However, a coating may perform a useful function by preventing the absorption of an accidently applied material (e.g., oil).

Conclusions and Recommendations

Sheet cork has been demonstrated to be an effective and efficient material for external thermal protection of aerospace vehicles such as Minuteman during launch and ascent both for the main vehicle surfaces and for base areas that are exposed to convective/radiant heating environments. Cork merits consideration for the external insulation of other current and future vehicles. The physical property data presented in this paper are believed to be valid for the design of presently conceived aerospace vehicles under boost-phase conditions. The potential for the use of cork in the high-shear environments experienced on re-entry also appears to be good for certain applications. Of course, the thermal environment for any new design must be carefully determined before one can judge the direct applicability of the data given herein.

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Measurement of Free-Air Properties from Onboard a Large Launch Vehicle

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An analysis is made of techniques to measure the pressure, density, temperature, and wind of the free atmosphere from onboard a large launch vehicle of the Saturn or Nova type. The flow field surrounding an accelerating launch vehicle of this size is so complex that it is impossible to make a measurement adjacent to the vehicle and convert it by shock and expansion theory to the free-air value. Consequently, attention was concentrated on remote measurement techniques wherein the instruments are located aboard the vehicle, but the actual measurement is made at a point outside the shock layer. All techniques used or proposed to be used to make aeronomical measurements with conventional sounding rockets or supersonic aircraft were evaluated. Measuring density by monitoring the change of differential absorption of solar radiation as the launch vehicle ascends is promising. The use of very short wavelength ultraviolet light to induce visible fluorescence is found to be very promising. Use of this more sophisticated technique will avoid the problem of backscatter from natural aerosols and meteoritic dust and will afford a much higher operation ceiling. A pitot-static tube is recommended for pressure, and a fixed cone with orthogonal static-pressure ports is recommended for wind.

Nomenclature

α = angle of attack
ρ = density
g = acceleration of gravity
h = altitude
H = scale height
I = intensity
k = mass absorption coefficient
$\bar{M}$ = mean molecular weight
R = gas constant
T = ambient temperature

Subscripts

0 = at the reference altitude (usually sea level)
1 = lower altitude
2 = upper altitude

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Introduction

A STUDY was recently completed at Electro-Optical Systems, Inc. (EOS) on techniques to measure the temperature, pressure, density, and wind above 30 km from onboard a large launch vehicle.[1] It was not the characteristics of the shock layer that were sought, but rather the characteristics of the free atmosphere, undisturbed by the passage of the launch vehicle. The purpose of the measurement is to obtain a more complete picture of the environment during the flight test of Apollo and Gemini launch vehicles than is possible with measurements made within the shock layer. This may appear at first to be an insignificant problem, since the making of aeronomical measurements from rockets is an everyday occurrence, but the problem is more difficult than is encountered with aeronomical soundings. The flow field surrounding an accelerating launch vehicle of the Saturn or Nova type is so complex that it is very difficult to make a measurement adjacent to the vehicle and convert it by shock and expansion theory to the free air value.

Vehicle and Measurement Location

Figure 1 illustrates the complicated, nonisentropic flow field around a Saturn at Mach 1.93, and probably is typical of

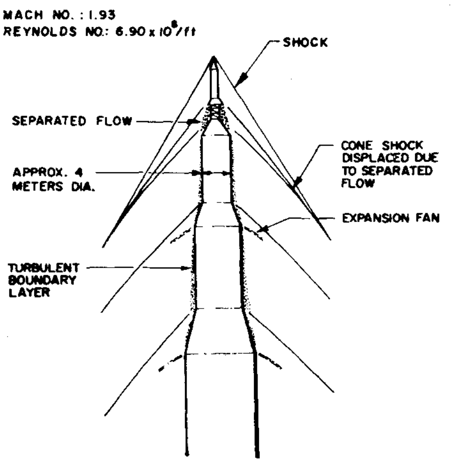

Fig. 1 Shock and boundary layers around the Saturn.

