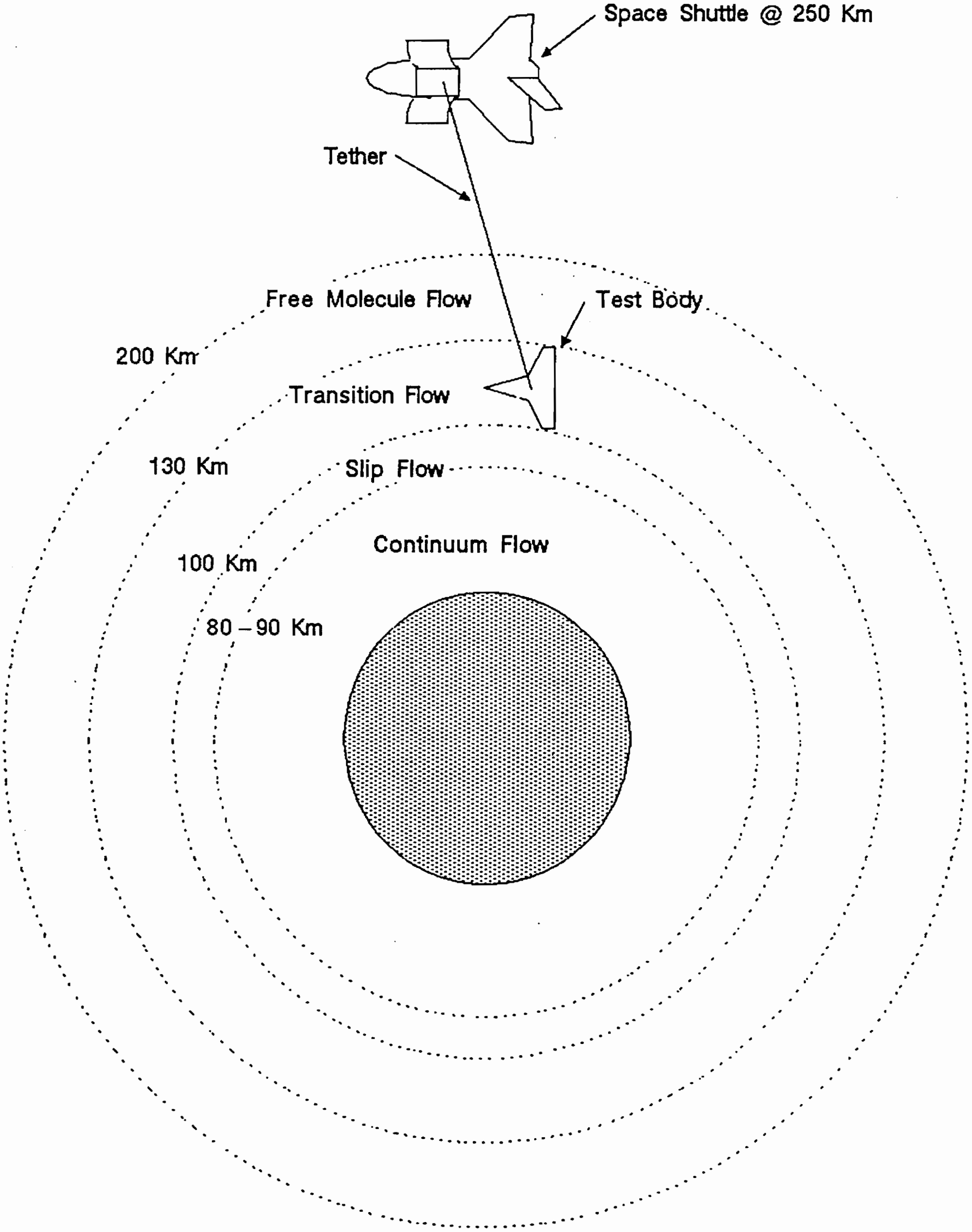


Figure 2.4 – 1 Tether "Facility" Concept

concept could collect data in the transition and free molecule flow regimes. Various models could be towed to collect aerodynamic and aerothermal data. [7]

This concept is in the early development stages. An operational "facility" could be available in the near-term given sufficient funding. The primary issue is the presence of dynamic noise in the tether. Definition and development of the concept are ongoing and are being coordinated with other tether concepts to minimize the cost of overcoming the issues and developing functional tethers. [7]

3.0 Hypersonic Programs

3.1 Introduction

This section describes several hypersonic vehicle development programs and outlines the important performance parameters and mission characteristics that will influence the vehicle's test program. The programs reviewed include:

- National Aerospace Plane (NASP)
- Aeroassisted Orbital Transfer Vehicle (AOTV)
- Planetary Aerocapture/Aeromaneuver Vehicle
- Generic Very High Speed Missile

Each of these programs addresses a set of stressing requirements. A small subset of these requirements are unique to each program while the majority of the requirements and concerns are common to all hypersonic vehicles. The following sections will review each of the aforementioned programs and will present the test parameters of each. In addition, any ongoing work in the area of computer simulation for each program will be briefly discussed.

3.2 National Aerospace Plane (NASP)

The NASP project is developing an experimental hypersonic aircraft designated the X-30 which will be capable of taking off and landing like a traditional aircraft. It will operate in one of two modes; as a hypersonic cruise aircraft, or as a single stage to orbit space launch vehicle. The anticipated flight envelope is shown in Figure 3.2-1. The goal of NASP is to advance the state of hypersonic aircraft technology and to demonstrate the feasibility of an operational hypersonic vehicle.

> Adequate ground test facilities to test components of the X-30 above speeds of Mach 8 for sustained periods do not exist. Thus, the X-30 is being developed as a "flying test bed" to validate the enabling technologies and computational fluid dynamic flight simulations at speeds between Mach 8 and 25. [10]

3.2.1 Planned Performance

As shown in Figure 3.2-1, the NASP vehicle will operate across a broad spectrum of Mach numbers in a rather large flight envelope. It will fly from subsonic to hypersonic (Mach 28) speeds and from ground level to 121,920+ meters in altitude. As a result, the aircraft's

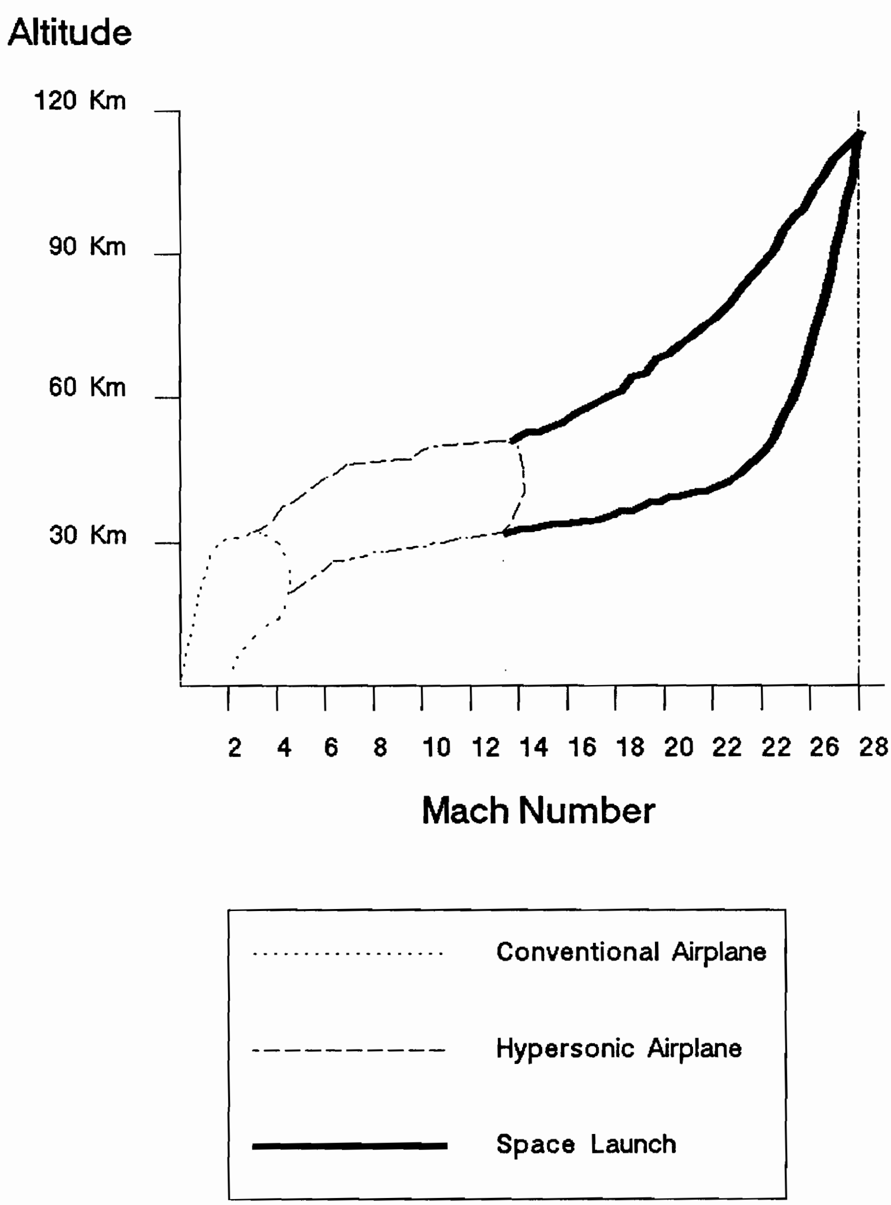


Figure 3.2 – 1

This figure shows the anticipated flight envelope of the National Aerospace Plane as it flies through the conventional aircraft regime, the hypersonic cruise regime, and the single stage to orbit regime.

performance characteristics are diverse. For instance, the NASP nosecone is expected to reach 3000 K while the leading edges of the wings and tail could reach 2200 K. [4,6.] These temperatures, sustained over long (2+ hours) hypersonic flights, will necessitate an active cooling system capable of prolonged and very reliable operation.

To achieve the desired speeds at the desired altitudes, NASP has chosen the integrated engine/airframe design shown in Figure 3.2-2. In this configuration the engine performance is dramatically affected by the forebody design and the engine thrust is affected by the afterbody design. Also, because NASP is an airplane that must be able to operate in the subsonic, supersonic, and hypersonic regimes, the design must include control surfaces able to withstand the stressing operating environment. Successful integration of all of the required components into a fully operational vehicle will require a great deal of aerodynamic analysis and testing. [10]

The propulsion technology required for NASP is considered to be the most critical enabling technology. [10] Because of the speeds desired, a scramjet (supersonic combustion ramjet) must be used. However, existing facilities do not meet the requirements for testing scramjets above Mach 8 for sustained periods. [10]

Figure 3.2 – 2 NASP Design Configuration Proposal

To control the NASP hypersonic vehicle, an advanced avionics system that relies heavily on computerized controls must be developed. It must provide an interface between the pilot and the control surfaces in such a way as to minimize the pilot-sensed differences between the various flight regimes (subsonic, supersonic, and hypersonic). The avionics system required will depend heavily on the aerodynamic design of the vehicle. Much of the aerodynamic test data will be utilized in avionics development and a great deal of testing will be required to validate the avionics system design due to the various forces encountered throughout a typical flight envelope and the extreme conditions involved (temperature, velocity, etc.)

3.2.2 Parameters of Concern

The primary parameters of concern for testing the NASP design include the aerodynamic performance (i.e. lifting and control surfaces,determination of force coefficients, body forces, etc.), the engine performance (coupled with the forebody and afterbody performance), and the surface heating effects. These parameters are interrelated because of the integrated engine/airframe design. Small changes in any portion of the design potentially affect many other components. For example, a change in the nosecone to reduce the maximum temperature encountered may affect the engine inlet performance, thereby reducing the engine thrust and efficiency, the control surfaces, affecting aircraft stability and maneuverability, and the lifting surfaces, affecting lift and therefore payload. The environmental and mission characteristics that affect these parameters include:

- velocity
- density
- temperature
- vehicle material
- flight path

These will be discussed in the following paragraphs.

The velocities of interest range from low subsonic (M = 0.3) to hypersonic (M = 28). The densities encountered throughout the flight envelope span several orders of magnitude from the dense air at sea level to the near-vacuum of 120,000 m. The temperatures anticipated, given these velocities and densities, are in excess of 3000 K. These high temperatures affect the flow over the surface and require some form of active cooling system capable of prolonged operation. In addition, when operating as a hypersonic aircraft, the X-30 will remain in the atmosphere traveling at Mach 8 for two or more hours.

This becomes very stressing and results in very high heat loads over extensive surfaces. Therefore the flight path or mode of operation becomes very important. All of the aforementioned concerns affect the design and performance of all aspects of the X-30. This requires a thorough understanding of the interactions and interrelationships between vehicle features, mission characteristics, and environmental forces.

3.2.3 Planned Test Program

The NASP test program will utilize as many existing test facilities as possible. These include materials facilities, engine test facilities, shock tunnels, and ballistic ranges. However, these facilities have limited capabilities and advanced computational fluid dynamic (CFD) techniques will be used to fill the gaps. [10] These CFD codes will utilize supercomputer centers including the NASA Numerical Aerodynamical Simulator Cray 2 supercomputer facility. [10] The major drawback with using simulations is the lack of actual experimental data that may be used to validate the codes.

The NASP program office is planning to use the capabilities of the Dryden Flight Research Facility to develop the X-30 flight systems, avionics controls, air data systems and sensors, and the flight path and flight pattern simulations. [10] These development efforts will utilize the available aerodynamic test and simulation data to enhance the design confidence.

To test the propulsion system, the NASP project office awarded two contracts in October 1986 for the construction of two Engine Test Facilities. These two facilities are expected to fill the current need for highly capable hypersonic propulsion ground test facilities. The facilities will provide the capability to perform full-scale testing of scramjets up to Mach 8 for sustained periods. [10]

The NASP Joint Project Office (JPO) is making every attempt to utilize existing capabilities to test and develop the X-30. However,

> ... ground test facilities have very limited capability and productivity and are expensive to build. For example, wind tunnels and shock tunnels can only measure the effects of a change in one variable (such as velocity, temperature, or pressure) at a time. Since only one or two tests can be run each day in a wind tunnel, productivity is low. The cost savings of using existing facilities are significant. According to a JPO official, the cost of building a new shock tunnel, for example, could total hundreds of millions of dollars. [10]

The "flying testbed" philosophy has been adopted because of the lack of highly capable, hypersonic test facilities. In addition, because of the long construction times involved, the X-30 would not benefit from a new facility. However, a second generation NASP would benefit from such a facility and it is on that basis that the current NASP program was reviewed.

3.3 Aeroassisted Orbital Transfer Vehicle (AOTV)

The Aeroassisted Orbital Transfer Vehicle (AOTV) is a concept proposed by NASA to utilize the lift and drag forces encountered in a planetary atmosphere to perform orbit transfers. The AOTV mission would include one or more aerobraking passes through the atmosphere to perform orbital plane changes or to reduce the vehicle's orbital velocity. Taking advantage of these atmospheric forces drastically reduces the vehicles fuel requirements. In fact,

> ... the use of aeroassist results in large payload gains.... An aeroassisted vehicle can place approximately twice as much payload into... orbit as an all-propulsive vehicle. Furthermore, aeroassist has been shown to yield similar payoff gains for missions to other planets. [9]

Figure 3.3-1 diagrams a nominal single-entry GEO to LEO AOTV orbital transfer while Figure 3.3-2a shows one proposed AOTV configuration.

