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RAMJET TECHNOLOGY

Chapter 6
RAMJET FUEL SYSTEMS

by

W. GARTEN, JR.
and
G. B. SWARTZ

The Johns Hopkins University
Applied Physics Laboratory

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(Manuscript submitted for publication
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6. RAMJET FUEL SYSTEMS

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6.1 INTRODUCTION

The ramjet-fuel system comprises those elements of the ramjet power plant necessary for the delivery of fuel into the engine air stream in the quantity necessary to maintain the engine thrust required for missile flight. Since ramjet engines under development thus far have used liquid-hydrocarbon fuels almost exclusively, this chapter will be restricted to this type of fuel.

The thrust level of an engine may be controlled by varying one of two factors, the fuel flow or the air flow. The adjustment of the fuel flow affords the simplest means for accomplishing this operation. Variation of the air flow provides an alternate means for such adjustment but is undesirable because of the massiveness of the devices required.

Generally liquid-fuel systems must provide storage, pressure, flow-control, and distribution of the fuel into the air stream at the points of injection. Although conventional engineering techniques already exist for the design of an adequate fuel system, a proper understanding of the boundary conditions peculiar to the ramjet power plant are necessary for making the proper choice among these techniques. This discussion of ramjet-fuel systems will draw principally from the experience gathered in designing fuel systems for missiles where high maneuverability and rapid change of maneuver are required. An effort will be made, however, to distinguish between the peculiar needs of a particular missile and the general ramjet requirements.

Techniques best adapted to Mach numbers over the range 1.5 to 3.0 are given in this chapter since this is the most useful Mach number range for the ramjet. The special environmental conditions imposed on the fuel system as a necessary part of flight in this Mach number region are basic to the design of the system. An effort is made here to set forth the major components of this environment.

The ramjet engine usually attains the high Mach number flight regime with the aid of a rocket booster. If the boost time is less than five seconds the fuel system is under a high acceleration load for this period. If the boost time is appreciably greater than five seconds it may prove profitable to have the ramjet engine supplement the thrust of the rocket during the latter portion of the boost period. This will require the fuel-control system to function under the rocket-acceleration load. In either c se the fuel system must be capable of providing a flow of fuel for full engine thrust a few tenths of a second after booster separation as the economics of booster design dictate that waste of boost velocity be held to a minimum. In cases where the booster rocket plugs up the duct exit until separation, a sudden drop in duct pressure at separation is followed by an abrupt rise when burning starts. The flow control and fuel pressurizing equipment must maintain proper fuel flow in spite of these changes in system-pressure drop (from pump outlet to injector).

Other problems are also encountered as a result of operation in this flight regime. For example, a further requirement upon the fuel system is that the feedback of duct-pressure oscillations through injector-outlet pressure or structural vibration must not be allowed to set up regenerative oscillations in the fuel flow. Operation at high Mach numbers means that the missile is immersed in air at high temperatures and under conditions productive of high convective heat transfer. Additional problems are introduced as a result of climb to high altitudes and possible variability of thrust and speed. These factors lead to wide ranges of flow over which the fuel system must be operable with corresponding complications in respect to accuracy and stability of control.

In addition to boost-phase acceleration, proper fuel-system design must take into account possible roll, lateral acceleration, and vibration of the missile airframe. Roll velocities of several revolutions per second, roll acceleration of $500^{\circ}/sec^2$, lateral maneuver of over 10 "g", and vibration of several "g" up to 1000 cycles per second are possible in missile application. Fore and aft acceleration of a few "g" maximum during flight must also be considered, especially in regard to fuel-control stability. Another significant design factor is the change in the missile center of gravity as fuel is consumed.

In addition to the physical environment, the fuel-system design should include a number of other factors, a few of which are discussed here. The general high performance required of missiles (partly due to the high energy expenditure required to achieve and sustain supersonic flight speed) dictates that weight and space requirements of components be held to a minimum. This condition makes it necessary to consider the interrelation of fuel-system and over-all design and emphasizes that a high degree of over-all integration is needed for efficient missile design. Reliability and ease of testing expresses another prime need in a missile-fuel system. Once the system has been factory tested the probability of malperformance should be exceedingly small. This can be assured only through proper design. In this connection, one factor which lessens the difficulty of achieving a satisfactory design is the short operating lifetime of the missile components. Improvement in the design is facilitated if provision is made for simplified testing after shipment from the factory.

A further general consideration necessary to proper ramjet-engine development concerns the role which the fuel system plays in such development. While steady progress is being made toward making the design of combustors and diffusers an exact science there remains a certain amount of art in the design of each of these engine components. Further, the testing of these engine components is a costly operation. In order to proceed with airframe design, it is necessary to specify the diffuser and, to a lesser extent, the combustor at an early stage in the engine program. The fuel system must therefore adjust to the detailed performance of these components, as later determined, in order to secure optimum engine performance. Sufficient flexibility must be provided in the functional relation between fuel flow and speed and atmospheric conditions of flight to permit adjustment at a late stage in engine development.

This general discussion of boundary conditions pertinent to the design of ramjet fuel-system components is meant to supply a framework within which the problems peculiar to each component of such a system may be discussed. Since detailed design of most of the components is simple once the pertinent boundary conditions are properly understood, this exposition will not treat these components beyond a qualitative discussion of the applicable boundary conditions except in the case of the fuel-flow control.

6.2 FUEL TANKS

The basic fuel-tank configuration is principally determined by general airframe-design requirements and by the effect on aerodynamic performance of missile center-of-gravity changes resulting from expenditure of fuel. Fuel-system needs can usually be satisfied by special provisions within the framework of this basic specification. Two simple fuel-tank configurations which can serve as illustrations are shown in Fig. 6.2-1. Figure 6.2-1(a) shows a cylindrical tank using a pressurizing gas between the tank shell and a plastic fuel bladder. The bladder contains the fuel and a central outlet tube. In Fig. 6.2-1(b) is shown an annular tank design with a plastic bladder immersed in the fuel. The bladder is inflated with a pressurizing gas. As shown in this cross-sectional view of the tank, two sectors of the annular region are used to permit simple installation and attachment of the fuel bladder. The latter arrangement also requires a system of perforated ducting strips laid on the inner tank body to insure against momentary interruption of flow from a partially filled tank. All of this rigid ducting represents volume of fuel which must be carried but is not available to the engine. Bladder and tank design must take proper account of loads developed under boost acceleration, lateral maneuver, and roll.

In these configurations a plastic bladder is used to separate the fuel from a pressurizing gas. This method of construction is dictated by a combination of the following factors: (a) high accelerations which could isolate the fuel from the outlet, (b) flight at high altitudes where air, which was absorbed when the fuel was originally stored at sea level, could be released to give momentary interruption of fuel flow with the consequent problem of restarting the engine, and (c) the need to prime a centrifugal pump rapidly during the boost phase.