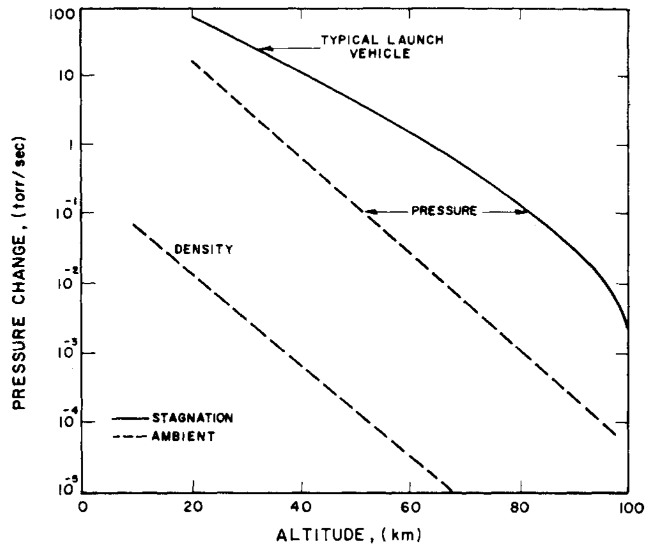

Fig. 2 Rate of change of pressure and density.

the flow field around any large launch vehicle. The sketch was taken from a wind-tunnel photograph. Vehicles of this type reach peak Mach numbers of 5 to 6 at an altitude of around 75 km.

The first aspect of the problem to be considered is whereabouts in the surrounding flow to take the sample of air whose properties are to be measured. With the exception of static pressure and possibly total temperature, the sample must come from outside the boundary layer. Wind-tunnel tests have indicated that for a vehicle like the Saturn the boundary layer is turbulent and has a thickness of about 8 cm at forward stations and as much as 35 cm at aft stations. In addition, such thick boundary layers are prone to separate. Thus, in order to reach outside the boundary layer, an ideal instrument mounted anywhere but on the nose tip must have a range of at least 8 cm.

Making the measurement within the shock layer was considered next. As Fig. 1 indicates, the shock-layer flow at the nose tip is relatively simple and steady. However, the flow aft of the tip is much less certain. Assuming that the flow field pictured is representative, the large number of detached, curved shock waves in the flow aft of the nose stations would make hopeless the task of analytically tracing a streamtube forward to the point where it entered the bow shock.

The conversion of a measurement made in the shock-layer flow into its corresponding value in the free air requires an estimate of the Mach number and the angle of the upstream shock wave(s) through which the streamtube being investigated has passed. Using cone theory, we found that an error in overestimating a shock wave angle by as little as 5% will cause, grossly, a positive error of about 10% in the conversion of a pressure measured within the shock layer to free air pressure. A 10% underestimation of shock wave angle resulted in a negative error of about 17% in free air pressure.

The conclusion from this investigation is that it is vital that measurements be made in as well-defined and simple a flow field as possible. A highly precise measurement of the local flow would be of little value if a 10 to 17% error is incurred in converting it into the corresponding value in the free air. This conclusion led us to concentrate our effort on remote field measurement techniques. Here, the instruments are located aboard the vehicle but the actual measurement is made some distance away, out in the undisturbed atmosphere.

Instrument Performance Requirements

Before discussing specific instruments and measurement techniques, let us examine some of the performance requirements that must be met by them. These requirements are imposed by the environment and include things such as dynamic range, accuracy, response, vibration tolerance, etc. Only those performance requirements that are uniquely related to the flight testing of large liquid-fueled launch vehicles are discussed.[1]

Dynamic Range

For those instruments that measure the ambient or free atmosphere directly, the required dynamic range is simply the variation of the property over the altitude interval of principal interest, i.e., 30 to 100 km. The range for each property is set forth in the column of the table labeled "Ambient" (Table 1).

The second column specifies the dynamic range that would be required of an instrument that measured the stagnation or total value. The range of this latter property is dependent upon the vehicle speed, and the cited ranges are based on a representative trajectory for a large, liquid-fueled launch vehicle.

Instruments that measure the ambient pressure or density must have a dynamic range at least ten times greater than their stagnation counterparts. This is because the increasing vehicle ascent speed offsets in part the natural decrease of the free air pressure and density with altitude. Temperature is an exception since the ambient temperature does not vary a great deal, whereas compressional heating greatly increases the stagnation temperature as the vehicle speed increases.