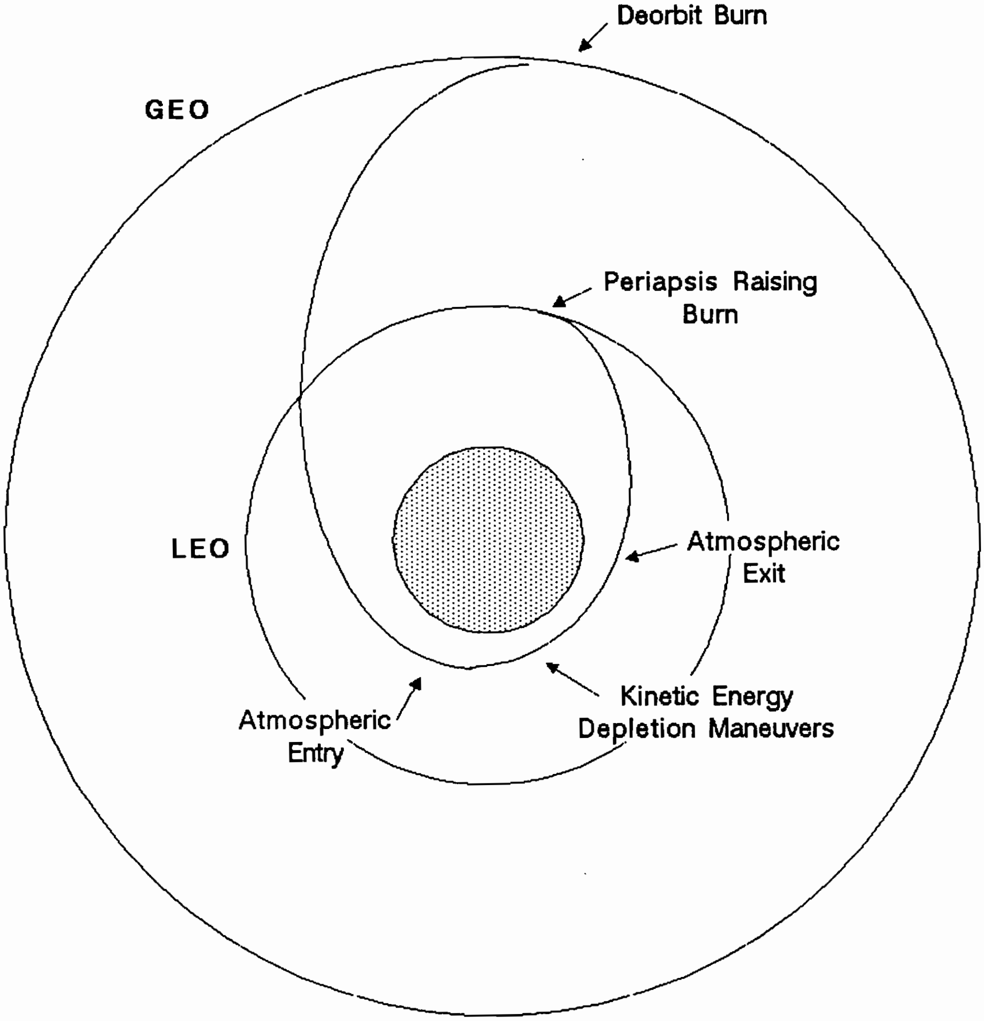


Figure 3.3 – 1 Nominal GEO to LEO AOTV Orbit Transfer

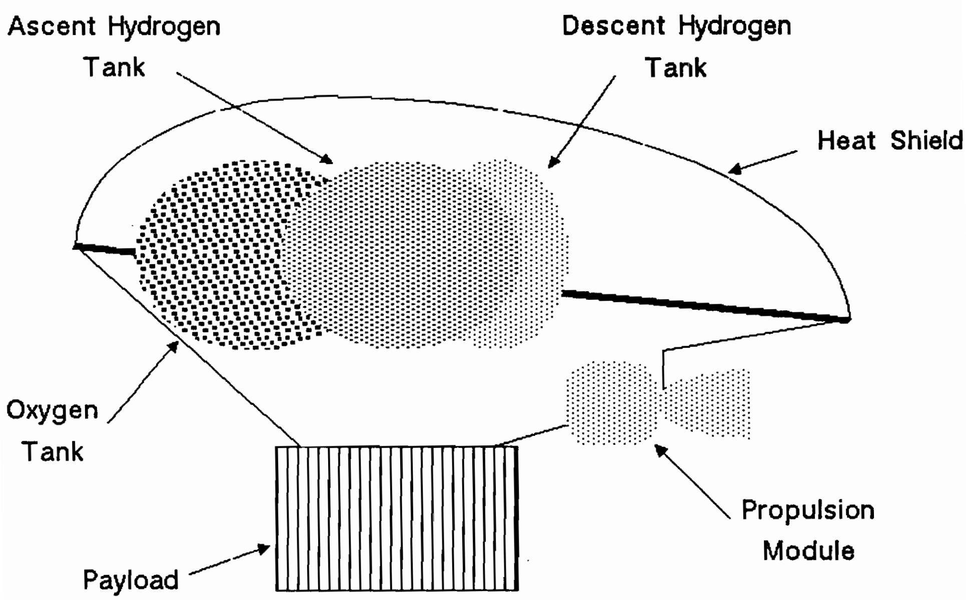


Figure 3.3 – 2(a) AOTV concept proposed by Johnson Space Center

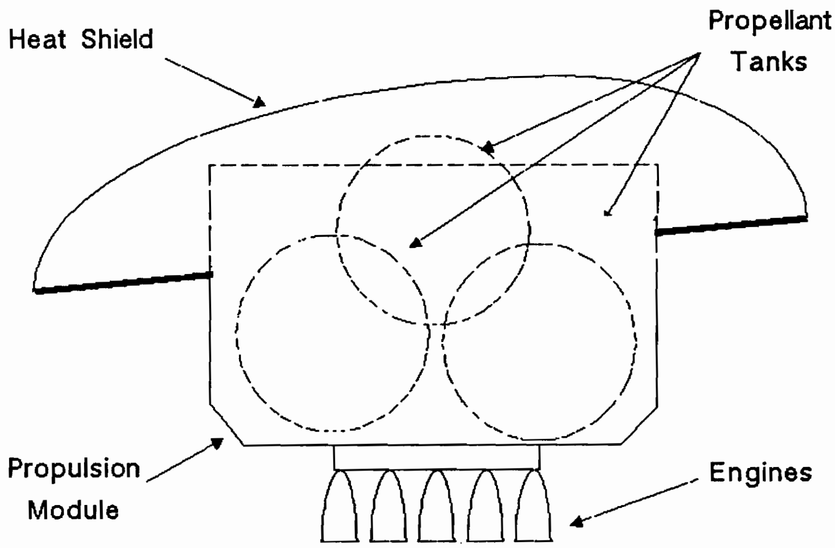


Figure 3.3 – 2(b) Proposed Aeroassist Flight Experiment (AFE) vehicle

3.3.1 Planned Performance

There are two primary concerns with aeroassisted orbital transfer. They are thermal protection and aerodynamic forces. Each of these concerns varies with the aerobraking scheme used. The single-entry scenario requires, in most cases, one fairly deep penetration into the atmosphere, including maneuvers to "scrub" excess velocity, and concluding with a lifting maneuver to exit the atmosphere. The deep penetration is required to "scrub" all of the excess velocity in one pass. It results in higher temperatures and aerodynamic forces.

The multiple entry scenario relies on two or more passes through the atmosphere. By increasing the number of passes, the required velocity decrease per pass is reduced. This reduces the time spent and the distance traveled in the atmosphere (per pass) thereby reducing the heating loads and aerodynamic forces when compared to the single-entry scenario. [9] However, the multiple-entry scenario takes more time to complete.

In either scenario, the temperature and aerodynamic forces are concerns that must be addressed synergistically and simultaneously. There is a strong coupling between the temperature and the aerodynamic forces on the body of the vehicle. For example, to reduce the heat loading for a plane change maneuver by up to a factor of three, the designer maximizes the lift coefficient (C_L) to make a short, quick plane change instead of maximizing the lift to drag ratio (L/D) which gives a long gradual turn. Maximizing lift reduces the total heat transferred dramatically because the body is not in the atmosphere for extended periods but it increases the g loading and the maximum heating rate. Conversely, an increased L/D increases the total heat transferred but decreases the g loading and the maximum heating rate drastically. [9]

Typical AOTV's operate at angles of attack up to 35 degrees. This results in maximum stagnation point temperatures approaching 2800 K and aerodynamic loads of several g's for the design illustrated. These parameters are reduced for the multiple-entry scenario. [9]

3.3.2 Parameters of Concern

The parameters of concern for the AOTV are centered around the thermal protection system. The question of expected heating loads is one not easily answered. The problem lies in the relative velocities expected on an AOTV mission and its affect on the atmosphere. The chemical composition and internal energy of the atmospheric gases are a strong function of velocity. The gases electronic structure can be substantially rearranged into higher energy states. The net effect of which is an increase in the gas temperature near the body and therefore a decrease in the "amount" of thermal energy carried away by the flow. This in turn increases the heating load and therefore the cooling requirements.

Another concern of hypersonic flow is the dissociation of atmospheric gases into chemically reactive components. This generally occurs above Mach 10 for the Earth's atmosphere. When the atmosphere breaks down, the surface of the body can act as a catalyst for a variety of reactions which degrade the surface. This affects the body cooling and the vehicle's aerodynamics. [10] The convective aerodynamic heating associated with these surface catalyst effects are not well characterized and further investigation is necessary. [11]

A third area of concern for AOTV missions is in the area of vehicle aerodynamics. The parameters that must be thoroughly understood and predicted include:

- drag
- lift
- stability
- control thruster plume interactions

A designer must be able to predict, with some confidence, the lift and drag of an AOTV before a mission in order to plan the mission scenario. The stability of a vehicle must also be understood throughout the flight envelope and the potential effects of reaction control thrusters, and their placement, must be examined. Any vehicle limitations must be known before planning a mission to insure the success and the safety of the crew and/or payload.

To summarize, the parameters of primary concern include electron excitation of the atmospheric gases below Mach 10, the dissociation of these gases above Mach 10, and the aerodynamic problems affecting lift, drag, and stability. The first two, namely the electrical excitation and the dissociation of the atmospheric gases, primarily affect the heating of the body. (These also affect the flow field around the body.) Because of the increased heat loads on the vehicle resulting from excitation and dissociation the cooling requirements are more stressing. The last area of concern, namely the aerodynamic forces, affects the body shape, control techniques, and mission planning. Trade-offs must be made between mission requirements and limitations imposed by these severe aerodynamic forces.

3.3.3 Planned Test Program

Several unique testing concerns exist. They include:

- rarefied flowfields
- viscous interactions
- high temperature interactions (electrical excitation and dissociation)
- plume interactions
- wake flows [9]

Several proposed AOTV tests, including simulations, ground tests, and flight tests, address many of these issues.

Simulation of an AOTV flight is a difficult task due to the high temperature effects mentioned above. Electrical excitation and dissociation are difficult to handle because of the uncertainty involved in predicting exactly where each happens. It is also difficult to accurately predict the location of transition from laminar to turbulent flow which affects drag, lift, and body cooling. All of which are important parameters in vehicle design. [11][10] In addition the state-of-the-art nonequilibrium chemical and thermodynamic models are inadequate for the current needs. [11]

Ground testing is difficult because of the limitations of existing facilities. Large uncertainties in aerodynamic performance and heat transfer/loading exist. [11] In addition, the types of low density, rarefied flight conditions encountered in the atmosphere are difficult to simulate. [9] These limitations require the performance of partial simulations that result in further uncertainties. Unfortunately partial simulations are the only means of ground testing available at this time. The AOTV program will perform ground testing, utilizing existing facilities to the extent possible.

The Aerobraking Flight Experiment (AFE) has been proposed as a solution to the problem caused by the lack of high confidence simulations and models and highly capable facilities. The goal will be to collect data in the hypersonic, high-altitude regime of interest. This will be accomplished by transporting a raked-off, blunted elliptic cone test vehicle into space via the Space Shuttle (See Figure 3.3-2b). Included in this package will be a propulsion module that will "lob" the body away from the Earth and push it back into the atmosphere, attaining the desired entry velocity. Temperatures up to 1670 K are expected with an energy flux of 40 W/cm^2. [11]

The AOTV program could benefit greatly from the AFE. However, while the AFE is expected to provide a great deal of data, it is susceptible to failures that could nullify its utility. Therefore, the AOTV program would benefit more from a facility capable of simulating the full range of effects, especially a low density, "clean", "quiet", hypersonic flow.

3.4 Planetary Aerocapture/Aeromaneuver Vehicle

The Planetary Aerocapture/Aeromaneuver Vehicle (PAAV) is similar to the AOTV (Section 3.3). The difference lies in the mission, which translates into a difference in flow parameters encountered. The PAAV is envisioned to be a vehicle that utilizes atmospheric drag to deplete sufficient energy (i.e. kinetic energy or velocity) for an interplanetary vehicle to be captured by the target planet's gravity. The aeromaneuver portion of the mission involves using drag and lift to raise an orbit, lower an orbit, or perform plane change maneuvers. By utilizing these forces, an interplanetary probe can save as much as 50% of the propellant weight necessary for a non-aerobraking vehicle. [4]

The potential exists for using this technique on missions to many of the planets and moons in the solar system including:

- Mars
- Saturn
- Uranus
- Titan

Many aspects of the proposed missions are discussed in the following paragraphs.

3.4.1 Planned Performance

The performance requirements for a PAAV depend greatly on the target planet. The anticipated entry velocities and angles for several planets are shown below:

Planet	Velocity	Angle
Mars	5.36 Km/s	14.5-16.5 deg
Titan	8, 10, 13 Km/s	23.0-33.0 deg
Uranus	30.0 Km/s	8.0-12.0 deg

The varying performance requirements depend not only on entry velocity and angle but on varying atmospheric properties including composition, density, etc. Different gases behave differently so that the design of each PAAV will be similar but unique.

One of the most important subsystems of a PAAV is the thermal protection system (TPS). Heat transfer rates ranging from 250 w/cm^2 to 4100 w/cm^2 are expected. This requires the ability to withstand high temperatures for short periods of time. The corresponding typical time of heating ranges from 150 s to 400 s. [4] There are several types of TPS including ablators and insulators. An example of an ablator is the one used on the Apollo spacecraft. A special layer covered the bottom of the spacecraft and upon reentry ablation occurred thus adding mass to the flow and in effect pushing the boundary layer away from the body. An example of an insulator is the U.S. Space Shuttle. Here, permanent silicon tiles protect the vehicle from the heat of reentry. Both methods offer many benefits including years of operational experience. However, "the long heating times associated with the use of high performance ablators, fall outside the demonstrated capability of current entry vehicle applications and thus need to be experimentally demonstrated." [4]

Another important subsystem is the vehicle's guidance and control system. Flight computers capable of withstanding the shocks of planetary entry are available and are not the problem. Questions arise from the uncertainties involved with the vehicle control surfaces. Because of the high velocities, many extraordinary phenomena occur. For instance, if the velocity is high enough, the atmospheric gases begin to dissociate and a plasma sheet envelopes the windward side of the craft. This affects the control surfaces via heating, extreme forces, and sudden changes in fluid properties. The overall effect on the vehicle must be understood and therefore must be tested.