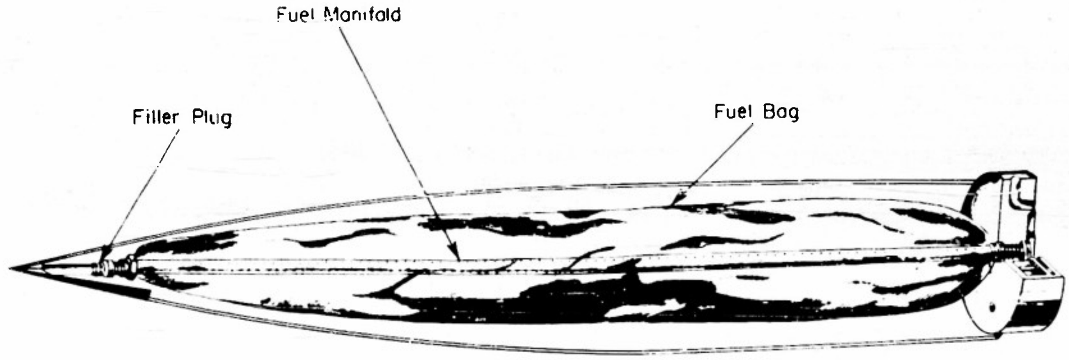

Fig. 6.2-1(a) CYLINDRICAL FUEL TANK CONFIGURATION

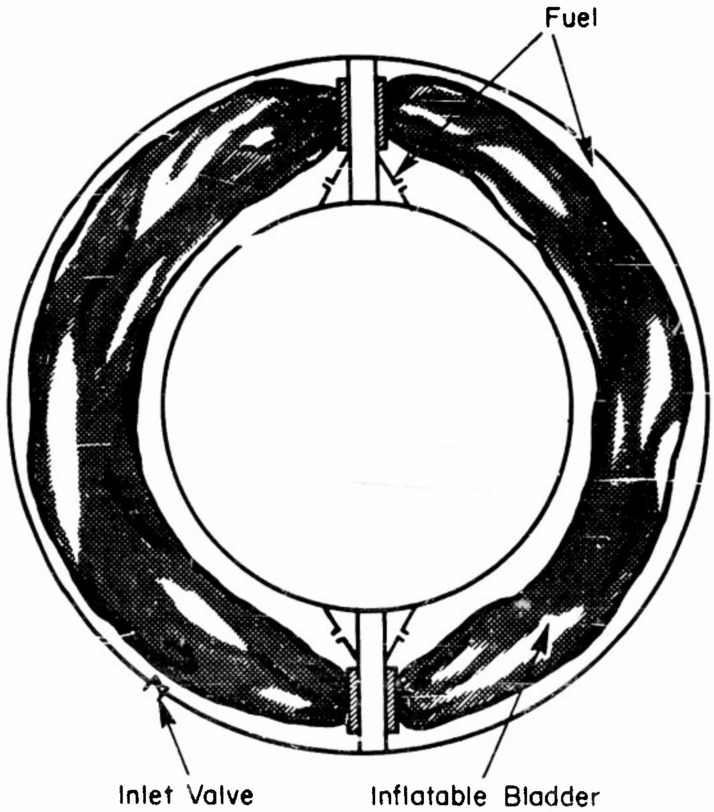

Fig. 6.2-1(b) ANNULAR FUEL TANK CONFIGURATION

Low maneuverability, low fuel volatility, and high available ram pressure at high altitudes might combine in some applications to make such a provision unnecessary. With hydrocarbon fuels and at velocities around 2000 ft/sec, these bladders may be made from nylon cloth coated with neoprene rubber. At higher velocities special silicone materials may be required to prevent burn-through under unfavorable conditions. Major material problems in regard to fuel bladders may arise if special liquid fuels are used.

The heating of components, resulting from immersion of the airframe in an air stream at high stagnation temperature, is a general problem of supersonic flight. In the ramjet fuel system this problem centers in the fuel tank. The temperature level of the entire fuel system in most designs is determined by conditions in the tank, since the fuel flows from the tank through the remainder of the system. Recovery factors and heat transfer coefficients for the calculation of expected fuel temperature may be found in Refs. [1] and [2]. In view of the high values of the heat transfer coefficient under the air flow conditions encountered in supersonic flight, it is possible for a small amount of thermal insulation of the tank wall to effect a large change in heat input to the tank fuel for short and medium ranges of flight. Insulating coatings or a double fuel bladder arrangement offer attractive forms for introducing such insulation. In some long-range designs it may prove to be economically feasible to make use of the fuel-turbine exhaust air to keep the fuel temperature appreciably below air stream stagnation temperature.

The major problems of fuel-system design resulting from air stream heating are: (a) the need for high pressure and consequent high stress of the fuel tank to prevent boiling, (b) the danger of cavitation of a centrifugal pump, (c) the need for special materials for fuel bladder construction, and

(d) the possibility of boiling in the fuel injector. The particular considerations of a specific application will strongly condition the choice of means for solving these problems.

6.3 FUEL PRESSURIZATION AND PUMPING

The need to inject fuel into a high pressure region in the ramjet engine plus the losses incurred in ducts, fuel control, filter, and injection nozzles requires that the fuel system provide means for pressurizing the fuel. This requires either fuel-tank pressurization or fuel pumping or both. Gas pressurization of the fuel tank provides the simplest means for pressurizing fuel in short-range missiles where weight and packaging considerations are not too pressing, for instance, in engine test vehicles. For optimum designs a pump must provide the major share of pressure at low altitude in order that the weight of the fuel-tank wall will not be excessive. The use of a pump introduces the problem of vapor lock caused by fuel boiling or the release of dissolved air. To solve this problem, some means of tank pressurization must be provided [3]. The resulting pressure must be greater than the total of the vapor pressure of the fuel, the solution pressure of gases dissolved in the fuel, the pressure drops in the ducting to the fuel pump inlet, and the pressure needed to avoid cavitation at the pump inlet. For low volatility fuels (kerosene, for example) pressurization by use of the free stream impact pressure may provide a simple system usable up to about 30,000 feet or even higher if special procedures can be used to assure that the quantity of dissolved air is appreciably below its normal storage value. Higher altitude operation requires the use of a more complicated tank-pressurization system. Systems based on the use of stored gas, vapor pressure of stored liquids, and an air pump driven by the fuel pump prime mover have all been studied [4,5,6]. The use of an air pump should provide the best system for extended range. For shorter ranges the stored-gas system is competitive with the air pump. The system based on the use of a volatile liquid is unattractive in view of the heat exchange required and the narrow tolerance with respect to missile storage temperature.

Figure 6.3-1 shows a fuel tank pressurization system based on the use of stored gas or diffuser-duct impact pressure, whichever is the higher. This arrangement provides adequate pressurization regardless of the flight path. The stored nitrogen is sufficient to permit trajectories which terminate at high altitudes. The use of diffuser-duct pressure provides ample pressure for trajectories which terminate at low altitudes without requiring nitrogen storage for other than high altitude needs. In the early part of flight, tank pressure is controlled to a fixed value above free stream impact pressure. This acts to prime the pump during the boost phase.

Bleeding a portion of the high pressure duct air and discharging it into the ambient pressure freestream is the most readily available source of power for the ramjet engine fuel pump. The most compact device available for converting this potential power into increased fuel pressure when the fuel flow rate is high is a combined single-stage air turbine and centrifugal pump mounted on the same shaft. Characteristic curves, showing efficiency and pump-pressure rise as functions of fuel flow, of such a unit, designed by the Bendix Aviation Corporation for use in a Bumblebee missile are shown in Fig. 6.3-2. This unit, designed in 1951, weighs 9.25 pounds and has an outer casting diameter of seven inches and a thickness of five inches. In general the design follows normal engineering practice. Features requiring special attention in design of this unit include an adequate low-net positive-suction head to avoid cavitation for the inlet pressures available, a shaft seal between turbine and pump, and sufficient strength and internal clearance to permit efficient operation when the unit is subjected to gyroscopic torques resulting from a radially mounted