Frequency and Time Constant

It is customary to specify gage requirements in terms of a frequency, for example, 20 cps. This type of specification would be artificial in the present situation because most of the quantities being measured change relatively slowly and uniformly with time. An exception is wind, the frequency response of which is intimately related to the dynamics of the particular vehicle. Because of the uniqueness of the wind

Table 1 Instrument dynamic range

Property	Ambient	Stagnation	Ref.
Pressure, torr	10–10^{-4}	80–10^{-1}	2
Temperature, °K	200–300	830–1500	2
Density, kg/m^3	10^{-2}–10^{-7}	10^{-1}–10^{-5}	2
Wind speed, m/sec	50–300	...	3
Wind shear, m/sec/m	0.03–0.113	...	3

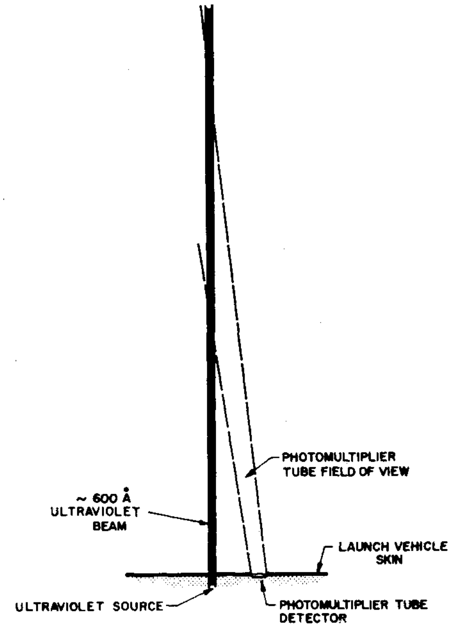

Fig. 3 Ultraviolet density gage geometry.

frequency requirement, no general specification will be made for wind.

This low rate of change is illustrated in Fig. 2. The figure shows the rates of change of the ambient and the stagnation pressure and density which would be seen by a rocket ascending vertically. The curve shown for stagnation pressure is typical of an actual large launch vehicle. It is seen that the maximum rate of change of stagnation pressure is only slightly greater than that of ambient pressure. However, a stagnation temperature instrument aboard a typical launch vehicle would have to follow a rate of change that is about seven times greater than that for a free-air temperature instrument. Since the rates of change are relatively low for all cases, we believe that the concept of time constant is the more useful parameter for a discussion of the dynamic performance of the pressure, density, and temperature instruments.

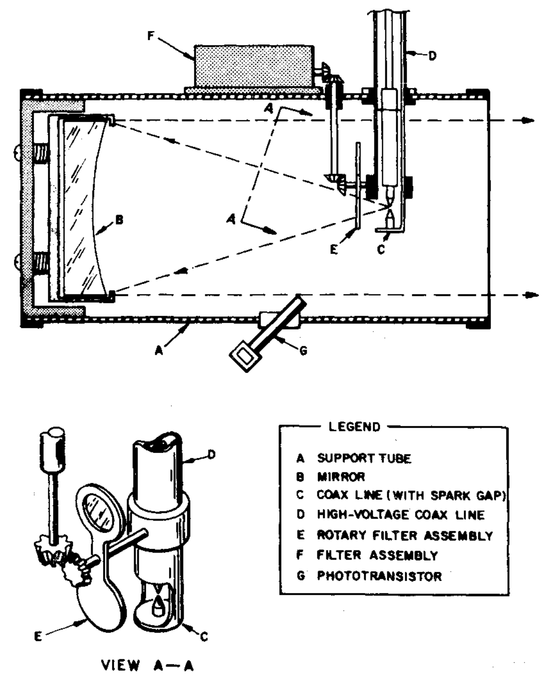

Fig. 4 Source configuration.

The time-constant requirements for pressure, temperature, and density were established by studying the error that would be caused by a given instrument lag. To establish an upper limit on time constant, a value of 1 sec was assumed and the reading error calculated. The error incurred with this large time constant was not excessive. Thus, it appears that a time constant of $\frac{1}{4}$ to $\frac{1}{2}$ sec would be entirely adequate.

Vibration and Shock

All instrumentation that is carried by a launch vehicle typically must be capable of operation in a vibration environment of 20 to 2000 cps and 7.5 to 12.0 g-rms. A typical shock specification is 40 g-rms for a duration of 6 to 11 msec. Since this frequency is well above that of any of the atmospheric parameters, it should be possible to distinguish the slowly varying atmospheric parameter signal from the rapidly varying vibration-induced noise.

Sustained Acceleration

Instrumentation that is carried aboard a large, liquid launch vehicle will be subjected to a transverse acceleration that will increase from about 0.1 g at 30 km, to about 4 g at first-stage burnout. Depending upon the particular vehicle, the first stage will burn out between 60 and 80 km. Lateral acceleration due to the turning of the vehicle along the flight path will impose negligible acceleration on the instruments. A typical test specification is 20 g for at least 1 min.