The mission envisioned for the PAAV will primarily involve high velocities and heat transfer rates. This places several requirements on many subsystems including the TPS and the guidance and control subsystems. The guidance and control system must withstand the high velocities and aerodynamic forces of aerocapture entry. More detailed information is necessary to demonstrate the capabilities of the heat shield and guidance and control subsystems. Acquisition of this data via ground testing and simulation is required.

3.4.2 Parameters of Concern

There are several parameters of concern affecting all aspects of PAAV performance. The primary parameters include:

- velocity
- entry angle
- atmospheric composition
- control forces
- thermal protection requirements

These primary parameters effect the following:

- Reynolds Number (Re)
- heat transfer rates
- time of heating
- Mach number (M)

The secondary parameters not only depend upon the flow but on the PAAV configuration and the atmosphere of interest.

The "dominant flow regime for the aerocapture vehicle is continuum flow with turbulent flow present over most of the vehicle at pull-up." [4] This leads to a transition from laminar flow to turbulent flow during the "flight" [4] which increases the difficulty in modeling the flow arround the vehicle. The Reynolds number encountered ranges from an anticipated low of 10^1 for Mars to a high of 10^8 for Uranus. [4]

The vehicle dependent concerns involve the TPS and the guidance and control system. As stated previously, the TPS will encounter heating rates up to 4100 w/cm^2. Internal heating must be kept to a minimum to protect the on-board avionics, hardware and propellant. This will require some advanced TPS involving either an ablative or an active cooling scheme. The proposed system(s) must undergo rigorous preflight testing to insure the TPS capabilities.

The guidance and control system will be greatly affected by the hypersonic velocity of an aerocapture/ aeromaneuver mission. An important part of the guidance and control system is the flight data sensors. Information about the flight conditions must be collected at all times to provide up-to-date information to the guidance and control system. The sensors used must be able to withstand the high thermal and structural loads associated with a deceleration of this magnitude. Another guidance and control system concern involves control surfaces and thrusters. Each of these typical methods of controlling autonomous vehicles has some difficulties associated with hypersonic flight. In the case of a control surface, the local heating is tremendous as are the aerodynamic forces and effects. For control thrusters, flowfield impingement is a primary concern. The interactions between a control thruster plume and a hypersonic flow are not well understood. Unexpected difficulties could result in sporadic control forces. Before either of these techniques could be used, thorough tests must be performed.

Many of the parameters of concern for PAAVs depend on the target planet and its atmosphere. A wide range of velocities, Re, heat transfer rates, etc. is expected for various solar system planets and moons. These properties in turn affect the crafts heating/cooling and guidance and stability. The effect is so great that these issues must be understood via testing (ground and/or flight) before a truly confident mission could be undertaken.

3.4.3 Planned Test Program

The testing of a PAAV, like the performance requirements, depends on the target planet's atmosphere. Ideally, each vehicle should be tested in a test gas that is optimized for the atmosphere of interest. However, this is prohibitively expensive. Therefore, test gases are selected to simulate certain aspects important to all planetary entries. An example of this is the selection of tetrafluoromethane (CF_4) to simulate high normal-shock density ratios.

> By testing configurations in a gas, like CF_4 which can produce high normal-shock density ratios such as those encountered in hypersonic reentry, certain aspects of real-gas effects can be simulated. [2]

One of the drawbacks of using gases other than the one(s) of interest is that flow tends to differ in other respects. Looking back to the use of CF_4, shocks occurring in a CF_4 flow are much closer to the vehicle than those occurring in air for the same Mach number. [2] This would affect some test results if the target atmosphere was air.

Because of the current facility limitations, PAAVs will be tested in several facilities by performing partial simulations that address a limited number of parameters of concern. In this way the test program will utilize the existing ground test capabilities. In addition, CFD models will be used extensively as a valuable design and engineering tool. Although these models have limitations, they are cost effective and more readily available than test facilities.

The aerocapture/aeromaneuver program would benefit from the AFE (see Section 3.3.3) proposed for the AOTV program. Data collected in the AFE would extend valid regions of the CFD models and would fill in some of the existing hypersonic data gaps. The AFE could also serve as a model for a similar flight experiment designed and performed by the aerocapture/aeromaneuver program.

The PAAV test program, as well as many other hypersonic vehicle test programs, must support a host of technological advances as shown in Figure 2.1-1. These advances must be fully tested and validated to be used with confidence. However, the current ground test and simulation capabilities cannot fully support the needed validation. The existing capabilities must be supplemented by either initiating a flight test program, or by developing a more capable ground test capability.

3.5 Generic Very High Speed Missile

Due to the never-ending cycle of technological advances in todays weapon systems, there has come about a perceived need for a very high speed missile to be used for a variety of missions. These missions include air defense, tactical missile defense, and strategic ballistic missile defense. Many technical challenges face the development of such a hypersonic missile. Concerns about sensor window cooling, vehicle stability and control, etc. require significant resources to fully characterize the problem, and to devise, test, and implement a solution. One of the needed resources is a highly capable hypersonic test facility. (Please note that due to the classification of some/all of the programs addressing the aforementioned missions, this section will outline the performance and describe the important parameters of a single, generic system that would require similar research, development, and testing.)

3.5.1 Planned Performance

The nature of the missions listed above require the operation of an hypersonic missile throughout the Earth's atmosphere for several minutes. This presents a broad range of flight conditions that includes hypersonic flight in the thick, dense sea level atmosphere to the rarefied gas, free-molecule flow regime of the extreme upper atmosphere at Mach numbers in the range from 8 to 15. Because of the severe heating effects associated with these speeds, and the plasma sheet generated by ionization, some form of mass additive or ablative cooling system is required. This is complicated by the need to "see" through the plasma sheet, the added mass, or the ablated material both for communication and to utilize the missile's sensor.

All types of sensors or communication devices must operate through some form of "window" that is transparent to the electromagnetic energy used by the unit, yet still protects the fragile components of the unit. An example is the special radome material used on conventional aircraft to cover the on-board radar unit. The radome is transparent to radar waves while being solid enough to protect the radar unit. This window must be protected from the severe heating encountered. In addition, the radiation from the ionized atmosphere is likely to affect the electromagnetic radiation used by the sensor and communication systems. Characterization of "seeing" and communicating through a plasma sheet is incomplete due to the lack of experimental and operational data from ground facilities, flight tests, and operational flights. [10]

Coupled with traveling at high Mach numbers is the need to maneuver quickly. A missile must be capable of performing multiple g turns in the last seconds, or endgame, of an engagement. This requires the use of some form of attitude control jet. Aerodynamic control surfaces can not be used because of the need to operate in the upper atmosphere (>40 Km) where they will have little or no effect.

3.5.2 Parameters of Concern

The primary concerns are the effects of aerodynamic heating on the sensor and communication systems. In the several minute flight of an hypersonic missile, the nosecone temperature will reach several thousand Kelvin. This may require a mass additive or ablative thermal protection system because of the need to add mass to the flow at the surface to, in effect, push the thermal boundary layer away from the body.

However, the combination of this added mass and the dissociation and ionization effects at speeds greater than Mach 10 will complicate the operation of the missile sensor and communication systems. The radiation from the atmospheric gases will inhibit the transmission of radio waves (as seen by the Apollo and Space Shuttle missions). It will also affect the functionability of the sensor. Extraneous noise will be introduced in certain spectral bands and other bands will be completely blocked. Techniques to either cool the flow over a portion of the body or redirect the plasma sheet must be developed and used in conjunction with the careful choice of spectral bands to be utilized.

Another concern for a missile traveling at these speeds is body stability and control. While missile flight through the atmosphere is fairly well understood, maneuvering at hypersonic speeds with some form of mass addition thermal protection system is not as well understood. The forces generated on a quickly turning body in a Mach 10 flow have some level of uncertainty due to the localized dissociation/ionization effects resulting from differing flow conditions over the body. In addition, the effects of attitude control jet plumes and the mass addition from the TPS on the already severe flowfield have not been fully addressed.

3.5.3 Planned Test Program

The current test program utilizes existing capabilities in unique and innovative ways. Aerodynamic, aerothermal, and propulsion facilities will be used to the maximum extent possible and practical. These test will likely utilize facilities at Arnold Engineering Development Center (AEDC), NASA Ames Research Center, and several contractor-built test facilities. All of the ground testing will be closely coupled to intensive CFD efforts utilizing the most up-to-date techniques available.

Even with the ground test and simulation effort outlined above, an extensive flight test program is necessary. The existing capabilities do not satisfy the needs of a hypersonic missile test program. The flight tests are necessary to fully validate the TPS concepts and the stability and control techniques. It should be pointed out that the flight test program has objectives other than verifying the TPS and control system and indeed much of the flight program addresses these other needs. However, a reduced flight test program would still be required to fully validate the proposed TPS and control concepts.

4.0 Comparison of Capabilities and Requirements

4.1 Introduction

This section will compare the requirements of the programs discussed in Section 3.0 with the capabilities of existing hypersonic test facilities. The purpose of this section is to determine if the existing hypersonic testing capabilities satisfy the hypersonic program requirements. The result will be the identification of the test facility shortfalls for the projected hypersonic program needs.

4.2 Projected Hypersonic Test Requirements

The programs reviewed for this study were discussed in Section 3.0. They include:

- National Aerospace Plane (NASP) - Section 3.2
- Aeroassisted Orbital Transfer Vehicle (AOTV) - Section 3.3
- Planetary Aerocapture/Aeromaneuver Vehicle (PAAV) - Section 3.4
- Generic Very High Speed Missile - Section 3.5

In order to determine the generic, nationwide facility shortfalls, a generic, program-wide set of requirements must be developed. These requirements must include all requirements regardless of their uniqueness.

The first parameter of interest is the Mach number. Table 4.2-1 summarizes the expected operating ranges for several parameters including the Mach number. Figure 4.2-1 provides a graphical presentation of the vehicles' expected operating ranges. As can be seen, the Mach number range for the programs reviewed spans from the subsonic, through the supersonic, and well into the hypersonic regimes. Because of the topic of this document, the Mach number range from 5 to over 53 is the range of interest.

The next parameter of interest is the Reynolds number. Based on the projected dimensions of the vehicles under study, their velocities, and their operating environments, the expected Reynolds number range is 8.7×10^0 to 1.5×10^9, as shown in Table 4.2-1. This range becomes very important in scaling test models and comparing test results. It should be noted that a desire in performing wind tunnel tests is to match the Reynolds number of the test to the expected flight Reynolds number.

Table 4.2 – 1 Program Requirements Summary

	Mach #	Re (x 10^6)	Max Temp (K)	Gas
NASP	0 – 28	0.009 – 1500.0	3000	Air
AOTV	> 10	0.000015 – 2.0	3500	Air
PAAV	20 – 53	0.0000087 – 140.0	3500	CO_2, NH_3, N_2, CH_4
Missile	8 – 15	0.0001 – 60.0	2000	Air
Summary	0 – 53	0.0000087 – 1500.0	< 3500	Air, NH_3, N_2, CH_4, CO_2

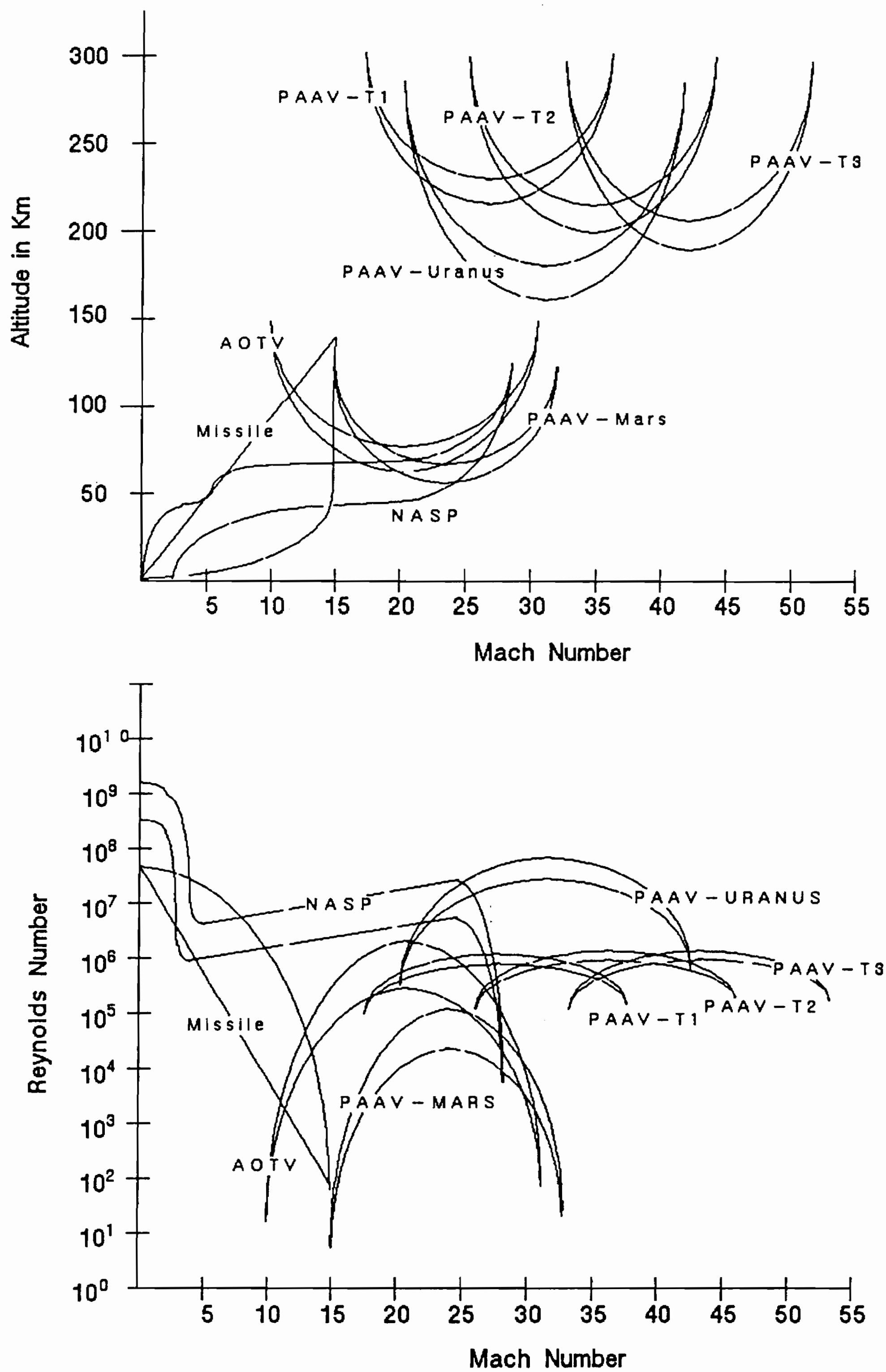


Figure 4.2 – 1 Vehicle Flight Regimes

This provides a means of comparing tests or flows that appear dissimilar but actually involve the same flow processes and interactions. It is sometimes difficult to match Re. One solution is to alter some of the flow parameters (velocity, density, etc.) to match or closely match the Reynolds number.

The model size and its relation to the Reynolds number becomes a significant issue. For instance, to test a one half scale model of a vehicle designed to operate at Mach 8 requires a velocity corresponding to Mach 16. However, higher Mach number tunnels generally have small test sections and models smaller than one half scale are required. This in turn increases the velocity requirement, leading to the need for more capable facilities.