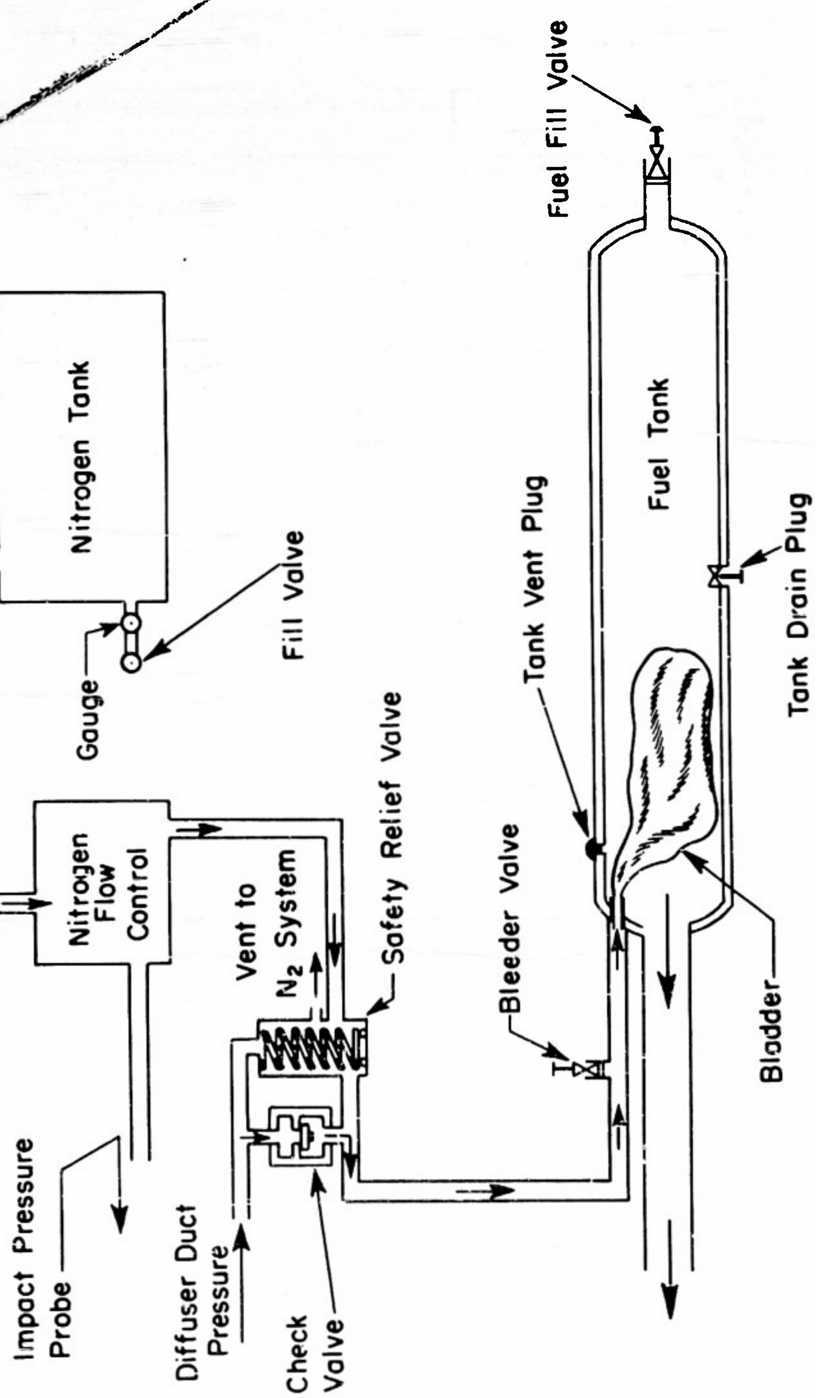

Fig. 6.3-1 FUEL TANK PRESSURIZATION SYSTEM

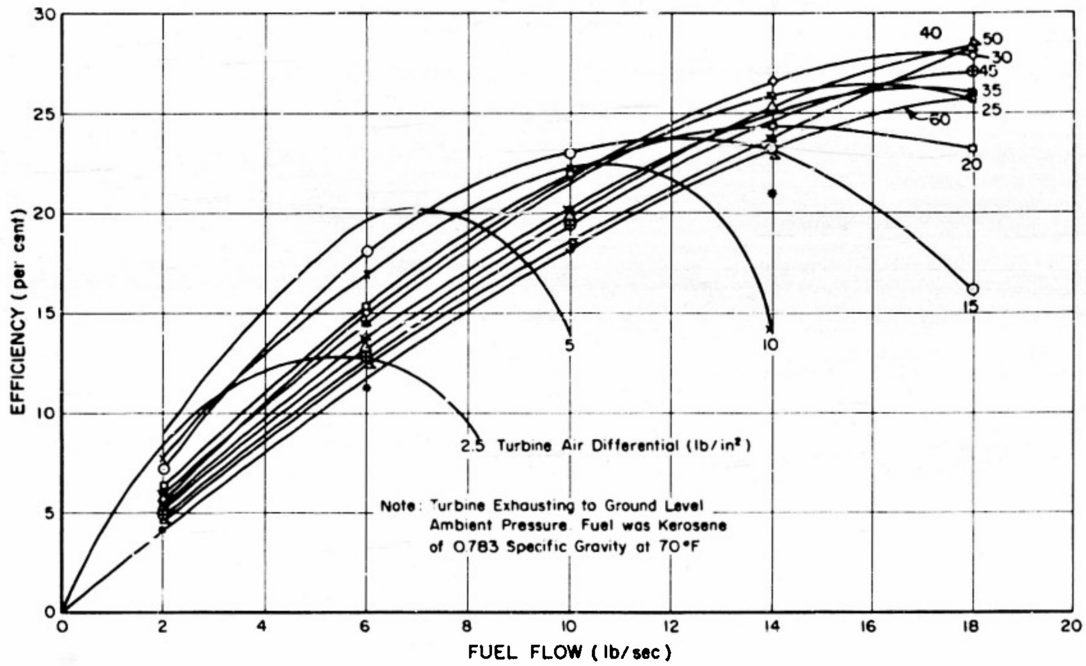

Fig. 6.3-2(a) OVER-ALL EFFICIENCY OF TALOS FUEL PUMP

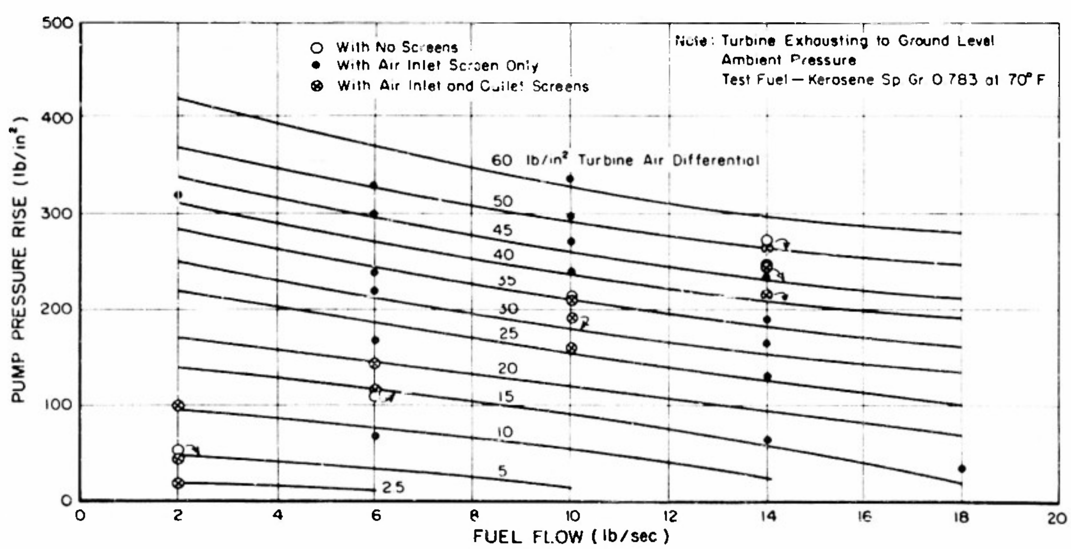

Fig. 6.3-2(b) PUMP-PRESSURE RISE AS FUNCTION OF FUEL FLOW OF TALOS FUEL PUMP

shaft in a missile rolling at instantaneous rates as high as three revolutions per second. Some economy in bearing design is possible in missile applications because of the short lifetime of approximately five hours maximum (including all testing).

While this unit represents an adequate ramjet fuel-pumping system, it must be recognized that electrical and hydraulic power are also needed in a missile. Since an air turbine also provides an efficient power source for these needs, the most economic power source would be one turbine with two pumps and a generator to supply the required outputs. Such a power system has not appeared practical for the high-maneuverability Bumblebee missiles. The requirement for stable electrical power during boost and the great variability of the hydraulic duty cycle make independent power supplies the more attractive choice. In general, the degree to which missile power supplies can be integrated must be decided in the light of the whole missile design, with fuel-system requirements representing only one of several considerations.

6.4 FUEL DISTRIBUTION

The ramjet fuel injector represents a point of overlapping interest to both the fuel-system and combustor engineer. The discussion in this section shall be confined to the requirements for securing an equal flow of liquid fuel from a number of discreet injection points. A solution to this problem requires that the distribution of fuel to the injection points be kept within a given tolerance in the presence of a number of upsetting factors without making excessive demands on the fuel pump or the fuel-pressurization system. In the Bumblebee program, injection design is generally aimed at an over-all flow tolerance at a single injection point of ±20 per cent of the average value for the injector. No extensive testing program has been carried out to establish this tolerance, but a number of injectors designed on this basis have proved adequate in both static and flight tests. Upsetting factors include the evolution of dissolved gas vapor because of reduced pressure or heating and the effect of lateral acceleration due to maneuver on fuel flow from injection points situated differently in this force field.

Evolution of gas vapor in the injector may result, in an extreme case, in violent spurting of the fuel or, if the evolution is slow, in the entrainment of a number of small bubbles in the fuel lines. The latter case is dangerous because it enhances the effect of lateral acceleration and makes the fuel flow responsive to pressure in the region of the injector. This pressure is controlled by the rate of burning at the combustor and is consequently a function of fuel flow. Therefore, the mechanism for an unstable oscillation is present once the volume of entrained bubbles becomes sufficiently large. Even without entrained bubbles, fuel distribution may be seriously affected if pressure developed in the fuel lines by lateral maneuver is comparable to the loss of fuel pressure which occurs between the point of division of flow to the injection points and the release of the fuel into the air stream.

The Talos missile uses a kerose -type fuel to permit adequate vaporization in the combustor while maintaining a low vapor pressure in the injector. Fuel atomization is obtained from the effects of air blast when the fuel is released at a low velocity into the high speed air stream. Fuel flow over a range of about forty to one must be handled by this injector. The main fuel is injected through sixteen low pressure spring-loaded nozzles which are fed from a central distribution valve through individual tubes. The nozzles serve to maintain pressure in these tubes at a level which prevents the formation of bubbles. The central distributor valve consists of a pair of close fitting, hollow cylinders into which sixteen matched holes have been radially drilled. The inner cylinder is spring loaded and is free to move axially. Fuel pressure acting against this load results in simultaneous opening of the matched holes and equality of flow to all points of injection in spite of poor matching of the individual spring-loaded nozzles or lateral maneuver of the missile.

Since the pilot injector releases the fuel into a region of low air velocity, the nozzle is required to atomize the fuel. Fuel flow is over a range of ten to one in the pilot-fuel system. The pilot injector consists of sixteen matched spring-loaded nozzles fed from a manifold. These nozzles have a linear relationship of flow to pressure drop and are set to give an adequate pressure to avoid bubbling at the low flow condition. A filter in the fuel system immediately downstream of the pump serves to protect both nozzles and fuel controls from foreign materials which may have gotten into the fuel.