Instrumentation that is carried aboard a solid-propellant sounding rocket must be capable of withstanding an acceleration of about 26 g. It has been reported that an acceleration of around 60 g is imposed on instrumentation during the jettisoning of a dropsonde. Therefore, the acceleration environment will impose no unusually severe requirements on the instrumentation.

Temperature

It is a requirement that all instrumentation be capable of operation at an elevated temperature and not subject to significant temperature error due to a zero shift, sensitivity shift, etc. At this time it is not possible to specify the heat input to the instruments because it is dependent upon the configuration and location aboard the launch vehicle. An instrument located at a stagnation point probably would reach a peak temperature of about 85% of the peak air temperature, or around 1300°K. The minimum temperature would be influenced by the proximity of the cryogenic fuel tanks. A typical test condition is a temperature range of ±85°C.

Density Measurement

We have concluded that the most desirable measurement technique is remote measurement. Here the instruments are located aboard the launch vehicle, but the actual measurement is made remotely in the undisturbed air outside the shock layer. This method avoids placing the instruments in the nose tip where they are lost when the escape rocket is jettisoned.

Two remote-measuring density devices, which show good prospects, are the ultraviolet air density gage and the solar absorption air density gage. An effective, remote-measuring, density instrument would be particularly valuable, because if the air density can be measured to great heights, the pressure and temperature can be calculated from the density profile and the air composition. There will be an increasing need for density and similar data at high altitudes as the manned space flight programs progress.

Ultraviolet Air Density Gage

A technique that utilizes the rearrangement radiation from molecular nitrogen shows great promise for the remote measurement of density. A column of air is excited to visible fluorescence by shining a beam of ultraviolet light through it as shown in Fig. 3. The intensity of the fluorescence, as a function of the amount of energy fed into the ultraviolet source, is a measure of air density. The device detects the fluorescence of molecular nitrogen when bombarded by 500 to 700 Å photons, and it is capable of measuring atmosphere densities at a distance up to 3 m from the vehicle.

The instrument would consist of an ultraviolet source and a photomultiplier tube detector with focused optics. Figure 4 illustrates an ultraviolet source that utilizes a carbon arc. Recent work has indicated that a linear dynamic pinch-type source might be superior to the carbon arc, and its use is being studied. A sequence of filters and source modulators is utilized to make certain that only gas molecules can contribute to the signal and not reflections from dust particles or ice crystals. This latter characteristic is important because other ultraviolet densitometers rely on absorption and reradiation. Both the bombarding light and the returned signal are of the same wavelength. Thus, the presence of dust, ice, or particles from entrained exhaust could cause a large error due to reflection from their surface. Accurate measurements in the region of nacreous and noctilucent clouds might be very difficult.

We believe that the rearrangement radiation-type ultraviolet gage will provide accurate density measurements at altitudes where the conventional absorption-reradiation type cannot because of lack of returned signal. The EOS technique utilizes a frequency where the most probable cross section is from 100 to 1000 times greater than at other practical frequencies. Thus, the return signal is orders of magnitude greater and the operational ceiling correspondingly greater. Moreover, the return signal is in the visible- and very-near-ultraviolet portion of the spectrum where a photomultiplier (P-M) tube detector is the most sensitive. In general, a P-M tube is ten times more efficient with visible light than with ultraviolet.

Filterphotometer Air Density Gage

Another very promising density technique is to measure the variation of the solar energy in selected ultraviolet wavelength bands as the rocket ascends, and thus determine the absorbing air mass between the rocket and the sun. Data taken in this way are little affected by the pressure field distortion caused by shock waves or by local contaminants. The absorbing air mass may be converted into air density with the knowledge of the relative atmospheric composition and ultraviolet absorbance.

Theory

Atmospheric density can be determined by measurements of optical absorption of the sun's radiation in various spectral regions in accordance with the equation:

$$\ln(I_2/I_1) = k\,H\,\rho_2 \exp\{[(h_2 - h_1)/H] - 1\} \qquad (1)$$

The absorption coefficient depends on the molecular species responsible for the absorption, which in turn determines the choice of wavelength. From about 40- to 90-km alt the absorption is almost exclusively due to molecular oxygen. Since the composition of the atmosphere appears to be constant up to about 90 km, a determination of the amount of absorption by molecular oxygen will yield an accurate measure of the density at the instrument. In order to span such a wide range of altitude, it probably will be necessary to monitor the absorption in at least three and possibly four different wavelengths, beginning with 2000 Å at 40 km and diminishing to 1800 Å at 90 km. Above 90 km, oxygen is dissociated so that measurements based on molecular oxygen are no longer useful. At high altitudes, one can use the absorption of soft x rays which is proportional to total air mass above the instrument and is independent of molecular aggregation.

