THE DESIGN AND IMPLEMENTATION OF A SUPERSONIC INDRAFT TUBE
WIND TUNNEL FOR THE DEMONSTRATION OF SUPERSONIC FLOWS

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ABSTRACT

The Design and Implementation of a Supersonic Indraft Tube Wind Tunnel for the Demonstration of Supersonic Flows

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Historically, the endeavor of scale testing flight vehicles at supersonic Mach numbers, especially for long durations, has required the development of closed-loop wind tunnels, which are extremely expensive both to build and operate due to the high complexity and incredible power required to drive such a system. The intermittent blowdown wind tunnel, indraft tunnel, and shock tunnel have alleviated many of these cost requirements to some degree, whilst facilitating testing at very high Mach numbers and enthalpies; however, these systems require the handling of gases at pressures and temperatures that can be prohibitive for many university settings. The Ludwig tube provides a simple, elegant method for producing testable supersonic flows at price points significantly lower than the aforementioned test-system architectures. Unfortunately, the spacial footprint and moderate cost required for driver tube and nozzle hardware can make it difficult to implement for many non-research universities.

In this thesis, a new supersonic test system architecture is conceived, designed, implemented, and validated for the purpose of making supersonic aerodynamic testing capability attainable for most universities, by combining properties of the Ludwig Tube and indraft wind tunnel to reduce the cost needed to produce this capability. This system, the Indraft Tube Tunnel, requires no long driver-tube or test-section hardware, aside from a vacuum chamber. Furthermore, it is safe to operate, as high pressure containment systems are not required for the Indraft Tube Tunnel System. It is designed and operated to draw stagnant atmospheric air through a converging-diverging nozzle to achieve a steady-state Mach number of 2.5. Sufficient pressure ratio to reach the desired Mach number is attained by evacuating the vacuum chamber

and placing a thin cellophane diaphragm across the inlet of the nozzle, thus separating the vacuum section from ambient atmosphere. To initiate gas flow, the diaphragm is mechanically burst with a puncture device.

This design requires much less hardware to implement than a typical Ludwig tube, and had an operating cost of less than one dollar per test. Using this method, steady, uninterrupted Mach 2.44 is attained for a duration of 13.6 ms and a test section diameter of 7 inches. The standard deviation of the Mach number measurements is .08 Mach. A shadowgraph imaging setup is used to view and measure the angle of oblique shockwaves on a simple wedge test-model. The Indraft Tube Tunnel is novel in the field of high-speed aerodynamic testing, and may be implemented by other universities to produce supersonic flows with a relatively small investment in hardware and laboratory space.

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Then I heard the voice of the Lord saying, "Whom shall I send?
And who will go for us?" And I said, "Here I am. Send me!"

-Isaiah 6:8

That I may go wherever you send me Lord, unto and beyond the ends of the Earth.

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Chapter 1

INTRODUCTION

The field of Aerospace Engineering undertakes the challenge of designing vehicles which must operate in very demanding environments. In order to facilitate these designs, it is necessary to perform testing in these extreme environments, which is oftentimes accomplished by reproducing flight conditions in a controlled, ground-based environment such as a wind tunnel. Among other things, wind tunnels allow the behavior of a design geometry for a given flight envelope to be determined much more economically and safely than through flight testing, rendering them irreplaceable in the design process of atmospheric flight vehicles.

Wind tunnel testing of supersonic vehicles is particularly difficult as the challenges associated with the power, facility, and material requirements necessary for the production of sufficient flow speeds and temperatures are both technically difficult and prodigiously expensive to overcome. As such, several general test-system architectures have been developed and utilized with great success, all of which attempt to maximize test-system performance, minimize cost, or provide a balance between the two. Although the largest, most advanced tunnels are limited to very large research institutions such as NASA Ames Research Center in California or the Air Force Research Laboratory in Ohio, the advent of intermittent testing architectures such as blow down wind tunnels, shock-tunnels, and Ludwig Tubes have reduced the monetary cost and physical footprint required to perform supersonic aerodynamic testing, making the endeavor realizable for most Universities.

Even the Ludwig Tube, which elicits an associated cost and physical footprint orders of magnitude more favorable than continuous-flow supersonic wind tunnels—albeit through a great reduction in achievable flow regimes—requires more physical

space and funding than is readily available at some universities, particularly those which do not prioritize research. The development of a new type of wind tunnel which combines and modifies characteristics of existing tunnel architectures may present a significant decrease in cost and space requirements, making supersonic testing attainable for a greater number of universities.

1.1 Statement of Purpose

The California Polytechnic State University San Luis Obispo Aerospace Engineering Department has possessed a small atmospherically exhausted blow-down supersonic wind tunnel for a number of years. For a variety of reasons which include rising maintenance costs as the facility ages and a restructuring of the available facility space, the system is quickly falling into obsolescence. A new apparatus is required both for laboratory instruction and student research. Therefore, a new facility must be designed and built to provide a replacement for and improvement on the existing supersonic wind tunnel.

1.2 Statement of Objectives

This thesis will detail the conception, design, and validation of a system to produce supersonic flows for aerodynamic testing compatible with existing facility space at California Polytechnic State University, San Luis Obispo for a budget of under \$10,000. The requirements of the system are as follows:

- Test section of greater than 6" diameter
- Mach number capability in excess of 2
- Schlieren or shadowgraph flow imaging compatibility

- Ability to position models in the flow at a range of orientations
- Cost per test of less than \$1

1.3 Approach

This thesis will present a brief historical background on the field of supersonic aerodynamic testing with an emphasis on short duration wind tunnels. Three possible design solutions to satisfy the needs of the Cal Poly Aerospace Engineering Department will then be analyzed and evaluated against the design requirements of the project. The comprehensive design of the chosen system architecture will then be presented in detail, followed by an accounting of the fabrication of the apparatus. Experimental validation of the system's operation will then be presented. The document will be concluded with an overview of the facility and its current and future capabilities.

The final design utilizes a large vacuum chamber owned by the Cal Poly Aerospace Engineering Department to act as a dump tank and free jet test section for a short duration supersonic wind tunnel which operates by applying the same basic principles which govern Ludwig Tubes to an indraft wind tunnel architecture. This novel apparatus will henceforth be referred to as the Indraft Tube Tunnel or ITT.

Chapter 2

BACKGROUND

In this chapter, a background of the many regimes in which flight vehicles operate will be given, as well as the specific types of wind tunnels used to test vehicles for operation in these flight regimes. The operational parameters, design constraints, and cost requirements inherent to each test-system architecture will be highlighted in the interest of selecting one to be designed and built for the California Polytechnic State University, San Luis Obispo Aerospace Engineering Department.

2.1 Flight Regimes

Before detailing the many types of wind tunnels in use, it is useful to gain an understanding of the regimes in which vehicles are required to fly, as the express purpose of wind tunnel testing is to represent the flight parameters in which a vehicle operates.

The flight regimes commonly described are subsonic, compressible, transonic, supersonic, and hypersonic. Progressing through these flight envelopes, the methods required to analyze and describe the flows become increasingly more complex, and fewer assumptions can be made to simplify the flow's governing equations. These regimes will be discussed in depth, and for easy reference Tab. 2.1 is included with a brief overview of the differences between each flight regime, and some characteristics typical of vehicles operating in each envelope.

2.1.1 Wind Tunnel Power

The power required to operate a wind tunnel is primarily dependent on flow velocity, which means that increasing from one flight regime to another also requires a sig-

Table 2.1: Comparison of Flight Regimes

	Incompressible	Compressible	Transonic	Supersonic	Hypersonic
Mach Number	0 - .3	.3 - .7	.7 - 1.4	1.4 - 5	5+
Assumptions	$\rho = \text{constant}$	$\rho = P/RT$	$\rho = P/RT$ Shocks	$\rho = P/RT$ Shocks	$\rho = P/RT$ Shocks Real Gas
Propulsion	Turbojet Turbofan Propeller	Turbojet Turbofan Propeller	Turbojet Turbofan	Turbojet RAMjet	SCRAMjet RAMjet
Wing Type	Straight	Straight Swept	Swept Delta	Swept Delta	Delta
Material	Aluminum Composite	Aluminum Composite	Aluminum Composite	Aluminum Composite Titanium	Titanium Ceramic Inconel Ablative

nificant increase in power input to the system. The actual power input required to run a tunnel varies based on the specific architecture of the system, but a reasonable way to estimate this power is by finding the jet power of the flow, which begins with calculating the kinetic energy in flow [18]:

$$KE = \frac{1}{2}mU^2 \quad (2.1)$$

Where KE is kinetic energy, U is flow velocity, and m is mass. The second driving factor in power requirements for a tunnel is the amount of air that is moving through the tunnel; the mass flow rate:

$$\dot{m} = \rho UA \quad (2.2)$$

Where $\dot{m}$ is mass flow rate, A is cross-sectional area, and ρ is density. Without considering the method of power input to the airflow, and assuming that no energy is

recovered in the flow as through a diffuser and closed-loop cycle, the power required to run a wind tunnel at a particular velocity can be described as the jet power of the flow[18]:

$$P_{jet} = \frac{1}{2}\rho AU^3 \quad (2.3)$$

Therefore the jet power, P_{jet} , increases proportionally to velocity cubed, meaning that power requirements dominate the operating costs in high-speed aerodynamic testing. This exact relation is a good order of magnitude approximation of power requirements to run a tunnel, but a more in-depth analysis would be necessary to find the power requirements for a specific system, which would take the characteristics of that system into account. Regardless, the sentiment is still valid, and should be kept in mind when considering the design of a high speed wind tunnel [18].

2.1.2 The Incompressible Flow Regime

In the incompressible flow regime, a vehicle's velocity is low enough that the pressure increase associated with flow stagnation and diversion due to the vehicles airframe are not high enough to constitute a significant change in density, and is therefore called incompressible. Although analytical methods for modeling this type of flow as a compressible gas are perfectly valid, simplifications can be made by assuming a constant density. This allows a direct relation to be made between stagnation pressure, static pressure, and velocity along a streamline for an inviscid, irrotational flow, as can be seen in Bernouilli's Equation [1]:

$$P_0 = \frac{1}{2}\rho U^2 + P \quad (2.4)$$

Where P_0 is total pressure, or pressure a flow would reach if it were isentropically decelerated to zero velocity, P is static pressure, and ρ is density. This assumption is

typically valid for Mach numbers up to .3, although it may be valid for higher Mach numbers if the density change in the flow is low enough [18].

$$\rho_1 U_1 A_1 = \rho_2 U_2 A_2 \quad (2.5)$$

Since density is constant:

$$U_1 A_1 = U_2 A_2 \quad (2.6)$$

Wind tunnels that test flows in this regime are the most simple and economical to run when compared to those that operate at higher speeds. They are typically driven or drawn with fans, and due to their low speed, the pressure and power they require are relatively easy to produce for indefinite test durations [18].

2.1.3 The Compressible Flow Regime

As a vehicle's airspeed increases, the pressure increase in the flow encountering the vehicle is great enough to cause a significant increase in the density of the flow, so the flow is called compressible. Compressible flows require a different set of equations — see eqs. 2.9 - 2.11— to describe flow dynamics. At this point it becomes necessary to introduce the concept of sound speed and Mach number. The speed of sound is defined as the velocity at which pressure waves can propagate through a medium, be it solid or fluid [1]. As can be seen in eq. 2.7 below, this velocity, a , is determined by the static temperature T , ratio of specific heats γ , and Specific Ideal Gas Constant R [1]:

$$a = \sqrt{\gamma R T} \quad (2.7)$$

Accordingly, the higher the temperature of a gas, the faster the speed of sound through it will be. As will shortly be seen, whenever compressibility in a moving flow must be taken into account, it is preferable to describe a flow's state as the ratio between its velocity and speed of sound [1]; this value is known as the Mach number, M :

$$M = \frac{U}{a} \quad (2.8)$$

The compressible behavior of a gas is more elegantly described using Mach number than velocity, as it allows simple relationships between pressure, temperature, density, and Mach number to be proven. To do so, the concept of total temperature, T_0 , must also be introduced as the temperature a flow would reach if it were isentropically stagnated. The following equations are commonly referred to as the Isentropic Relation Equations [1]:

$$\frac{T_0}{T} = 1 + \frac{\gamma - 1}{2} M^2 \quad (2.9)$$

$$\frac{P_0}{P} = \left(\frac{T_0}{T} \right)^{\left(\frac{\gamma}{\gamma - 1} \right)} \quad (2.10)$$

The Isentropic Relation Equations gained their name because if it can be assumed that entropy does not change from one position of the flow to another, the stagnation conditions T_0 and P_0 are conserved, so knowing the overall stagnation conditions and Mach number at any point in the flow allows the static condition of the flow to be found, and vice-versa. Once these two static conditions of the flow are known, the density can easily be found using the ideal gas law, which relates pressure, temperature, and density of a gas at any one state [1].

$$P = \rho RT \quad (2.11)$$

The technology required to produce a subsonic, compressible wind tunnel is almost identical to an incompressible tunnel, with the primary difference being increased power and pressure requirements. This increase of power required makes closed-loop architectures more common in moderate Mach number subsonic tunnels.

2.1.4 The Transonic Flight Regime

The transonic flight regime is notable because, even though a vehicle may be traveling subsonically, the local Mach number at certain locations on the aircraft may exceed

Mach one, resulting in a host of problems including a large increase in drag due to shock-wave induced flow separation, and loss of aircraft stability [17]. Transonic tunnels can typically provide Mach numbers from .5 to 1.4, allowing the testing of aircraft throughout this entire range [18].

2.1.5 The Supersonic Flight Regime

At subsonic Mach numbers, meaning velocities under Mach one, pressure waves can travel freely both up and downstream through a flow. At the sonic condition and beyond, however, the velocity of pressure propagation is slower than the flow velocity, so information cannot travel upstream. When the flow encounters an obstacle and must undergo a compression turn, the presence of the obstacle cannot be communicated upstream, so the gas is forced to change state discontinuously in order to turn or stagnate; the boundary at which this occurs is called a shock wave. Shock waves are assumed to be infinitely thin regions in space across which nearly everything important about a flow changes. Shockwaves introduce a marked increase in the entropy in a flow, resulting in a loss of stagnation pressure, but conserving stagnation temperature, as even though flow through a shockwave is anisentropic, it remains adiabatic, and energy is conserved [1].

The presence of shockwaves in a flow is arguably the most significant difference between the subsonic and supersonic flight regimes, the phenomenology of which dominates the design of supersonic wind tunnels and vehicles alike.

Normal Shockwaves

The Mach number, density, velocity, and pressure across a shockwave are defined in the following equations [1]:

$$M_2^2 = \frac{1 + [(\gamma - 1)/2]M_1^2}{\gamma M_1^2 - (\gamma - 1)/2} \quad (2.12)$$

$$\frac{\rho_2}{\rho_1} = \frac{U_1}{U_2} = \frac{(\gamma + 1)M_1^2}{2 + (\gamma - 1)M_1^2} \quad (2.13)$$

$$\frac{P_2}{P_1} = 1 + \frac{2\gamma}{\gamma + 1}(M_1^2 - 1) \quad (2.14)$$

Where the subscripts 1 and 2 respectively denote that the value is before or after a normal shock. Because total temperature is conserved across a shockwave, the isentropic relation equation for temperature, eq. 2.9, can simply be evaluated using the pre-shock total temperature and post-shock Mach number. fig. 2.1 depicts a bowshock, which is seen in front of blunt-bodies traveling supersonically. The region of flow directly in front of the model undergoes a true normal shock, whereas flow in the far-field undergoes an oblique shock, which will be discussed shortly.

Oblique Shockwaves

As was discussed above, whenever a supersonic flow encounters an obstacle, a shock must occur in order for the flow to slow down and turn. Not every shock is normal and results in a subsonic flow, however. If a flow turns into itself gently enough, an oblique shock is formed, across which the flow decreases in Mach number without becoming subsonic, and changes direction. An oblique shock can be formed by a vehicle moving supersonically through air as is shown on the left side of fig. 2.2, or by encountering an angled wall, as can be seen on the right side of the same figure.

Figure 2.1: Bow shock caused by a blunt body in a supersonic flow, and an ideal normal shock, which describes the strongest region of a bow shock [1]

An oblique shock is not as strong as a normal shock starting from the same Mach number, meaning that the loss in total pressure and increase in entropy demanded by the presence of a shockwave is not as high. In an ideal two-dimensional case as shown in fig. 2.2, the flow has a turn angle, θ , so that it flows parallel to the wall or wedge in its path. The turn angle is the same as the ramp angle in this two-dimensional case. The shockwave is always formed at a steeper angle, β . The shockwave traces out the area of influence of the obstacle; any flow in front of the wave is undisturbed, and the flow turns discontinuously at the wave (fig. 2.2).

To calculate the shockwave angle of an oblique shockwave, the turn angle θ and the Mach number M must be known. The following equation relates these values to the shockwave angle β [1]:

$$\tan \theta = 2 \cot \beta \left[\frac{M^2 \sin^2 \beta - 1}{M^2(\gamma + \cos 2\beta) + 2} \right] \quad (2.15)$$

Therefore, knowing the Mach number and turn angle of a geometry allows the shock-wave angle to be easily calculable. Conversely, the Mach number can also be determined by evaluating the relation for a known or observed shockwave angle and

Figure 2.2: Oblique Shock [1]

turn angle. This process will be used to determine Mach number for the validation experiment of the Indraft Tube Tunnel in 7.2.2.

Flow with Area Change

In a steady, quasi one-dimensional analysis of a gas flow, the mass flow rate between any two points 1 and 2 must be constant, just as was shown in eq. 2.5. However, for a compressible flow, the density is no longer constant, so a different relation in terms of Mach number must be used. It is found that at subsonic Mach numbers, a decrease in cross-sectional area of a gas flow will increase Mach number, and an increase in cross-sectional area will do just the opposite [1]. This relationship holds true until a flow's area is decreased to the point of reaching Mach 1; this is called choking the flow.

Once a flow becomes choked, further decreases in area will never cause the Mach number to exceed one. If the area is increased again after the flow becomes choked, however, the flow can increase in Mach number, going supersonic. This is how a converging-diverging nozzle works, and is the primary method by which steady su-

personic flows are produced for high speed wind tunnel testing [18]. It is valuable to refer to the area at which a flow of a given Mach number and area would choke as the * condition. For a converging-diverging nozzle this is referred to as the throat area. This star condition is useful as it is unchanged for a flow as long as the stagnation conditions of that flow, P_0 and T_0 do not change. The following equation relates the ratio of a flow area to the * condition based on the flow's Mach number [1]:

$$\left(\frac{A}{A^*}\right)^2 = \frac{1}{M^2} \left[\left(\frac{2}{\gamma+1}\right) \left(1 + \frac{\gamma-1}{2} M^2\right) \right]^{\frac{\gamma+1}{\gamma-1}} \quad (2.16)$$

Where A^* is the area required to choke a flow of Mach number M , Area A , and ratio of specific heats γ . An expression directly comparing the Mach numbers and areas at two points in a flow is as follows [20]:

$$\frac{A_1}{A_2} = \frac{M_2}{M_1} \left[\frac{1 + \frac{\gamma-1}{2} M_1^2}{1 + \frac{\gamma-1}{2} M_2^2} \right]^{\frac{\gamma+1}{2(\gamma-1)}} \quad (2.17)$$

2.1.6 The Hypersonic Flight Regime

At significantly higher flow velocities, colloquially defined in free-flight as Mach 5 and above, is the hypersonic flight regime. At such high Mach numbers the isentropic relation equations begin to break down for air, as the incredible increase in temperature associated with a normal or even oftentimes oblique shock at these Mach numbers results in the excitation of vibrational and rotational modes in Nitrogen and Oxygen, even inducing a chemically reacting flow at very high temperatures [1]. The internal energy of a flow, e , can be defined as the average of all the forms of kinetic energy held by each particle in the flow, whether the motion be translational, rotational, or vibrational. If the internal energy of a flow is overwhelmingly composed of translational kinetic energy, and the flow is chemically unreacting, it can be assumed that the flow is thermally perfect, meaning that the internal energy and enthalpy are only dependent on temperature, and the specific heats of the gas, C_v and C_p , are constant

for a given temperature and pressure[1]. Further, if the specific heats of the gas are unchanging with regard to temperature and pressure, the gas is called calorically perfect [1]. In the hypersonic flight regime, these assumptions break down and more complex analytical methods are required to describe the gas flow.