6.5 FUEL FLOW CONTROL

In order to produce the desired missile-flight performance the ramjet fuel control must make proper use of the thrust capabilities of the ramjet engine. The fuel flow must be controlled in order to secure engine operation within a desired regime over the desired range of missile-flight conditions and to produce the thrust necessary to accomplish the missile-flight objective. The objective of this section is not a detailed analysis of the multi-variable problem of missile design optimization of which the fuel-control requirements are only one aspect. Only the techniques used to conform to a particular set of requirements are discussed.

Ramjet fuel-control requirements are generally related to one of two sets of missile requirements, either the short-range antiaircraft missile or the long-range bombardment missile. The short-range missile is characterized by a short-flight duration, high-acceleration boost, high maneuverability with accompanying rapid variations of drag, and a wide range of possible flight paths. Thrust rather than efficiency is the prime consideration, with a small fuel consumption (relative to total missile weight). The long-range missile will probably have a lower acceleration requiring a longer duration of boost, low maneuverability, a predetermined flight path, high fuel to total missile weight ratio, and high engine efficiency.

The closed-loop type of ramjet engine fuel-air mixture control seems to prevail in engines both in use or under development today for the control of missile-flight performance or the attainment of a desired engine operating condition. The term "closed loop" means, briefly, that any variation in a controlled quantity, such as flight Mach number, will cause a change in the controlling quantity, such as fuel-air mixture, and this latter change in turn produces a further variation in the controlled quantity, thus forming a "closed loop". Wide variation exists in the choice of means for determining and specific mechanisms for attaining the desired engine operation and flight performance.

The fuel controls used in Bumblebee ramjet engines (short-range missile) obtain their sense of engine inlet air-flow parameters from the measurement of missile flight variables. The desired fuel flow (or the limits of a range of desired fuel flow) is derived from these measurements. The relationship between desired fuel flow and the missile flight variables is determined by ground tests. The fuel-flow control measures the flow of fuel and compares this with the derived value of required fuel. A sense of difference between desired and measured values of fuel flow serves to activate a fuel-flow controlling mechanism (a throttle in the fuel line or in the turbine-pump air line) to change the fuel flow in the direction which will reduce this difference. No direct measurement is made of the engine reaction to the fuel flow.

The basic engine-control system may be divided into three parts: the measurement of flight variables, the derivation of a sense of desired fuel flow from these variables, and the use of a feedback fuel-flow control device to attain this desired fuel flow. Since the second of these is the means for satisfying the engine requirements, it will be discussed first in relation to the simple properties of the ramjet engine.

Fuel Mixture Control

Several ramjet engines (including the Bumblebee ramjet engine) employ a fuel-control system in which a sense of the engine inlet air-flow parameters is obtained from measurement of missile-flight variables. This class of fuel controls is discussed in this section. The following equation serves to illustrate the principles upon which this class of fuel controls is based [7].

$$m_a \phi S_a = p_2 A_2 \left[1 - (C_{db} - 2) \frac{\gamma M_2^2}{2} \right] . \qquad (6.5\text{-}1)$$

C_{db} is the coefficient of burner drag; the numerical subscript refers to the station indicated on the ramjet-engine schematic diagram shown in Fig. 6.5-1. The quantity ϕ is a function of Mach number and is the thrust correction for the effect of the exit nozzle. This equation indicates that the stream thrust at the combustion-chamber exit is equal to the difference between the stream thrust at the diffuser exit (Station 2) and the burner drag.

In many ramjet applications, in particular if the diffuser is of the normal shock type and high thrust is a primary need, the fuel-flow control requirements are met if the ratio of engine-air flow to fuel flow is made constant. This may be accomplished if the fuel flow (m_f) is controlled in proportion to the absolute static pressure immediately ahead of the combustor or in proportion to some more convenient pressure comparable in magnitude to that pressure. In practice, the burner-drag coefficient is nearly equal to 2. If C_{db} is assumed to be 2, and if m_f is proportional to p_2, then Eq. (6.5-1) reduces to

$$\frac{m_f}{m_a} = K \frac{\phi}{A_2} S_a \qquad (6.5\text{-}2)$$

where K is a proportionality constant.

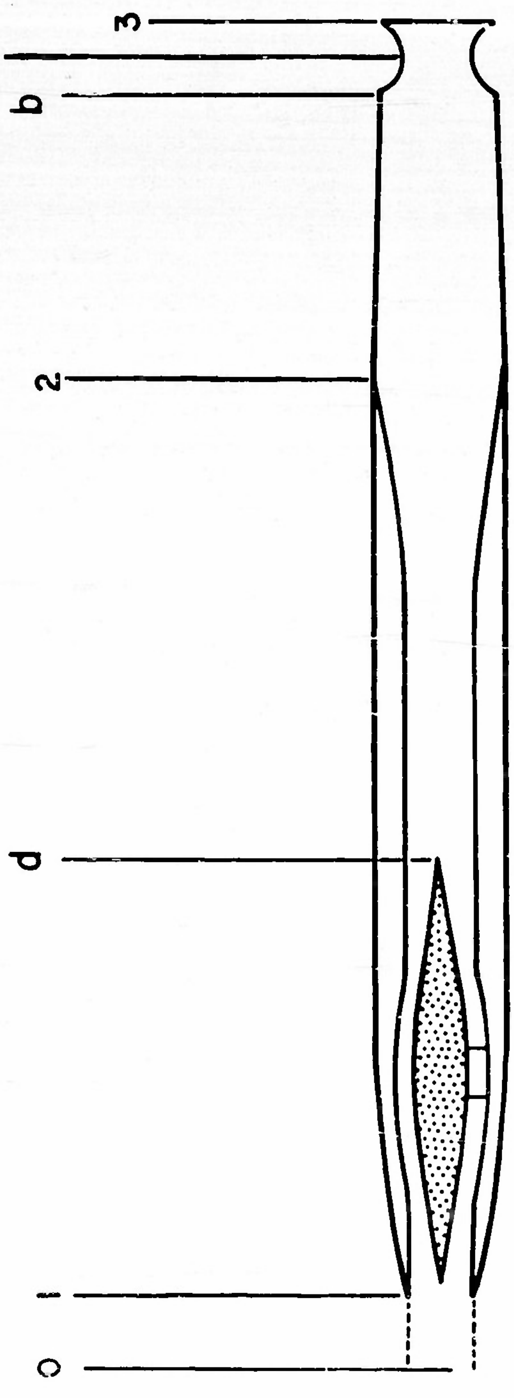

Fig. 6.5-1 SCHEMATIC DIAGRAM OF A RAMJET ENGINE

Since, for a given fuel and combustor, S_a is strongly dependent on the fuel to air ratio (but only weakly dependent on ambient atmospheric conditions and flight Mach number over a considerable range), this expression effectively relates the fuel to air ratio to the value of K chosen for the control mechanism (a principle of fuel control originally proposed by the Esso Laboratories [8]). Bendix fuel controls for Bumblebee missiles with normal shock diffusers make use of a variant of this principle in which free stream pitot pressure is used in place of p_2. For spillover conditions (normal shock ahead of diffuser entrance) these two pressures differ by the diffuser losses which are nearly proportional to free stream pitot pressure and by the dynamic pressure in the region ahead of the combustor which is a small quantity at the maximum thrust operating point. Deviations of C_{db} from the value 2, necessary to establish Eq. (6.5-2), only act to produce small effects generally not important to the control accuracy required for shorter-range missiles (±5 per cent or more for Bumblebee missiles).

If the coefficient of burner drag in Eq. (6.5-1) is assumed, as previously, to be 2, then the mass airflow is

$$m_a = \frac{p_2 A_2}{\phi S_a} . \qquad (6.5\text{-}3)$$

Replacing p_2 by the ram pressure p_T, as previously indicated for the Bendix fuel controls, the expression for mass airflow reduces to

$$m_a \sim \frac{p_T}{S_a}$$

If both sides of this expression are multiplied by $\frac{m_f}{m_a}$ one obtains the expression for required fuel flow, $m_f \sim \frac{p_T}{\frac{m_a}{m_f} S_a}$.

Since S_a is primarily dependent on air to fuel ratio, the fuel flow required for constant air to fuel ratio is $m_f \sim p_T$. This proportionality is designated as the fuel metering function.

An alternate form of expression for this fuel metering function is

$$m_f \sim \frac{p_T}{\sqrt{p_o}} \sqrt{p_o}. \qquad (6.5\text{-}4)$$

This modified form has been used as the basis for Bumblebee fuel controls for two reasons. In the first place, a device for realizing the first half of this function was in existence very early in the program; secondly, the form specified by Eq. (6.5-4) allows greater flexibility, since it provides for two-variable control rather than just one if the function $m_f \sim p_T$ were used directly. The first half $\frac{p_T}{\sqrt{p_o}}$ of this fuel metering function is controlled by means of the air-piston (1) shown in the Bendix fuel meter schematic diagram, Fig. 6.5-2; the second half $\sqrt{p_o}$ represents the contoured altitude valve (4).