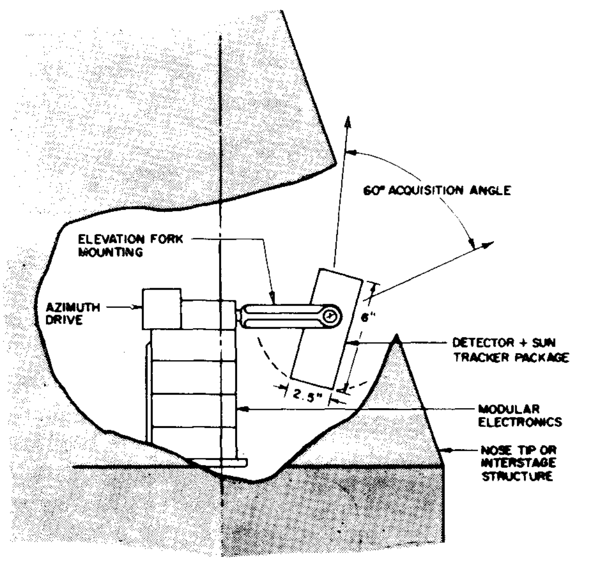

Fig. 5 Instrument and solar tracking mounting.

The instrument should be located in the nose tip or the first interstage of the launch vehicle in order to have the best view of the sun at all times. The instrument and solar tracking mounting is depicted in Fig. 5. It consists of a fork-mounted detector and solar tracking sensor that is free to move through 60° of elevation and 90° of azimuth. If the viewing port interferes with the aerodynamic performance of the missile, a lithium fluoride window may be used; however, the x-ray detector must be mounted such that its protective window can be removed when the detector begins operation.

A suggested configuration for the detector and sun sensor assembly is shown in Fig. 6. The x-ray detector is mounted in front of the chopper motor and has an unobstructed view of the sun. The chopper wheel contains filters in those openings which pass in front of the photomultiplier. A clear reference notch for data identification is also located in the chopper disk. As the wheel rotates, each filter in turn is placed in front of the photomultiplier, as well as a reference signal that may be identified by its shorter duration. The Lyman-α detector is also chopped at the same rate as the photomultiplier.

Backscatter Density Gage

Another method of avoiding the disturbed flow field near the launch vehicle is to use the backscatter of nuclear radiations to measure density. The radiation coming from a nuclear source interacts with the air and some of the radiation is scattered back to a detector. The amount of interaction or scattering is primarily a function of the density of the air, and thus the amount of radiation detected is a good measure of the density of the air.

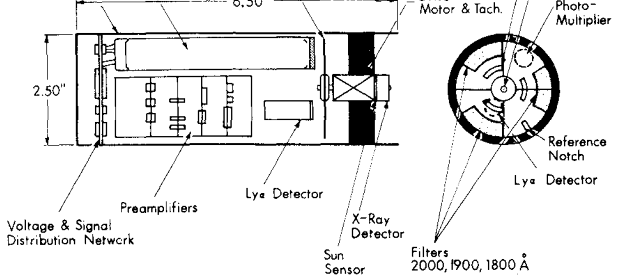

Fig. 6 Detector and sun-sensor assembly.

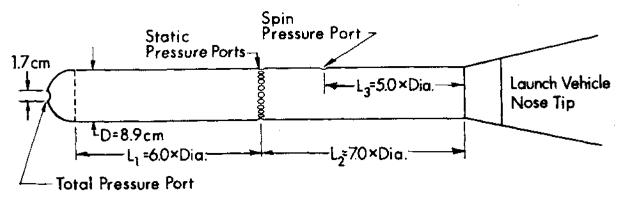

Fig. 7 Configuration of a pitot-static tube.

Density has been measured in the past with low-energy beta particles up to an altitude of about 18 km, where the count rate was about 300/sec. The source used was 4 mC Pm^{147}. In order to obtain the same count rate at 70-km alt, the radiation source would have to be 10^5 times as intense or a 400-C source. A source of this strength would be very difficult to implement aboard a manned launch vehicle. This requirement of a strong radioactive source proved to be a chronic disadvantage of this type of backscatter technique and led to its eventual elimination from consideration. We then sought out techniques that had a larger effective backscatter cross section.