To overcome this problem, alternative test gases are used and the velocity and model size are adjusted to match the Reynolds number. This leads to another problem, namely the modification of the flowfield and the improper simulation of potential "real gas" effects such as dissociation and ionization. The flowfield alterations severely affect the measurement of aerothermal heating effects. Temperatures as high as 3500 K are expected in the desired flight regimes. Temperatures that high affect the composition and characteristics of the surrounding flowfield. Inaccurate results are obtained by using gases other than the expected operating gas because of the differing qualities of different gases. In addition, temperatures this high are not possible in many types of facilities because of material constraints and real gas effects. Proper simulation of real gas effects is important because at the expected temperatures, the vehicle's thermal protection system is severely stressed. The problem can be alleviated by using one of the more exotic wind tunnel types, for example a hotshot tunnel where temperatures as high as 3900 K are possible. This however leads to very short run times and small models.

Another concern of hypersonic testing is the measurement of aerodynamic forces. Detailed knowledge of a vehicles reaction to changing attitude is necessary for guidance and control algorithm development. In addition, the effects of control surface deflections must be understood, in terms of guidance and control, heating, and loading. Sensitive multi-degree of freedom force and moment balances are needed as is the ability to change the vehicles pitch and yaw over $\pm$ 10s of degrees.

4.3 Existing Capabilities

There are numerous hypersonic test facilities in the United States. The majority are government owned facilities, but there are numerous privately owned facilities. Most of the existing facilities were built in the 60s, during the last rise in interest in hypersonic testing. And while many of these facilities have been upgraded in the last 15 years, they still lack some necessary capabilities.

The primary drawbacks of the existing facilities are the inability to simultaneously simulate all of the necessary flow parameters and the limited test section size. The inability to simulate the necessary parameters results in the need to test the same configuration in a variety of tunnels and to try to correlate the data. This reduces the opportunity to identify and to collect data on parameters that are coupled. The limited test section size often makes it necessary to use small models that cause the problems discussed in Section 4.2. It also limits the ability to fully test integrated vehicle designs like the NASP (i.e. testing the scramjet inlet design as it is coupled to the forebody).

Appendix A contains a summary of the capabilities of some of the more capable facilities available in the U.S. These capabilities will be briefly discussed below. Table 4.3-1 provides a summary of the capabilities presented in Appendix A and Figure 4.3-1 presents a graphical summary of all existing hypersonic facilities.

Table 4.3 – 1 Summary of Appendix A Capabilities

Section Size (m)	Mach	Re (x 10^6)	Temp Range (K)	Gas	Run Time (s)	Number of Facilities
< 0.61	5 – 22.2	0.17 – 7000	Amb – 1950	H_2, Air Freon 14, N_2	7 – 300 Continuous	6
0.61 – 1.22	5.5 – 40	0.001 – 75	470 – 6390	Air, N_2, H_2	< < 1 – 240 Continuous	9
1.22 – 2.44	4 – 10	0.3 – 4.7	390 – 1250	Air		1
> 2.44	4 – 18	0.06 – 22	1090 – 2033	CH_4 & Air, N_2, H_2	< < 1 – 240	4
Summary	4 – 40	0.001 – 7000	Amb – 6390	Air, H_2, N_2, Freon 14, CH_4 & Air	< < 1 – 300 Continuous	20

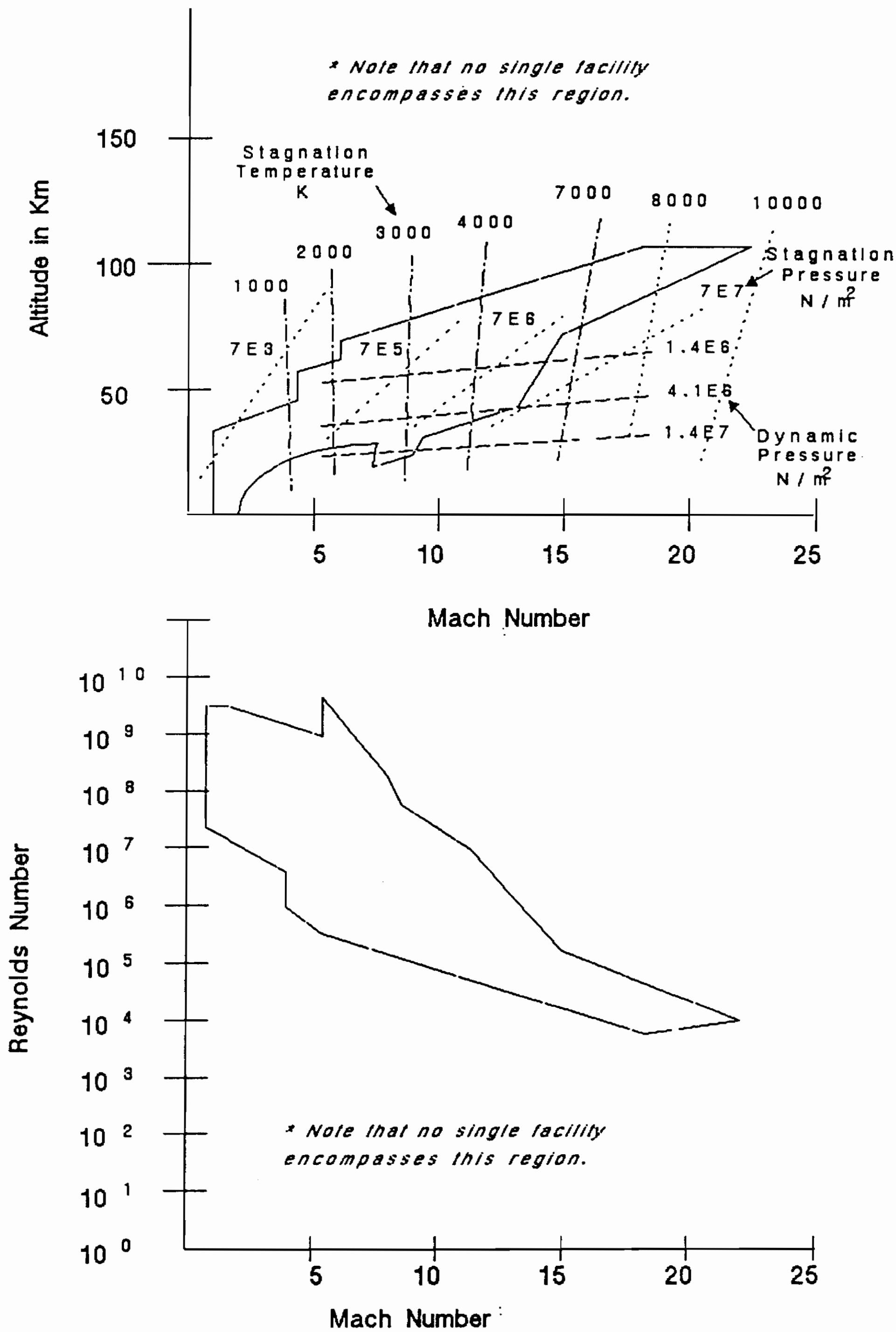
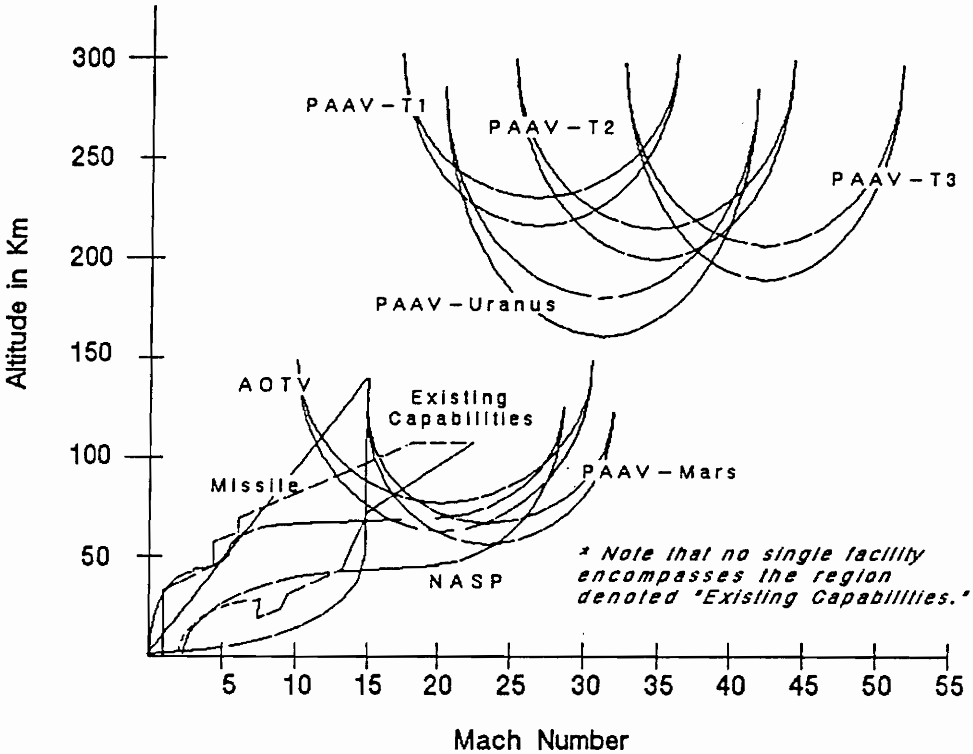


Figure 4.3 – 1 Summary of Current Ground Test Capabilities

It is evident from Table 4.3-1 that the majority of the existing facilities possess relatively small test sections. This characteristic is one of the primary drawbacks of the existing hypersonic facilities because of the resulting difficulty in matching Re as is demonstrated in Figure 4.3-2. Another characteristic apparent from Table 4.3-1 is that as the test section size increases, the other important parameters tend to decrease. This presents an interesting trade-off. Either small models can be tested in more capable facilities or large models in less capable facilities. This relationship is not surprising because when technology and resource limitations are imposed on the design and construction of any item, trade-offs must be performed. The limitations entering the picture here are primarily driven by inadequate resources. The trade-off is one between test section size and mass flow (and its relation to velocity). A large test section requires a greater mass flow to maintain a given Mach number when compared to a smaller test section. An increased mass flow requirement translates into increased demands on the flow source, whether it is a pressurized tank, a vacuum tank, a combination of both, a shock tube, etc. The increased demand translates into increased cost because of the need for more capable equipment. Therefore, the capabilities of some facilities are limited because of the limited funding available when they were built.



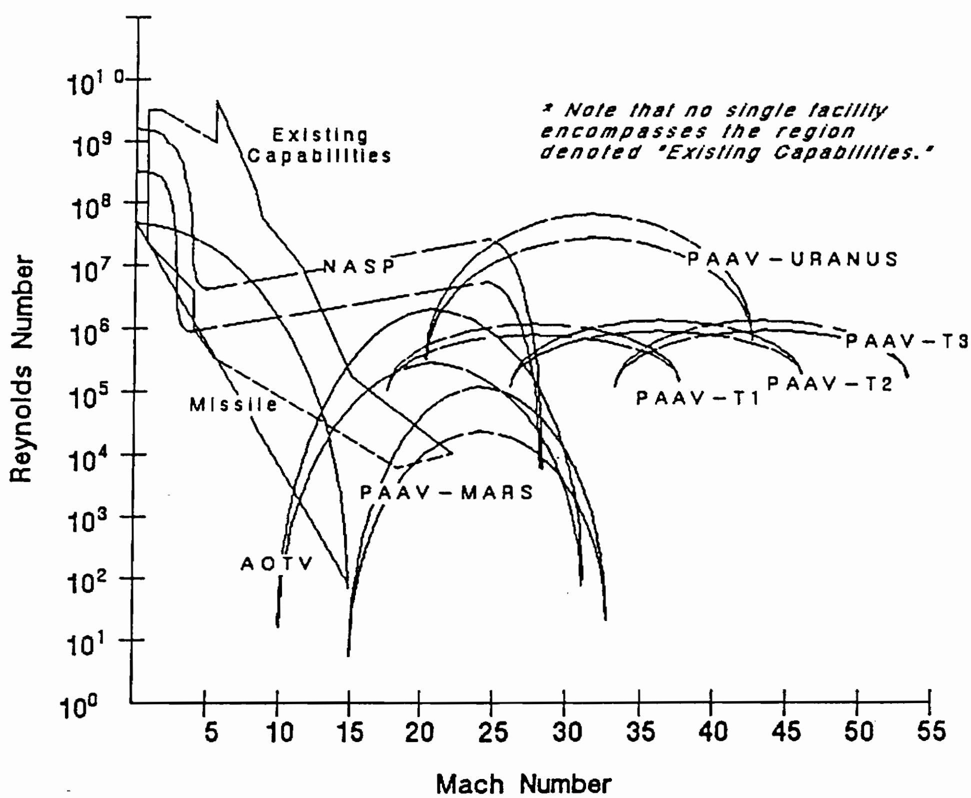


Figure 4.3 – 2 Existing Capabilities Vs. Program Requirements

The information in Appendix A provides evidence that although some of these facilities have been upgraded as recently as 1985, all of them were originally built in the late 50s and early 60s. That means that the majority of the existing hypersonic facilities are based on 30 year old technology. That fact in itself provides an impetus to construct a new facility incorporating the latest technology. This will serve to enhance the capability as well as the safety and reliability of the U.S. hypersonic capability.

4.4 Capability Shortfalls

A comparison of Table 4.2-1 and Table 4.3-1 will result in the identification of the projected hypersonic test facility shortfalls. These shortfalls are based on the hypersonic vehicle development programs discussed in Section 3.0 and the capabilities presented in Appendix A. While these programs and facilities are limited in number, they represent a reasonable cross-section of requirements and the most capable facilities. [3]

It is important to emphasize that the capabilities presented in Table 4.3-1 are summarized by test section size range. All of these capabilities do <u>not</u> exist at one facility. As was stated previously, there is currently no single facility that simultaneously simulates all of the characteristics of hypersonic flight. Test programs generally utilize tests in several facilities.

When comparing the summary lines of Tables 4.2-1 and 4.3-1, it is evident that for a few of the selected parameters, the capabilities satisfy, or nearly satisfy, the requirements. The required Reynolds number range is satisfied (1.5×10^9 required, 7.0×10^9 available). The available temperature capabilities exceed the requirements (3500 K needed, 6390 K available), and the Mach number range required is greater than the capabilities available (53 required, 40 available). However, these comparisons are somewhat deceiving.

Based on the comparison of the selected parameters, one would be lead to believe that the current capability is nearly sufficient and therefore does not require much enhancement. However, some important considerations are not addressed by these parameters. First, as has been previously discussed, the existing facilities cannot simulate all or even most of the important parameters simultaneously. This precludes accurate collection of some data because of coupling effects between flow parameters.

Another consideration not addressed by the selected parameters is the test section size. A small test section requires a small model and small, detailed, instrumented models are difficult to build. Effects of forebody components affect the performance of and flow around afterbody components. For example, the current NASP configuration would require a very detailed model capable of some form of injection in the scramjet so that the scramjet exhaust effects could be accurately determined. It is this type of vehicle component coupling that places severe demands on test facilities and model builders. Small models are increasingly difficult to build and test as coupling effects increase. Therefore, test section size is an important consideration and maximizing this characteristic is very desirable.

Still another important consideration is the test run time. As is apparent from Appendix A, most of the existing facilities have very short run times (many not more that a few seconds). This severly limits the ability to perform many types of tests.

The availability of the existing capabilities is also of great concern. As you can see from Appendix A, many of the most capable facilities are on standby status and would require time and money to be reactivated. Many more facilities, not shown in Appendix A, are also on standby status and are essentially unavailable. Of the facilities that are still active, most have one to two year backlogs, making them virtually unavailable.