From this figure, it may be seen that when the Mach number control valve (7) is closed (rich limit operation) one side of the air-piston is subjected to atmospheric pressure and the other side to a higher pressure derived from ram pressure. The force on the air-piston caused by this pressure differential is transmitted through a force pin to a fuel-piston (2) which exerts an oppositely-directed force on the force pin. This is caused by the differential between the unmetered and

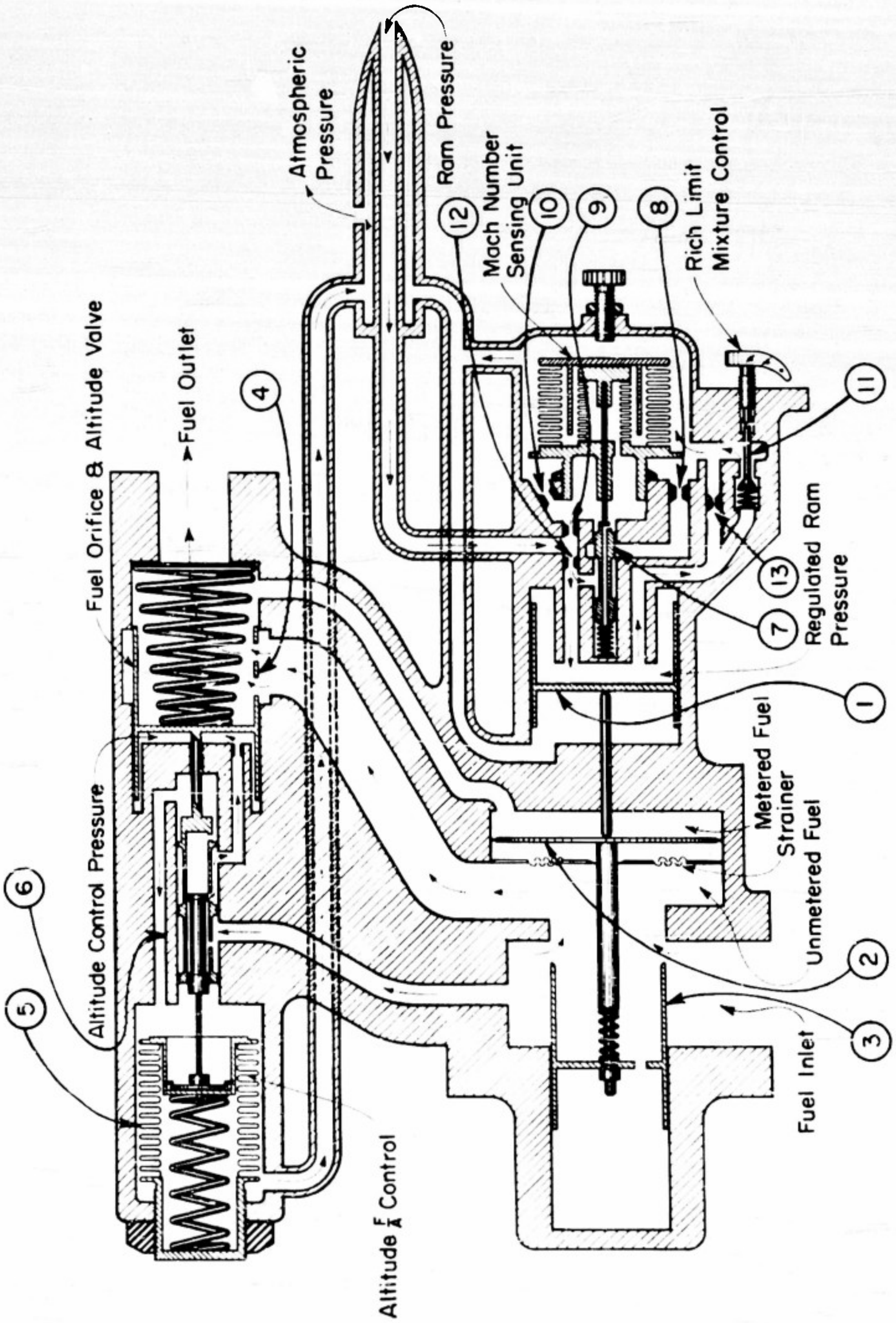

Fig. 6.5-2 SCHEMATIC DIAGRAM OF MODEL BB-5 FUEL METER

metered fuel pressures (pressure drop across the altitude valve). These forces are in balance when the ratio of the air-piston Δp to the fuel-piston Δp is equal to the ratio of fuel-piston area to air-piston area. A change in the air-piston Δp will then cause a force unbalance which tends to produce a movement of the poppet valve (3) which controls the fuel flow. As a result of the changed fuel flow, the fuel metering pressure differential either increases or decreases until a new force balance is achieved. Since the square of the fuel flow is proportional to the fuel-piston Δp (at constant altitude) which in turn is proportional to air-piston Δp, it is evident that this fuel meter will maintain fuel flow in proportion to the square root of the air-piston differential pressure.

To fulfill the first half of the condition specified above [Eq. (6.5-4)], the air-piston differential pressure must therefore be proportional to $\frac{p_T^2}{p_o}$. This is done by placing two restrictions, (12) and (13), in series in the ram tube. The up-stream restriction (12) is under sonic conditions, while the pressure drop across the downstream restriction (13) is equal to the air-piston Δp. The mass-air flow (m_{a_α}) through (12) is then

$$m_{a_\alpha} \sim \rho_o a_o \backsim \frac{p_T}{\sqrt{T_o}} . \qquad (6.5\text{-}5)$$

Assuming further that the pressure in the region between the restrictions is only slightly greater than atmospheric so that compressibility of the air need not be considered, the pressure drop across the downstream restriction (13) is

$$\Delta p_\beta \sim \rho_\beta v_\beta^2$$

since $\rho_\beta v_\beta A_\beta = m_{a_\alpha}$, then

$$\Delta p_\beta \backsim \rho_\beta \frac{(m_{a_\alpha})^2}{\rho_\beta^2} \sim \frac{(m_{a_\alpha})^2}{\rho_\beta} .$$

But $\frac{1}{\rho_\beta} \sim \frac{1}{\rho_\alpha} \sim \frac{T_\alpha}{p_\alpha} \sim \frac{T_o}{p_o}$. In addition, from Eq. (6.5-5), $m_{a_\alpha}^2\, T_o \sim p_T^2$. Therefore

$$\Delta p_\beta \sim \frac{p_T^2}{p_o} . \qquad (6.5\text{-}6)$$

It is evident, then, that the application of this pressure differential Δp_β across the air piston will make it possible to maintain a constant fuel-air ratio under spillover flight conditions.

The assumption concerning the incompressibility of the air in the region between the restrictions implies, in a strict sense, that the flow velocity in this region must be zero. If this were the case, the pressure drop across the downstream restriction would be zero, thus invalidating this method. This difficulty is overcome by choosing the size of the downstream restriction so as to compromise between the error caused by compressibility and a low value of pressure drop across the restriction.

The second part $\sqrt{p_o}$ of the fuel metering function specified by Eq. (6.5-4) is controlled by the contoured altitude valve (4) in series with the pressure regulating poppet valve (3). The positioning of this altitude valve is accomplished through the use of a servomechanism pilot valve in accord with changes in atmospheric pressure sensed by the evacuated bellows (5). By controlling the orifice area of the altitude valve so that it will vary in proportion to the square root of atmospheric pressure, the fuel flow as determined by the first half of the fuel metering function is modified in accord with the second half of the function to produce a constant air-fuel ratio. The proportionality constant of Eq. (6.5-4) is controlled by the choice of the area of the contoured altitude valve, the ratio of the areas of the fuel and air pistons, and the level of the air-piston differential pressure.

The preceding discussion indicates the simple general approach to the ramjet fuel-control problem adopted in the Bumblebee program. Adaptation of the control mechanisms developed in the early stages of the Talos missile program involved a final step wherein the simple devices, developed to conform to a simple representation of the ramjet engine, were corrected to fit the needs of the engine as determined by ground testing. This process involved principally the use of a mixture ratio varying as a function of altitude. Major factors which appear in the final engine but were not considered in the simplified analysis are variations of burning efficiency with pressure and air temperature, judgment in the balance between missile thrust and range capability in arriving at a desired rich limit setting, and the split-up of fuel flow between the pilot and main burner in the dual fuel system. The mechanisms developed have sufficient flexibility to make this final adjustment to the engine.

In addition to controls of the type used in the Bumblebee program or alternate mechanizations of these same principles of control, fundamentally different approaches to the ramjet fuel-control problem are being studied. One such approach consists of taking the maximum diffuser pressure as a criterion for a maximum thrust operating point. The United Aircraft Corporation developed a control of this type for the Meteor program [9]. This type of control offers a simple means for securing maximum engine thrust independent of engine design to a first approximation; thus, it holds the possibility of a universally adaptable mechanism for engines where thrust is of paramount importance. Several factors act to limit the adaptability of such a control. Three of these are listed below:

1. High thrust up to the point where fuel consumption is excessive for unit increase of thrust will usually be more desirable than peak thrust.