As such, the energy which is converted from the velocity of the flow (kinetic) is converted into internal energy of the flow, a significant amount of which goes to molecule vibrations rather than translations, resulting in a smaller increase in temperature than the isentropic relation equations would suggest. Put simply, the stagnation temperature for the flow is no longer conserved across a shockwave, and these phenomena, called real gas effects, must be described with more complex models [1].

In aerodynamic testing, significant effort must be spent achieving high velocities as well as high Mach numbers, making the endeavor of hypersonic testing particularly arduous. The upper reaches of this regime can only be achieved intermittently through the use of Shock Tunnels due to the incredible power requirements involved as well material limitations of the facility, although several continuous-flow facilities do exist at Mach numbers in excess of 5.

2.2 A Review of Supersonic Wind Tunnel Testing Methods

Several test-system architectures have been developed for supersonic testing; they can be grouped into two general subcategories: continuous and intermittent [18]. Since this thesis is focused on the design and implementation of supersonic testing methods, subsonic and transonic system architectures will be ignored in the following section.

As it is with any endeavor, cost is a major constraint when designing a wind tunnel. This cost can be divided into two categories: initial capital investment and operating cost. The facilities, hardware, and assembly costs present a large initial

capital investment to a project, and the primary driving factor in the cost of operation is typically energy requirements [18]. Two primary methods used to reduce the cost of tunnel operation are decreasing the mass flow rate by minimizing the cross-sectional area of the tunnel test section, and operating intermittently such that very high power requirements last for a short duration, resulting in a low overall energy requirement to run the tunnel. Additionally, high flow velocities allow for much shorter test durations to be used than for low speed testing [18].

While intermittent testing presents a massive reduction in operating cost, it can also be difficult to fill large test matrices if test durations are short, especially if the time it takes to reset the system for another test is very long, as these can often be on the order of hours for intermittent testing methods [18]. Steady-state wind tunnels like the Unitary Plan Tunnel at NASA Ames can run indefinitely, so models can be tested at many different attitudes and test conditions relatively quickly [18]. In the case of intermittent testing, however, the tunnel must cease operation for an extended period of time as the facility is reset by refilling pressure vessels, evacuating vacuum spheres, and resetting hardware such as burst diaphragms and valves. The primary methods for performing intermittent high speed wind tunnel testing are through the use of the blowdown wind-tunnel—a special case of which is the Ludwieg Tube—and the Shock Tunnel, all of which will be studied in later sections. The schematics describing each of the test system architectures discussed in this section were based heavily on those from references [1] and [18]. As can be seen in tab: 2.2, continuous flow closed-loop tunnels are by far the largest and most expensive tunnels to operate, while intermittent testing methods can greatly reduce both facility and operational costs. The three basic types of intermittent tunnels are the blowdown wind tunnel—which has a sub-type known as an indraft tunnel—shock tunnel, and Ludwieg Tube, originally called a Tube Tunnel by Hans Ludwieg in the paper describing its initial design [14]. Each of these basic architectures will now be described in detail. For

Table 2.2: Comparison of Supersonic Wind Tunnel Basic Architectures. Approximate values extrapolated from tunnel characteristics presented in: [18], [14], [11], [15], [17], and [7], as well as from personal experience of the author.

	Closed Loop	Blowdown	Shock Tunnel	Tube Tunnel
Test Time	Unlimited	1-100 s	.5 - 5 ms	50-200 ms
Facility Cost, \$	$10^6 - 10^8$	$10^4 - 10^6$	$10^5 - 10^7$	$10^4 - 10^5$
Cost Per Run, \$	$10^2 - 10^4/\text{hr.}$	$10 - 10^3$	$10 - 10^3$	$10 - 10^2$
Test Cases Per Day	$10 - 10^3$	$1 - 10^2$	$1 - 10$	$1 - 10$
Test Section Width	$1 - 12 \text{ ft}$	$1 - 10 \text{ ft}$	$.5 - 3 \text{ ft}$	$.5 - 3 \text{ ft}$

reference, a basic, high level overview of the characteristics of each system architecture is included in tab: 2.2, which presents several key performance parameters for each test-system architecture. Included are test time, facility cost, cost to run a test, number of test points that can be gathered per day, and test-section size.

2.2.1 The Continuous Flow Supersonic Wind Tunnel

As was earlier stated, the continuous flow supersonic wind tunnel is by far the most expensive and complex method for performing supersonic flow testing. Although it is quite common in the case of low-speed wind tunnels to draw air from and exhaust it to atmosphere, which is feasible due to the low jet power of the low-velocity flow; the power required to do so for a supersonic flow would be enormous. Instead, continuous flow supersonic tunnels typically include a return loop, so that the flow entering the compressor already has a significant amount of energy. After a starting period in which all of the air in the flowpath is accelerated from rest, the power required to run a closed-loop tunnel is significantly lower than it would be without reclaiming the flow [18]. This return tunnel, requires a large number of subsystems to maintain the tunnel's operation including vacuum storage, pressure storage, air-dryers, heaters,

coolers, and flow straighteners; for simplicity's sake, many of these subsystems have been omitted from fig. 2.3, but should be considered in the detailed design of such a system. Of particular necessity, however, is the presence of turning vanes when the flow necessarily turns so as to return to the compressor's inflow, as is detailed in fig. 2.3. Omitting turning vanes at the corners in a flowpath would cause significant flow separation and a loss in total pressure in the system, which would have to be recovered by the compressor at great additional power cost [18].

The compressor increases the total pressure of the flow to that which is required for the test case. It can take a significant amount of time to start a continuous flow supersonic wind tunnel as the compressor itself, which must be quite massive for a large flow rate, reaches the necessary speed. It is also known that a compressor has a maximum pressure ratio it can achieve at a given flow velocity in the tunnel; its performance increases as the flow speeds up [18].

In order to provide a particular test condition, it is often necessary to pump the pressure in the loop either up or down from standard atmospheric pressure [18]. This requires large pressure or vacuum storage tanks, as well as auxiliary compressors or vacuum pumps to maintain the correct pressure in the tunnel as large circuits inevitable leak[18]. One notable addition to a continuous flow wind tunnel is a diffuser. It is known that the process of accelerating a supersonic flow through expansion is a nominally isentropic process, but that forcing the flow to undergo a compression, forcing it to slow down or turn causes a shock wave, which necessitates an increase in entropy, and therefore loss in stagnation pressure [18]. The more violent the deceleration of a supersonic flow, the closer it comes to a normal shock, which is the maximum increase in entropy possible for a supersonic flow, and if the flow is not decelerated to subsonic before encountering the compressor, a normal shock is unavoidable and potentially harmful to the compressor itself [18]. Since the purpose of a compressor in a continuous flow supersonic wind tunnel is to maintain the

Figure 2.3: Simplified diagram of a Closed-Loop Supersonic Wind Tunnel [18]

proper stagnation pressure in the flow, the greater the losses in bringing the flow back down to subsonic speeds after passing through the test section, the greater the power required by the compressor to maintain the test condition. Therefore, a diffuser is typically implemented, the purpose of which is to gradually decrease the flow Mach number to below 1 after the test section so as to minimize losses. Just as the nozzle takes time to start when the tunnel is first turned on, so too does the diffuser [18].

Because these systems are so large and expensive to produce, they are typically designed with variable nozzle geometry so as to allow testing at different Mach numbers, thus maximizing the utility of the necessary hardware. A case in which this concept is very well implemented in the Unitary Plan tunnel at NASA Ames, which uses one compressor to power the flowpaths for three separate test sections, allowing testing from Mach .2 to 3.5, including the transonic regime, although current capability of the tunnel is currently limited to Mach 2.55 [7]. For reference, the current operation

test sections are 11 ft.x 11 ft. and 9 ft.x 11 ft., and the power required to drive one loop of the system is on the same order of magnitude as the power consumption of a city of 150,000 people [7].

2.2.2 The Blowdown Wind Tunnel

A blowdown wind tunnel is the primary type of intermittent tunnel in use today [18]. As can be seen in fig: 2.4, it consists of a pressure vessel, flow regulator, test section, diffuser, and usually a vacuum chamber, although it is possible to exhaust to atmosphere given a high enough stagnation pressure to achieve the necessary pressure-ratio across the nozzle. Upon the initiation of a test, the valve between the nozzle and the pressure chamber is opened which initiates the flow of air through the system. It passes through the pressure regulator, which limits the total pressure to the desired test condition. If the pressure regulator were not included, the total pressure seen

Figure 2.4: Basic layout of an intermittent blowdown wind tunnel [18]

in the test section would change continually throughout the test. These regulators can be quite expensive and temperamental, especially for vthe high mass flow rates that would be required for large test sections [18]. It should also be noted that even though the stagnation pressure is constant through a test, the stagnation temperature

decreases over time since the air in the high-pressure reservoir cools over time as it decreases in pressure [18]. This problem can be mitigated through complex heating methods [14].

After leaving the pressure regulator, the flow passes through a diffuser and enters the settling chamber, also known as a plenum, which is a section of large area compared to the duct leading from the regulator, and which usually contains flow straighteners to condition the flow [18]. The conditioned flow then enters a converging-diverging nozzle, which accelerates it to the proper Mach number as it enters the test section. The flow is then usually brought down to a subsonic Mach number through a diffuser, then enters the dump tank or is exhausted into the atmosphere, depending on the tunnel's design [18].

Systems exist which heat the flow as it leaves the large pressure vessel, thus raising the stagnation temperature and increasing the Mach and Reynolds numbers that can be achieved [18]. Blowdown tunnels can reach well into the hypersonic testing regime using this method, although it demands a significant increase in cost and complexity to the system [18].

The Indraft Wind Tunnel

There exists a concept for a blowdown wind tunnel which does not include a pressurized section upstream of the nozzle, but rather draws air directly from atmosphere, through the nozzle and test section, and into the vacuum dump tank, which is shown in fig. 2.5. This system is known as the Indraft Wind Tunnel [18]. A major advantage of an indraft supersonic wind tunnel is the total temperature of the flow does not change over the course of the test. Additionally, total pressure remains absolutely constant without the need of a regulator, therefore Reynolds number also remains constant [18]. The primary disadvantage is that Reynolds number cannot be con-

trolled as the stagnation conditions of the flow are subject to atmospheric conditions [18].

Figure 2.5: Indraft Wind Tunnel Diagram [18]

2.2.3 The Ludwig Tube

The simplest and least expensive architecture for producing supersonic flow testing is a Ludwig Tube, called a Tube Wind Tunnel by Hubert Ludwig in his paper outlining its conception [14]. A Ludwig Tube consists of a driver tube, nozzle, test section, diaphragm or fast-acting valve, and dump tank as seen in fig. 2.6. A diaphragm is placed either between the nozzle and the tube, or the nozzle and the test section, and everything on the dump tank side of the diaphragm is brought to vacuum, while everything on the tube side of the diaphragm is pressurized, usually with air or nitrogen. At the start of a test the diaphragm is burst, the pressure differential between the vacuum section and the pressurized section accelerates the compressed gas through the converging-diverging nozzle, causing an expansion wave to propagate down the tube starting at the nozzle end; the head of these expansion waves necessarily travels down the tube at the speed of sound of the resting gas in the tube (fig. 2.7) [15]. Until the expansion wave propagates down the tube, reflects off the end, and returns to the nozzle entrance, the flow conditions are completely

Figure 2.6: Basic layout of a Ludwieg Tube [1]

unchanged with time if viscosity is neglected [14]. When the expansion wave returns to the nozzle, however, the stagnation conditions of the flow drop sharply, resulting in a test condition at a significantly lower Reynolds number. If the dump tank has a large enough volume, several tests can be run with a single diaphragm burst by counting the flow after each reflection as a different test condition.

After exiting the nozzle and reaching the designed Mach number, the flow enters the test section where it interacts with an aerodynamic model. The flow then exits the test section and expands into a dump tank in order to keep the pressure low enough for the flow of gas to continue through the nozzle.

As with the blowdown and continuous flow tunnels, Mach number is usually limited either by water condensation or air liquefaction in the test section; however, if sufficiently dry air is used in the driver tube, which is especially easy to accomplish due to the relatively low pressure storage volume of a tube tunnel compared to a blowdown tunnel, worries of water condensation in the test section can be completely eliminated. The gas in the driver tube can be initially heated to delay the onset of liquefaction to higher Mach numbers, although this significantly increases the complexity of an otherwise simple system. Additionally, increasing the temperature in

Figure 2.7: Ludwieg Tube Operation [1]

the driver tube also increases the speed of sound in the gas, and since the expansion waves caused by the diaphragm burst travel at the speed of sound of the resting driver-tube gas, the time it takes for the waves to reflect and reach the nozzle can decrease significantly, decreasing the useful test time as a result [14]. Additionally, Reynolds number can be easily changed by either changing the composition of the gas in the driver tube or by simply increasing the pressure, and therefore density of the flow.

It would seem desirable to simply increase the driver tube length of a Ludwieg tube to immense proportions, thus guaranteeing that the expansion wave will take a very long time to return to the nozzle entrance. Unfortunately, doing this would cause a severe loss in stagnation pressure as time passes due to boundary layer growth in the tube [14]. As time passes, the air that reaches the nozzle has been traveling for a longer distance, and therefore has a significantly thicker boundary layer than the flow entering the nozzle initially [14]. Logically, by increasing the tube's diameter, the ratio of boundary-layer thickness to pipe diameter decreases, limiting the negative effects of boundary layer growth; by keeping the tube length between 50 and 100 tube diameters long, the loss in stagnation pressure can be limited to below 1% [14].

Ludwig tubes have the advantage of generally being inexpensive to produce and operate, are relatively simple in design, and have the longest test durations by far of any variety of shock tunnel, sometimes exceeding 100 ms. Due to the onset of liquefaction in air at low temperatures, there is a limit of achievable Mach number of around 4 without additional heating of the gas. Additionally, the expanded flow is typically very cold (as its total temperature is room temperature), so velocities representative of a supersonic or hypersonic flight regime cannot be reached. Both of these limitations may be mitigated by heating the gas prior to bursting the diaphragm; this method allows for high velocities and Mach numbers higher than 4, but adds a very large amount of complexity and cost (both operational and capital) to the system.

Placing the diaphragm between the nozzle and the pressurized section maximizes the Mach number that can be reached, as stagnant air does not have to be evacuated from the nozzle before reaching the desired steady flow can be produced. This method can add perturbations to the flow due to the irregular orifice created by the diaphragm remnants, and so may be undesirable. Placing the diaphragm after the nozzle makes for a much smoother flow, but limits the achievable Mach number, as the nozzle start time is greatly increased in this case, to the point of potentially never fully starting depending on the Mach number and dump tank size and pressure [15].

Diaphragm Design

Although they may still be quite high, the pressure seen in Ludwig Tubes are typically much lower than those in shock tunnels, so the diaphragms required are often much less sturdy. In fact, it is often reasonable to forgo metals completely and instead use polymers such as Mylar or cellophane [10]. Some of these materials have yield strength on par with lower grade steels, and nearly eliminate the concern of large pieces of debris breaking off and damaging the tunnel or test model, as there is so

little material in the diaphragm that the prospect of such damage is unlikely. The concern remains, however, that pieces of the diaphragm may disturb the flow in the test section [15].

Diaphragms are typically burst via physical means such as a mechanical puncture device or gradually raising the pressure in the driver tube to above the rupture pressure of the diaphragm [10]. The former can introduce significant perturbations in the flow if great care is not taken, while the latter introduces uncertainty in test time unless a double diaphragm system is utilized with a fast-acting valve. Running a high current through a thin wire or layer of conductive material in contact with the diaphragm can also cause it to heat and rupture [10]. Another method exists in which the diaphragm can be caused to fail due to the application of high intensity electromagnetic radiation such as a laser. This would cause a small portion of the diaphragm to rapidly increase in temperature until the mechanical properties of the diaphragm degrade to the point of failure, causing a rupture. This method would eliminate the need to introduce a foreign object into the flow, and, given a laser with a sufficiently high intensity, can be on the order of milliseconds. Additionally, since high intensity lasers in the visible spectrum would add a significant safety hazard to lab operations, it is possible to use a laser of sufficiently high intensity in the infra-red spectrum, virtually eliminating the risk to vision damage.

2.2.4 The Shock Tunnel

Similarly to Ludwig tubes, Reflection Shock Tunnels contract and expand a high pressure gas through a nozzle and pass it over a model. Reflection Shock Tunnels, however, achieve their high pressures and temperatures dynamically, rather than statically. A shock tunnel consists of a high-pressure section and low pressure section separated by a diaphragm, as well as a test section, converging-diverging nozzle, and

dump tank, which are separated from the pressurized sections by a second diaphragm (fig. 2.8). The high-pressure gas is commonly referred to as the driver. The low-pressure gas is referred to as the test gas or driven gas. Downstream of the diaphragm is a long tube, which is occupied by the test gas and is often of constant cross section (fig: 2.8).

Figure 2.8: Shock tunnel in its initial state before the diaphragm is ruptured [1]

Through some method of excitation, the driver section is brought to a high pressure (and oftentimes high temperature), and the diaphragm separating the driver (1) and driven (4) sections is broken. This allows the driver gas (which is at a much higher pressure than the driven gas) to accelerate down the tube. This gas quickly exceeds the speed of sound of the driven gas, so a traveling normal shock wave forms in the driven gas [1]. This wave accelerates to a constant speed and propagates through the driven gas as it is driven by the light driver gas [1]. At the same time, a series of expansion waves travel backwards from the diaphragm, to the back of the driver tube (fig. 2.9) [1]. It is by varying the initial pressure and soundspeed ratios, P_4/P_1 and a_4 that the characteristics of this shock amy be varied [10].

The shock wave imposes on the test gas a high total pressure and high enthalpy. When the traveling shock hits the nozzle end-plate, it reflects back down the tube,

Figure 2.9: Shock tunnel during operation as the shock wave propagates down the tube [1]

leaving behind it a region of stationary gas at very high pressure and temperature denoted in fig. 2.10 as region (5). This gas then accelerates through the nozzle before entering the test section. At this stage, the temperature and pressure in sections (1) and (4) are unchanged, section (3) is at a lower temperature and pressure than it started, and its flow velocity is the same as that for the contact line and section (2) (fig. 2.9). The gas in section (2) has a higher temperature, pressure, and velocity than when it started [1].

Figure 2.10: Shock tunnel during operating after the shock has reflected and gas begins flowing through the nozzle [1]

Driver Methods

As a general rule, it is easier to obtain high Mach numbers with gases that have a low molecular weight such as helium or hydrogen, since the maximum strength shock is attained when the ratio of the speed of sound in the driver section, (4), to that in the driven section, (1) [10].

$$ShockStrength \propto \frac{a_4}{a_1} \quad (2.18)$$

Since the speed of sound of a gas is dependent on the ratio of specific heats and the gas' ideal gas constant, as is seen in eq. 2.7. Since the molar mass of a gas plays a large role on the specific ideal gas constant of the gas, generally speaking, the lower the molar mass of a gas, the higher the speed of sound at a given temperature. To show this relationship, a list of common driver gases has been compiled in table 2.3, and the speed of sound at 300 Kelvin has been calculated for each.

It should be noted that this relationship is different than that which governs the expandability of a gas, or the propensity to expansion that it has. As can be seen in eqs. 2.9-2.10, the higher the ratio of specific heats γ , the smaller the decrease in temperature and pressure is required to reach a particular Mach number. Accordingly,

Table 2.3: Properties of Common Driver Gases

Gas	Cp, kJ/kg K	γ	MM	R, J/kg K	a, m/s
Air	1.01	1.4	28.967	287	347
H_2	14.32	1.4	2.016	4154	1321
He	5.19	5/3	4.003	2077	1019
Ar	.52	5/3	39.948	208.1	322.6

a higher temperature driver gas will have a higher speed of sound a_4 than it would at room temperature, and therefore will form a stronger shock.