Considering these facts, namely the current inability to simultaneously simulate all or most of the parameters important for hypersonic testing, the severely limited test section size, the limited availability of capabilities, and the age of the existing capabilities (primarily late 50s and early 60s vintage), a logical conclusion supports the development and construction of a new, highly capable facility. This facility should have a large test section and should be able to match several of the important hypersonic flow parameters. Also, a run-time on the order of minutes to 10s of minutes is desirable. This would allow prolonged operation for dynamic testing, heat transfer testing, etc.. The following section will outline the plan for the facility proposed by this document. This facility proposal is intended to satisfy as many of the program requirements identified in the preceding sections.

5.0 Needed Facility Parameters

5.1 Introduction

The purpose of this section is to outline a proposed highly capable facility that would greatly enhance the U.S. hypersonic ground test capability. The discussion of the facility proposal will include a description of the analysis performed and a presentation of the results of this analysis. The facility hardware (nozzle size, tankage, etc.) and the capabilities provided (Mach number, mass flow, etc.) will be discussed.

Based on the qualities of the wind tunnel types presented in Section 2.0, the pressure-vacuum wind tunnel concept with constant mass flow was selected for this proposal. The pressure-vacuum tunnel provides the greatest flexibility in terms of operating times and pressures, test section size, Mach number, etc. and allows for larger pressure ratios across the tunnel; an important parameter for very high speed flows. The following sections describe the efforts undertaken to design a highly capable pressure-vacuum wind tunnel. Included in this section is a brief discussion of the estimated cost and time required to build the proposed facility.

5.2 Analysis

The objective of this analysis was to outline a facility that could meet all, or most, of the program requirements. Through the course of the analysis, many trade-offs were performed. The driving force in making the decisions discussed below was the desire to maximize the overall facility capability, not to maximize any single parameter. The result of this "philosophy" is a facility that does not satisfy all of the program requirements but greatly extends the current test capability. It was recognized early in the analysis that all of the requirements may not be satisfied by one facility due to the variety of gases required and the extreme Reynolds and Mach numbers required.

At the outset of the analysis, a set of design goals was formulated. These goals were intended to satisfy most of the requirements and to provide a starting point for the design. These goals include:

- Air as a test gas
- Test section with a 3 m width
- Mach number range from 5 to 25
- Reynolds number range from 10^3 to 10^8
- Axisymmetric nozzle
- Temperatures approaching 3500 K (limited by real gas effects)
- Avoidance of low-density flow, and
- the ability to perform runs of up to 1 minute

Once these were established, one goal was selected as the fixed design driver. In this analysis, the 3 m wide test section was selected. Two cases were investigated, namely the 3 m x 3 m square test section and the 3 m diameter circular test section. This established the tunnel throat size as a function of the Mach number via equation 5-1.

5-1 $$\frac{A}{A^*} = \frac{1}{M}\left[\frac{2}{(\gamma+1)}\left(1+\frac{(\gamma-1)}{2}M^2\right)\right]^{\frac{(\gamma+1)}{2(\gamma-1)}}$$

A = Area
A* = Throat Area
M = Mach Number
γ = Ratio of Specific Heats

Next, a series of Mach numbers were selected (5, 8, 11, 16, 20, 25) in the range of interest. A review of literature suggested that the maximum total temperature allowable to avoid significant dissociation of the test gas was in the 2200 K to 2400 K range. Temperatures in this range result in dissociation of approximately 1% of the O_2 present [12]. This is generally acceptable because some degree of recombination occurs between the heater and the test section. Initially a temperature of 2300 K was selected. This was later increased to 2350 K to allow Mach 25 flow without liquefaction (discussed later in this section). A temperature of 2350 K does not satisfy the requirement, or goal, of temperatures to 3500 K. Heating the test gas to 3500 K, however, results in significant dissociation of O_2. Recombination to an acceptable level of dissociation in this case would not occur and high temperature testing is left for those specialized high temperature tunnels. The selected values (i.e. T_t, M), combined with the fixed test section size and the selection of air as a test gas, become the baseline flow parameters.

The next step involved characterizing the mass flow through the test section as a function of the total pressure. The aforementioned parameters were treated as constants. This resulted in equation equation 5-2.

5-2 $$\dot{m} = AMP\left(\frac{\gamma}{RT}\right)^{1/2}\left[1+\frac{[\gamma-1]}{2}M^2\right]^{\frac{-[\gamma+1]}{2(\gamma-1)}}$$

$\dot{m}$ = Mass Flow Rate
P = Stagnation Pressure
R = Gas Constant
T = Stagnation Temperature

Equation 5-2 represents the mass flow (and indirectly the density) in the test section as a function of the baseline parameters and the total pressure. In this way, densities can be determined for each baseline flow condition and a density equivalent altitude could be found. The density equivalent altitude is defined here to mean the altitude in a standard U.S. atmosphere at which the atmospheric density equals the test section density. This is important in that it provides another way of comparing the facility capability with the program requirements. In addition, the Reynolds number can be calculated for each baseline flow condition. A concurrent calculation performed after each variation in total pressure and therefore mass flow was the calculation of the mass flow at the throat given the same conditions using equation 5-3.

5-3 $$\dot{m}^* = \frac{\Gamma P A^*}{[\gamma R T]^{1/2}}$$

$\dot{m}^*$ = Mass Flow at Throat

$$\Gamma = \gamma [1 + \frac{(\gamma-1)}{2}]^{\frac{-(\gamma+1)}{2(\gamma-1)}}$$

This calculation was performed to insure that the mass flow through the test section is less than or equal to the maximum mass flow throught the throat.

Once some of the important flow parameters were calculated (density, total pressure, Mach number, temperature) another check was performed. This check determined if the calculated and selected flow parameter would result in the liquefaction of the test gas. (This lead to the identification of the lower bound for several of the final parameter ranges.) The check performed involved several steps. First, the graphs presented in Figure 5.2-1 were developed. (Note that Figure 5.2-1b only includes the lower range of total temperatures. This was

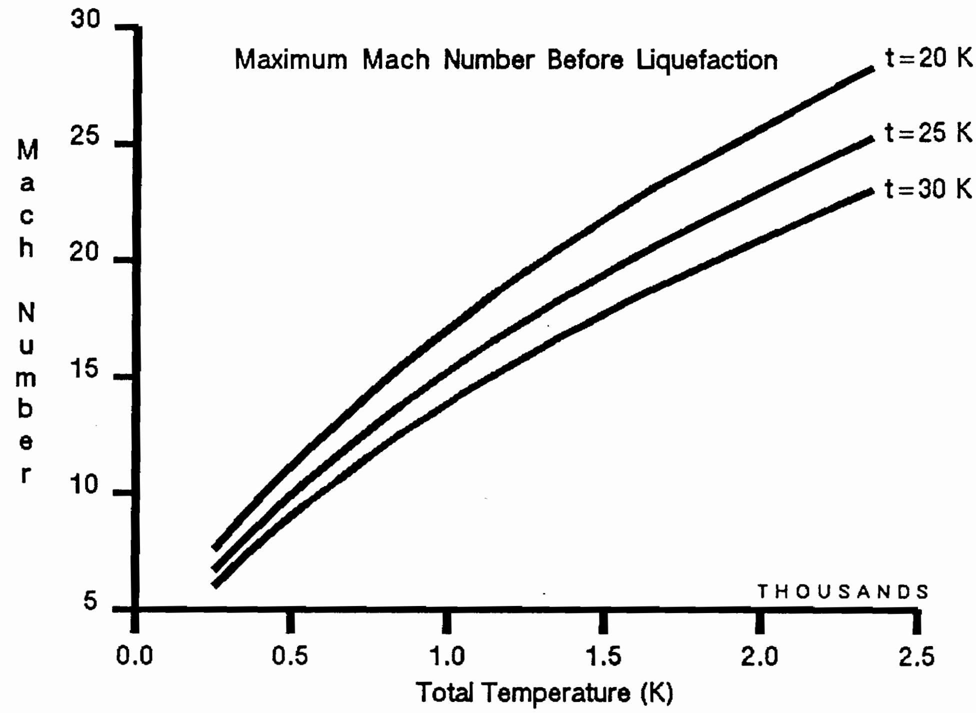


Figure 5.2 – 1a Total Temperature Vs. Mach Number

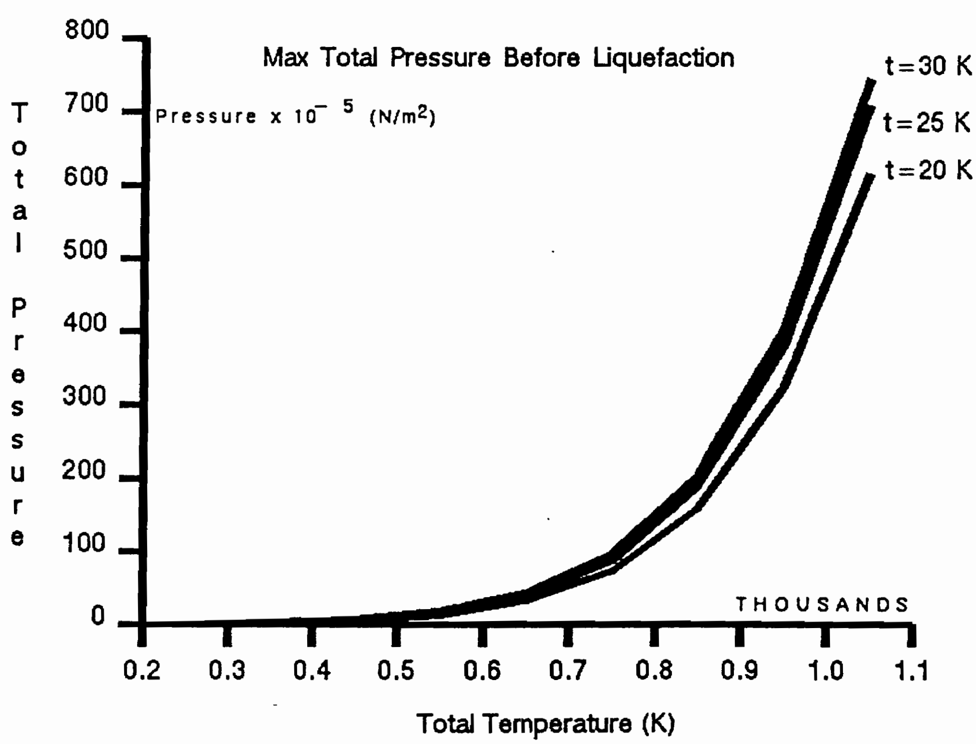


Figure 5.2 – 1b Total Temperature Vs. Total Pressure

done to enhance the detail of the plot and has no ill effects because the pressure increases exponentially and at 1650 K is in excess of 13,000 atmospheres.) The families of curves shown define the maximum Mach number and total pressure allowable given the test chamber static temperatures associated with each curve. Data used to generate these curves are found in Reference 1 and define the static temperatures and pressures at which liquefaction occurs. The maximum Mach number can then be found using equation 5-4 and the static temperature, the total temperature, and the corresponding ratio of specific heats (gamma).

5-4 $$M_{max} = \sqrt{\left(\frac{T}{t} - 1\right)\frac{2}{(\gamma-1)}}$$

t = Static Temperature

Once the Mach number is found, the total pressure can be found using the static pressure and equation 5-5.

5-5 $$P_1 = p_1 \left(1 + \frac{(\gamma-1)}{2} M^2\right)^{\frac{\gamma}{(\gamma-1)}}$$

p = Static Pressure

This process can be done for any combination of total and static temperatures. A static temperature of 25 K was selected as the minimum static temperature allowable.

It was after performing these calculations that the initial limit of 2300 K was raised to 2350 K as noted previously. You will notice that at lower total temperatures, the total pressure limit is fairly small when compared to the sea level air pressure. This constrains the total pressure limit at lower temperatures. Also note that the larger total temperatures correspond to extremely large total pressures that in no way limit the tunnel operating pressure.

After reviewing the current capabilities for storing high pressure gases and reviewing the resultant flow parameters for various pressure selections, a maximum tunnel operating pressure was determined. The selection of the maximum total pressure allowed for the definition of many other tunnel parameters including the mass flows, the densities, the density equivalent altitudes, the Reynolds numbers (based on test section size), the pressure tank size and the vacuum tank size.

The tankage sizing was performed by determining the flow conditions that would most stress the tankage capabilities ability to maintain the necesary pressure ratio. A maximum run time, at the most stressing conditions, of 1 minute was selected. This corresponds to one of the initial goals established at the outset of the tunnel design. This run time is intended to provide test durations adequate for any type of test. Run times of less than 1 minute are possible. The 1 minute limit only holds under the most stressing conditions. The tank (pressure and vacuum) capacities were determined based on mass flow, the desired maximum run time, and the pressure ratio (including the pressure loss due to the heater, etc.) required to maintain the desired Mach numbers. Once the tankage was sized, the run times for the remaining flows were determined.

Data from Reference 1, and extrapolations from those data, lead to the determination of the following pressure ratios for the Mach numbers of interest:

Mach 5	-	15
Mach 8	-	70
Mach 11	-	280
Mach 16	-	1000
Mach 20	-	2500
Mach 25	-	3500

These pressure ratios must be maintained for the duration of the test run. Once the pressure ratio across the tunnel drops below the values stated above, the Mach number of the flow drops and the test is terminated.

Once all of the tools were in place, the tunnel hardware and capabilities were developed. It should be noted that isentropic flow is assumed in the test section and upstream of the test section. Also, losses due to friction were neglected because this is a first order tunnel design.

5.3 Description of Facility and Capabilities

This section will outline the proposed facility and its capabilities based on the analysis discussed in the previous section. Figure 5.3-1 presents a sketch of the proposed facility. Tables 5.3-1 through 5.3-4 display the facility parameters and facility capabilities. These figures and tables will be discussed more fully in the following paragraphs.