2. Pressure may not maximize in some acceptable engine designs since an increase in fuel flow may generate an increase in diffuser pressure up to a point where diffuser instability brings about a discontinuous drop in diffuser pressure.
3. Universal applicability may be seriously compromised by the need for rapid response in such a control. If the dynamic properties of the relationship between fuel flow and engine thrust are a significant consideration in the stability of the control then universal use is not possible and the control must be matched to the engine in full-scale tests.

In general, this type of control should be given consideration for its promise of simplicity of mechanism for cases where it is applicable. Some degree of flexibility in the mechanism can overcome, in a large part, the objections raised above.

Another noteworthy principle of fuel control is the control of the position of the normal shock in the diffuser, or more generally, the control of diffuser operating regime by direct measurement. While no serious effort has thus far been made to develop such a control, some thought has been given to devices applicable to this purpose [10]. This control would find its application in long-range vehicles where highly optimized diffuser design and the need for most efficient use of the engine would require that heat release be adjusted to maintain the diffuser at its most efficient operating point. A control system where the rate of climb is used to control flight at design Mach number and a shock-position control is used to adjust fuel flow for most efficient engine operation, should result in efficient flight of a long-range vehicle.

A third approach to the fuel-control problem is based on the direct measurement of air flow in the engine air duct.

Pitot static tubes, hot wire anemometers and spinning propellers are among the means suggested for securing such measurement. Generally this approach suffers from the fact that a single representative point of measurement is not available for the range of engine operating conditions. The alternative of using a number of points of measurement significantly increases the complexity of the required equipment, making this approach appear unattractive for most applications.

6.6 MISSILE FLIGHT CONTROL

In addition to fuel-mixture control, it is also usually necessary that the fuel control provide means for maintaining a desired missile flight speed. This section presents the modifications in the basic Bumblebee fuel-metering system which have been made in order to achieve missile flight control.

In this category, the first additional requirement which has been placed upon the system is that the flight Mach number be maintained constant. To do this, a Mach number sensing unit, shown in Fig. 6.5-2, is included. The general function of this unit is to maintain the thrust of the missile equal to the missile drag at the required Mach number by controlling the fuel-air ratio.

The essential element of the Mach number sensing unit is an evacuated double-bellows assembly which has one fixed end; the other (free) end is connected to the balanced air-piston by-pass valve (7). The larger bellows is subjected to atmospheric pressure, the smaller bellows to ram pressure; the area ratio of the two bellows being equal to the ratio of ram pressure to atmospheric pressure at the desired limiting Mach number. With such an arrangement, the position of the by-pass valve remains fixed regardless of altitude as long as the design Mach number is maintained as the ratio of ram pressure to atmospheric pressure is constant for a given Mach number. Any tendency for the Mach number to increase above the design value causes the by-pass valve to open. As a result the regulated ram pressure is bled down and the fuel-air ratio is decreased.

By properly selecting bleeds at (9) and (10), it is possible for the Mach number sensing unit to sense a higher Mach number than the design value. The pressure in the small bellows consequently reaches a value less than full ram pressure at the design Mach number; thus rich limit flow conditions are maintained until a higher Mach number is reached.

Lean and rich air-fuel ratio limits are controlled through the proper use of bleeds, the lean limit through bleed (8), the rich limit through a manually adjustable bleed (11) in series with the ram bleed (12).

The operation of the Mach number sensing unit is based upon the fact that the ratio of ram to static pressure is fundamentally a function of Mach number. It is evident that, if the pressure ratio applied to the evacuated double bellows were a function of velocity rather than Mach number, this unit could function as a velocity-control device.

Mach number, and therefore ram to static pressure ratio, may be expressed as a function of velocity and stagnation temperature. If the ram-static pressure ratio can be modified in such a way as to remove its dependence upon stagnation temperature at the desired control velocity, then a pressure ratio which is a function of velocity only may be obtained.

A velocity-control system based upon the above general principles and produced by Bendix for the Talos missile is shown in Fig. 6.6-1. A comparison of this figure and Fig. 6.5-2 shows that the principal modification is the use of a bellows-powered contoured variable bleed which reduces the ram pressure before it is applied to the smaller of the evacuated double bellows described in the preceding section. The motion of the bellows which powers the variable bleed is proportional to the differential pressure across the bellows. This differential pressure in turn is the output of a stagnation temperature pick-up unit, described in a later section of this chapter. The bleed is contoured to give a relationship between stagnation temperature and bleed area so that at constant velocity there is no dependence of the pressure ratio across the bleed upon stagnation temperature. This pressure ratio then is a function of velocity only and when applied to the evacuated double bellows makes it possible for the latter to function as a velocity-control device.

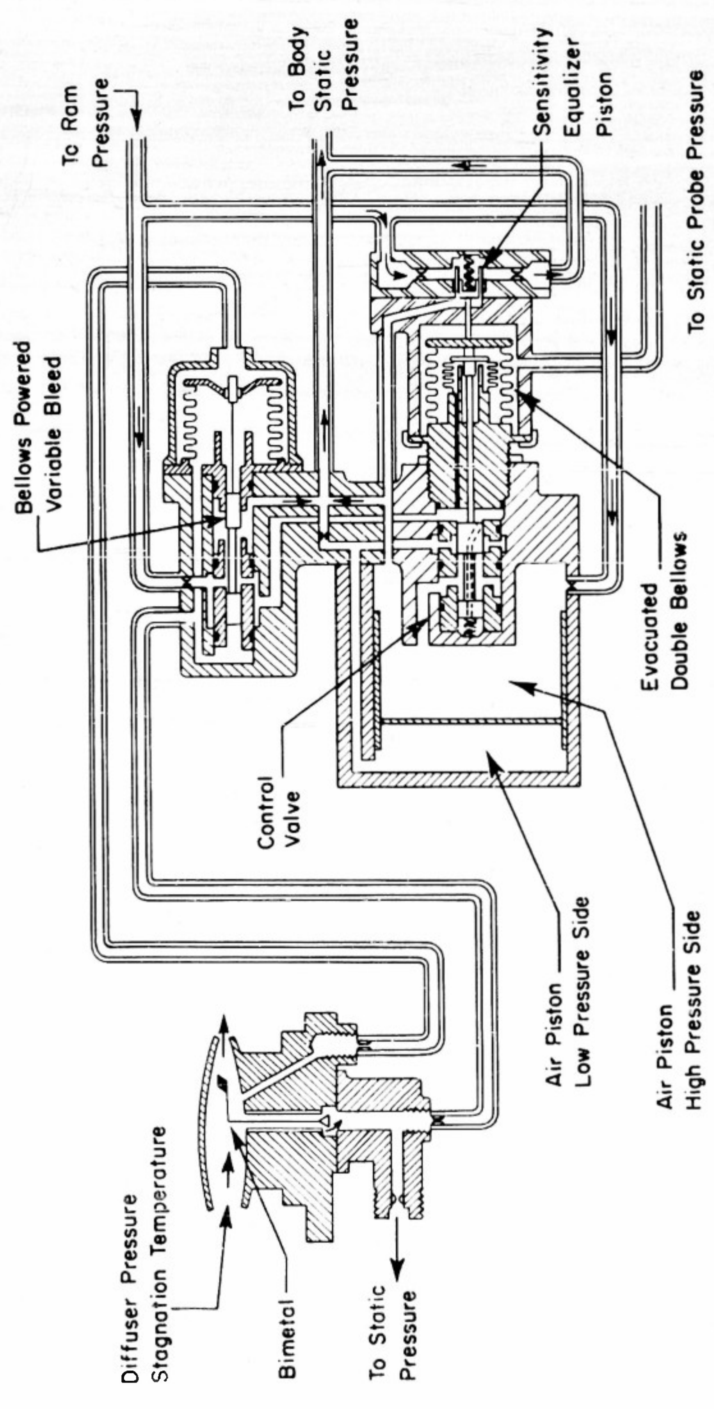

Fig. 6.6-1 SCHEMATIC DIAGRAM OF VELOCITY-CONTROL SYSTEM

The sensitivity of this sensing device is a function of altitude, decreasing by a factor of ten in going from sea level to 60,000 feet. This is due to the fact that the valve position for off-design Mach number is determined by the interaction of the bellows elasticity and the unbalance forces on the bellows, the latter being proportional to atmospheric pressure at a given Mach number.

A device which has been designed to act as a sensitivity equalizer is included in the velocity-control schematic diagram in Fig. 6.6-1. This device consists of a small piston whose resultant force is applied to the double bellows. One side of this small piston is subjected to high-side air-piston pressure in such a manner as to act in the same direction as ram pressure on the double bellows. The pressure on the other side of the small piston is created by bleeding down ram pressure to give a pressure midway between the rich and lean limit values of high-side air-piston pressure. As a result, the travel of the air-piston by-pass valve is prevented from becoming excessive with a given velocity overshoot at low altitude; thus high altitude sensitivity may be increased with respect to what it would be without the sensitivity equalizer, while sea-level sensitivity is held to some given maximum value dictated by control-stability requirements.