Many tunnels are also defined as being either high enthalpy or low enthalpy. High enthalpy tunnels attempt to produce both high Mach numbers and high velocity flow as would be seen in supersonic or hypersonic flight, whereas low enthalpy tunnels merely attempt to achieve high Mach numbers while allowing the temperature to get very low in some cases [18]. A high enthalpy tunnel will always have a higher velocity flow than a low enthalpy tunnel given the same Mach number and driving gas.

As was previously mentioned, the two most common light gases used for shock tunnels are Hydrogen and Helium. Hydrogen is monetarily inexpensive to use and can reach higher Mach numbers than any other driving gas, but it can be dangerous to use because of its high flammability. This becomes a major safety concern in high enthalpy Hydrogen tunnels, also known as Hot Hydrogen tunnels, because Hydrogen is often pressurized and heated to above its auto-ignition temperature [5]. Any leak in a system like this would be disastrous and could result in loss of life if proper safety precautions are not taken.

One way to obtain a high enthalpy, Hydrogen-driven flow while mitigating the risk of having pressurized hydrogen above its autoignition temperature is to use a mixture of Hydrogen and Oxygen at room temperature, and ignite the mixture; using this method also has the added benefit of not requiring a device to burst the diaphragm, as the diaphragm can easily be designed to burst from the high-pressure spike caused by a detonation [5]. Helium has a very close molecular weight to Hydrogen and therefore can also be used to achieve very strong shocks. As a noble gas, it is inert, and therefore does not pose a fire hazard as Hydrogen does. Helium is approximately 50 times more expensive, however, and large tunnels sometimes capture and reuse helium after every test.

Air or nitrogen can be used as a driver gas, but as can be seen in tab. 2.3, this limits the shock strength, and therefore stagnation conditions that can be obtained

in the driven gas. This would be the least expensive driver option, however, as only an air compressor would be necessary.

The driver gas can also be pressurized using a physical piston, as is famously done with the T5 tunnel at GALCIT. This piston is typically large and heavy, and is driven by a compressed gas at very high pressures [5]. The piston is released, gains kinetic energy, and rapidly compresses the driver gas, bringing it to a very high temperature and pressure. This method is very energy efficient, but mechanically difficult and expensive to implement [5].

Diaphragm Design

The diaphragms used to separate the high pressure, low pressure, and vacuum sections of a shock tunnel are typically constructed of high strength materials such as stainless steel or Aluminum [10]. They are often scored, usually in a cross pattern, to ensure that the diaphragm bursts at the desired pressure, and that the diaphragm opens sufficiently wide to allow the passage of the driving gas down the tube. Special care is taken in the design to ensure that fragments of the diaphragm do not come off and flow into the test section where they may damage test section and model hardware, in addition to disturbing the flow [5].

Diaphragm bursting can be controlled in a number of ways; the diaphragm can be precisely calibrated and tested to burst at a certain pressure, it can be physically punctured with an arm that retracts once it punctures the diaphragm wall, and a small pyrotechnic charge can be mounted to the diaphragm, which would cause it to burst when detonated [5]. Additionally, a double-diaphragm system exists in which two diaphragms are placed close to one another, with the ability to depressurize the space in between the diaphragms. This space is rapidly pressurized, causing the downstream diaphragm to burst first, quickly followed by the upstream diaphragm.

Using a detonation driver decreases the precession needed for the burst pressure of the diaphragm.

2.2.5 Test Section Design and Data Acquisition

Reynolds Number

Reynolds number is an important relation to consider in the design of an aircraft. It is dimensionless and, as can be seen in the following equation, relates the inertial forces present in a flow (density, velocity), with friction, viscosity. Therefore, the higher the Reynolds number, the less of an effect viscosity has on the flow [17].

$$Re = \frac{\rho U L_c}{\mu} \quad (2.19)$$

where Re is Reynolds Number, ρ is density, U is velocity, L_c is a characteristic length (usually the length of the test model or diameter of the test section) and μ is the kinematic viscosity.

With few exceptions, the test sections of wind tunnels used to test aircraft are much smaller than the aircraft itself, and so a scale model must be built simply so it can fit in the tunnel. As can clearly be seen in the above equation, decreasing the size of a model will also decrease the characteristic length, which changes the Reynolds number. Therefore, it is often necessary to modify the test condition so as to attempt to match Reynolds number in the tunnel as closely as possible to that seen by the full-sized aircraft in free-flight. By matching Reynolds number, the boundary layers in the flow are said to be representative of those on the full-sized aircraft [18].

There are several ways typically used to change the Reynolds number in a wind tunnel. As has already been shown, changing the model size can change the characteristic length, so it is oftentimes beneficial to attempt to test in the largest configuration possible to relax the need to vary other parameters of the flow. Velocity can

be changed, and since it is dependent both on Mach number and speed of sound, as is seen in eq. 2.7, and since the speed of sound is dependent on temperature, increasing the temperature of the flow necessarily increases the flow velocity at a particular Mach number, increasing the Reynolds number. Additionally, the density of the flow may be varied, which can be accomplished by decreasing or increasing the stagnation pressure in the flow; this is an especially useful technique in blowdown wind tunnels. The test gas composition can also be changed.

Wind tunnels typically list their Reynolds number capabilities, but do so by dividing their attainable Reynolds number at a particular Mach number by the test section width dimension, giving values in Reynolds number per foot or per meter. In this way a customer is able to easily determine the Reynolds number of a test article by simply multiplying the characteristic length of the test article by the Re/L value given by the tunnel.

In the case of indraft style wind tunnels, Mach number can be controlled, but temperature and density are fixed because total temperature and total pressure are both standard for atmosphere. This makes it very difficult to match Reynolds number to a flight condition, especially at very high Mach numbers. Although the Ludwig tubes reserve some Reynolds number control by varying tube pressure and therefore density and, in some cases, increasing stagnation temperature in the tube, Reynolds number matching is still a significant limitation at high Mach numbers. These effects, in conjunction with the relatively small size of Ludwig tube test sections compared to those commonly implemented for blowdown and continuous flow tunnels, mean that low temperature tunnels are much more valuable in modeling shockwave effects on models than in predicting viscous effects. This is not to say that low temperature tunnels are not valuable, rather that their value is limited to certain areas. Reflection shock tunnels have a high degree of control over Reynolds number given their ability to attain extreme temperatures as well as Mach numbers; it is for this reason that

shock tunnels are used almost exclusively in hypervelocity testing in applications such as atmospheric reentry [12].

Liquefaction and Water Condensation

Two potential limiting factors in achievable Mach number in high speed testing are the effects of liquefaction and water condensation. As can be seen in eqs. 2.9 and 2.10, bringing a flow from low to high Mach number caused the temperature, pressure, and density of the flow to decrease. As temperature decreases, the capacity of air to contain water vapor also decreases. Below a certain point, this vapor condenses, which can cause a fog to form, obscuring the model and preventing the use of flow imaging techniques, as well as damaging hardware and causing incorrect total pressure readings in the flow, as the condensed water is not compressible [18]. The point at which water condenses is known as the dew point, as is defined in the following equation [18]:

$$\log_{10}P = \frac{-605.4}{T} + 4.114 \quad (2.20)$$

Where P is pressure in atm, and T is temperature in °R. It is important to account for and attempt to prevent water condensation in the test section; as such, almost all high speed flow facilities implement air dryers to pull water out of the flow, or, in the case of Ludwieg tubes and shock tunnels, pre-dessicated air is typically used. It is important to note that the negative effects of condensation do not always come into play simply with temperature drop [18]. Since significant adverse effects require the formation of water droplets, and droplets take time to form from an evenly distributed vapor, these effects oftentimes do not present themselves during very fast cooling periods, such as through a short converging-diverging nozzle [18]. Although eq. 2.20 is true for the static case, significantly more cooling is achievable than this relation would suggest when applied rapidly, although the precise amount varies depending on the

application [18]. Additionally, decreases in temperature of less than 55 R° are very unlikely to cause test-section condensation, regardless of the starting humidity.

The second phenomenon, liquefaction, occurs when the working gas is cooled to the point at which its constituents change phase from a gas to a liquid [18]. In order to prevent this occurrence, the working gas must be heated to raise the total temperature to the point at which the static temperature never drops below the liquefaction limit. When accelerated from standard temperature, this limit is typically reached between Mach numbers of 4.2 and 4.6 for air [18]. It is possible that significant liquefaction do not necessarily skew some data, such as force and moment measurements on models. It is recommended to attempt to prevent liquefaction wherever possible [18].

Shadowgraph and Schlieren Imaging

Two common techniques for identifying shock structures in supersonic testing are shadowgraph and Schlieren photography. For these setups, a light source is collimated and passed through the test section [18]. Since light refracts when it passes through regions of varying density, and shockwaves cause sharp changes in density, when the collimated beam passes through a shockwave, shock structures can be easily identified when the beam is refocused into a camera; a basic setup can be seen in fig. 2.11 [18]. For a Schlieren setup, a knife edge is placed at the focal point of the lens that focuses the beam of light into the camera; this way, any light deflected by changes in density in the flow hits the knife edge and is completely blocked. A shadowgraph is simply the same setup without the knife edge. Since all of the light from the image is collected in the case of a shadowgraph, more large changes in density can be seen, but smaller changes are difficult to pick up; Schlieren photography, on the other hand, is very sensitive, and even the most minor changes in density can be seen [18]. Focusing

mirrors are often used in lieu of lenses for these setups, especially when large beam widths are needed.

Figure 2.11: Diagram of a basic Schlieren setup.

Errors in image transmission from one lens to another in any optical setup is known as aberration [8]. Three forms of aberration which may be of particular concern to shadowgraph and Schlieren imaging techniques are as follows:

- Comatic aberration, or coma, which arises from misalignment of the image from the lenses and may cause images to be distorted, unfocused, and exhibit a kind of tail not present in the perfect image.
- Spherical aberration, which always arises from the uncorrected use of a spherical focusing lens, but can also occur with other lens designs, shows up as a general defocusing of the image, because it causes a longitudinal offset of the focal point, meaning that light rays traveling through the lens at different distances from the longitudinal axis of the lens are focused on different focal lengths, but all back onto the same axis. This produces ring in an image, severely distorting it

[8]. It is essentially impossible to collimate light using a spherical lens without a more complicated optical system in place to account for this aberration [8].

- Collimating light from a point source produces divergence in the collimated beam which is dependent on the angle of light traveling from the point source to the lens. This angle is determined by the relative diameter of the lens to the point source, and the distance apart [8]. The effects of beam divergence become greater the farther the beam travels. To reduce this divergence and subsequent loss of sharpness, a larger lens to focal length diameter is required, or a smaller point source [8].

Chapter 3

PRELIMINARY WORK AND ANALYSIS

A significant amount of work was required to bring existing hardware and infrastructure into satisfactory operation for this project. The state of these facilities, particularly the Cal Poly Aerospace Engineering Department's recently acquired large vacuum chamber, which were necessary for the realization of this project, were in an advanced state of disrepair at its inception. While the contents of this section are not directly related to the focus of this thesis, it was necessary for the implementation of the project.

3.1 Vacuum Chamber Refurbishment

All possible supersonic wind-tunnel test architectures considered for this project required the use of a large vacuum chamber to operate as a dump tank. As the large vacuum chamber recently acquired by the Cal Poly Aerospace Engineering Department was missing much of its necessary hardware such as functioning gaskets and flange covers, and a large portion of the chamber was in a state of advanced oxidation, an effort was undertaken to return the chamber to operability, thus enabling its use as a dump tank initially, and in the long term, a fully operational vacuum chamber. To accomplish this task, the sealing surfaces of the many flanges were reconditioned, and replacement flanges, fasteners, and gaskets were designed and sourced. A vacuum feed-through was also added to the system. In its final state, this chamber will have dual operation as a supersonic wind tunnel and high vacuum chamber, so it is necessary when bringing it into operation to take into account that both modes of operation and the requirements they will entail.

Figure 3.1: Vacuum Chamber in its Original State. Pictured are all four access flanges.

3.2 Rust Management

When received, the chamber was coated in a layer of white paint, which protected most of it from the elements, as it sat outside for years without protection from the wind and rain. Unfortunately, the surfaces which were not painted, the sealing surfaces of the many flanges and the door, pictured in figs. 3.1 and 3.3, are also the most important for function as a vacuum chamber. The quality of the seal that can be attained on each of these access points is directly dependent on the surface finish quality on each and the material and shape of the sealing gasket. The presence of rust represents a massive problem in this area, as iron oxide has a different density than iron, which means that a thick oxide layer is able to form as the oxide expands and flakes off, making the surface finish poor and unsuitable for vacuum operation.

This oxide layer was easily removed with light abrasives, but the biggest problem to seal quality lies in the deep pitting which was left by the rust on the the chamber's many flange faces. In order to ensure a surface quality sufficient to attain a seal, the

Figure 3.2: Vacuum Chamber in its Original State

faces were sanded down in progressively finer grits of sand paper. This process was arduous, but after much labor and polishing, the surfaces became much improved and, once the new flanges and gaskets were installed, vacuum was successfully pulled on the system. The details of this test can be seen in fig. 6.9.

3.3 Flange and Fastener Design and Selection

Each of the four flanges built into the chamber as access ports are of the following standard sizes, and so solid covers with matching bolt-hole patterns and sizes were sourced to cover them (tab. 3.1). The method for defining the flange and gasket dimensions which was used in tab. 3.1 can be seen in fig. 3.4.

3.3.1 Flange Cover Material Selection

In order to prevent further oxidation from forming in the new chamber hardware, it was decided that the material must be corrosion resistant. The two primary materials considered were 6061 Aluminum and 304 Stainless Steel, due to their widespread

Figure 3.3: Vacuum Chamber after only minor refurbishment, showing the large door which gives access to the chamber’s internals. The door rim was in particularly poor condition at the outset of this project.

Table 3.1: Flange sizes; all dimensions in inches.

Quantity	Pipe	OD	Bolt Holes	Bolt Hole Dia.	Bolt Circle	Thickness
1	20	27-1/2	20	1-1/4	25	1-3/8
2	10	16	12	1	14-1/4	1/2
1	6	11	8	7/8	9-1/2	1/2

availability from suppliers. 304 stainless has a comparable yield strength that of 6061 T6 Aluminum; 31.2 ksi compared to 35 ksi, respectively. The major disadvantage was the significantly higher density of 304 stainless compared to 6061 Aluminum. The density of 6061 Aluminum is 2700 kg/m^3 compared to 8000 kg/m^3 for 304 stainless. Since the thicknesses required for structural integrity would be similar due to the similar yield strength, the total mass of material for the steel would need to be significantly higher. This is a minor concern for the smaller flange plates needed for the chamber, but the largest plate is $1 \frac{3}{8}$ " thick and 27.5" in diameter, so the

Figure 3.4: Diagram showing the dimensioning layout of ASME standard flanges and full-face gaskets to match.

mass would be 107 kg for Stainless Steel, compared to 36 kg for Aluminum. Because a sizable nozzle also needed to be attached to large flange plate, and that assembly needed to be fitted to the chamber, the total mass would have been too high to handle without mechanical lifting equipment if stainless steel was used.

The surface finish required for vacuum sealing is routinely stated to be 32 rms or root mean square, in units of micro-inches [3]. This surface finish is easily attainable with either material, but Aluminum is much easier to mar and deform than stainless, so stainless flange plates will retain their surface finish much more easily. In the long term, especially for high vacuum applications, it is recommended that 304 or 316 stainless steel be used to seal all flanges on the chamber except for perhaps the largest one, but 6061 Aluminum will be acceptable for this project.

Figure 3.5: Vacuum Chamber after replacement of large 27.5” diameter flange cover.

3.3.2 Gasket Selection

When selecting gaskets and flanges for high vacuum application, there are several things that need to be considered. There are two basic types of flange/gasket combinations that can be used: CF flanges, and plain-face flanges with elastomeric (rubber) gaskets. CF flanges use precisely machined Flanges with a sharp knife edge machined into the face [3]. When compressed together, the knife edges bite into a soft metallic gasket such as copper, creating a seal. CF flanges are the standard for high and ultra-high vacuum applications as they produce far lower leak rates than elastomerically sealed gaskets; they are capable of sealing to pressures below 10^{-13} Torr [3]. Unfor-

Figure 3.6: Vacuum Chamber after replacement of three side flange covers.

Unfortunately, each gasket can only be used once, as tightening a flange pair permanently deforms the gasket, making it unusable once the flanges are opened¹. This can cause high cost of operation if the flange in question must be opened and closed often [3]. It is necessary that both flange faces have a machined knife edge that can bite into the metallic gasket, which means that the flange faces on the chamber itself would have to be expertly machined, which would be expensive to accomplish. It is, however, possible to use CF style flanges on the feed-throughs that pass through the flange covers themselves. This would require custom machined flange covers with the knife edges machined into the surface.

Elastomerically sealed flanges use an elastomer (rubber) as the sealing gasket. This gasket can either take the form of an O-ring, or a full-face gasket as seen in Fig. 3.4 [3]. Elastomeric seals present three potential modes of leaking: leaks between the gasket and the flange, gas transmission through the elastomer, and outgassing

of the elastomer itself [3]. Outgassing of the gasket itself can be mitigated by good material selection, as well as by baking out the gasket prior to use. Buta-N or butyl rubber gaskets are the cheapest to purchase (approximately 5-10 dollars per gasket), and allow for sealing to moderate vacuum levels, but allow gas permeation and have moderate outgassing preventing ultra high vacuum levels [3]. In order to achieve ultra high vacuum levels it is recommended to use fluorelastomers or fluorocarbons, a common version of which is Vitron, which are much more expensive (200 dollars per gasket) [3]. Fluorocarbon gaskets allow 20x less water permeation than butyl gaskets [3]. It is therefore recommended that butyl gaskets be purchased to be used during initial refurbishment of the chamber and operation as a wind tunnel, in which the vacuum requirements are relatively low, and a set of fluorocarbon gaskets be purchased when the chamber is ready to operate as high level vacuum chamber.

3.4 Vacuum Feed-Through Selection

The most important feed-through required was a simple large-port gas feed-through to allow a vacuum pump to be attached to the chamber. Since the Space Environments Laboratory at Cal Poly possesses a large number of standard KF-40 vacuum fittings for vacuum pumps, bellows, and instrumentation, it was decided to drill a 1 inch diameter hole in the Aluminum replacement window and attach a KF-40 baseplate feed-through. This allowed for a vacuum pump to be connected via bellows, and for pressure measurements to easily be taken of the chamber over time.

Chapter 4

DESIGN SELECTION CRITERIA, METHODOLOGY, AND ANALYSIS

As discussed in the background section, the primary methods for producing a supersonic flow for the purposes of testing that were considered are a closed loop compressor driven tunnel, blow down wind tunnel, shock tunnel, Ludwig Tube, and the novel Indraft Tube Tunnel. Due to their high requirements in space, capital investment, and cost of operation, the compressor-driven and blow-down wind tunnels were quickly eliminated. To best fit the needs and requirements of the Cal Poly Aerospace Engineering Department, further analysis was required to choose between a reflection shock tunnel, Ludwig Tube, and ITT. This Analysis is presented in the following sections.