Beginning with the storage tanks in Figure 5.3-1, the analysis resulted in the selection of a maximum pressure of $7.8\text{x}10^7$ N/m^2 and a volume of at least 60,300 m^3. These values are based upon the selection of a maximum tunnel operation pressure of $6.895\text{x}10^7$ N/m^2 and a maximum run time of at least 1 minute for any set of parameters. It was determined that the most stressing conditions from the standpoint of the pressure tanks occur for a Mach 5 flow at the maximum pressure. From Table

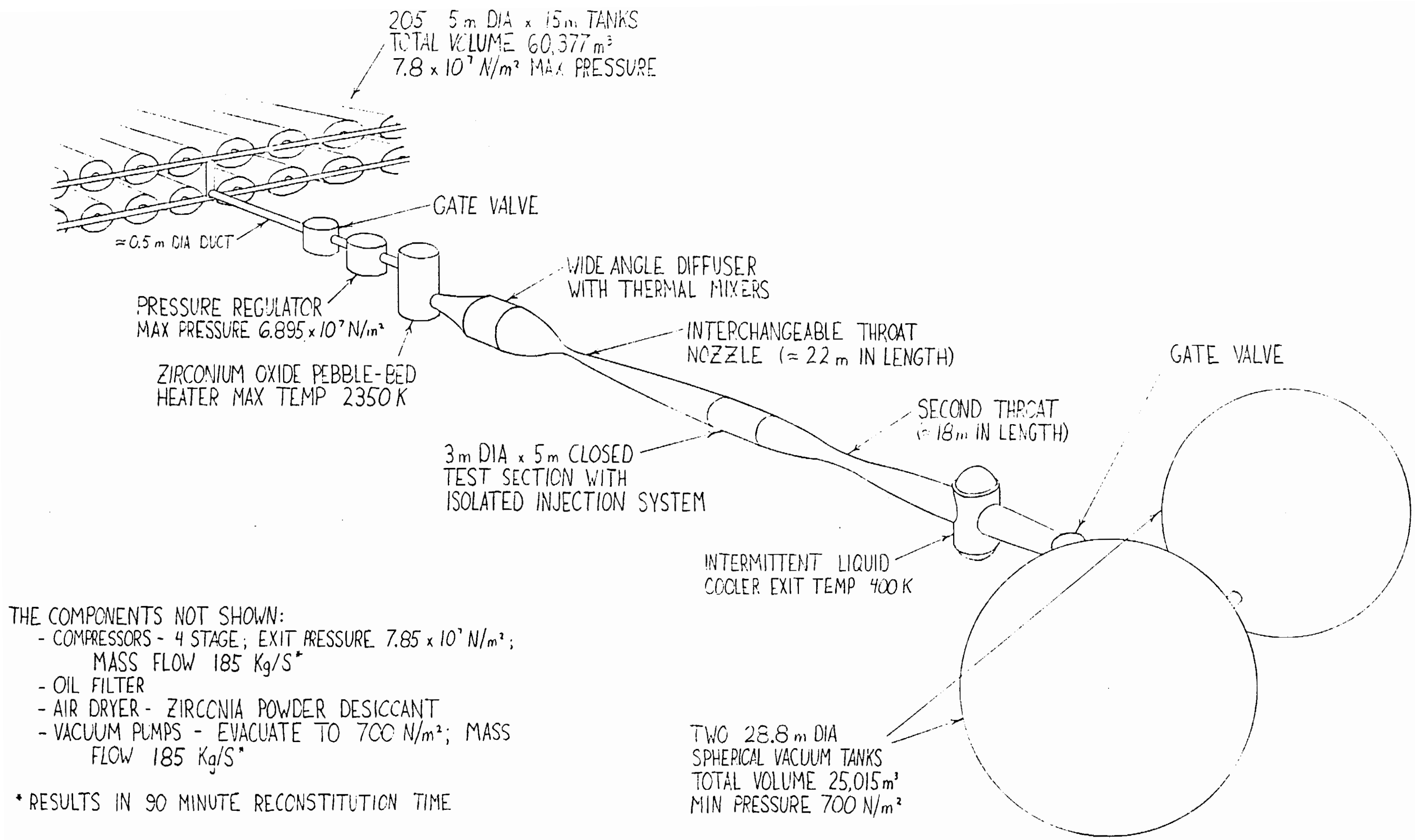


Figure 5.3 — 1 Proposed Hypersonic Wind Tunnel

Table 5.3 – 1 Proposed Tunnel Capabilities for T = 2350 K

Mach Number	Mass Flow (kg/s)	Throat Area (m^2)	Density (kg/m^3)	Altitude (m)	Reynolds Number	Run Time (min)
5	8064.9	1.4E – 1	2.5E – 1	13200	6.2E7	2.07
8	557.8	1.0E – 2	1.1E – 2	33100	4.1E6	6.41
11	75.53	1.4E – 3	1.3E – 3	47830	5.8E5	11.84
16	6.46	1.2E – 4	6.1E – 5	71920	4.5E4	38.75
20	1.45	2.6E – 5	1.2E – 5	82460	1.1E4	69.06
25	0.32	5.8E – 6	2.0E – 6	92990	1.2E3	222.15

Table 5.3 – 2 Proposed Tunnel Capabilities for T = 1670 K

Mach Number	Mass Flow (kg/s)	Throat Area (m^2)	Density (kg/m^3)	Altitude (m)	Reynolds Number	Run Time (min)
5	13512.3	1.6E−1	4.8E−1	5880	9.5E7	1.24
8	1314.7	1.2E−2	2.9E−2	26850	9.4E6	2.72
11	234.0	1.8E−3	3.8E−3	40110	1.5E6	3.82
16	28.4	1.8E−4	3.2E−4	59460	1.7E5	8.82
20	7.9	4.2E−5	7.1E−5	70850	4.5E4	12.68
25	N/A	N/A	N/A	N/A	N/A	N/A

N/A = Mach Number Not Achievable at this temperature

Table 5.3 – 3 Proposed Tunnel Capabilities for T = 1030 K

Mach Number	Mass Flow (kg/s)	Throat Area (m^2)	Density (kg/m^3)	Altitude (m)	Reynolds Number	Run Time (min)
5	16591.6	1.9E – 1	7.5E – 1	4520	1.1E8	1.01
8	1541.7	1.8E – 2	4.3E – 2	24340	1.1E7	2.32
11	264.5	3.1E – 3	5.4E – 3	37730	1.8E6	3.38
16	N/A	N/A	N/A	N/A	N/A	N/A
20	N/A	N/A	N/A	N/A	N/A	N/A
25	N/A	N/A	N/A	N/A	N/A	N/A

N/A = Mach Number Not Achievable at this temperature

Table 5.3 – 4 Proposed Tunnel Capabilities for Minimum Temperature

Mach Number	Minimum Temp	Maximum Pressure @T min	Mass Flow (kg/s)	Throat Area (m^2)	Density (kg/m^3)	Altitude (m)	Reynolds Number	Run Time (min)
5	Amb	0.7E5	46.8	2.8E – 1	3.8E – 3	40110	3.1E5	356.6
8	400	4.5E5	31.9	3.6E – 2	1.4E – 3	47240	2.0E5	112.1
11	700	63.2E5	48.8	5.1E – 3	1.2E – 3	48480	3.0E5	18.3
16	1150	689.5E5	24.3	3.0E – 4	3.0E – 4	59980	1.3E5	10.3
20	1650	689.5E5	2.8	4.2E – 5	2.5E – 5	77840	1.8E4	35.8
25	2330	689.5E5	0.33	5.9E – 6	2.0E – 6	80000	1.2E3	217.43

5.3-3, these conditions result in a mass flow of 16591.6 Kg/s, the maximum mass flow encountered. The tankage scheme selected uses 205 cylindrical tanks, each with a diameter of 5 m and a length of 15 m for a total pressure tank volume of 60,377 m^3.

In order to maintain a constant mass flow, heat must be added to the high pressure storage tanks because as the test gas expands, it cools. There are many ways to maintain a constant (or nearly constant) temperature in the pressure storage tanks. A very simple, innovative, and widely used method is to place lightly crumpled tin cans in the storage tank. The addition of these cans provides an easy, inexpensive method of storing thermal energy that can be added to the gas as it expands. The cans in no way interfere in the operation of the tunnel and "the cheapest cans are those intended to contain dog food." [1] Data from Reference 1 indicate that for the pressure tanks selected (length to diameter ratio of 3.0), inclusion of the metal heat sink (tin cans) leads to a value of the coefficient n in equation 5-6 between 1.05 and 1.15.

5-6
$$\frac{T_i}{T_f} = \left(\frac{P_i}{P_f}\right)^{\frac{(n-1)}{n}}$$

T_i = Initial Stagnation Temperature
T_f = Final Stagnation Temperature
P_i = Initial Stagnation Pressure
P_f = Final Stagnation Pressure
n = Polytropic Expansion Exponent

This results in a small temperature drop in the pressure tank (for the pressures of interest, a drop of approximately 1%).

As was stated, a maximum operating pressure of 6.895×10^7 N/m^2 was selected. This value was selected to satisfy the required/desired density in the test section for various Mach number flows. Also, it was felt that this value is close to the technology limit in terms of high pressure storage and handling. The additional pressure in the storage tanks is required to maintain the operating pressure throughout the test run and to compensate for the pressure los between the storage tanks and the nozzle. This pressure (7.8×10^7) pushes the high pressure requirement even closer to the technology limit.

The heater installed downstream of the pressure regulator will maintain the test gas at a constant temperature with a maximum temperature of 2350 K. A pebble-bed heater using zirconium oxide is proposed.

Axisymmetric converging-diverging nozzles with throat sizes dependent upon the desired Mach flow are recommended. Several nozzles are required for this facility as proposed and their throat areas are shown in Tables 5.3-1, 2, 3, and 4. These nozzles will be contoured so as to minimize the formation of shocks and they will be designed so that they may be easily interchanged.

The test section selected is cylindrical with a 3 m diameter and is 5 m long. A circular cross section was chosen over a square 3 m x 3 m cross section because of the reduced area (7.0686 m^2 versus 9.0 m^2) and therefore the increased density in the test section. In addition, the 3 m diameter circular cross section achieves the design goal of a 3 m wide test section.

A closed test section was chosed for the proposed facility. In addition, a model injection system that can be sealed off from the tunnel during tunnel start-up and can be sealed off from the exterior during injection was selected. These choices have many advantages such as the tunnel start is not hindered by the model and support, the model and balance are not subjected to the loads associated with starting and stopping, the model installation is accomplished outside the tunnel, and the model may be easily cooled between test runs.

The maximum model size obviously depends on the cross sectional area of the model. The presence of the model in the test section has the same effect as a nozzle throat, namely a reduced area. Therefore, in order to maintain the desired flow conditions in the test section, a maximum model cross section must be determined. In addition, the model ideally should not extend into the tunnel boundary layer. This is not always possible in hypersonic facilities because of the large turbulent boundary layers common at high Mach numbers. Table 5.3-5 presents estimates of the boundary layer thicknesses, the usable cross sectional area (based on criteria discussed below), and the resulting estimate of the maximum allowable model cross section and width. Note that for Mach 25 operation, the boundary layer extends beyond the centerline and a uniform flow core does not exist. A further discussion of this follows. The boundary layer estimates are based on experimental data (and extrapolations from these data) from reference 14. These data were used to formulate an equation for a turbulent boundary layer in an axisymmetric flow with a boundary layer of zero thickness at the sonic line in a nozzle throat. A completely turbulent boundary layer was assumed, even though the Re is low at high Mach numbers, because of the lack of transition information. The boundary layer thicknesses reported in Table 5.3-5 were calculated 2m from the beginning of the test section at the model test station.

While it is desirable to perform wind tunnel tests in a uniform flow, as can be seen from Table 5.3-5, this is

Table 5.3 – 5 Test Section Characteristics and Maximum Model Dimensions

Mach Number	Boundary Layer (m)	Usable Area (m^2)	Allowable Area (m^2)	Maximum Width (m)
5	0.1707	5.987	0.753	2.761
8	0.3566	4.912	0.618	2.501
11	0.5486	3.913	0.491	2.232
16	0.9144	2.323	0.292	1.720
20	1.2802	1.146	0.144	1.208
25	1.8087	0.691	0.087	0.938

* Calculated for T = 2350 K, P = 689.5E5 N/m , and 5% velocity non – uniformity

sometimes difficult for high Mach number tests. In fact, the flow conditions implied by the data in Table 5.3-5, namely a non-uniform velocity profile due to a boundary layer that extends beyond the tunnel centerline, is common in existing hypersonic facilities. This condition does not render the proposed hypersonic facility totally useless because of the fact that turbulent boundary layer velocity profiles are fuller than laminar profiles. Using the velocity profile that is implicit in the empirical formula used for the boundary layer calculations, at 20% of the boundary layer thickness from the wall the velocity is already approximately 80% of the freestream velocity. At 50% of the boundary layer thickness the velocity is 90% of the freestream velocity. And at 70% of the boundary layer thickness the velocity is 95% of the freestream velocity.

In the case where the boundary layer thickness exceeds the test section radius, the usable cross sectional area and the maximum model width depend on the tests tolerance to flow non-uniformity. For the data presented in Table 5.3-5, the usable cross sectional area, the maximum model width, and the maximum model diameter is based on a velocity profile non-uniformity of 5%.

The maximum allowable model cross sections shown in Table 5.3-5 are based on data from reference 1 that asserts that the ratio of the maximum model diameter to the square root of the usable test section area should equal approximately 0.4. Again, the usable test section area allows for a 5% velocity non-uniformity.

The cooler installed downstream of the second throat will maintain an exit temperature of 400 K. This temperature was selected to sufficiently cool the test gas prior to entering the vacuum tanks while eliminating the need for extremely sophisticated cooling equipment. The recommended method of cooling the flow is through an intermittent cooler. An intermittent cooler consists of banks of metal rods filled with a suitable liquid to remove the required thermal energy. The exit temperature can be controlled by varying the cooling fluid flow.

The vacuum tanks were sized to accommodate the pressure ratio and mass flow requirements of the most stressing operating condition from the vacuum tank viewpoint. This corresponds to a Mach 5 flow at 1030 K with a mass flow rate of 16591.6 Kg/s. This temperature was determined by examining equation 5-2.. Holding A, M, and P constant (baseline parameters discussed earlier) we see that to maximize the mass flow one would maximize the total pressure and minimize the total temperature. However the maximum total pressure is defined as 689.5 x 10^5

N/m^2. In addition, from Figure 5.2-1b, the minimum total temperature allowable to avoid liquifaction at the pressure stated above is approximately 1030 K. This, therefore, is very near the maximum mass flow available in the proposed tunnel. To maintain a Mach 5 flow, a pressure ratio of 15 must be preserved. This requirement, coupled with the mass flow rate under these conditions, requires a vacuum tank volume of 24,868.9 m^3 and an initial pressure of 700 N/m^2. This pressure is approximately 0.007 atmospheres and is not difficult to achieve or maintain. Two 28.8 m diameter spherical tanks are used providing 25,015 m^3.

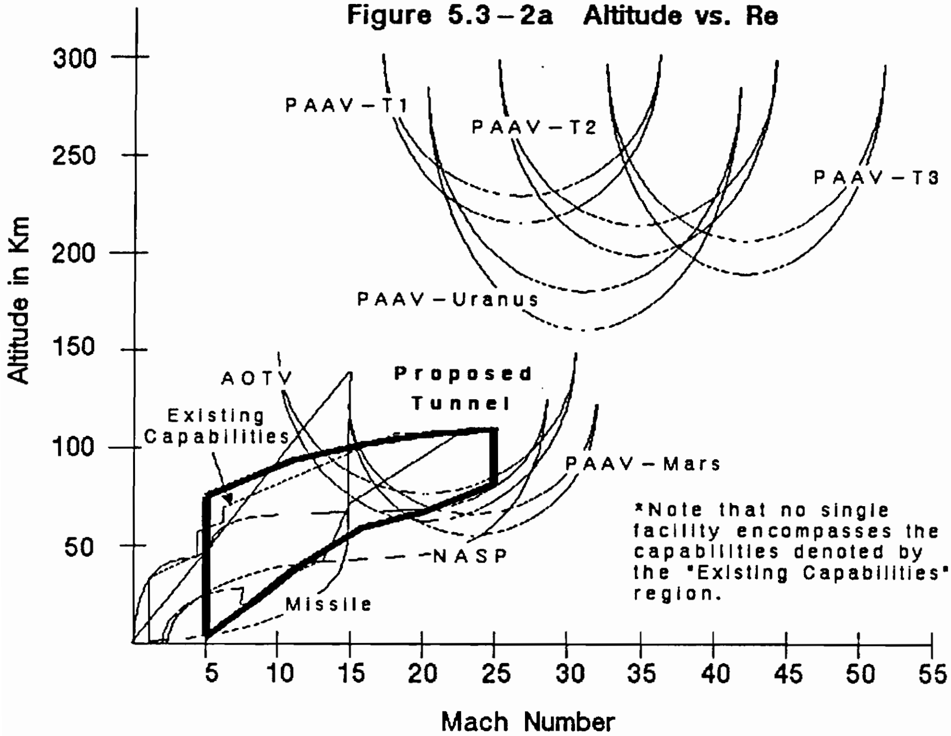
There are five tunnel components not shown in Figure 5.3-1. They are the compressors, an aftercooler, an oil filter, an air dryer, and the vacuum pumps. The compressors required are four stage compressors with an exit pressure of approximately 7.85×10^7 N/m^2 (excess pressure to allow for pressure drops across aftercooler, oil filter, and air dryer). It is recommended that these compressors provide a combined mass flow of 185 kg/s. This mass flow is based on the desire to reduce the recycle time of the tunnel to no more than 90 minutes. The mass flow rate of 185 kg/s was calculated by dividing the mass of air that flows through the tunnel in 1 minute at the maximum mass flow rate by 90 minutes. A recycle time of 90 minutes will allow for three to four test runs per eight hour shift. The number, type, and model of the compressors used is best determined by a more detailed trade study and is therefore beyond the scope of this document.