The interrelation of the various components discussed above is shown in the schematic diagram of the Talos propulsion system (Fig. 6.6-2).

Since the fuel-flow control system must operate in the force field generated by the maneuvers and longitudinal accelerations of the missile, care must be exercised in the orientation of the critical moving elements of the system. These elements are generally oriented with the direction of motion along the missile axis to avoid the high and variable lateral accelerations. The Mach number bellows assembly is oriented to give motion under positive acceleration equivalent to increase of flight Mach number, thus creating a degenerative feedback in the Mach number control loop.

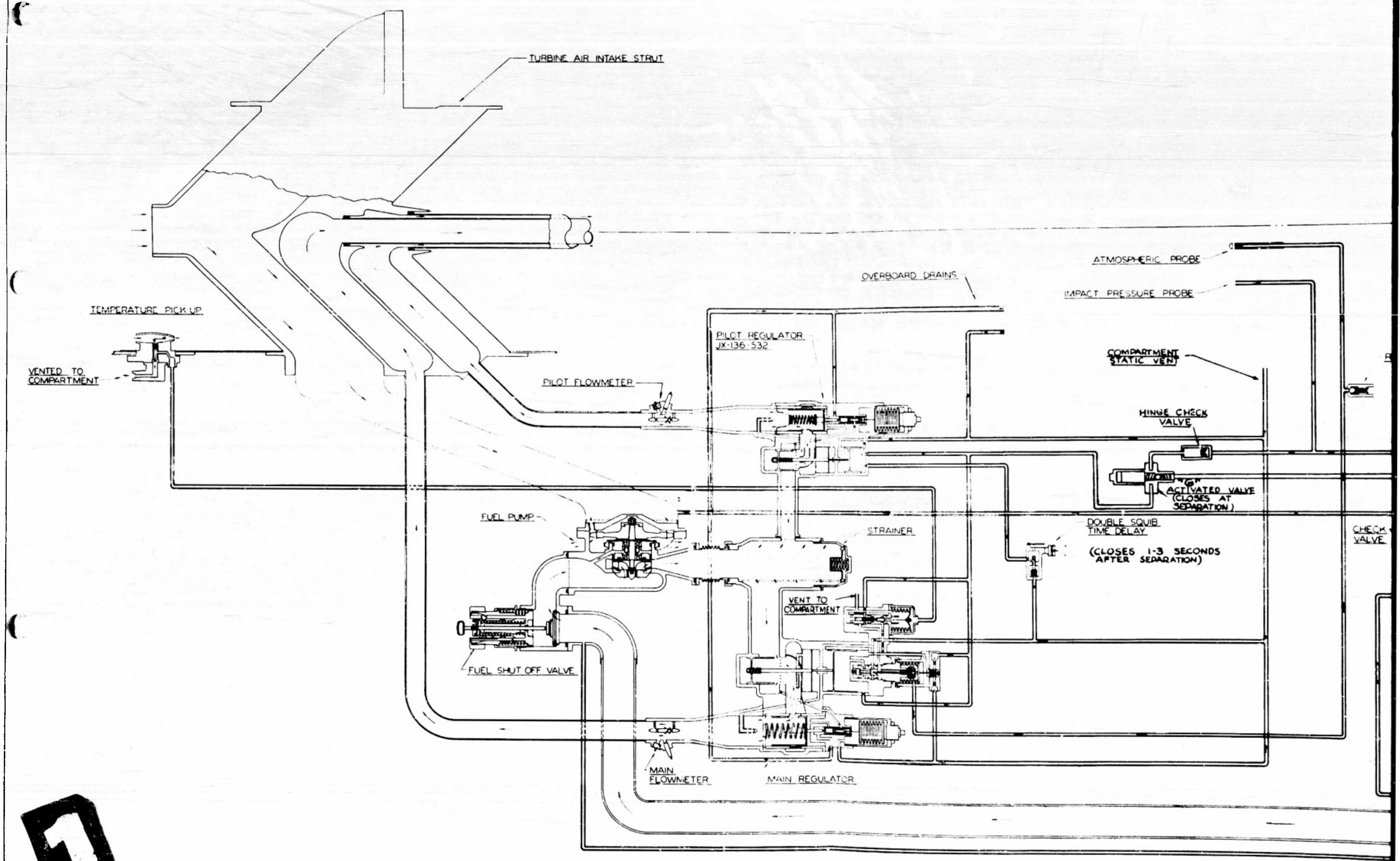

Fig. 6.6-2 SCHEMATIC DIAGRAM OF THE TALOS PROPU

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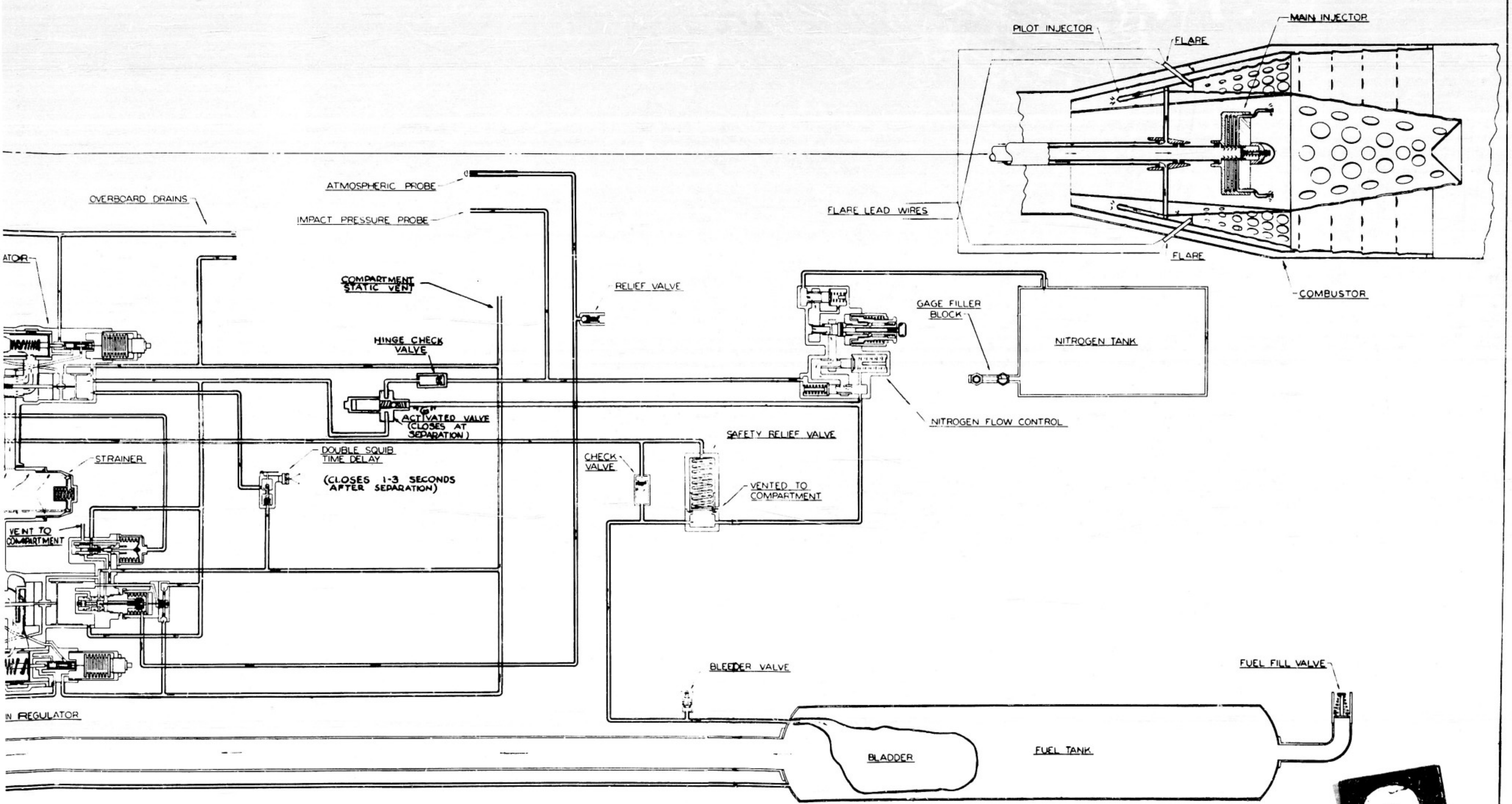

Fig. 6.6-2 SCHEMATIC DIAGRAM OF THE TALOS PROPULSION SYSTEM

6.7 MEASUREMENT OF CONTROL VARIABLES

Fuel flow control needs may be satisfied if the fuel flow is controlled in proportion either to the absolute static pressure immediately ahead of the combustor or to the free stream pitot pressure. It is also necessary to use ambient static pressure as part of the Mach number control system. In addition, velocity control requires a knowledge of the air-stream temperature. It is therefore necessary to have some means of measuring fuel-flow and air-stream pressures and temperature.