4.1 An Analysis of a Reflection Shock Tunnel

4.1.1 Governing Equations of a Shock Tube

The initial composition, pressure, and temperature of the gases in sections (1) and (4) determine the behavior of the entire shock-tube system [10]. Although the series of equations required for this analysis begins with analyzing the shock relations for a traveling shockwave, and ends with obtaining what is called the Shock Tube Equation, (eq. 4.1) efficient analysis of an actual system begins with the Shock Tube Equation, which finds the shock strength in terms of $\frac{P_2}{P_1}$ given the initial pressure ratio $\frac{P_4}{P_1}$, and the gas properties in regions (1) and (4) [10]:

$$\frac{P_4}{P_1} = \frac{P_2}{P_1} \left[1 - \frac{a_1}{a_4} (\gamma_4 - 1) \frac{\frac{P_2}{P_1} - 1}{\sqrt{2\gamma_1 [(\gamma_1 + 1)\frac{P_2}{P_1} + (\gamma_1 - 1)]}} \right]^{\frac{-2\gamma_4}{\gamma_4 - 1}} \quad (4.1)$$

Note that this general equation allows for different driver and driven gases to be used, which is valuable due to the benefit of light gas drivers. Also for easy reference, figs. 2.9 and 2.10 are included here. The Mach number of the shockwave, M_s , can then be

Figure 4.1: Shock Tunnel [1]

found in terms of $\frac{P_2}{P_1}$, followed by the shockwave velocity, W , in the lab frame [1]:

$$M_s = \sqrt{\frac{\gamma+1}{2\gamma} \left(\frac{P_2}{P_1} - 1 \right) + 1} \quad (4.2)$$

$$W = a_1 M_s \quad (4.3)$$

Temperature and pressure in region (2) can also be found [1]:

$$\frac{\rho_2}{\rho_1} = \left[\frac{1 + \frac{\gamma+1}{\gamma-1} \left(\frac{P_2}{P_1} \right)}{\frac{\gamma+1}{\gamma-1} + \frac{P_2}{P_1}} \right] \quad (4.4)$$

$$\frac{T_2}{T_1} = \frac{P_2 \rho_1}{P_1 \rho_2} \quad (4.5)$$

$$u_p = W \left(1 - \frac{\rho_1}{\rho_2} \right) \quad (4.6)$$

Shock Reflection

The strength of the reflected shockwave as it returns back along the tunnel, passing through gas from region (2), is described as follows [1]:

$$\frac{M_R}{M_R^2 - 1} = \frac{M_s}{M_s^2 - 1} \sqrt{1 + \frac{2(\gamma - 1)}{(\gamma + 1)^2} (M_s^2 - 1) \left(\gamma + \frac{1}{M_s} \right)} \quad (4.7)$$

Where M_R is the reflected shockwave Mach number. The reflected normal shockwave behaves just as the incident shockwave does, with the exception that the flow it propagates into is not stagnant in the laboratory coordinate frame; since the shock relation equations must be evaluated in the wave's reference frame, however, nothing fundamental about the analysis changes. The stagnant flow properties in section (5) may be found by relating the following equations to the properties already found for section (2). First, eq. 4.2 should be reordered and relabeled to allow for the shock strength to be found in terms of the reflected shock Mach number [1]:

Figure 4.2: Shock Tunnel [1]

$$\frac{P_5}{P_2} = 1 + \frac{2\gamma}{\gamma + 1} (M_R^2 - 1) \quad (4.8)$$

Now that the reflected shock strength $\frac{P_5}{P_2}$ is known, eqs. 4.4 and 4.5 can be easily modified to relate density and temperature in regions (2) and (5) [1]:

$$\frac{\rho_5}{\rho_2} = \left[\frac{1 + \frac{\gamma+1}{\gamma-1} \left(\frac{P_5}{P_2} \right)}{\frac{\gamma+1}{\gamma-1} + \frac{P_5}{P_2}} \right] \quad (4.9)$$

$$\frac{T_5}{T_2} = \frac{P_5}{P_2} \frac{\rho_2}{\rho_5} \quad (4.10)$$

Since the flow in region (5) is stagnant, this temperature, pressure, and density constitute the stagnation conditions for the nozzle flow and in the test sections. Since the reflected shock Mach number is relative to the moving flow in region (2), the reflected shock velocity W_R in the laboratory frame is expressed as [1]:

$$W_R = U_p - a_2 M_R \quad (4.11)$$

Note that $U_p = U_2 = U_3$, and that since the wave is moving in the opposite direction of the incident flow in region (2), the velocity is negative in the laboratory frame.

Nozzle and Test Section Flow

As was previously mentioned, the stagnated flow in region (5) defines the stagnation conditions for the nozzle and test section, thus modeling the flow behavior from this point onward is trivial and akin to that of a blowdown wind tunnel or rocket nozzle.

The mass flow rate of gas through the nozzle is determined by its stagnation conditions and the nozzle throat area, as was shown in 4.18. The total mass of gas in region (5)—which is the same as the mass of gas in region (1) prior to the start of the test—coupled with this mass flow rate determines the test duration. Once all of the gas in region (5) has been ingested by the nozzle, driver gas enters, changing the test conditions and ending the test.

The total mass of test gas, m , may be calculated from the volume and density of the driver tube:

$$V_{tube} = L_{tube} \frac{\pi D_{tube}^2}{4} \quad (4.12)$$

where V_{tube} , L_{tube} , and D_{tube} are the driver tube volume, length, and diameter, respectively.

$$m_{driven} = V_{tube}\rho_1 \quad (4.13)$$

where m_{driven} is the total mass of driven gas in region (1) and, accordingly, in region (5). By neglecting adverse effects of boundary layer growth in the tube or potential penetration of the driver gas into or around the edges of the stagnated driven gas in region (5), the useful test duration can be calculated as:

$$t_{test} = \frac{m_{driven}}{\dot{m}^*} \quad (4.14)$$

In the following subsections, these analytical methods will be used to analyze the behavior of two different types of reflection shock tunnel—one using Helium, the other air as the driver gas—and each will be traded for design compatibility for implementation at Cal Poly San Luis Obispo.

4.1.2 Consideration of a Helium Driven, Mach 5 High Enthalpy Shock Tunnel

It is possible through the use of a Hydrogen or Helium driver gas to attain reasonably high temperatures and pressures in the driven gas as seen in figs. 4.4 and 4.3, which accordingly allows high velocities to be reached in the test section. Because of the safety concerns that come with handling high pressure Hydrogen, only Helium is considered for use in this system. Note that in these plots, four curves of constant driver pressure are used, so as the ratio $\frac{P_4}{P_1}$ increases, the value of P_1 necessarily decreases, as does the attainable pressure in region (5). Driver pressure is held constant along each line as a shock tube system is typically limited by maximum pressure and temperature. Notice that with room temperature helium, temperatures in excess of 3500K are easily attainable (fig: 4.4). Achieving such a temperature without dynamic methods such as occur in a shock tube such as this is extremely difficult, which lends to their incredible usefulness in testing vehicles which must perform in the hypersonic

Figure 4.3: Post Reflection Stagnation Pressure for a Helium-Driven Shock Tunnel, with Several curves of constant Driver Gas Pressure, P_4

flight regime, especially for hypervelocity flows which require test-section velocities well above what can be achieved in a blowdown or continuous flow wind tunnel.

4.1.3 Consideration of an Air Driven, Moderate Enthalpy Shock Tunnel

As has been discussed, high pressure Helium is much more expensive than air or Nitrogen, and so it may be valuable to run a reflection shock tunnel with air as a driver gas for laboratory demonstration, but use more expensive gases such as Helium for research purposes when high enthalpies are necessary. For a large range of P_4/P_1 and at several initial driver pressures, P_4 , a reflection shock tube was analyzed for total temperature, total pressure, and shock strength capability. This is meant to be a relatively quick method for gaging the capabilities of such a tunnel as it would be used at Cal Poly.

Figure 4.4: Post Reflection Stagnation Pressure for a Helium-Driven Shock Tunnel; Temperature is Independent of absolute pressure

In comparison to the Helium driven tunnel, the ability of air-driven tunnel to produce both high stagnation temperatures and temperatures is greatly diminished. As can be seen in fig. 4.5, the strength of the shock air can produce with high initial pressure ratios is not nearly as favorable as it is for Helium in fig. 4.3. Much higher pressures in region (1) are required to produce high stagnation conditions in in the air-driven case.

4.1.4 Concluding Statements on the Validity of a Shock Tunnel

As can immediately be seen, the ability to produce high stagnation conditions post-reflection is far superior with a Helium driver compared to air. In terms of high pressure handling, the use of Helium allows for a much lower initial driver pressure to obtain the same stagnation pressure as using air as a driver would require.

Figure 4.5: Post Reflection Stagnation Pressure for an Air-Driven Shock Tunnel, with Several curves of constant Driver Gas Pressure, P_4

Figure 4.6: Post Reflection Stagnation Pressure for an Air-Driven Shock Tunnel; Temperature is Independent of absolute pressure

Helium is clearly a superior driver gas to use when compared to air; however, the pressures required to produce useful, strong shocks are very high in both cases. Therefore, the cost to produce a facility using either gas will be much greater than the expense of using Helium as a driver. The primary difference in each architecture is the far more extreme temperatures attainable through the use of Helium.

The true beauty and benefit of using the shock tunnel is the ability to achieve extremely high temperature, high velocity flows that exceed the abilities of nearly any other facility to produce. Limiting such a facility such as would be necessary to make it affordable for implementation at Cal Poly eliminates the primary benefit of shock tunnels. Additionally, the extremely short test times cannot be avoided, and making smaller, simpler systems merely exacerbates this problem.

Additionally, the diaphragms necessary to contain high pressure differentials on the order of 50-100 atmospheres in conjunction with an accurately replicable system for bursting such a diaphragm requires a significantly high level of complexity as would be prohibitive given the scope of this project.

4.2 An Analysis of a Ludwieg Tube

4.2.1 Governing Equations of a Ludwieg Tube

The unsteady expansion wave generated at the initiation of a test for a Ludwieg tube is isentropic, but the stagnation conditions of the nozzle flow are not the same as the pressure and temperature at the beginning of the test as one might think [15]. As has been stated, an unsteady expansion wave must necessarily be coupled with a shockwave moving in the opposite direction. This is exactly the same process as occurs in a reflection shock tunnel, except that the emphasis and analysis is typically directed toward the properties of the driving gas rather than the driven. Considering

a shock tube, the driver gas, which is the working fluid in a Ludwieg tube, effectively acts as a piston, doing work on the driven gas. This is an irreversible process, and results in a loss of total pressure and temperature. The actual stagnation conditions for the nozzle and test-section flow in a Ludwieg tube are dependant only on the ratio of tube area to nozzle throat area and the gas characteristics, and can be found using the following process [15]. First the Mach number of flow in the tube must be solved for [15]:

$$\frac{A_t}{A^*} = \frac{1}{M_t} \left(\frac{\gamma + 1}{2} \right)^{-\frac{\gamma+1}{2(\gamma-1)}} \left(1 + \frac{\gamma-1}{2} M_t^2 \right)^{\frac{\gamma+1}{2(\gamma-1)}} \quad (4.15)$$

Where M_t is tube Mach number, A_t is tube area, and A^* is the nozzle throat area. The ratio of post-expansion stagnation conditions to tube fill conditions are as follows [15]:

$$\frac{T_0}{T_t} = \frac{1 + \frac{\gamma-1}{2} M_t^2}{\left(1 + \frac{\gamma-1}{2} M_t^2 \right)^2} \quad (4.16)$$

$$\frac{P_0}{P_t} = \left(\frac{T_0}{T_t} \right)^{\frac{\gamma}{\gamma-1}} \quad (4.17)$$

Where T_0 and P_0 are the stagnation temperature and pressure of the flow passing through the nozzle, and T_f and P_f are the pressure and temperature to which the tube was filled before bursting the diaphragm.

4.3 An Analysis of a Tubeless Ludwieg Tube

A Ludwieg tube was considered in which there was no tube upstream of the nozzle, but instead air was drawn directly from atmosphere, through a converging diverging nozzle, and into an evacuated dump tank. For this analysis the large vacuum chamber owned by the Aerospace Department will be considered as the dump tank hardware. Primary concerns for this variation on a conventional Ludwieg Tube are test duration,

Reynolds Number capability, test section sizing and instrumentation compatibility, and Mach Number limitations via air liquefaction and water condensation.

4.3.1 Initial Transient Behavior of the Indraft Tube Tunnel

Upon initiation of a test, the ITT will generate an expansion wave upstream and a shockwave travelling downstream exactly like a Ludwieg Tube. The primary difference is that there is no end to the tube. Upon exiting the inlet bell, the expansion wave changes from a linearly propagating wave of constant strength, to a spherical expansion wave, which decreases in strength according to the inverse square law.

Although validation of stagnation pressure in the flow was not possible with the experimental setup, it is reasoned that the stagnation pressure in the initial transient flow, if measured, would be markedly lower than would be expected if the flow isentropically accelerated through the nozzle.

The starting behavior of the Indraft Tube Tunnel may be different depending on the location of the diaphragm. The can either be placed in between the inlet bell and the inlet adapter, which is approximately 3.5 ft. from the nozzle contraction, or between the extension and the nozzle, which is 5 inches from the nozzle contraction. If the former location is chosen, assuming an instantaneous burst of the diaphragm and formation of a shockwave in the mostly evacuated nozzle, the shockwave velocity would be approximately 900 m/s, reaching the nozzle in just over a millisecond, then reflecting from it. Similarly to the weakening of expansion waves once exiting the inlet bell, it is also predicted that the reflected shock will exit the inlet and weaken significantly shortly thereafter. On its passage through the nozzle extension, the shockwave changes the stagnation conditions of the air present, thus changing the behavior of the nozzle even long after the shockwave exits the nozzle completely. This may cause a significant decrease in useful test duration, as the flow at the

beginning of a run will have one set of stagnation conditions, but air ingested by the inlet bell, which underwent a weakened shock (due to the inverse square law) may have gradually changing stagnation conditions until the shock becomes so weak as to no longer significantly affect the flow.

Locating the diaphragm at the inlet bell location is far preferable for ease of installation, as the inlet bell is external to the tank; the nozzle location, on the other hand, requires entry to the vacuum chamber and removal of the nozzle, which—although relatively easy—proves more difficult than disassembly and reassembly of the inlet bell. Since the nozzle diaphragm location is very close to the converging section of the nozzle, and a fully formed shockwave would take only .1 ms to reach it, it is predicted that the diaphragm's rupture time will prevent a full shockwave from forming, resulting in behavior much closer to a conventional Ludwig Tube.

It is hoped that any shock or expansion wave reflection from any obstacle upstream of the inlet bell will be so weak upon returning to the inlet bell as to have little effect on the nozzle's performance.

4.3.2 Considerations on Test Duration

The test duration is the amount of time the test rhombus immediately downstream of the nozzle exit remains at the design condition (Mach number, Temperature, Pressure, Density). Because the propagation of expansion wave from the nozzle mouth will not encounter the end of a tube as would occur in a typical Ludwig Tube, the test duration is no longer limited by this reflection as a conventional Ludwig Tube would be, as the chamber is essentially drawing from an infinite volume (the Earth's atmosphere). It is therefore necessary to investigate the nozzle's interaction with the chamber, as well as the starting process to predict test duration.

There are several modes of limitation which can affect the test duration of a supersonic wind tunnel with this basic design. Included in this list are rise in chamber pressure to above the nozzle exit pressure, and shock reflection from the walls of the chamber itself.

Immediately after the diaphragm bursts, expansion waves propagate out of the nozzle and into the atmosphere. As they pass, the previously stagnant air begins flowing through the nozzle. Immediately after the diaphragm bursts, a shockwave propagates through the nozzle, and partially reflect from its converging section, which is likely to affect the time scale in which the nozzle starts [15]. The more pronounced this affect is, the shorter the total test duration will be.

As air flows into the chamber, the density and therefore pressure inside the chamber increase proportionally to the mass flow rate of air through the nozzle. Because mass is conserved through the nozzle, the mass flow rate of air into the chamber is the same as the mass flow rate evaluated at any point in the nozzle.

Because this nozzle chokes the flow at the throat, the mass flow rate can most easily be evaluated at the nozzle throat, where the Mach number is known to be one:

$$\dot{m} = \rho^* U^* A^* \quad (4.18)$$

Where ρ^* , U^* , and A^* are the flow density, velocity, and area, respectively at the nozzle throat. A^* may be determined by using the following equation, designed nozzle exit area, and design Mach number to solve for the throat area.

$$\left(\frac{A}{A^*}\right)^2 = \frac{1}{M^2} \left[\frac{2}{\gamma + 1} \left(1 + \frac{\gamma - 1}{2} M^2 \right) \right]^{(\gamma+1)/(\gamma-1)} \quad (4.19)$$

In order to find the velocity and density at the nozzle throat, it is first necessary to find the static temperature and pressure at the nozzle throat by evaluating equations 2.9 and 2.10, respectively, at the throat Mach number of 1. Then density, ρ , can be found by using the ideal gas law, eq. 2.11, at the throat temperature and pressure.

Finally, the velocity can be found using the known throat Mach number of 1 and evaluating the speed of sound. The Reynolds number of the test flow can be calculated using the following equation,

$$Re = \frac{\rho UL}{\mu} \quad (4.20)$$

where ρ can be determined by evaluating equations: 2.10, 2.9 and 2.11 simultaneously at the nozzle exit.

$$Re = \frac{\rho M \sqrt{\gamma RT} L}{\mu} \quad (4.21)$$

Where Re is Reynolds number, and L is characteristic length. During initial steady state operation the nozzle will behave as an under-expanded rocket engine, because the exit pressure from the nozzle will be significantly higher than the initially evacuated chamber. As the pressure rises, the degree to which the nozzle is under-expanded will decrease, until it is optimally expanded, eventually over-expanded. When the nozzle flow becomes over-expanded, the flow around the outside of the nozzle will be forced to undergo a compression turn and a shock will stand against the nozzle wall, constricting the size of the test rhombus. At this point in time the test condition will be considered disturbed, and the test point is no longer valid for the original volume swept by the test rhombus. The rate at which the chamber fills and pressure increases is proportional to the mass flow rate through the nozzle, which is most easily calculated at the nozzle's throat:

Eq. 4.18 may be solved after evaluating eqs. 4.15-4.17, treating the nozzle extension as though it were a Ludwieg tube and finding the stagnation conditions of the flow through the nozzle. Once the mass flow rate is known, assuming that it is constant until the chamber repessurizes and the nozzle becomes unchoked, the average density in the chamber can be calculated at any point in time after the start of the test as:

$$\rho_{chamb}(t) = \frac{m_0 + \dot{m}t}{V_{chamb}} \quad (4.22)$$

where m_0 is the mass of air initially in the chamber, calculated from the pressure and density via the ideal gas law, $\rho_{chamb}(t)$ is average density in the chamber at any point in time, t , since the test started, and V_{chamb} is chamber volume.

If the flow is assumed to be adiabatic, which is a fair assumption due to the short duration in which this occurs, and the low-pressure air in the chamber is assumed to be stagnant, which may be a poor assumption, the temperature of air in the chamber can be assumed to be atmospheric. Therefore, the pressure in the chamber is:

$$P_{chamb}(t) = \rho_{chamb}(t)RT_0 \quad (4.23)$$

Where $P_{chamb}(t)$ is chamber pressure at any point t during the test, and T_0 is stagnation temperature. Assuming that shock reflections within the chamber do not significantly perturb the flow, the test point will continue unchanged until the point at which the pressure in the chamber exceeds the nozzle exit pressure. At this point, the nozzle will transition from operating as an under-expanded rocket engine with an expanding plume, and will begin to operate as an over-expanded rocket engine. At this point, the flow along the edges of the nozzle be forced to undergo a compression turn inward, naturally forming a shock in the nozzle and reducing the size of the test rhombus [13]. As time goes on, this shock will grow until the nozzle becomes completely unstagnated, where all flow exiting the nozzle is subsonic. For the purposes of the preliminary analytical description of this system, the test duration will be the time from diaphragm burst to the point at which the flow becomes over-expanded.

Several assumptions had to be made for this analysis to be valid, so it is expected that these predictions are overly optimistic, and that the actual useful test time when taking into account nozzle starting time and perturbations within the chamber itself will be a fraction of that predicted. The computational methods that would be required to attempt an accurate prediction of the dynamic nature of the problem of nozzle starting and reflections within the chamber could constitute a thesis in

and of themselves, and even then would be difficult to trust without experimental verification. Therefore, the analysis in the preceding equations is presented as a first order of magnitude prediction of test duration, but the actual test duration can be found only through experimental validation of the system, and is likely to be lower. In Fig. 4.7, five different nozzle exit diameters were analyzed, each for a range of

Figure 4.7: Test duration of the Indraft Tube Tunnel for 5 different nozzle exit diameters and a range of Mach numbers.