The compressors empty into aftercoolers which are simply ducts that direct the high pressure air at a low velocity through a manifold of small, cooling pipes containing a cooling fluid. The aftercooler is required to reduce the air temperature to the point that the oil filter and air dryer are effective. The oil filter is a common device that removes oil and other impurities that were introduced in the compressor. The air dryer removes excess water from the air. The dryer recommended is an absorption dryer using zirconia powder as a desiccant. The number of aftercoolers, oil filters, and air dryers depends on the number of compressors but the tunnel should have one aftercooler, oil filter, and air dryer for each compressor.

The vacuum pumps must be able to evacuate the vacuum tanks to a pressure of approximately 700 N/m^2. Common commercially available pumps are recommended. The vacuum pumps should be able to remove air from the tanks at a rate of 185 kg/s to be compatible with the compressor capacity. The selection of specific brands and models is beyond the scope of this preliminary design.

As was discussed in Section 5.2, the mass flow rate was determined by the parameters selected including the Mach number, the test section cross-sectional area, the ratio of the specific heats (defined by the total temperature), and the total pressure. The mass flow rates presented in Tables 5.3-1 through 5.3-4 represent the maximum rates achievable for each Mach number.

The densities and in turn the density altitudes are determined by the mass flow rate, the velocity, and the cross-sectional area of the test section. The densities shown in Tables 5.3-1 through 5.3-4 represent the maximum densities achievable under the specified conditions. The density altitudes calculated are however the lowest altitudes achievable for each Mach number flow. This is because the atmospheric density decreases with increasing altitude. Figure 5.3-2a presents a plot of the altitude versus Mach number envelope covered by this facility. The upper and lower curves represent the maximum and minimum altitudes that can be simulated based on the density in the test section during a test run. Figure 5.3-2b shows the variation of Re with M for the proposed tunnel. The Re was calculated for this plot based on a length equal to one fourth of the maximum allowable model width as reported in Table 5.3-5. The values presented in Table 5.3-5 were calculated at the maximum temperature (2350 K) conditions.



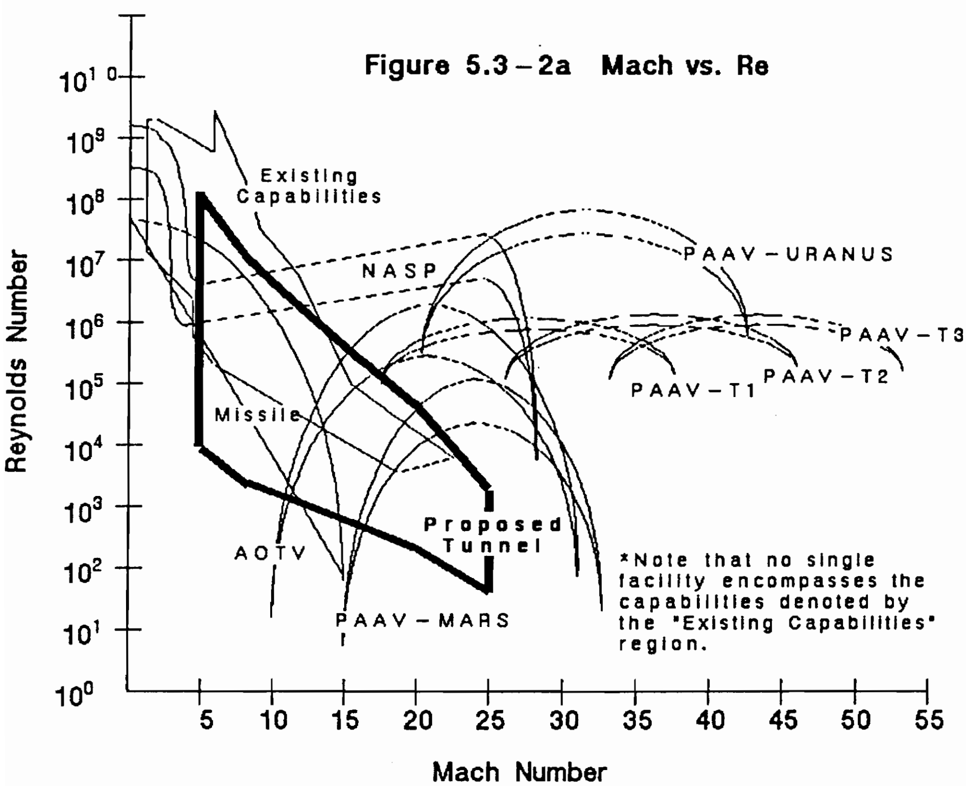


Figure 5.3 – 2 Proposed Tunnel Capabilities

As can be seen from Figure 5.3-2, the proposed hypersonic facility does not satisfy all of the requirements of the programs reviewed in Section 3. It does however enhance the current hypersonic capabilities and it satisfies <u>many</u> of the programs requirements over sometimes large portions of the anticipated flight regimes. The proposed facility would be very useful for development of the National Aerospace Plane, the Aeroassisted Orbital Transfer Vehicle, selected variations of the Planetary Aerocapture/Aeromaneuver Vehicle, and the generic high speed missile. More importantly, the proposed facility is able to assemble these needed capabilities at one location, in one facility.

The throat areas calculated for each Mach number are also shown in Tables 5.3-1 through 5.3-4. These were formulated using the appropriate Mach number and the appropriate ratio of specific heats as defined by the total temperature used. As you can see, the throat areas for the higher speed flows are small. They do, however, allow a sufficient mass flow to support the reported test section mass flow. It is assumed that some form of cooling will be required at the nozzle throat, but a detailed analysis of this was not performed.

The run times are based on the tankage sizing and mass flows determined and represent the maximum run time available for the specified conditions. These run times include the time required to start the tunnel and to allow the flow to settle. This means that the time available for data collection will be slightly less than the run time reported. The start-up and flow settling time is estimated to be on the order of 10 seconds.

Additional items would be required for the operation of the proposed facility. These included flow visualization equipment, data collection and reduction equipment, pumps, compressors, and computers as well as a power source, a building or buildings, maintenance workers, technicians, and administrators. These elements were not analyzed in detail for this effort but due to the advanced nature of this tunnel it is important to equip the facility with the latest hardware to fully exploit the tunnels capabilities. For instance, on-site computer resources should be abundant and should provide real-time or near real-time data reduction and analysis. There should be a variety of data collection techniques so that the proper system will be available for nearly any desired test. And there should be a well-trained, qualified staff on hand to provide expert support.

5.4 Cost and Schedule Estimates

This section will provide a rough order of magnitude cost estimate for the facility described in the preceding sections. This estimate is based on a review of the reported replacement costs for the facilities listed in Appendix A and a review of other comparable hypersonic facility proposals. A comparison of the costs, weighted by the capabilities provided, was performed. The results are discussed below.

The estimates for the replacement cost of the facilities in Appendix A were developed by the managers of the facilities and can therefore be confidently taken as accurate. The data obtained for the hypersonic facility proposals were derived by the originators of the tunnel designs and are based on a costing of the proposed tunnel components. Based on these data, the proposed facility would cost approximately 275M 1985 dollars. This cost does not include the cost of the land, the supporting utilities (power, water, etc.), or the personnel required for tunnel operation. It does include all structures, storage/vacuum tanks, nozzles, computers, compressors, and other hardware associated with the wind tunnel.

It is projected that this facility would require two and one half years to build and another one year to test and calibrate. This time estimate includes all construction and facility outfitting that is included in the cost estimate. The estimate would be increased if the basic utilities, such as power, water, etc. were not available and had to be built. Because of these potential added expenses, it is recommended that this facility be collocated with other similar test facilities.

6.0 Summary

The facility proposal outlined in the preceding section represents a preliminary analysis and design intended to establish a starting point for the development of a highly capable, new hypersonic wind tunnel. This section will summarize the analysis and design as well as the requirements that drive the design.

There are several current or planned programs that would benefit from the construction of a more capable hypersonic tunnel. Four of these programs were discussed in Section 3.0. They are:

- National Aerospace Plane (NASP)
- Aeroassisted Orbital Transfer Vehicle (AOTV)
- Planetary Aerocapture/Aeromaneuver Vehicle
- Generic Very High Speed Missile

The test requirements of these programs are presented in Section 3.0 and are summarized in Section 4.0, Table 4.2-1. These requirements exceed the current test facility capabilities causing a capability shortfall. This shortfall is identified and defined in Section 4.0 where, the capabilities of the 20 most capable facilities in the U.S. are summarized and compared to the program requirements. Information on the 20 facilities is presented in Appendix A.

A solution to the identified shortfall is addressed in Section 5.0 where a top level design for a proposed highly capable hypersonic wind tunnel is discussed. The proposed tunnel is a pressure-vacuum constant mass flow wind tunnel capable of operating at Mach numbers ranging from 5 to 25 and temperatures up to 2350 K. It is capable of simulating altitudes as low as 4500 m and as high as 103,000 m. Maximum run times range from 1 minute to 6 hours, depending on the speed and operating pressure selected. Figure 5.3-1 presents a diagram of the proposed facility and Tables 5.3-1 through 5.3-4 and Figure 5.3-2 summarize its capabilities.

Preliminary cost and schedule projections for the construction of the proposed facility are discussed in Section 5.4. These projections are based on replacement cost data for the 20 facilities presented in Appendix A and several comparable new facility proposals (see reference 10). The estimated cost is 275M 1985 dollars. The facility construction is projected to last two and one half years. One additional year will be required for testing and calibration of the equipment.

In conclusion, this document attempts to clarify and quantify the hypersonic test facility shortfalls that exist today. These shortfalls are identified in terms of the inability to satisfy hypersonic program testing requirements. A means of satisfying a majority of the identified hypersonic testing requirements is presented in the form of a preliminary facility design. This proposal is intended to stimulate interest in the enhancement of the U.S. hypersonic test capability.

References

1. Pope, Alan and Kennith L. Goin, High-Speed Wind Tunnel Testing, Robert E. Krieger Publishing Company, Huntington, New York, 1978, (Originally published by John Wiley and Sons, Incorporated, New York, 1965).

2. Calloway, R.L., "Real-Gas Simulation for the Shuttle Orbiter and Planetary Entry Configurations Including Flight Results," AIAA Paper 84-0489, January 9, 1984.

3. Aeronautical Facilities Catalogue - Volume 1, Wind Tunnels, National Aeronautics and Space Administration, Office of Aeronautics and Space Technology, January, 1985.

4. Florence, D.E., "Aerothermodynamic Design Feasibility of a Generic Planetary Aerocapture/Aeromaneuver Vehicle," AIAA Paper 81-1127, June 23, 1981.

5. Rumple, JoAnne, "Computer to Shift Hypersonic Flight from Realm of Imagination to Reality," United States Air Force Aeronautical Systems Division, Office of Public Affairs News Release, Wright-Patterson Air Force Base, March 10, 1989.

6. Moss, James N., "Flow-Field Analysis for Future Planetary Entry," AIAA Paper 79-0918, May 8, 1979.

7. Tethers in Space Handbook, National Aeronautics and Space Administration, Office of Space Flight, Advanced Programs, August, 1986.

8. United States General Accounting Office, National Aero-Space Plane: A Technology Development and Demonstration Program to Build the X-30, Report to Congressional Committees, April, 1988.

9. Walberg, Gerald D., "A Review of Aeroassisted Orbit Transfer," AIAA Paper 82-1378, August 9, 1982.

10. United States Air Force Scientific Advisory Board, Report of the Ad Hoc Committee on Requirements for Hypersonic Test Facilities, May 1, 1989.

11. Roberts, B.B., "Systems Analysis and Technology Development for the NASA Orbit Transfer Vehicle," AIAA Paper 85-0962, June 10, 1985.

12. Moss, James N. and Carl D. Scott, Editors, Thermophysical Aspects of Re-Entry Flows, Volume 103 of "Progress in Astronautics and Aeronautics," American Institute of Aeronautics and Astronautics, Inc., New York, 1985.

13. Schetz, Joseph A., Foundations of Boundary Layer Theory for Momentum, Heat, and Mass Transfer, Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1984.

14. Handbook of Supersonic Aerodynamics: Section 17 - Ductc, Nozzles and Diffusers, NAVWEPS Report 1488 (Volume 6), The Chief of the Bureau of Naval Weapons, Department of the Navy, January 1964.

APPENDIX A

Summary of Existing Hypersonic Capabilities

Tunnel (Date Built)		
NASA Langley 8 Ft. High Temperature Tunnel (64/80/85)		
Type		
Open Jet Blowdown – Burns methane in air under pressure		
Test Section Size	**Mach Range**	**Dynamic Pressure**
2.44 m dia. x 3.66 m	4 – 7.2	1.20 – 8.62 x 10^4
Run Time	**Temperature**	**Stagnation Pressure**
15 – 120 seconds	1333 – 2000 K	1.03 – 1.66 x 10^7
Test Gas	**Reynolds Number Range**	**Operational Status**
Methane and Air	0.3 – 22 x 10^6	1 Shift (backlog)
Data Acquisition		
100 channels, 5 Hz filter, 40 samples per second 250 channels, 10 Hz filter, 4 samples per second Real-time graphics display		
Comments		
Conical-contoured axisymmetrical nozzle. Model mounting is semispan or sting with insertion after tunnel start. Single-stage air ejector for low-pressure or high-altitude simulation.		

Tunnel (Date Built)		
Calspan Corporation 96 In. Hypersonic Shock Tunnel (63)		
Type		
Shock tunnel with hydrogen-nitrogen or helium-air driver		
Test Section Size	**Mach Range**	**Dynamic Pressure**
Variable .61 m to 2.44 m dia.	6.5 – 24	to 9.58 x 10^5
Run Time	**Temperature**	**Stagnation Pressure**
Microseconds	720 – 6390 K	0.069 – 1.38 x 10^8
Test Gas	**Reynolds Number Range**	**Operational Status**
Air	0.001 – 75 x 10^6	Operational
Data Acquisition		
48 channel or 128 channel digital data system. HP 9836 computer for reduction. Schlieren and shadowgraph, high speed movie, electron beam, microwave interferometer, X-band pulse doppler radar.		
Comments		
Models are sting mounted. 4 axisymmetric nozzles; 24" contoured for Mach 8, 48" contoured for Mach 16; 48" and 72" conical.		