The measurement of static pressure is most easily accomplished by using some form of pitot-static tube whereby static pressure is measured by means of taps in the sides of a pitot tube. Stagnation pressure is measured with a pitot tube by means of an orifice at its nose. In the case of a diagonal shock diffuser impact, stagnation pressure may be determined through an opening at the tip of the diffuser innerbody. In any case, if free stream static pressure and impact or stagnation pressure are essential quantities in the fuel-control system, then a very important problem is the effect a variation in the missile angle of attack will have upon such pressure measurements.

Experimental data relating to this effect are discussed in Ref.[11]. For a pitot tube with a hemispherical head, the stagnation pressure remains essentially unchanged up to a four-degree angle of attack; at a nine-degree angle of attack the decrease is still less than two per cent but becomes somewhat greater at higher angles of attack. In addition, the general trend of the decrease appears to be independent of Mach number in the Mach number range from 1.50 to 2.00. It is interesting to note that for a pitot tube with a conical rather than a hemispherical head, the stagnation pressure does not fall off at higher angles and for practical purposes is constant over the angle of attack range from zero to ten degrees. For the case where stagnation pressure is measured by means of the diffuser innerbody opening, a variation in angle of attack in the range from zero to ten degrees produces no measurable effect.

The results are somewhat different for the static pressure measurements. Here, as one might expect, the static pressure variation with angle of attack for side-taps differs considerably from the case where taps are located on the top and bottom of the tube. For this reason, the four static taps are usually manifolded so as to yield an average pressure around the tube, thereby decreasing the sensitivity to angle of attack variation. Experimental data for a hemispherical-headed tube with manifolded static taps indicate less than a two-per cent variation in static pressure at a four-degree angle of attack; however, a pressure fall-off as high as ten to twelve per cent is possible at a ten-degree angle of attack. Practically, however, this problem is not too serious since ordinarily, if the missile exceeds a five-degree angle of attack, it will do so only for relatively short periods, such as when executing a quick maneuver during homing.

An essential component of the velocity-control mechanism is a variable-area bleed powered by a bellows whose movement is proportional to the pressure differential across it. This pressure differential in turn is the output of a temperature pick-up unit and is essentially proportional to stagnation temperature and independent of altitude and diffuser pressure [12]. This temperature-sensing device is shown schematically in Fig. 6.6-1 and is described in Ref.[13].

The temperature element is a bimetallic strip, located in the diffuser air stream and connected by means of a wire to a check valve of area "A". The temperature element exerts a pull F on the wire which is opposed by a force due to the pressure drop $P_H - P_M$ across the valve. By providing a small bleed vented to a low pressure P_L, the check valve is then able to meter the flow and thus maintain $F = A(P_H - P_M)$. A change in the absolute level of P_H, P_M or P_L does not produce any noticeable effect upon the pressure differential $P_H - P_M$ as long as $A(P_H - P_L)$ is greater than F. Operation of this device as a constant head valve for a constant temperature is made possible by keeping the bleed to low pressure small and valve travel negligible. Consequently, when a change in the measured temperature occurs, the force F changes and produces in turn a change in the pressure differential $P_H - P_M$. The method of including this temperature controlled pressure differential in the velocity control scheme has been discussed.

One method of measuring fuel flow is by means of a conventional orifice meter whereby fuel-flow rates are determined from a knowledge of the pressure differential across a restriction in the flow and the area of the restriction. The reliability of this type of meter depends on streamlined flow through the meter. This is a very difficult condition in missile usage because of the sharp turns in the fuel ducts. This is the principal reason why orifice or venturi meters are generally not used in Bumblebee missiles. The fuel-flow measuring function of the Bendix fuel controls is the one exception to this. Here fuel flow is determined by measuring the pressure drop across the contoured altitude valve. Special devices are required at both the upstream and downstream point of measurement to secure a valid pressure. This measurement constitutes the most critical point of design in these fuel controls.

The measurement of fuel flow, as practiced in the Bumblebee program, has been accomplished in a highly successful manner by means of a device referred to as a turbine flowmeter, developed at the Physical Science Laboratory of the New Mexico

College of Agriculture and Mechanic Arts. One distinct advantage of this flowmeter is that the volume rate of flow is a linear function of frequency over a range of 40 to 1. The flowmeter measures volume rates of flow and may be built in different sizes. For example, both the pilot and main flowmeters in the Talos fuel system have the same basic design but differ principally in the size of the bore and thus in the "flow versus frequency" characteristics.

One form of the turbine flowmeter is shown in Fig. 6.7-1 and is discussed in Ref.[14]. Fuel flows through the meter housing causing a vaned rotor to revolve. The rotor axle is mounted on jeweled bearings at its ends and is suspended in the center of the bore through the housing, parallel to the flow direction. The volume rate of flow through the meter determines the rotational frequency of the rotor shaft. This frequency is measured by a beam of light directed perpendicular to the axis so that it strikes a section of the axle which has been milled flat. As the rotor turns, the light beam to a photocell on the opposite side of the axle is interrupted with a frequency equal to the rate of rotation. A variation of this method which is more convenient from an assembly point of view is to use the reflected beam so that the photocell may be located on the same side of the housing as the light source. The instrument must be calibrated before it is used; the calibration curves (volume rate of flow versus frequency) form straight lines passing through the origin. Factors affecting the slope of these calibration curves are the diameter of the bore, the shape of the rotor blades, and the angle between the rotor blades and the axle.

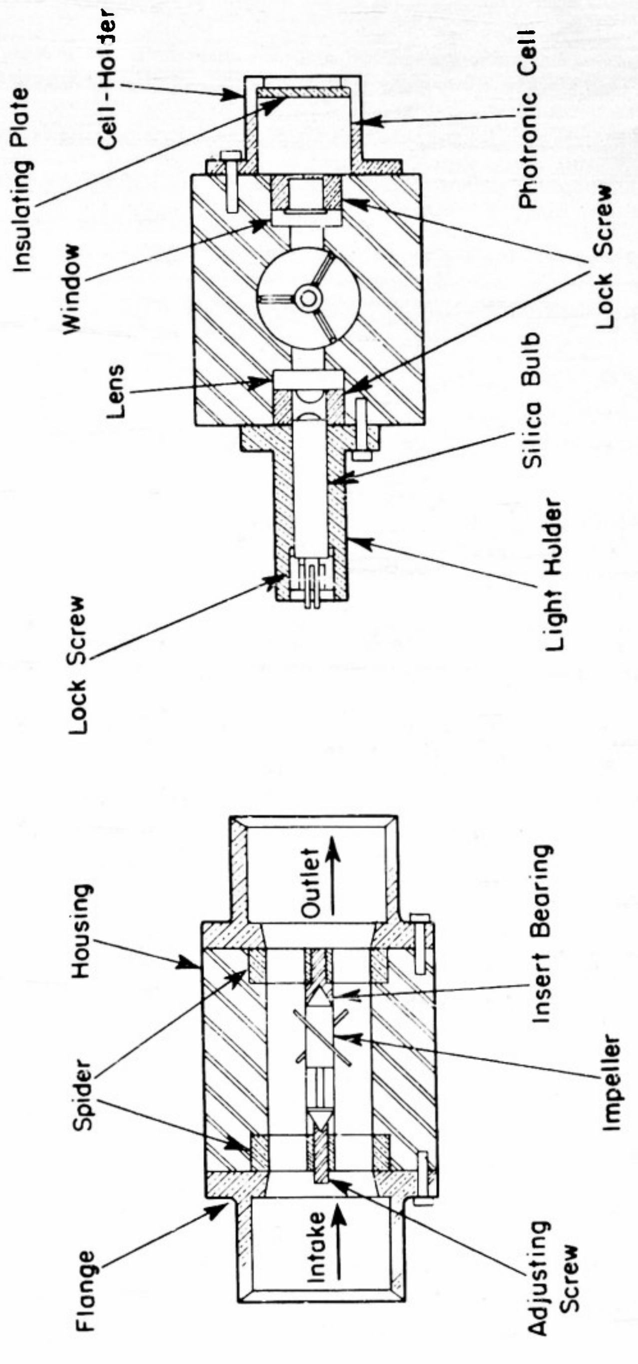

Fig. 6.7-1 TURBINE FLOW METER

NOMENCLATURE

Symbol	Definition	Unit
A	cross-sectional area	ft^2
a	sonic velocity	ft/sec
C_{db}	burner drag coefficient referred to A_2	
K	proportionality constant, Esso fuel control function	ft^2/sec
M	Mach number	
m_a	mass air flow rate	slugs/sec
m_f	mass fuel flow rate	slugs/sec
p	static pressure	lb/ft^2
p_T	ram pressure	lb/ft^2
S_a	air specific impulse	sec
T	temperature	^oR
V	velocity	ft/sec
γ	ratio of specific heats of air	
$\phi(M)$	thrust function	
ρ	air density	$slugs/ft^3$

Subscripts

α	upstream restriction in ram tube	
β	downstream restriction in ram tube	
o	free stream conditions	
2	diffuser exit	

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