Mach numbers from 2 to 4.5 for air. In order to change Mach number for each curve of constant exit diameter, the throat area was varied to obtain the proper $\frac{A}{A^*}$. Since the throat area governs mass flow rate into the chamber, it also largely governs test duration. The smaller the nozzle throat area, the longer the test duration. Since higher Mach numbers require lower exit pressures, the point at which the nozzle becomes overexpanded is reached sooner. Even though an increase in Mach number for a constant exit area necessitates a decrease in throat area and therefore mass flow

rate, the dominating trend is that higher Mach numbers result in shorter test times (fig. 4.7).

Since the viewing window in the chamber is 8 inches in diameter and the primary method of validation of the system was decided to be shockwave visualization, nozzle exit diameters of greater than 8 inches were eliminated from contention. A nozzle exit diameter of 7 inches was chosen as it would allow for visualization of flow phenomena at the edges of the nozzle for startup and terminating cases, would allow for potential test durations in excess of 100 ms up to Mach 2.6, which is competitive with other Ludwig Tube systems.

A design Mach number of 2.5 was selected as validation of operation up to this Mach number would allow for testing for the vast majority of supersonic aircraft in use today. Increasing the Mach number much higher would increase the chances of water condensation playing a role in tests. It also would allow for test durations easily comparable with Ludwig tubes currently in use for similar test section dimensions.

A Discussion of Chamber Reflection

It was found at the California Institute of Technology Ludwig Tube, upon starting with a Mach 4 nozzle and an upstream diaphragm experienced significant shock reflection in the dump tank, which affected the test section [15]. In order to mitigate the problem, baffles were placed in the dump tank to prevent a strong shock from reflecting off of the back of the dump tank [15]. This issue only presented itself from the reflection of the shock caused by the initial diaphragm burst, the phenomenology of which is discussed in 4.1. This is likely because the flow immediately following the startup shock was traveling supersonically, and accelerated through the nozzle and into the dump tank, significantly increasing its strength upon reflection. This may

also explain why the phenomenon was seen only using the Mach 4 nozzle and not the Mach 2.3 nozzle, even with the same diaphragm burst location [15].

A similar reflection is possible to be seen for the Indraft Tube Tunnel, especially given the fact that the test section is contained within the dump tank itself. If shock reflections disturb the flow to too great an extent, a similar baffle system may need to be implemented.

4.3.3 Discussion on Reynolds Number Capability

One obvious disadvantage of the vacuum drawn supersonic wind tunnel is the lack of ability to control inflow gas characteristics. Not only is it not possible to significantly change the stagnation conditions for the test as desired, temperature and humidity variations from one day to the next can change the conditions of the test. The one thing that can be controlled is the Mach produced, which is dependent solely on the nozzle geometry. Therefore, the primary value of this system is in demonstrating the production of shockwaves on test models, but the prospect of matching Reynolds number with a supersonic flight vehicle is likely not feasible. Boundary layer phenomena may be observed and measured, however.

Chapter 5

DETAILED SYSTEM DESIGN

It was decided that the pressures necessary for a reflection shock tunnel were prohibitively high for operation at Cal Poly and for use in undergraduate testing applications. Additionally, the cost of heavy wall tube and the flange connections to operate at high pressure, especially at the tube diameters that would be desirable for longer test times, were well beyond the scope of funding for this project. Finally, even on the longer end of attainable test times for a reflection shock tunnel, being on the order of 5 ms, would make data acquisition much more difficult than that for an Ludwig Tube or Indraft Tube Tunnel [11]. Put simply, the design and implementation of a reflection shock tunnel is beyond the reasonable scope of this project.

In order to select between a Ludwig Tube or Indraft Tube Tunnel, it is necessary to consider the performance of each relative to each-other. For a Ludwig Tube, in order to maximize test duration without causing significant boundary layer growth in the tube, the driver tube typically has a tube length of no more than 50 diameters, unless the tube diameter is exceptionally large, in excess of 1.5 feet [14]. For the given standard tube area to throat area ratio for a 7 inch exit diameter, Mach 2.5 nozzle is 5.5 in, which means the minimum tube length is 21 feet. This length is quite easy to find space for, will limit the test duration as the expansion wave will reach the end in a mere 18.6 ms. It is generally advised to use a longer tube, which necessitates a larger diameter.

Since the Indraft Tube Tunnel is capable of using nozzles with exit diameters up to approximately 12 inches, if a Ludwig Tube with a similarly sized nozzle is considered with a tube diameter of 8 inches and an L/D of 60, the tube length nearly doubles to 40 feet, which significantly increases the required laboratory space. Therefore a small

Ludwig Tube which competes with the Indraft Tube Tunnel in required lab space, but increasing the test time and nozzle dimensions requires a much larger system, whereas the footprint of the Indraft Tube Tunnel remains unchanged. At some point, however, increases in nozzle size with reduce the test duration so far as to necessitate a large vacuum chamber. It was then decided to design and build an Indraft Tube Tunnel rather than a Ludwig Tube, as it will provide similar capabilities at a fraction of the cost and space required. Fig. 5.1 shows the overall layout of the new Cal Poly ITT.

Figure 5.1: Simplified section view of the final assembly design including the full nozzle assembly and vacuum chamber, not including validation test hardware. All dimensions in inches.

5.1 Nozzle Design

For the sake of simplicity, and to increase the likelihood of a proper starting of the nozzle, a simple conical geometry was selected in lieu of a more intensive design utilizing the method of characteristics and viscous effects. Velocity vectors across the exit plane of a conical nozzle will not be strictly parallel to the long axis of the nozzle

except along the the centerline of the nozzle itself ; this non-uniformity is generally non-ideal for aerodynamic testing. However, was determined that a simple nozzle would sufficiently prove the concept of the Indraft Tube Tunnel, as well as provide a means for supersonic aerodynamic testing to the Cal Poly Aerospace Engineering Department, while simultaneously allowing for more complex, ideal nozzles to be designed and implemented in the future.

Figure 5.2: Nozzle section view detail

5.1.1 Test Rhombus Characteristic Selection

A test rhombus is commonly referred to in supersonic testing as the volume in which testing can occur at a desired test point. In a typical supersonic wind tunnel with walls of the same diameter as the exit of the converging-diverging nozzle, this rhombus is typically defined by the location of the shock generated by the leading edge of the test article after it reflects off of the test section wall. In the case of a freejet test section, the test rhombus is defined by the volume swept out by the Mach wave from the trailing edge of the nozzle.

Once the flow exits the nozzle into the vacuum chamber, which has a much lower pressure than the flow, it will undergo a Prandtl-Meyer expansion turn away from the centerline of the flow, causing the Mach number of the flow to increase and the flow direction to change. This effect, however, isn't seen within the test rhombus because it cannot affect anything within the Mach wave angle. Therefore any test article placed within the test rhombus will experience the designed test point for the duration of the test.

Mach wave is the minimum area of effect allowed in a supersonic flow. Since the flow is moving faster than the speed of sound propagation in the fluid itself, the maximum area of effect possible in the flow from a perturbation is defined as [1]:

$$\mu = \arcsin \frac{1}{M} \quad (5.1)$$

where μ is the Mach angle. Therefore, for a flow with a Mach number of exactly 1, the Mach wave angle is 90 deg, and for the designed Mach number of 2.5, the Mach wave angle is 23.6 deg. The test rhombus forms a cone with a maximum diameter of 7 inches, a length of 8.02 inches, and a half angle equal to the Mach wave angle, 23.6 deg. The complete test rhombus can be seen in fig. 5.3, and the viewable area through the window can be seen in fig. 5.4. It must here be noted that this analysis assumes that the flow exiting the nozzle is uniform across the exit plane and traveling parallel to the long axis of the nozzle. Since the nozzle used for the validation testing is conical, the flow exiting the nozzle plane is not uniform, and is parallel to the nozzle wall near the edges of the nozzle, which is angled at 12 deg. Therefore a test rhombus more representative of this nozzle would be traced at 23.6 deg – 12 deg, or 11.6 deg. What can be seen through the window in this figure is the effective test section for an imaging test. The nozzle exit plane and exit diameter were designed in an attempt to maximize the possible test model size.

Figure 5.3: View of the entire test rhombus after the nozzle exit, with dimensions in inches.

Mach Number

A Mach number of 2.5 was selected to demonstrate the tunnel's ability to reach moderate Mach numbers while simultaneously attempting to minimize the effects of water condensation eliminate liquefaction of the gas. At a later date the limits of the chamber may be more closely approached with a higher Mach number, or one tailored for a specific test condition or study.

Diameter

The window through which the test rhombus will be viewed has a diameter of 8 inches, which means that the maximum test section possible in this apparatus is also 8 inches in diameter, assuming that data is taken via the window; in theory, a much larger test section could be used, so long as data was gathered via other methods. Since the primary goal of this thesis is to validate the operation of this apparatus, a nozzle exit diameter of 7 inches was selected so that the entire test rhombus would be in the view of the window, and the performance of the chamber itself could be better analyzed.

Figure 5.4: Effective test section viewable through the window, with dimensions in inches.

5.1.2 Nozzle Inlet Design

The mass flow rate through the inlet and nozzle extension assuming a uniform velocity profile is calculated to be 2.274 kg/s, with a Mach number of .19 with stagnation conditions of standard sea level temperature and pressure. Since the Mach number is well below .3, an incompressible analysis may reasonably be expected to yield satisfactory results in describing the flow behavior through the inlet system [4].

There exists a phenomenon in which flow entering a pipe from a large volume undergoes a flow separation, as it must turn sharply around a corner. Such flow separation in the case of this apparatus would be annular in shape. In order to reduce the flow separation around this corner it is necessary to blunt the corner to encourage the flow to stay attached. This phenomenon has been greatly studied with relation to pipe flow, and is usually described with respect to a loss coefficient, K , which is used to calculate the head loss caused by a type of pipe junction [4]. If nothing is done to condition the flow entering a pipe from a large volume, as it does

for the Indraft Ludwig Tube, there will undoubtedly be significant flow separation, which can introduce unsteadiness to the flow in the test section, as well as abnormal boundary layer growth [17].

Rather than perform an in-depth analysis of a wide array of inlet-bell geometries, it was reasoned that a geometry that has been found to have a very low loss-coefficient would therefore necessitate a similarly low occurrence of flow separation. Therefore, qualitative data were found relating pipe diameter, D , and inlet bell radius of curvature, R , to the loss coefficient K in Table 5.5 [4]. These specific results are valid for an incompressible flow entering a pipe protruding from a wall [4]. For the inlet bell, an R/D of .2 was selected, which would give a loss coefficient of .03, minimizing the aforementioned negative affects of flow separation, fig. 5.5.

Figure 5.5: Basic layout of a pipe inlet protruding from a wall where R is inlet curve radius, U is flow velocity, and D is pipe diameter [4]

A schematic showing basic dimensions for the Inlet Bell can be seen in fig. 5.6. It can be seen that the bell radius of curvature fits with the selected R/D . In addition to adding the radius of curvature detailed in fig. 5.5, the end lip was rounded so as to further mitigate potential for flow separation. The outer diameter was sized to be

under 11” to limit the cost of manufacturing, as this dimension drives the amount of raw stock needed to machine the part.

Figure 5.6: Basic schematic of the Inlet Bell.

5.1.3 Nozzle Extension

Because the window on the vacuum chamber which is needed to perform Schlieren photography is centered approximately one meter from the flange face of the chamber, and the nozzle’s exit plane needs to be within several inches of the window, the nozzle itself must also be nearly a meter in length. A nozzle of the required length with the area ratio necessary to achieve a Mach number of 2.5, an exit diameter of 7 inches, would have a very small half angle on the order of 1-3 degrees on both the converging and diverging sections of the nozzle. Such gradual changes in area lead to significant boundary layer growth in the nozzle, which would likely yield adverse results to the quality of the supersonic test section [18]. It was therefore decided to design a conventionally proportioned nozzle, but introduce a long straight section with no

Figure 5.7: Simplified section view of the nozzle extension showing major dimensions. All dimensions in inches.

area change, which would essentially act as an extension to the nozzle so the test rhombus could be viewed through the viewing window. Manufacturing the internal contour on a single piece of this length would be less cost effective, if even possible by conventional methods, so it was decided to split the nozzle into at least two segments: a nozzle, and an extension.

Because the throat diameter of the nozzle design is 4.311 inches, the mass flow rate is 2.27 kg/s, which means that given a small enough flow path diameter in the extension, the flow velocity leading to the nozzle itself could contribute to significant viscous losses, so the extension diameter was sized to keep the Mach number under .2.

Additionally, the nozzle assembly behaves as a cantilevered beam under its own weight, so the smaller the diameter of the flow path and supporting structure, the weaker the nozzle will be, as well as more susceptible it will be to bending moment, potentially causing misalignment of the exit plane of the nozzle. For these reasons with would seem beneficial to have the largest diameter possible, but this is limited by the material and manufacturing costs of producing it. Based on material cost and

availability in comparison to total funding, an internal flow path diameter of 7.550" was selected, with a wall thickness of approximately .22".

5.1.4 Boundary Layer Growth

Of course even with boundary layer minimization methods in place such as large port area to minimize flow velocity for as long as possible in the nozzle, a significant boundary layer will be ingested by the nozzle, and continue to grow as the flow expands and accelerates. A portion of the boundary layer will be subsonic, a portion will be supersonic, and beyond the boundary will exist an inviscid core to the flow. The actual Mach number and test rhombus diameter will be determined by this inviscid core, so it is typically necessary to account for boundary layer growth in the design of the nozzle.

Since the nozzle being designed to demonstrate the validity of a tubeless Ludwig tube is simple and conical, and the test condition itself is arbitrary inasmuch as it does not matter precisely at what test condition the facility operates, as long as stable supersonic operation is proved, boundary layer growth in the nozzle can likely be ignored. There will be some error in the predicted Mach number, but this error should be inconsequential to the success of the project as a whole.

In the future, other methods can be employed to improve the quality of the test flow, such as an in-depth nozzle design using method of characteristics, further analysis of the boundary layer, and eventually bleed holes may be inserted between the nozzle extender and the nozzle itself, so as to suck down the boundary layer and vent it into the chamber, so as to minimize the size of the boundary layer ingested by the nozzle. The scope of such an undertaking may warrant the creation of another thesis project.

5.1.5 Structural Analysis

In the design of a cantilevered nozzle system such as this, it is important calculate the peak stress in the structure, and compare them to the yield strength of the material to determine that the design has a sufficient factor of safety for operation. The material used in the fabrication of this nozzle system is 6061-T6 Aluminum, which has a yield strength of 35000 psi. It was selected primarily due to its high strength-to-weight ratio, especially when compared to common steels, along with its widespread availability and cost-effectiveness. In the following subsections stresses will be considered at several key locations of interest along the nozzle geometry for both pressure and gravitational loads.

Fastener Selection

A design was initially conceived to split the nozzle into two parts (extension and nozzle) in order to improve machinability, especially of the nozzle and throat contour. The first iteration of this design implemented classical bolted flanges to connect and seal the nozzle pieces together. This method resulted in two relatively long 1.5 ft. pieces of stock, with an outer diameter of approximately 14", which is both large and heavy, especially given the weight of fasteners at the midpoint of the nozzle caused the total assembly mass of the nozzle to exceed 120 lb.

A second method was considered implementing V-band retainer rings and flanges as are commonly used on automotive exhaust and turbo charger systems. In a V-band system two angled flanges with flat faces are mated together and a retainer ring fits around them, pressing them together. Tightening the ring results in an increased clamping force between the two flange surfaces. This method has several benefits: Clamping force is evenly distributed around circumference of the geometry, V-Band retainers are much lighter and stronger than bolt fasteners, and assembly

and disassembly of sections of the nozzle and diaphragm system are much faster, especially with the implementation of a quick release latch on the retainer V-band. With a standard bolted flange, all bolts and nuts would need to be individually removed to replace the diaphragm, resulting in a significant increase in turnover time between shots, thus decreasing the number of shots possible per day, and impacting the feasibility of using the apparatus in an instructional lab setting.

The strength of a V-Band retainer defined with a pressure rating. A series of equations are outlined by the manufacturer to reduce the bending moment, axial force, and pressure acting on the retained joint to loadings [2]. These loads are then added and converted to an operating pressure, which is specified by the particular band that has been selected.

$$L_m = \frac{4 * M_b * D^2}{\pi * D_f l^2} \quad (5.2)$$

$$L_p = \frac{P * D_f l}{4} \quad (5.3)$$

$$L_a = \frac{F_a}{\pi * D} \quad (5.4)$$

$$L = L_m + L_p + L_a \quad (5.5)$$

$$P_{op} = \frac{4 * L}{D} \quad (5.6)$$

Three V-Bands are implemented in the design, one to allow for the easy removal and replacement of diaphragms between the inlet bell and the bell adapter, one between the nozzle adapter and nozzle extension, and one between the nozzle and the nozzle extension (fig. 6.5). Stress due to loading on the inlet bell V-Band was deemed to be minimal compared to the other two mate locations. Similarly, the bending moment

on the adapter-extension interface must be higher than that on the extension-nozzle interface because it is supporting the entire nozzle. Therefore, only the latter must be analyzed, since it is the maximum loading case.

For the case of the nozzle design, using SolidWorks mass analysis of the part, the total mass was found to be 15 kg, and the center of mass was found to be 18.3” from the nozzle face, and the bending moment sustained by the flange was found to be 604 in-lb.

Determination of Maximum Load Case

Since the maximum load on a V-Band system is the case at which the effective load on the band is at a maximum, the loading due to internal pressure of the nozzle and bending moment must be the maximum load case.

The maximum pressure case was determined to be one atmosphere of pressure. It may be thought that the highest pressure would be seen as a result of the shock reflection upon rupture of the diaphragm as was demonstrated for reflection shock tunnels in Sec:ShockTunnel; it turns out, however, that since both the driven and driving gas in this case are air, no pressure can be reached that is higher than atmospheric; therefore, the maximum pressure load case on the nozzle is 14.7 psi.

5.1.6 Structural Analysis of the Nozzle Extension

The maximum stress in the walls of the nozzle componentry itself occurs at the location of maximum moment, which is the farthest distance from the center of mass of the nozzle. This moment can be calculated using the distance from the center of mass to the location of interest.

$$M_b = W * x \tag{5.7}$$

Where M_b is the bending moment at the location of interest, W is the weight of the nozzle, and x is the distance from the center of mass to the location of interest along the long axis of the nozzle. The area moment of inertia of the nozzle cross-section must then be calculated. Since the geometry at the location of interest is a hollow cylinder, the following equation can be used,

$$Z = \frac{r_o^4 - r_i^4}{r_o} \quad (5.8)$$

Where r_o is the cylinder outer diameter, r_i is the inner diameter, and Z is the area moment of inertia. The stress from bending moment can then be calculated as:

$$\sigma_b = \frac{M_b}{Z} \quad (5.9)$$

where σ_b is the peak stress in the cylinder due solely to bending moment. The stresses in the nozzle due to pressure loads must then be calculated. These are typically divided up into longitudinal stress and hoop stress. The hoop stress for a thin-walled pressure vessel can be calculated as,

$$\sigma_H = \frac{Pr}{t} \quad (5.10)$$

where σ_H is hoop stress, P is pressure, r is radius of the vessel, and t is the wall thickness. The expression to calculate axial stress is similar:

$$\sigma_L = \frac{Pr}{2t} \quad (5.11)$$

where σ_L is longitudinal stress. Hoop stress is orthogonal to both longitudinal stress and the stress due to bending moment, which puts the top half of the nozzle in tension, and the bottom half in compression. Therefore, the tensile longitudinal stress will relieve the stress state on the bottom half of the nozzle, and add to that of the top half, which means that the top of the nozzle at the location of minimum area and maximum moment is a candidate for peak stress. Additionally, as can be seen in 5.10 and 5.11, hoop stress is always double longitudinal stress, so if the stress caused by

bending moment is small enough, the peak stress in the system, by which the factor of safety is calculated, will be the hoop stress.