Tunnel (Date Built)		
Calspan Corporation 48 In. Hypersonic Shock Tunnel (59)		
Type		
Shock tunnel with helium and air drivers		
Test Section Size Variable 0.61 m to 1.22 m dia.	**Mach Range** 5.5 – 20	**Dynamic Pressure** 2.73×10^5
Run Time Microseconds	**Temperature** 610 – 3220 K	**Stagnation Pressure** 1.38×10^6 – 3.72×10^7
Test Gas	**Reynolds Number Range** 0.004 – 50.0×10^6	**Operational Status** 1 shift
Data Acquisition		
48 channel and 128 channel digital data system. HP 9836 computer for reduction. Schlieren and shadowgraph visualization systems.		
Comments		
Models are sting mounted. 4 nozzles; 24" contoured for Mach 8, 24 " conical, 48" contoured for Mach 16, 48" conical.		

Tunnel (Date Built)		
Naval Surface Weapons Center Hypervelocity Wind Tunnel (76/85)		
Type		
Blowdown		
Test Section Size 1.52 m dia.	**Mach Range** 10.0, 14.5	**Dynamic Pressure** 206.84
Run Time	**Temperature** 1090 – 2030 K	**Stagnation Pressure** 4.14×10^2 – 3.45×10^5
Test Gas Nitrogen	**Reynolds Number Range** 0.06 – 20.0×10^6	**Operational Status** 1 shift
Data Acquisition		
128 channel modifiable to 256 channels, 16 mm and 70 mm film. Shadowgraph and holography visualization.		
Comments		
Equipped for full-scale reentry body testing.		

Tunnel (Date Built)		
Naval Surface Weapons Center Hypersonic Research Wind Tunnel (71)		
Type Blowdown		
Test Section Size 0.61 m	**Mach Range** 18	**Dynamic Pressure** $2.63 \times 10^3 - 5.27 \times 10^3$
Run Time 4 minutes	**Temperature** 2055 K	**Stagnation Pressure** $2.07 \times 10^7 - 6.07 \times 10^7$
Test Gas Nitrogen	**Reynolds Number Range** $0.2 - 0.6 \times 10^6$	**Operational Status** 1 shift
Data Acquisition 128 analog channels. Stripchart recorders, plotters, and tape recorders are available. Heat transfer, force, moment, pressure, and oil flow measurements can be made.		
Comments Angles of attack between -15° and $+60^\circ$. High altitude reentry body testing performed.		

Tunnel (Date Built)		
NASA Langley Mach 20 High Reynolds Number Helium Tunnel (52/83)		
Type Pressure – Vacuum		
Test Section Size 1.52 m	**Mach Range** 16.5 – 18	**Dynamic Pressure** $1.92 \times 10^3 - 2.97 \times 10^4$
Run Time 5.0 seconds	**Temperature** Ambient	**Stagnation Pressure** $2.07 \times 10^6 - 1.38 \times 10^7$
Test Gas Helium	**Reynolds Number Range** $1.90 - 15.0 \times 10^6$	**Operational Status** 1 shift (backlog)
Data Acquisition 100 channel system, 40 samples/second. HP 9825 connected to 16 channel system. Electron beam flow visualization available.		
Comments Experience with 3 – D turbulent boundary layer studies.		

Tunnel (Date Built)		
NASA Langley Hypersonic Helium Tunnel – Aerodynamic Leg (58)		
Type Intermittent Blowdown		
Test Section Size 0.57 m	**Mach Range** 17.6 – 22.2	**Dynamic Pressure** $6.90 \times 10^3 - 2.87 \times 10^4$
Run Time 40 seconds	**Temperature** Ambient – 480 K	**Stagnation Pressure** $3.45 \times 10^6 - 2.07 \times 10^7$
Test Gas Purified Helium	**Reynolds Number Range** $1.10 - 11.30 \times 10^6$	**Operational Status** 1 shift (backlog)
Data Acquisition 100 channel system, 40 samples/second. Photo–optical and electron beam systems for 3–D flow visualization.		
Comments Three contoured nozzles for Mach 18, 22 and 26 flow. Aerodynamic force and moment, pressure, heat transfer, and flow field experiments.		

Tunnel (Date Built)		
NASA Langley Hypersonic Helium Tunnel – Fluid Mechanics Leg (58)		
Type Open Jet Blowdown		
Test Section Size 0.56 m or 0.91 m	**Mach Range** 20 or 40	**Dynamic Pressure** $6.90 \times 10^3 - 2.87 \times 10^4$
Run Time 20 seconds	**Temperature** Ambient – 480 K	**Stagnation Pressure** $6.90 \times 10^6 - 1.38 \times 10^7$
Test Gas Purified Helium	**Reynolds Number Range** $1.30 - 6.00 \times 10^6$	**Operational Status** 1 shift (backlog)
Data Acquisition 100 channel system, 40 samples/second. Photo–optical and electron beam systems for 3–D flow visualization.		
Comments Three contoured nozzles for Mach 18, 22 and 26 flow. Aerodynamic force and moment, pressure, heat transfer, and flow field experiments.		

Tunnel (Date Built) Arnold IEngineering Development Center – Hypersonic Wind Tunnel B (54)		
Type Continuous flow, closed circuit, variable density		
Test Section Size 1.27 m dia.	**Mach Range** 6 or 8	**Dynamic Pressure** $2.35 \times 10^3 - 2.87 \times 10^4$
Run Time	**Temperature** 390 – 750 K	**Stagnation Pressure** $1.38 \times 10^5 - 5.86 \times 10^6$
Test Gas Air	**Reynolds Number Range** $0.30 - 4.70 \times 10^6$	**Operational Status** Lightly scheduled
Data Acquisition Various control and data management computers. Schlieren system, shadowgraph system, high speed camera, IR camera, and force and moment balance.		
Comments Includes interchangeable contoured axisymmetric nozzles, model injection system, and captive store separation system.		

Tunnel (Date Built) Arnold Engineering Development Center – Hypersonic Wind Tunnel C		
Type Continous flow, closed circuit, variable density		
Test Section Size 0.64 m and 1.27 m dia.	**Mach Range** 4 or 10	**Dynamic Pressure** see below
Run Time	**Temperature** 915 – 1250 K	**Stagnation Pressure** see below
Test Gas Air	**Reynolds Number Range** see below	**Operational Status** lightly scheduled
Data Acquisition Various control and data management computers. Schlieren system, shadowgraph system, high speed camera, IR camera, and force and moment balance.		
Comments Interchangeable contour nozzles, full scale flight environmental testing of tactical missiles. Re: $0.40 - 1.30 \times 10^6$ (M=4); $0.30 - 4.70 \times 10^6$ (M=10) Dyn. Press.: $2.30 \times 10^2 - 7.04 \times 10^2$ (M=4); $1.92 \times 10^3 - 2.27 \times 10^4$ (M=10) Stag. Press.: $4.48 \times 10^5 - 1.38 \times 10^7$ (M=4); 1.38×10^7 (M=10)		

Tunnel (Date Built)		
NASA Langley Continuous Flow Hypersonic Tunnel (62/84)		
Type Pressure – Vacuum or Continuous Flow		
Test Section Size 0.79 x 0.79 m	**Mach Range** 10	**Dynamic Pressure** $2.87 \times 10^3 - 1.92 \times 10^4$
Run Time Continuous/1 minute	**Temperature** 1000 K	**Stagnation Pressure** $1.72 \times 10^6 - 1.24 \times 10^7$
Test Gas Air	**Reynolds Number Range** $0.40 - 2.40 \times 10^6$	**Operational Status** 1 shift/day (backlog)
Data Acquisition 128 channel system, 100 samples per second, force and moment balance.		
Comments Can run continuously but usually operated in intermittent mode for energy conservation. Electric resistance heaters used to prevent condensation.		

Tunnel (Date Built)		
NASA Ames 3.5 Foot Hypersonic Wind Tunnel (60/72)		
Type Pressure – Vacuum		
Test Section Size 1.07 m dia. x 3.05 m	**Mach Range** 5, 7, 10	**Dynamic Pressure** $8.14 \times 10^3 - 7.66 \times 10^4$
Run Time 0.5 – 4 minutes	**Temperature** 670 – 1920 K	**Stagnation Pressure** $3.03 \times 10^5 - 1.35 \times 10^7$
Test Gas	**Reynolds Number Range** $0.30 - 7.40 \times 10^6$	**Operational Status** Standby
Data Acquisition 228 channel system, 60,000 samples per second. Limited real – time data processing capability. Shadowgraph system, high speed cameras, and force and moment balance available.		
Comments Interchangeable contoured axisymmetric nozzles. Storage heater using aluminum oxide pebbles preheated by burning natural gas.		

Tunnel (Date Built)		
Grumman Aerospace Corporation 36" Hypersonic Wind Tunnel (62/67)		
Type Pressure – Vacuum		
Test Section Size 0.91 m dia.	**Mach Range** 8, 10, 14	**Dynamic Pressure** 5.03×10^3 – 3.80×10^4
Run Time	**Temperature** 555 – 1670 K	**Stagnation Pressure** 3.45×10^5 – 1.38×10^7
Test Gas	**Reynolds Number Range** 0.20 – 4.50×10^6	**Operational Status** Standby
Data Acquisition 100 channel system. Special cameras for heat paint transfer testing. Schlieren system.		
Comments Ideal for heat transfer testing. Able to perform 2 runs per hour. Vacuum system currently in use elsewhere.		

Tunnel (Date Built)		
Northrop Corporation 30" Hypersonic Wind Tunnel		
Type Pressure – Vacuum		
Test Section Size 0.76 m dia.	**Mach Range** 6, 10, 14	**Dynamic Pressure** 4.79×10^2 – 2.87×10^4
Run Time 30 seconds	**Temperature** 670 – 1780 K	**Stagnation Pressure** 2.76×10^4 – 8.96×10^6
Test Gas	**Reynolds Number Range** 0.02 – 3.50×10^6	**Operational Status** Inactive
Data Acquisition 256 channel system, dedicated computer, graphics CRT, high–speed pen plotter. Schlieren system.		
Comments 100,000 cu. ft. vacuum sphere, 2000 cu.ft. 3200 psia air supply. Interchangeable free jet or closed jet test sections.		

Tunnel (Date Built)		
McDonnel Douglas 2' Hypersonic Wind Tunnel (61)		
Type Intermittent Pressure – Vacuum		
Test Section Size 0.61 m	**Mach Range** 6, 8, 10	**Dynamic Pressure** 1.38×10^4 – 6.90×10^4
Run Time 60 seconds	**Temperature** 470 – 1370 K	**Stagnation Pressure** 1.72×10^6 – 1.55×10^7
Test Gas Air	**Reynolds Number Range** 1.20 – 11.20×10^6	**Operational Status** Standby
Data Acquisition 64 channel system. Data reduction between runs.		
Comments 1.5 runs per hour typical. Facility last operated in 1975 and is currently for sale.		

Tunnel (Date Built)		
NASA Langley Hypersonic CF_4 Tunnel (72)		
Type Real Gas Simulation Blowdown Tunnel		
Test Section Size 0.51 m dia.	**Mach Range** 6	**Dynamic Pressure** 23.94 – 71.82
Run Time 7 – 20 seconds	**Temperature** 590 – 700 K	**Stagnation Pressure** 6.90×10^6 – 1.72×10^7
Test Gas Freon 14	**Reynolds Number Range** 0.30 – 0.50×10^6	**Operational Status** 1 shift (backlog)
Data Acquisition 128 channel system, 100 samples per second. Pressure and aerodynamic force test capability.		
Comments CF_4 yields normal shock density ratio of approximately 12 permitting simulation of real gas effects at entry speeds for Earth and the terrestrial planets.		

Tunnel (Date Built)		
Wright Aeronautical Lab, Wright-Patterson AFB 20" Hypersonic (60)		
Type: Intermittent Pressure-Vacuum (Open Jet Test Section)		
Test Section Size: 0.51 m dia. x 0.76 m	**Mach Range:** 12 - 14	**Dynamic Pressure:** 1.92×10^3 - 5.75×10^3
Run Time: 3 - 5 minutes	**Temperature:** 1000 - 1110 K	**Stagnation Pressure:** 5.52×10^6 - 1.10×10^7
Test Gas: Air	**Reynolds Number Range:** 0.40 - 1.00×10^6	**Operational Status:** 1 shift (backlog)
Data Acquisition: 128 channel system, 100 samples per second. Water-cooled balance. On-site data reduction.		
Comments: Pitch control $\pm$ 20°. Model size 1'. Primarily used for ballistic reentry vehicle studies.		

Tunnel (Date Built)		
Naval Surface Weapons Center Hypersonic Wind Tunnel Number 8 (60)		
Type: Blowdown with Heated Flow		
Test Section Size: 0.43 m - 0.57 m dia.	**Mach Range:** 5 - 8	**Dynamic Pressure:** 33.52 - 2.87×10^3
Run Time: minutes to hours	**Temperature:** 330 - 810 K	**Stagnation Pressure:** 5.07×10^5 - 1.52×10^7
Test Gas: Air	**Reynolds Number Range:** 100 - 7000×10^6	**Operational Status:** 1 shift available
Data Acquisition: 128 channel high-speed analog-to-digital system. Schlieren and shadowgraph systems.		
Comments: High Re for accurate aerodynamic and aerothermodynamic testing. Ablation testing of Low Temperature Ablation materials. Free-flight and free-fall techniques available.		

Tunnel (Date Built)
Sandia National Lab 18" Hypersonic Wind Tunnel (62/75)

Type
Pressure – Vacuum

Test Section Size	Mach Range	Dynamic Pressure
0.46 x 1.22 m	5, 8, 14	9.58 – 3.16×10^2
Run Time	**Temperature**	**Stagnation Pressure**
30 seconds	345 – 1390 K	3.45×10^5 – 2.07×10^7
Test Gas	**Reynolds Number Range**	**Operational Status**
Air	0.20 – 9.70×10^6	1 shift

Data Acquisition
16 channel near–realtime data reduction system. Electronic Pressure Scanning System provides capability for measuring large number of pressures. Schlieren and shadowgraph flow visualization systems available.

Comments
Models up to 4" diameter. Primary testing: 6–DOF and surface pressure distribution. Mach 5 and 8 turbulent testing.

Tunnel (Date Built)
NASA Langley Hypersonic Nitrogen Tunnel (64)

Type
Blowdown

Test Section Size	Mach Range	Dynamic Pressure
0.41 m dia.	18	1.20×10^3 – 3.50×10^3
Run Time	**Temperature**	**Stagnation Pressure**
30 minutes	1670 – 1940 K	1.38×10^7 – 4.14×10^7
Test Gas	**Reynolds Number Range**	**Operational Status**
Nitrogen	0.17 – 0.40×10^6	1 shift (backlog)

Data Acquisition
100 channel, 40 samples per second. Electron gun flow visualization technique.

Comments
Routinely perform advanced aerospace transportation studies (i.e. Shuttle II). Measure forces, moments, performance, stability, control, pressure and heating.

VITA

Mr. Gregory Drauch was born on January 6, 1964 in Trenton, Michigan, a suburb of Detroit, Michigan. He was raised in Trenton where he graduated ninth in his class at Trenton High School. He went on to attend the University of Michigan and graduated Cum Laude in 1986 with a Bachelor of Science in Aerospace Engineering. After college Mr. Drauch moved to Northern Virginia and began his career with SYSCON Corporation. He later obtained a position with General Research Corporation where he is currently employed. He began his graduate studies in 1986, attending Virginia Polytechnic and State University as a part-time student. Mr. Drauch was married on October 29, 1988 and he and his wife, Amy, live and work in Northern Virginia.

Gregory Drauch