5.2 Machining and Manufacturing

Any design, no matter how creative or intensive, can be rendered ineffective by poor machining and tolerancing practices. The primary driving constraint of the manufacturing tolerances for this system is ensuring the flow enters the nozzle with uniformity, and the smallest boundary layer possible. Therefore, it is important to ensure that the flow is not significantly disturbed on its transit from the inlet to the nozzle. In particular, great care must be taken when determining the tolerance on the flow path diameter between the 6 parts, as well as ensuring locational alignment between them. Variations in diameter between each piece would cause either a forward or backward-facing step, which, if large enough, may asymmetrically impact the boundary layer, and will cause flow recirculation at those features as seen in figs. 5.8 and 5.9. The illustrations in figs. 5.8 and 5.9 depict an exaggerated change in diameter between parts in order to highlight the flow features caused by these changes in geometry.

Figure 5.8: Flow perturbation due to a sudden reduction in the flow-path diameter

Figure 5.9: Flow perturbation due to a sudden increase in flow-path diameter

Locational misalignment would cause a forward-facing step on one side, a backward-facing step on the opposite, which can be of a larger concern than a simple improper sizing of each piece in the flow-path 5.10. Therefore it is necessary ensure that the flow path diameter is close enough between each segment to prevent such a disturbance in the flow. While the strictest manufacturing tolerances may be desired to maximize

Figure 5.10: Flow perturbation due to improper alignment of two mated pipes

flow quality, they may also lead to drastic increases in machining cost and lead time; therefore, a healthy balance must be struck between performance and cost. Each mated piece in the flow-path geometry was given a locating feature with a specific

dimension and location tolerance . As can be seen in Fig. 5.11, each mate had one piece with a counterbore (shown on the right of the detail) and on had a shoulder machined from the outside. For this example, the shoulder on the Nozzle Extension acts as a shaft which fits into the counterbore. The depth of the counterbore and shoulder were designed such that interference between the two is impossible, so that when the two pieces are mated together the bottom of the counterbore and face of the extension are in contact.

The fit between the shoulder on the left and counterbore on the right of fig. 5.11 was used to locate the two pieces to one another and ensure a smooth flow-path, preventing misalignment such as is noted in Fig. 5.10. A moderate clearance locational fit of LC3 was selected, as it would ensure a clearance fit between mated parts, while also locating the flow-path between the parts sufficiently well without imposing extremely difficult and expensive tolerances [16]. It was considered to give a tolerance

Figure 5.11: Detail of the locating feature between the nozzle and extension, with tolerances included.

of concentricity between each part, but it was decided that setting the flowpath as a datum and locating each part to that datum would be sufficient. Detailed part drawings detailing all of the required tolerances can be found in App. A

Chapter 6

VALIDATION EXPERIMENT AND HARDWARE

In order to validate the operation of the Indraft Tube Tunnel, a simple experiment was devised to prove that the system operates as designed. Due to the rather large undertaking of conceiving, designing, building, and validating the operation of any decently sized supersonic wind tunnel in the pursuit of a Masters degree, it was necessary to simplify several of the aspects of this project, including the nozzle design and preliminary analysis of the system. As such, an in-depth characterization of the Tunnel was simply not possible in the time line allotted, as simply adding the instrumentation typically necessary for supersonic testing would be a project in and of itself. Therefore, it was decided to validate operation of the system by placing a simple two-dimension wedge of known angle in the test rhombus of the tunnel, and image the incident oblique shockwaves formed on the model through the use of a shadowgraph setup. Since oblique shockwaves are very well-understood and trivial to analyze, the shockwave angle was measured and compared to the predicted shockwave angle to determine both that supersonic flow was produced, and the Mach number of the flow.

As test duration is a significant factor in the viability of the design, the useful test duration was computed based on the number of frames shockwaves were observed, and framerate of the video.

6.1 Nozzle Hardware and Assembly

The entire nozzle assembly consisted of 6 pieces, all manufactured from 6061 Aluminum. They were the Inlet Bell, Inlet Adapter, Flange, Extension Adapter, Nozzle Extension, and Nozzle.

6.1.1 Flange

Installation of the large flange required at least four people both to lift and hold the flange and gasket into place, and to insert and tighten the 20 fasteners and matching washers. The flange alone weighs on the order of 65 pounds without other hardware and fasteners.

Figure 6.1: The completed flange with bolt pattern and center bore to match the flow path of the nozzle is shown. The sealing O-rings inside and outside the bolt pattern are also featured.

6.1.2 Inlet Adapter and Extension Adapter

The inlet adapter and extension adapter are bolted together through the large flange on the front of the vacuum chamber by a circular pattern of 10 1/2-20 bolts, which tap directly into the extension adapter. The inlet adapter with bolt pattern can be seen in Fig. 6.2. These two pieces were designed to be assembled and disassembled

Figure 6.2: The inlet adapter is shown mounted to the flange and extension adapter.

infrequently. The Extension Adapter pictured in Fig. 6.3 in particular was designed to be very strong and to withstand a large bending moment so as to prevent drooping of the nozzle, as well as to accommodate any future nozzle designs which may be much heavier than the current nozzle. At least two people are needed to install these pieces on the flange; one inside the chamber, and one outside. The Extension Adapter features 10 blind holes tapped for 1/2"-20 bolts to a maximum thread depth of 3/4".

Given the thickness of the Flange, depth of the holes in the Extension Adapter, and thickness of the Inlet Adapter, 2.5" long bolts are required, with one washer each, to properly assemble the pieces and compress the O-rings.

Figure 6.3: The extension adapter is shown mounted to the flange and inlet adapter.

6.1.3 Nozzle Extension

The nozzle extension is very light given its inner diameter of 7.550" and length of 21". It was also left short enough that future nozzles which may need to be longer than the current nozzle can be afforded sufficient room to still sit in front of the window.

It was reasoned that by designing an extension with a versatile mating connection, many different varieties of nozzle can be designed and installed in the future.

Figure 6.4: The manufactured nozzle extension, which places the nozzle at the correct location in front of the window.

6.1.4 Full Nozzle Assembly

The full nozzle is assembled with V-bands linking the parts together. The flowpath is sealed with an O-ring between each part. It is very quick and easy (less than five minutes) to remove the nozzle and extension components either for maintenance, storage, or replacement with a nozzle of different design.

6.2 Test Hardware

6.2.1 Diaphragm

The diaphragms used for this series of validation tests were made of cellophane and cut to a diameter of 8.5". Two thicknesses were purchased, one of .002" and the other .001", with a quantity of 500 each. Based on initial calculations based on the yield strength of cellophane and diameter of the port that the necessary material thickness to withstand one atmosphere of pressure was .004", so it was planned to use two .002"

Figure 6.5: The fully assembled nozzle is shown integrated with the chamber and flange, consisting of the nozzle, nozzle extension, and extension adapter, all joined together by V-bands.

thick diaphragms for each test, and if those calculations turned out to be either too conservative or not conservative enough, three could be used with a total thickness of either .003" or .005", respectively. As it turns out and will be discussed in depth, these diaphragms tended to slip out from between the inlet bell and inlet adapter at lower thicknesses, preventing sufficiently low pressures from being reached. As such thicknesses of at least .010" were necessary to prevent slippage, making the .001" diaphragms unnecessary.

In Fig. 6.6 significant concavity can be seen as the diaphragm bows inward. This phenomenon complicated the design of the method used to burst the diaphragm.

Figure 6.6: Cellophane diaphragms can be seen pressed between the inlet bell and adapter whilst under pressure.

6.2.2 Model Setup

Wedge Dimensions

A simple wedge-shaped test article was machined from 6061 Aluminum to cause the flow to undergo a simple two-dimensional compression turn, resulting in oblique shockwaves that can be easily observed and analyzed. The designed wedge had a half angle of 15 degrees, height of 2 inches, and a depth of 2.5 inches.

Figure 6.7: The wedge is shown mounted to the sting arm just in front of the nozzle exit plane

Sting Design

Due to the short timescale in which this project was performed, a rudimentary model mount and sting was devised utilizing 80-20 aluminum sections. This setup will need to be improved in the future so as to allow a model to be placed at angle of attack and sidesweep. It needed to be very stiff to limit vibration and movement under the force of the flow. Fortunately, because the flow is drawn from atmosphere, it is impossible for steady state pressure on the model to exceed one atmosphere, although starting loads exceeding this value may be possible depending on the strength of the shock caused at startup.

As can be seen in Fig. 6.7, the test model can be relatively easily translated within the test section by adjusting the mounts, but rotational adjustment is very limited, and modifications will need to be made to add that capability. Additionally, some vibration of the model, lens, and light source were observed mid-test, so stiffness will need to be taken into account.

6.2.3 Shadowgraph Setup

Because the chamber only has one viewing window, it was necessary to place a light source as well as focusing mirrors in the chamber itself. This posed several challenges, as all hardware had to be well secured to prevent damage by reflecting shockwaves within the chamber, and had to survive the vacuum environment. In particular, this limited power sources for the light to placing small piles (i.e. AAA batteries) within the chamber, purchasing a specialized vacuum rated battery, which would be very expensive, or running a power feed-through into the chamber.

Through careful experimentation, it was found that household alkaline batteries, as well as Ryobi-brand rechargeable lithium-ion batteries were able to survive and operate in the vacuum environment at which these tests were run. In initial tests, hand-held LED flashlights were used as a light source, but it was found that several brands of flashlight utilized pulse-width modulation to preserve battery life, which shows up as a strobing phenomenon under high-speed video. These lights, although strong enough for high-speed video at 3500 lumens, were unusable due to lost data.

Lens Setup

Due to funding prioritization of the nozzle and flange hardware, limitations in budget and available budget, two relatively small focusing lenses were used, which are not properly suited for a shadowgraph or Schlieren setup. A 50 mm parabolic plano-convex lens with a focal length of approximately 200 mm was used to collimate radial light emitted from a .060" diameter pinhole. This column of light passed through the test section, out the window, and through a second plano-convex lens, with a diameter of 70 mm and a focal length of 104mm, and a spherically shaped contour, which focused the beam onto the camera lens. As a result, significant spherical aberrations were observed due to the non-parabolic focusing lens. Additionally, for a

number of potential reasons, the degree of collimation attainable by the 50 mm lens was less than satisfactory, resulting in significant blurring both of the test model and imaged shockwaves. Since the camera lens used had a 76 mm, significantly larger

Figure 6.8: Schematic depicting the general shadowgraph setup for this test, including the beam path, light source, lenses, camera, model, and nozzle.

than the parabolic lens used to create the collimated beam of light through the test section, it was initially attempted to set the camera lens focus to infinity and omit the second focusing lens in the setup. Due to zoom limitations and difficulty obtaining proper alignment, this method produced very poor, small images. By incorporating the second lens, even with the introduction of spherical aberration, the image quality was much improved, and able to be enlarged. Regardless, significant improvements need to be made to the setup hardware in order to produce higher quality images.

High Speed Camera

A Vision Research Phantom V310 high speed camera was used to record images of the flow. This camera is capable of recording 1 megapixel (1200x800) images at sample rates up to 3250 fps. Much faster framerates, up to 500,000 fps are possible, but only by reducing the resolution of the image. Due to light limitations and a desire to keep the total recording time above 3.5 seconds to account for human error in the triggering process, a framerate not exceeding 2500 fps was used, with an exposure of 150 microseconds. The resolution was also limited in the tests to maximize filming duration whilst capturing the entire image.

When testing, the camera was set to record continuously, overwriting the high speed RAM on the camera. At the completion of the test, the camera was triggered, capturing the few second before the trigger signal was given, recording the test. In the future, with a more advanced triggering system linked to some phenomena related to actual initiation of the test, along with a stronger lightsource, significantly faster framerates can be reached, which would maximize the value of the achievable test time and aid in the determination of starting and unstaring phenomena for the nozzle.

6.2.4 Vacuum Hardware

The large 3.5 m^3 vacuum chamber discussed throughout this thesis was utilized as both the dump tank and test section. The center of an 8" diameter viewing wind is located approximately 1 meter from the large flange face. It is through this window that the test section was viewed.

A Welch Duo-seal vacuum pump model number 1374 was utilized to pump down the system. On initial pump-down, the chamber was brought to a minimum pressure of 1.64 Torr in 112 minutes. Although this pressure is several orders of magnitude

higher than what is typically expected to be achieved in a good vacuum chamber by a roughing pump, several additional improvements to the chamber are yet to be made as was discussed in 3.1. In operation, it typically took approximately 25 minutes to reach the minimum test pressure of 30 Torr, and 35 minutes to reach the optimal test pressure of 20 Torr. A model CVG101GA Worker Bee convectron gauge by Instru

Figure 6.9: Chamber pressure vs. time for the first full vacuum test of the chamber. .

Tech, Inc. was used to measure pressure in the chamber for each test. It presented great accuracy at pressures below 20 Torr, but read values that jumped up and down greater than 50% of the mean value during pump down, especially in the region of 30-50 Torr. Pressures were recorded at random points during the test without regard for noise in the data, which likely resulted in significant error in measurements, especially at the beginning of the test. It is likely that this great variance in pressure readings was caused by outgassing of oils and adhesives internal to the chamber. A convectron gauge is very sensitive to gas composition, as it infers pressure by measuring gas thermal conductivity, so significantly changing gas composition by

outgassing of materials, especially those not designed for use in a vacuum was the likely cause.

Chapter 7

EXPERIMENTAL RESULTS

In order to validate the operation of the Indraft Tube Tunnel, a simple experiment was devised to prove that the system operates as designed. This test was to image oblique shockwaves incident on a wedge, measure those shockwaves, and calculate Mach number based on the angles measured and wedge dimensions.

7.1 Experimental Setup

The experimental setup used to validate operation of the Indraft Tube Tunnel consisted of a shadowgraph setup with a light source, two focusing lenses, and a high speed camera. This optics system passed a collimated beam across through the test rhombus, around the test wedge, out the window, through the second focusing lens, and into the camera aperture.

7.1.1 Test Procedure

The full step-by-step procedure used to assemble the Indraft Tube Tunnel can be found in app. D, and the procedure to run the tunnel can be found in app. E. A truncated version of the full procedure is included here for reference.

- The model is installed on the sting, and the shadowgraph setup is tuned with the high-speed camera to ensure that the image produced is satisfactory.
- Then the chamber door is closed and secured.
- The diaphragm is installed and the inlet bell attached to the inlet adapter by V-band.

Figure 7.1: Schematic detailing the components of the chamber ready to test. Omitted in this view are the focusing lenses, light source, and camera.

- The diaphragm puncture device is armed.
- Range safety is positioned around the laboratory, and the vacuum pump is turned on.
- The pressure is monitored over time via convectron gauge.
- When the desired test pressure is reached, the diaphragm puncture device is triggered.
- The trigger on the high-speed camera is then triggered, recording the test.

7.2 Experimental Results and Validation

7.2.1 Diaphragm Burst Validation

The primary method initially envisioned to cause diaphragm rupture was to put a short loop of nichrome wire in contact with a point on the diaphragm and run a

current through it, causing it to heat up, melt the diaphragm, and cause a rupture. It was found that this method did not always result in a complete diaphragm rupture. Instead, the melted diaphragm formed a thick ring around the location of the nichrome, reinforcing the edges of the hole. This prevented the diaphragm from completely rupturing, and resulted in a slow leak through the diaphragm, seen in fig. 7.2. Additionally, deflection of the diaphragm under pressure made keeping the nichrome in contact with the diaphragm difficult without adhering the nichrome to the surface in some way. It was then decided to adopt a mechanical bursting method.

Figure 7.2: The nichrome filament after successfully melting through the diaphragm, but failing to make it rupture fully.

This mechanical burst method consisted of a weighted pendulum with a sharpened star-shaped puncture device protruding from the end. This pendulum was suspended above the diaphragm by a string. The same ignition circuit used for the original nichrome burst method was then used to burn the string at the desired time, causing the pendulum to fall, bringing the sharpened point in contact with the strained

diaphragm at velocity, and resulting in a 100% occurrence of puncture. Obviously this method is not ideal in the long run as the pendulum remains at the inlet of the nozzle throughout the test, potentially causing significant perturbation in the flow; however, it was deemed necessary to utilize a simple, reliable burst method for the success of validation efforts.

7.2.2 Shockwave Imaging and Observed Phenomena

The presence of aberration significantly degraded the quality of the recorded images. It is likely that all three of the aberrations discussed in 2.2.5 manifested themselves together to some degree, causing this suboptimal image quality. Upon investigation of the lenses in use, it was discovered when one of them was incapable of producing collimated light, that its curvature was spherical instead of parabolic, resulting in significant spherical aberration. This lens was then used to focus the collimated beam into the camera, limiting the negative affect of this aberration, but it was still present. Minor misalignments of the lenses and camera likely introduced coma, which typically presents itself more strongly around the edges of an image. Finally, a relatively long focal length of the collimating lens and the relatively large aperture of the light source combined to produce an imperfectly collimated beam which produced a slightly blurred image before reaching the second focusing lens 5 feet away.

Fig. 7.3 shows a wind off image of the model moments before the start of the test beside an image captured during steady-state supersonic operation. Notably, even in the wind off image, significant blurring of the test model is visible, likely due to a combination of spherical aberration and a poorly collimated beam. In the wind off image on the left, two primary images can be seen: a dark mass in the shape of the wedge, and a lighter, blurred outline of the wedge translated to the left of the image. Comparing this image to that of the wind-on case, it is clear that the most

easily identifiable shockwaves are aligned with the lighter, translated image, and much more difficult to identify line of the same angle can be seen originating at the point of the darker image. Essentially, it is posited that the true image is distorted, with

Figure 7.3: Shadowgraph image showing windoff state of the model on the left, and wind on, steady flow with established oblique shockwaves. Flow is moving from left to right.

aberration producing duplicate images offset from the model itself, and that both this blurred image and the fainter image can be used in conjunction with the image of the model geometry in the wind-off case, can be used to measured to determine shockwave angle, albeit with a greater variance that would likely present itself with a more optimal optical setup.

By taking these aberrations into account, the primary ways that the images were impacted, and a reliable way to measure the shockwave angle by comparing wind on and wind-off images has been identified. As can be seen in fig. 7.3, the image captured is very blurred, and there exist what appear to be oblique shockwaves originating from a point standing in front of the wedge outline, making the actual measurement of the shockwave angle difficult.

Mach Number Determination

The method used to validate the operation of the Indraft Tube Tunnel as designed was to put a model wedge of known half-angle at no angle of attack in the test rhombus of the nozzle. This forces the flow to undergo a compression turn and develop an analytically trivial oblique shockwave. As is described in 2.1.5. For a 15 degree half angle wedge in Mach 2.5 flow, the shockwave angle for air is 36.95 deg. As can be seen in fig. 7.4

The shockwave angle was measured by drawing a line through the center of the wedge, and middle of each oblique shock, as seen in fig. 7.4. As can be seen clearly in figs. 7.4, 7.3, and 7.5, due to the significant blurring of the image, it is difficult to define the exact location of the shock without some error. Therefore, in order to gain some idea of the error in measurement, the shockwave angle for every frame of steady state flow was measured, and a mean and standard deviation were computed for the entire set.

To determine the test-section Mach number, the 19 images collected depicting oblique shockwaves attached to the model were analyzed. Using an image of the wind-off case, a stencil was drawn over the model geometry, as the front of the model was visibly distorted when the wind turned on (Fig. 7.4). This stencil was then applied to every image, and a line was drawn through the middle of each visible oblique shockwave, one on top, one on the bottom as shown in 7.4. The angle from the centerline to the lines drawn along the upper and lower shocks were measured by protractor and recorded. The 19 raw images are included in app. C, and the images with the shockwaves measured is included in B

Using the $\theta - \beta - M$ equation, 2.15, with the known turn angle of 15 deg and the measured shockwave angles, the Mach number was solved for each individual measurement. The mean and standard deviation, σ , for both the measured shockwave

Table 7.1: The mean and standard deviation of the measured shockwave angle and calculated Mach number, computed for measurements of only the upper oblique shock, only the lower oblique shock, and for both combined. These data were gathered from 19 images.

	Mean Angle	σ Angle	Mean Mach	σ Mach
Upper Shock	38	0.83	2.42	0.065
Lower Shock	37.4	1.19	2.47	0.097
Comined	37.7	1.07	2.44	0.085

angle and calculated Mach number were calculated using three data sets: upper shock measurements, lower shock measurements, and all 38 measurements taken irrespective of whether the shock was the upper or lower shock. These data are presented in Tab. 7.1.

The mean of the measured shockwave angle was 37.7 deg, which is within 1 σ of the optimal value of 36.95 deg. The mean Mach number calculated from the measured shockwave angles was 2.44, which is within 1 σ of the design Mach number of 2.5. Improvements to the shadowgraph setup would undoubtedly increase the accuracy to which the operational Mach number can be determined. Other methods such as using a pitot probe could be used to determine Mach number in the future.

Shock Reflection

A traveling shockwave can be seen propagating from right to left into the test section toward the nozzle can be seen in Fig. 7.5. It was predicted in section **section of shock reflection** that this reflected shock may impact the test, but hopefully would not. It is clear that it does, but also that it halts just in front of the model, meaning that it does not travel into the nozzle causing an unstart. In fact, shockwaves can again be seen on the model shortly after in **fig with new shockwaves**.

Figure 7.4: Four images showing the technique by which the shockwave angles were measured. The top two show the windoff case and creation of the stencil, and the bottom two show the steady-state case with the shockwaves traced according to the method.

Detached Bow Shock

In this case, however, because the presence of the diaphragm on the model surface significantly changed the aerodynamic properties of the wedge, the visible shock is a detached bowshock rather than attached oblique shock as was seen on the model

Figure 7.5: Progression of startup shock reflection from the back of the chamber over six frames, each spaced 400 microseconds apart. Frames are depicted in order from left to right.

before the diaphragm became stuck. This bowshock can be clearly seen running vertically through the left-middle of Fig. 7.6. Asymmetry in the placement of the diaphragm on the model likely led to significant asymmetry on the model itself.

7.2.3 Test Duration and Observed Phenomena

The relationship between the initial pressure in the chamber and estimated test duration can be seen for three different Mach numbers, the mean measured Mach number and $\pm$ one standard deviation(σ) in Fig. 7.7. The analysis from was used here again, but instead of trading test duration with nozzle size as in Fig. 4.7, it is being used here in 7.7 to estimate the duration of supersonic flow achieved based on the starting pressure of the chamber, which in reality is always non-zero. It can be seen

Figure 7.6: The diaphragm stuck on the model after passing through the nozzle, changing its outline and blunting its nose. The detached bowshock can clearly be seen in the image.

that for the test pressure of 25 Torr, this estimate of nozzle operation ranges from 44 ms to 65 ms at the outer ranges of $\pm \sigma$, and 54 ms at the mean. 81 slides were counted from the appearance of the first frame in which a shockwave is observed to the last one, which for an interval of $400 \mu s$ between frames is a duration of 32.4 ms. However, this test duration did not constitute ideal operation in its entirety, as several phenomena occurred during the test which limited the useful test time.

Observed Test Phenomena

21 frames of steady state flow with attached oblique shockwaves were initially observed before the reflected shock passed through the test section, briefly stopping the flow of gas. Several frames were observed in which the flow appeared to be restart-

Figure 7.7: Ideal maximum test duration plotted against initial chamber pressure for the final design of the Indraft Tube Tunnel. Three Mach numbers, included the measured Mach number of 2.44 are included, all for a nozzle exit diameter of 7 inches. The chamber pressures at which this test termination is predicted to occur for $M = 2.44$, $M = 2.44 - \sigma$, and $M = 2.44 + \sigma$ are respectively 48.8 Torr, 55.8 Torr, and 42.7 Torr.

ing, but the passage and attachment of diaphragm material made any shockwaves that may have been present indistinguishable. After the diaphragm impacted and got stuck on the model, 13 more frames were observed with steady supersonic flow, just with a detached bowshock instead of attached oblique shocks. While the precise Mach number cannot easily be gleaned from observation of this bowshock, it is reasonable to assume that its presence necessitates a supersonic flow, which is likely of similar characteristics to that observed before the passage of the diaphragm. Since images were sampled once every $400 \mu s$, the useful test duration was 19 images, which corresponds to 7.6 ms for the used framerate of 2500 fps (Tab. 7.2).

7.2 presents the four observed test phenomena from test start as: ideal operation, in which steady, attached oblique shockwaves are observed, Reflected Shock, in which

the flow is temporarily disturbed by the reflection from the back of the chamber at startup, diaphragm disturbance, as the period of time in which diaphragm material is passing through the test section or attaching to the test model, detached bowshock, in which supersonic flow phenomena were observed in the form of a standing bowshock visible in front of the diaphragm-enveloped test model.

Furthermore, two analogues were created to combine several of these categories in an attempt to predict test duration under more ideal conditions. These analogues are: Steady State Supersonic Flow and Potential Steady State Test. The former is analogous to a case in which the diaphragm never enveloped the test model, and steady supersonic flow returned after the reflected shock and diaphragm matter ceased to disturb the flow; it is the summation of Ideal Operation and Detached Bowshock, and is the total time that steady supersonic flow was confirmed in test. The latter is analogous to a case in which diaphragm material did not adhere to the test model and cause a lengthy disturbance in the diaphragm became fully seated on the model and the bowshock formed. Partial or forming bowshocks were observed at several points during this period, but disturbances from the diaphragm seemed to delay the onset of steady flow. Therefore it is reasoned that without diaphragm attachment to the model, much of this time could reasonably be expected to contribute to ideal operation. This category is the summation of ideal operation, diaphragm disturbance, detached bowshock, and represents the most hopeful of estimates on potential test duration from the given starting pressure. Shockwaves observed records from the first observed shockwave to the last observed shockwave during the test, irrespective of other flow phenomena and flow steadiness.

Simply stated, the confirmed supersonic operation of the nozzle for this validation test at the given pressure was 13.6 ms, with potential for up to 18.4 ms if the problem of diaphragm disturbance in the test section can be mitigated. Therefore the useful test duration, given no diaphragm interference, can tentatively be 18.4 ms. In 7.3

Table 7.2: Test duration and timing of events, determined by number of frames counted.

Phenomenon	No. Frames	Sample Time [ms]	Duration [ms]
Ideal Operation	19	0.4	7.6
Reflected Shock	11	0.4	4.4
Diaphragm Disturbance	12	0.4	4.8
Detached Bowshock	15	0.4	6.0
Steady State Supersonic Flow	34	0.4	13.6
Potential Steady State Test	46	0.4	18.4
Shockwaves Observed	81	0.4	32.4

Table 7.3: Predicted test duration $\pm\sigma$ compared to the measured durations of observed shock phenomena and Steady State Supersonic Operation.

	Prediction	Observed Shock	Fraction	Steady State	Fraction
$mean + \sigma$	44	32.4	0.74	13.6	0.31
mean	54	32.4	0.6	13.6	0.25
$mean - \sigma$	62	32.4	0.52	13.6	0.22

the confirmed durations of Observed Shock Phenomena and Steady State Supersonic Operation are compared with predicted test duration for Mach numbers within 1σ of the mean measured value. Therefore, the presence of supersonic flow was observed on 52% – 74% of the predicted value, and good, steady state supersonic flow was observed for 22% – 31%. The fact that the presence of adverse flow phenomena exists in the chamber due to reflection and diaphragm ingestion, along with these data should be taken into account when designing future nozzles and predicting the potential test duration for variations on this design.

7.2.4 Diaphragm Slippage

On the majority of tests, it was found that the diaphragm or diaphragms slipped out of the clamp between the inlet bell and inlet adapter, causing a premature rupture. Initially it was found that adding more diaphragms than would typically be necessary prevented slippage, but after many tests these began to slip as well. A manufacturing defect was found in the inlet bell mating face, and that face was not flat as designed, but in manufacturing significant peaks and valleys were put in the surface in excess of .002". It was also found that diaphragm slippage location usually occurred at one particular orientation of the inlet bell, further reinforcing the conclusion that slippage was caused by a manufacturing defect in the piece.

As a solution, a thin layer of room-temperature vulcanizer, which is a common curable liquid gasket, was added to the surface of the Inlet Bell mating surface, but poor shear properties of the material either due to the inherent properties of the cured material or improper curing resulted in an increased incidence of diaphragm slippage of using bare aluminum to aluminum.

A .015" thick gasket was constructed from Buna-N and applied to the inlet bell mating surface via spray adhesive in an attempt to prevent the diaphragm from slipping. It was reasoned that adding a softer gasket material to one of the surface would fill the peaks and valleys in the mating surface, resulting in a uniform load on the diaphragm and preventing slippage. This technique was moderately successful, but ultimately unsatisfactory as slippage still occurred at even diaphragm thicknesses of .010". It is recommended that future diaphragm designs incorporate larger diaphragms that extend outside the V-band flange, as the available surface area appears to be insufficient for properly holding the diaphragms in place.

Figure 7.8: Typical behavior of a diaphragm after slipping out under pressure and failing. The diaphragm slipped out of the bottom left, and sudden unloading of one side caused the diaphragm as a whole to fail.

Chapter 8

CONCLUSION

8.1 Summary of Work

A new variety of short duration supersonic wind tunnel was conceived, designed, manufactured, and successfully tested at a Mach number of 2.44 and nozzle exit diameter of 7 inches, and a measured steady-state supersonic flow duration of 13.6 ms, with potential for much longer test durations. This system has comparable operating costs to a Ludwieg Tube, but eliminates the need for a long driver tube by drawing air from atmosphere and exhausting into a vacuum chamber. No condensation was observed in any of the tests run at this Mach number, in spite of the fact that the flow was brought well-below the static dew-point of the air. In order to accomplish this task, a 3.5 m^3 vacuum chamber was refurbished and reconditioned, a supersonic nozzle, inlet, and diaphragm burst system was designed and manufactured, and a shadowgraph system was devised and constructed with high-speed imaging to validate operation of the system. A model and mounting system were created and installed to produce easily identifiable shockwaves through this setup.

8.2 Future Work

The primary goal of this thesis was to validate operation of a newly conceived variation of the Ludwieg Tube to allow for inexpensive supersonic aerodynamic testing. As such, many steps were taken to limit the scope of the project in order to keep the workload reasonable and attainable. This project has a large potential for future improvements, and can become a remarkable piece of equipment given proper stewardship of its improvements. Most of all, it is hoped that future students will use this

tunnel to test creative new designs for supersonic flight, and that it will be a valuable asset for Cal Poly.

8.2.1 Data Collection

Instrumentation for the validation test was limited to a shadowgraph system, but there are many other methods for analyzing supersonic flows in wind tunnels that can be added to the Indraft Tube Tunnel.

Shadowgraph System

There were several problems with the shadowgraph system used to validate operation of the Indraft Tube Tunnel, including the use of small lenses which did not maximize the capability of the tunnel, one of which produced significant spherical aberrations in the focused image. A new Schlieren or Shadowgraph system should be implemented to allow larger models to be tested and sharper images to be collected. This task should include:

- New, larger lenses or focusing mirrors to take advantage of the test rhombus size
- New light source designed for operation in a vacuum, with high lumosity
- New high-speed camera with improved image resolution and zoom capabilities

Instrumentation

The installation of pressure transducers and thermocouples to respectfully measure pressures and temperatures in the test section and on the model significantly improve the breadth of testing that can be done with this tunnel.

Adding a balance for taking force and moment measurements on the models may be challenging given the short test durations and vibrations in the system, but may be a valuable addition to the tunnel.

It is conceivable to seed the incoming flow with particles to perform some form of Particle Image Velocimetry, as seeding of the flow for this wind tunnel would be much easier than for the majority of supersonic wind tunnels, since the air immediately upstream of the inlet bell may be seeded.

8.2.2 Hardware

Nozzle Additions and Improvements

The nozzle used for this test was a simple conical nozzle designed for a Mach number of 2.5 and had relatively relaxed tolerances on the throat and exit diameter dimensions. Future nozzles should be designed to provide different test points in the range of Mach 1.5-3.5, although Mach numbers much higher than 3 may be problematic for this system due to the short test duration. The nozzles may be designed using the method of characteristics and taking the boundary layer ingested by the nozzle into consideration to produce flow does not deviate from the central axis of the nozzle as is seen with conical nozzles.

Diaphragm Improvements

Significant consideration should be given towards improving the diaphragm and burst method designs used to operate the tunnel. The diaphragm slipping issue must be solved, as well as the issue of diaphragms getting stuck on the test model. Using thinner diaphragms that rupture into small pieces should help to mitigate this problem. An improved burst method may be desired which introduces fewer perturbations to

the flow upstream of the nozzle than the mechanical burst method currently in use. This endeavor may involve using a laser to melt a hole in the diaphragm, a thinner puncture device, or pyrotechnics.

Sting and Model Mounting

A new sting and model mounting system should be designed to increase stiffness over the current system, and to reduce vibration. Additionally, the ability to easily vary the angle of attack and sideslip angle of a model should be added. Having the ability to change these parameters remotely without opening the tunnel will greatly improve the testing rate for diverse test matrices.

Vacuum Chamber

A larger vacuum pump can be sourced and implemented in order to reduce pumping times for tests. As tested, the turn-around time is 30-40 minutes for a standard test, and up to 90 minutes for a lower pressure test point; 90% of this time is spent pumping down the chamber to the desired pressure.

A filter should be added between the vacuum pump and chamber to protect the pump from diaphragm and ingested debris. Additionally, air dryers may be added to protect the internals of the chamber from condensed water.

The large O-ring on the door should be replaced to improve vacuum capability, and the many flange faces should be polished to a better surface finish. Additional interchangeable feed-throughs should be added, including power and data transfer to and from the tunnel.

BIBLIOGRAPHY

- [1] J. D. Anderson. *Modern Compressible Flow: with Historical Perspective*. McGraw-Hill, 3rd edition, 2003.
- [2] Associated. V-band clamp/coupling load formulas. <http://www.clampco.com/techspec/engineering%20data%20specifications.asp>.
- [3] P. J. Bilton. *Vacuum Technology*. Las Positas College, 2nd edition, 2002.
- [4] R. D. Blevins. *Applied Fluid Dynamics Handbook*. Van Nostrand Reinhold Company Inc., 1984.
- [5] R. Brun. Shock tubes and shock tunnels: Design and experiment. Technical report, Universit dAix-Marseille, France, Sep 2009.
- [6] P. Danielson. Sealing materials require a careful choice. *R&D Magazine*, (32), 2002.
- [7] J. Dunbar. Unitary plan wind tunnel specification brochure. <https://www.nasa.gov/centers/ames/orgs/aeronautics/windtunnels/index.html>. Accessed: 4/29/2018.
- [8] E. Hecht. *Optics*. Pearson, 5th edition, 2017.
- [9] P. G. Hill and C. R. Peterson. *Mechanics and Thermodynamics of Propulsion*. Addison-Wesley Publishing Company, 2nd edition, 1992.
- [10] D. F. Hornin. *Shock Waves in Chemistry and Physics*. John Wiley & Sons, Inc., 1962.
- [11] H. G. Hornung. Performance data of the new free-piston shock tunnel at galcit, 1992.

- [12] H. G. Hornung. Personal interview regarding the galcit t5 schock tunnel and ludwieg tube, March 2017. A tour of GALCIT’s T5 Reflected Shock Tunnel and Ludwig Tube was given by Dr. Hans G. Hornung in which he gave an overview of the design and operation of each of these facilities.
- [13] D. K. Husel and D. H. Huang. *Design of Liquid Propellant Rocket Engines*. National Aeronautics and Space Administration, 2nd edition, 1967.
- [14] H. Ludwig. Tube wind tunnel, a special type of blowdown tunnel. Technical report, North Atlantic Treaty Organization, Sep 1957.
- [15] C. A. Mouton. *Transition between Regular Reflection and Mach Reflection in the Dual-Solution Domain*. PhD thesis, California Institute of Technology, 2007.
- [16] E. Oberg, F. D. Jones, H. L. Horton, and H. H. Ryffel. *Machinery’s Handbook*. Industrial Press Inc., 29th edition, 2012.
- [17] A. Pope. *Wind-Tunnel Testing*. John Wiley & Sons, Inc., 2nd edition, 1954.
- [18] A. Pope and K. L. Goin. *High Speed Wind Tunnel Testing*. Robert E. Krieger Publishing Company, 1978.
- [19] M. Rudroff, A. Lodes, and R. Curry. The generation of triggered shock-waves in shock-tubes with exploding wires. *Pulsed Power Conference(PPC)*, 2011.
- [20] G. P. Sutton and O. Biblarz. *Rocket Propulsion Elements*. John Wiley & Sons, Inc., 8th edition, 2010.

APPENDICES

Appendix A

DRAWING PACKAGE

The following eight pages show the full drawing package created for submitting the many requests for quotes required to find the right machinist, and properly communicate to the machinist the important features and dimensions for the project. The first two drawings detail the overall design of the nozzle and the fits made between the parts, and the following six constitute the detailed drawing for each individual part.

- NOTE
- ALL ITEMS FROM AL 6061 T6
 - ITEM 3 IS INCLUDED FOR REFERENCE, BUT IS NOT BEING QUOTED
 - ITEM 1 IS ALSO INCLUDED FOR REFERENCE, BUT MAY BE REQUESTED FOR FUTURE QUOTE

ITEM NO.	DRAWING NUMBER	PART NAME	DESCRIPTION	QTY.
1	3	Inlet Bell		1
2	4	Inlet Adapter		1
3	5	Flange		1
4	6	Extension Adapter		1
5	7	Nozzle Extension		1
6	8	Nozzle		1

Indraft Ludwieg Tube	
Title: EXPLODED NOZZLE ASSEMBLY	
Dwg. #: 1	Scale: 1:4
Drawn By: DANIEL JOHNSON Checked By: ERIN MCMURCHIE	
Date: 5/2/2018

- NOTES**
 UNLESS OTHERWISE SPECIFIED
 1. ALL DIMENSIONS IN INCHES
 2. TOLERANCES
 X.X = ±.1
 X.XX = ±.01
 X.XXX = ±.005
 ANLGES = ±.5°
 3. BREAK SHARP EDGES .005 MAX
 4. Material 6061 T6 Aluminum

DETAIL AF
SCALE 1 : 1

SECTION AB-AB
SCALE 1 : 3
118

Indraft Ludwig Tube	
Title: INLET BELL	
Dwg. #: 3	Scale: 1:3
Drawn By: DANIEL JOHNSON Checked By: ERIN MCMURCHIE	
Date: 5/2/2018	

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Aerospace Engineering

10X ϕ .531 THRU
 ϕ .003 B A

ϕ 8.650 H8 +.003 / -.000
 ϕ .003 B A

- NOTES
 UNLESS OTHERWISE SPECIFIED
 1. ALL DIMENSIONS IN INCHES
 2. TOLERANCES
 X.X = $\pm$.1
 X.XX = $\pm$.01
 X.XXX = $\pm$.005
 ANLGES = $\pm$.5°
 3. BREAK SHARP EDGES .005 MAX
 4. Material 6061 T6 Aluminum

SECTION V-V
 SCALE 1 : 3

DETAIL W
 SCALE 1.5 : 1

Indraft Ludwig Tube	
Title: INLET ADAPTER	
Dwg. #: 4	Scale: 1:3
Drawn By: DANIEL JOHNSON Checked By: ERIN MCMURCHIE	
Date: 5/2/2018	

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 Aerospace Engineering

- NOTES
- UNLESS OTHERWISE SPECIFIED
 - ALL DIMENSIONS IN INCHES
 - TOLERANCES
 - X.X = $\pm .1$
 - X.XX = $\pm .01$
 - X.XXX = $\pm .005$
 - ANLGES = $\pm .5^\circ$
 - BREAK SHARP EDGES .005 MAX
 - Material 6061 T6 Aluminum
 - ALL GLAND DIAMETERS DIMENSIONED BY ID

Indraft Ludwig Tube

Title: FLANGE

Dwg. #: 5

Scale: 1:5

Drawn By: DANIEL JOHNSON

Checked By: ERIN MCMURCHIE

Date: 5/2/2018

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Aerospace Engineering

10X $\phi .453 \nabla 1.000$
1/2-20 $\nabla .750$

$\oplus \phi .005$ B A

NOTES

UNLESS OTHERWISE SPECIFIED

1. ALL DIMENSIONS IN INCHES
2. TOLERANCES
X.X = $\pm .1$
X.XX = $\pm .01$
X.XXX = $\pm .005$
ANLGES = $\pm .5^\circ$
3. BREAK SHARP EDGES .005 MAX
4. Material 6061 T6 Aluminum

SECTION T-T
SCALE 1 : 3 121

DETAIL U
SCALE 1.5 : 1

Indraft Ludwig Tube

Title: NOZZLE EXTENSION ADAPTER

Dwg. #: 6

Scale: 1.5:1

Drawn By: DANIEL JOHNSON

Checked By: ERIN MCMURCHIE

Date: 5/2/2018

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NOTES

- UNLESS OTHERWISE SPECIFIED
1. ALL DIMENSIONS IN INCHES
 2. TOLERANCES
 $X.X = \pm .1$
 $X.XX = \pm .01$
 $X.XXX = \pm .005$
 $ANLGES = \pm .5^\circ$
 3. BREAK SHARP EDGES .005 MAX
 4. Material 6061 T6 Aluminum

$\phi 7.550 \pm .002$ THRU

DETAIL AE
SCALE 1.5 : 1

SECTION AC-AC

DETAIL AD
SCALE 1.5 : 1

Indraft Ludwig Tube

Title: NOZZLE EXTENSION

Dwg. #: 7 Scale: 1:3

Drawn By: DANIEL JOHNSON
Checked By: ERIN MCMURCHIE

Date: 5/2/2018

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Aerospace Engineering

NOTES

- UNLESS OTHERWISE SPECIFIED
- ALL DIMENSIONS IN INCHES
- TOLERANCES
 - X.X = $\pm .1$
 - X.XX = $\pm .01$
 - X.XXX = $\pm .005$
 - ANLGES = $\pm .5^\circ$
- BREAK SHARP EDGES .005 MAX
- Material 6061 T6 Aluminum

Indraft Ludwig Tube

Title: NOZZLE

Dwg. #: 8

Scale: 1:3

Drawn By: DANIEL JOHNSON

Checked By: ERIN MCMURCHIE

Date: 5/2/2018

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Aerospace Engineering

Appendix B

PROCESSED OBLIQUE SHOCK IMAGES

Figure B.1: Raw image of one taken during operation of the tunnel with shockwave stencil included. Framerate: 2500 fps, $175 \mu s$ exposure. 19 of 19.

Figure B.2: Images of 6 frames taken during operation of the tunnel with shockwave stencil included. Framerate: 2500 fps, 175 μs exposure. 2-7 of 19.

Figure B.3: Raw images of 6 frames taken during operation of the tunnel with shockwave stencil included. Framerate: 2500 fps, 175 μ s exposure. 8-13 of 19.

Figure B.4: Raw images of 6 frames taken during operation of the tunnel with shockwave stencil included. Framerate: 2500 fps, 175 μs exposure. 14-19 of 19.

Appendix C

RAW OBLIQUE SHOCK IMAGES

Figure C.1: Raw image of 1 frame taken during operation of the tunnel.
Framerate: 2500 fps, $175 \mu s$ exposure. 1 of 19.

Figure C.2: Raw images of 6 frames taken during operation of the tunnel. Framerate: 2500 fps, 175 μ s exposure. 2-7 of 19.

Figure C.3: Raw images of 6 frames taken during operation of the tunnel. Framerate: 2500 fps, 175 μ s exposure. 8-13 of 19.

Figure C.4: Raw images of 6 frames taken during operation of the tunnel. Framerate: 2500 fps, 175 μ s exposure. 14-19 of 19.

Appendix D

INDRAFT TUBE TUNNEL ASSEMBLY PROCEDURE

The following is a procedure for assembling the Indraft Tube Tunnel, which includes assembly of the chamber itself, as well as the the flowpath.

Safe Operating Procedure for Indraft Tube Tunnel Chamber and Flowpath Assembly

Risk Assessment

Risk	Consequences	Mitigation
Flange plate is dropped during assembly	<ul style="list-style-type: none">• Harm caused to feet or toes• Harm caused to appendages sticking through the V-band holes	<ul style="list-style-type: none">• Multiple diaphragms with a total factor of safety of 4 are used• Participants always wear close-toed shoes• Participants shall never stick appendages through the flange cover holes while it is pressed up against a flange• Participants will transport the 27-1/2" flange via cart, and only lift it into place; they shall not carry it from one location to another

Personal Protective Equipment

All participants are required to wear safety glasses and close-toed shoes at all times. Butyl gloves are to be worn when handling denatured alcohol and vacuum grease.

Safety Callouts

At no point while any flange is being assembled is a participant permitted to put a finger or other appendage through the bolt hole of both flange and flange plate. Each smaller flange requires at least 2 people in its assembly, and the large 27.5" diameter flange cover requires at least 4 people

List of Equipment

List of Fasteners. All dimensions in inches. Note that two wrenches of each nut size will be required for assembly, except for the 1/2"-20 Bolts.						
Fastener Type	Fastener Size	Wrench Size	Length	Material	Quantity	Fastened Pieces
Bolt	1/2"-20	3/4	2-1/2"	316 SS	10	Inlet Adapter-27.5" Flange Cover- Extension Adapter
Washer	.532" ID	1.25" OD	N/A	18-8 SS	10	Inlet Adapter-27.5" Flange Cover- Extension Adapter
Bolt	7/8"-9	1-5/16"	3"	18-8 SS	24	16" Flange Covers
Nut	7/8"-9	1-5/16"	N/A	18-8 SS	24	16" Flange Covers
Washer	.938" ID	2" OD	N/A	18-8 SS	24	16" Flange Covers
Washer	.938" ID	2" OD	N/A	Aluminum	24	16" Flange Covers
Bolt	1-1/8"-7	1-11/16"	4"	18-8 SS	20	27.5" Flange Cover
Nut	1-1/8"-7	1-11/16"	N/A	18-8 SS	20	27.5" Flange Cover
Washer	1.188" ID	2.5" OD	N/A	18-8 SS	40	27.5" Flange Cover
Bolt	3/4"-10	1-1/8"	2-3/4"	18-8 SS	8	11" Flange Cover
Nut	3/4"-10	1-1/8"	N/A	18-8 SS	8	11" Flange Cover
Washer	.812" ID	1.75" OD	N/A	18-8 SS	8	11" Flange Cover
Washer	.812" ID	1.75" OD	N/A	Aluminum	8	11" Flange Cover
Bolt	1/4"-20	3/16" Allen	1-1/4"	18-8 SS	12	Windows

List of Flange Covers. Dimensions in inches. All manufactured from 6061 Al. Each flange has a matching full-face gasket with the same bolt pattern and dimensions, with an ID equal to the Pipe Size							
Pipe Size	OD	ID	# Bolt Holes	Bolt Hole Dia.	Bolt Circle Dia.	Thickness	Quantity
10"	16"	Solid	12	1"	14-1/4	1/2"	2
20"	27-1/2"	7.550"	20	1-1/4"	25"	1-3/8"	1
6"	11"	Solid	8	7/8"	9-1/2"	1/2"	1

List of Consumable Materials

- Butyl Gloves
- Denatured Alcohol
- Kim Wipes
- Vacuum Grease

Procedure

1. Pre-Setup and Cleaning

- 1.1. Clean all flange surfaces with denatured alcohol and Kim Wipes until no residue is seen on the Kim Wipes
- 1.2. Lay out paper towels or Kim Wipes to set cleaned hardware on in preparation for assembly
- 1.3. Clean all flange covers on both sides, taking special care on the sealing surfaces
- 1.4. Clean all flange cover holes using the same method
- 1.5. Clean all full-face flange gaskets front and back using the same method
- 1.6. Clean all O-rings and O-ring channels

2. Flange Assembly

When possible, Aluminum washers shall be placed between the bolt and Aluminum flange covers to reduce wear on the covers. Stainless steel washers shall be used in between the nut and tank flanges. No Aluminum washers exist for the bolts on the 27-1/2" flange, so ensure that the smooth side of the Stainless-Steel washers are in contact with the Aluminum flange face. Use Fastener table for bolt, washer, nut, and wrench information.

- 2.1. Place the full-face gasket over the 11" flange cover and line up the bolt holes
- 2.2. Place Aluminum washers over the 8 bolts
- 2.3. Insert two of the bolts into the flange cover to keep the gasket aligned
- 2.4. With two people, lift the flange cover onto the flange
- 2.5. One person presses the cover against the flange while the other inserts bolts and hand tightens the nuts on the backside. Ensure that a washer is placed between each nut and the flange.
- 2.6. When all bolts are hand tight, the flange cover may be released, and the bolts can be tightened down
- 2.7. To tighten the bolts, a socket wrench is required on the bolt side, and a combination wrench is required for the backside
- 2.8. Tighten in a star pattern, ensuring not to overtighten; do not use a cheater bar
- 2.9. Repeat this same process for the two 16" flange covers

3. 27-1/2" Flange Assembly

For this procedure, at least four people are required. See Fastener table for bolt, washer, nut, and wrench information.

- 3.1. Place the full-face gasket over the 27-1/2" flange cover and line up the bolt holes
- 3.2. Place washers smooth-side up over the 20 bolts
- 3.3. Position the flange cover so that the side with 2 O-ring grooves is facing the inside of the chamber
- 3.4. Insert 4 of the bolts into the flange cover to keep the gasket aligned
- 3.5. With 4 people, lift the flange cover onto the flange

- 3.6. Do not place fingers through the fange cover and flange at any point. If the flange slips or is dropped, it could cause significant harm to an appendage that is sticking through the holes
- 3.7. 2 people press the cover against the flange while the other 2 people insert bolts and hand tightens the nuts on the backside. Ensure that a washer is placed between each nut and the flange.
- 3.8. When all bolts are hand tight, tighten four of the bolts on opposite sides with the wrench
- 3.9. The flange cover may now be released, and the rest of the bolts are tightened in a star pattern
- 3.10. To tighten the bolts, a socket wrench is required on the bolt side, and a combination wrench is required for the backside
- 3.11. Tighten in a star pattern, ensuring not to overtighten; do not use a cheater bar. The final product is shown in the figure below:

4. Nozzle Adapter Installation

Two people are required for these steps. Use the O-ring table for O-ring information.

List of O-Rings. Dimensions in inches				
Dash Number	Width	Material	Mated Junction	Quantity
266	1/8"	Buna-N	Inlet-Inlet Adapter; Extension-Extension Adapter Nozzle-Extension	3
268	1/8"	Buna-N	Inlet Adapter-Flange; Flange-Extension Adapter	2
275	1/8"	Buna-N	Flange-Extension Adapter	1
8.25" ID	3/16"	Buna-N	Window	2

- 4.1. Lightly grease the three O-rings that seal the mate between adapters and large flange cover with vacuum grease. No excess grease should be visible on the O-rings.
- 4.2. Install the O-rings with two on the inside of the chamber, and one on the outside. The figure below shows the final product from the inside of the chamber for this step
- 4.3. One person enters the chamber with the extension adapter and presses it into the counterbore on the large flange

- 4.4. One person on the outside looks through the holes in the large flange and instructs the person inside the chamber to rotate the extension adapter until the bolt patterns line up

- 4.5. One person on the outside places the inlet adapter on the flange and rotates it until the bolt patterns line up
4.6. A third person inserts the 10 ½"-20 bolts with one washer each, and hand tightens them
4.7. They are then tightened in a star pattern until the O-rings are depressed
4.8. The final product is seen in the figures below showing the inside and outside of the flang

5. Nozzle Assembly

See the O-Ring table for information on the proper O-rings to use

- 5.1. Two people enter the chamber with V-band and inlet adapter
5.2. Lightly grease a cleaned O-ring and install it on the Extension Adapter

- 5.3. Fully open the V-band and place it over the Extension Adapter
- 5.4. One person lifts the extension into place and presses it on
- 5.5. The other seats the V-band over the lip between the two pieces, and tightens the nut with a 9/16" wrench
- 5.6. Tighten until difficult to turn and the extension does not flex when loaded
- 5.7. Release the extension, and perform the same steps for the nozzle to mount it to the extension
- 5.8. The below figure shows the final product of the assembled nozzle:

Appendix E

INDRAFT TUBE TUNNEL STANDARD OPERATING PROCEDURE

The following is a procedure for safely running the Indraft Tube Tunnel in the Cal Poly Aerospace Engineering Propulsion Laboratory.

Safe Operating Procedure for Indraft Tube Tunnel Pressure Test

The following procedure outlines the risks, risk mitigation techniques, and process for safe operation of the Indraft Tube Tunnel. It assumes that the separate assembly procedures have already been followed.

Risk Assessment

Risk	Consequences	Mitigation
Diaphragm Bursts Unexpectedly	<ul style="list-style-type: none">• Hearing Damage to participants• Damage to glass windows/doors	<ul style="list-style-type: none">• Multiple diaphragms with a total factor of safety of 4 are used• Participants always wear hearing protection• Participants never cross the nozzle inlet plane while pump is running• .002” total of Mylar, .004” total of cellophane required• Inlet bell will be pointed away from any glass or sensitive structures• Range safety must halt pumping process if someone enters hallway without hearing protection after 100 Torr is passed.
C-Clamps fall when door contracts under vacuum	<ul style="list-style-type: none">• Someone could be hit on the head	<ul style="list-style-type: none">• While the chamber is pumping down, periodically tighten the c-clamps so that they are snug.
Diaphragm Bursts as Designed	<ul style="list-style-type: none">• Hearing loss to participants• Hearing loss to passers-by• Damage to glass windows or doors	<ul style="list-style-type: none">• Participants always wear hearing protection• Participants never cross the nozzle inlet plane while pump is running• Participants wear a face shield after the pressure in the chamber drops below 100 Torr• Range Safety gets stationed outside the lab in the corridor, and ensures that there is no one outside without hearing protection when the diaphragm is burst• Warn everyone in the adjacent labs (wind tunnel, machine shop, space environments, structures) that a test is going to occur

Personal Protective Equipment

All participants are required to wear safety glasses and close-toed shoes at all times.

All participants are required to wear hearing protection whenever the vacuum pump is on.

Safety Callouts

At no point while the chamber is under vacuum is a participant to cross the nozzle inlet plane.

All participants are required to stay in the control room once the pressure in the chamber passes below 200 Torr, resulting in a FOS of 6. A participant may approach the chamber after this limit so long as they are wearing a face shield and ear protection, and do not cross the nozzle inlet plane.

Procedure

1. Pre-Setup

- 1.1. Check oil level on vacuum pump. Fill oil to line if not full
- 1.2. Ensure that the interior of the chamber is free of loose debris
- 1.3. Close the chamber door and pinch shut with several c-clamps at different locations around the door until no light can be seen in the crack (see below).

- 1.4. Connect vacuum pump to KF-40 flange on the aluminum window of the chamber via bellows
- 1.5. Ensure that vacuum cutoff valve is in-line with the pump and closed
- 1.6. Ensure that a convectron gauge is also in-line with the pump, plugged in, and operational. Note: the convectron gauge reads in units of Torr.
- 1.7. See figure on right for example of vacuum setup.
- 1.8. Fully open the tent flaps on both sides of the chamber
- 1.9. Clear area of loose debris and large objects within 10 ft. of inlet bell
- 1.10. Clean area to reduce ingestion of dirt and small debris by the inlet

2. Diaphragm Inspection and Installation

- 2.1. Place V-band over the inlet adapter and allow it to rest on the straight section (see diagram)
- 2.2. Select 2 .002" and 1 .001" thick cellophane diaphragms or 1 .002" thick mylar diaphragm and inspect for deformities, punctures, tears, or scratches. Any marring in the surface of the diaphragm can cause a significant decrease in diaphragm strength, making it unusable. See diaphragm on right.
- 2.3. Ensure that the O-ring on the inlet bell adapter has been cleaned and lightly lubricated with vacuum grease
- 2.4. Press the diaphragms together against the O-ring, making sure to center them on the O-ring. The diaphragms should stick to the grease on the O-ring
- 2.5. Press the Inlet Bell against the diaphragms, making sure that it clears the locating lip on the inlet adapter before applying pressure. Note: from this point until the V-band is fastened, one person's sole job is to hold on the inlet bell.
- 2.6. Ensure that the diaphragm does not stick out any of the edges of the V-band flange
- 2.7. Completely remove 9/16" nut from V-band and slide it over the two lips
- 2.8. Press the V-band as tight as possible against the two lips to ensure that it does not scratch when tightened as seen below

- 2.9. Tighten the nut on the V-band with 9/16" wrench until difficult to turn and diaphragm distorts slightly under the light. Do not overtighten!

- 2.10. The diaphragm is now installed. See below figure for final product.

3. Diaphragm Trigger Installation

- 3.1. Suspend puncture device above diaphragm with string so that the point will hit the diaphragm near its middle
- 3.2. Attach nichrome filament to string with good contact so that the string will be burned when the nichrome resistively heats. See figure below.
- 3.3. Run the nichrome ignition wires to the power source, ensuring that there is no slack near the inlet

bell so that the wires to not get sucked in

- 3.4. Flip the switch on and test continuity with multimeter
- 3.5. If there is continuity, flip switch to off, close safety cover, and move on to procedure 4 (see below figure)
- 3.6. If continuity is not attained, check circuit for breakage and retest for continuity

4. Pumping

4.1. CLOSE ALL VENT VALVES

- 4.2. Clear the yard of unnecessary personnel
- 4.3. Close all doors leading from the yard except for the glass door leading to the control room.
- 4.4. Plug in convectron gauge to reader, turn on
- 4.5. All participants must don hearing protection
- 4.6. Open the vacuum valve a small amount, allowing air to flow slowly into the vacuum line
- 4.7. Incrementally open valve, ensuring that oil does not become ejected from the vacuum pump
- 4.8. Plug in vacuum pump to outlet in the yard
- 4.9. Walk around the door, inspecting for leaks. Readjust c-clamps as necessary
- 4.10. When the pressure begins dropping, the seal will improve substantially. Note: the c-clamps may become loose as the pressure drops, so tighten them periodically to ensure they don't fall off.
- 4.11. Listen for pops or bangs. It is possible that one or more diaphragms may pop during the pumping process. The test may continue after one pop, but the pump must be turned off and the diaphragms inspected after the second pop
- 4.12. When the pressure drops below 100 Torr all personnel must leave the yard and enter the test cell. Participants may reenter the yard if wearing a face shield in addition to hearing protection and safety glasses.
- 4.13. Close Vacuum Valve and notify range safety
- 4.14. Range safety must warn labs (wind tunnel, machine shop, space environments, and structures) not to enter the corridor without hearing protection as a test will occur in the next few minutes
- 4.15. In order to proceed and further decrease pressure, range safety must state that the corridor is clear of pedestrians
 - 4.15.1. Pedestrians may pass if they are wearing ear protection, but the valve must be reclosed if a pedestrian passes through without hearing protection
- 4.16. When the desired pressure is reached (15 Torr for this test), notify range safety and enter firing procedure

5. Firing

- 5.1. Range Safety must be stationed with at least one participant at either end of the long hallway next to the propulsion lab
- 5.2. When notified of readiness to test, range safety must warn the labs that a firing countdown is about to occur; all lab doors must be closed before the all-clear can be called
- 5.3. When the hallway is clear of unprotected people, range-safety sounds the all-clear, and the test conductor may proceed
- 5.4. Close the vacuum valve
- 5.5. Unplug the vacuum pump from the wall
- 5.6. Ensure that the ignition switch is off and the safety cover is closed
- 5.7. Connect one of the trigger wire leads to the positive, and one to the negative terminals on the battery car
- 5.8. All personnel other than range safety must enter the control room
- 5.9. Close control room door
- 5.10. Give count-down
- 5.11. Open safety cover of switch
- 5.12. Flip switch to initiate diaphragm burst
- 5.13. A loud boom followed by approximately 3 seconds of rushing wind should follow
- 5.14. In the event that the diaphragm does not puncture within two seconds of flipping the switch, flip the switch to off, close the cover, check the lead connection to the batteries, and return to step 4.16
- 5.15. The test controlled yard area is now inert and may be entered