Electron Beam Density Gage

A higher effective backscatter cross section could be obtained by using an electron beam as the illuminator to cause a column of air to fluoresce. The geometry and operation of the device is similar to that of the ultraviolet density gage portrayed in Fig. 3. The source of the electron beam is an electron gun, and the detector is a photomultiplier tube that senses the fluorescence of the air.

Although initially the electron beam densitometer appeared very promising, our ultimate conclusion was that the technique is not practicable below an altitude of 75 km. At the lower altitudes, the divergence of the beam is so great that the collimation is destroyed. Moreover, below 75 km, the electron gun must be operated in a vacuum that requires bulky differential pumping machinery. Its ceiling is about 130 km.

Pressure

There appears to be no practicable device or technique that can measure the pressure of the atmosphere at a remote point. The best method to determine the pressure of the free atmosphere directly appears to be to measure it with a pitot-static tube mounted on the nose tip. It can be determined indirectly by calculation using a measured density profile and the atmospheric composition.

The pitot-static instruments have been used in several forms to secure data in the mesosphere and thermosphere. The configuration can be a cone such as that discussed by LaGow et al.[4] or a cylinder such as discussed by Ainsworth et al.[5] There is no need to go into detail on the principle of operation of a pitot-tube here. For details, the reader is referred to the foregoing references. The cylindrical configuration similar to the one depicted in Fig. 7 was used by Ainsworth et al., and this type is recommended here.

The Range Development Support Directorate at the Pacific Missile Range currently is sponsoring the development of a pitot-static tube for use aboard the boosted Arcas up to about 110 km. J. Horvath and others of the University of Michigan have successfully flown pitot tubes aboard Nike-Yardbird rockets at Wallops Island and have made at least one flight aboard a Saturn. Hopefully, these activities will settle the controversy as to whether a pitot tube having the conventional nose-tip shape shown in Fig. 7 can be used in the slip and free molecule flow regions. These development programs should be watched with interest, because this device would be directly applicable to the present case.

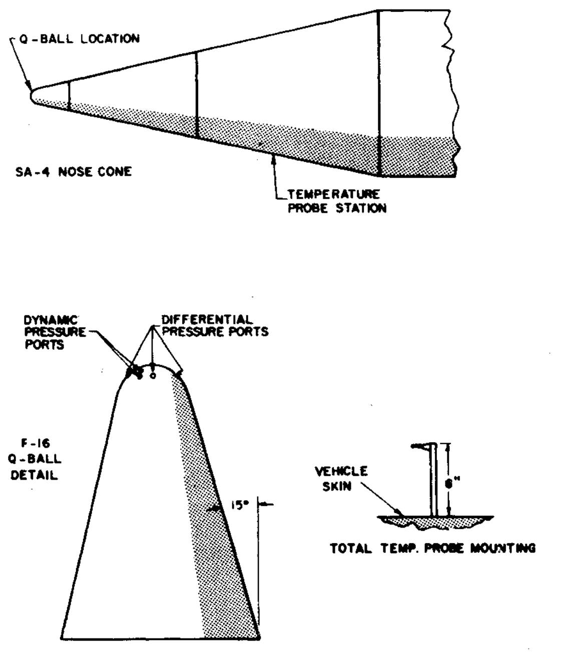

Fig. 8 Saturn nose cone.

Temperature

Total Temperature Tube

A number of measurement techniques have been suggested for use aboard sounding rockets, but temperature rarely has been measured directly except with a dropsonde. Of those few techniques which have been used, the total temperature tube is deemed the best choice for direct measurement. It must be located near the nose tip and can be used with confidence only in continuum flow. However, all temperature techniques, except possibly calculation from the density profile, have these same shortcomings. The total temperature tube at least has the advantage of being a commonly used instrument. It has been reported that two having the configuration shown in Fig. 8 were successfully flown on Saturn SA-4.[6]

The problem with total temperature tubes is that in rarefied air it is difficult to achieve a constant value of the recovery factor. Not only are the air molecules brought to rest and their energy absorbed in heating the sensor, but additional high-energy, high-speed molecules strike the sensor. If a sufficient number do this, the recovery factor becomes uncertain and has been known to rise as high as 1.2. In spite of these difficulties, a well-designed total temperature tube is the most promising instrument for the onboard measurement of temperature.

Acoustic Meter

It is possible to utilize a measurement of the speed of sound waves to determine the ambient air temperature. This is due to the well-known proportionality of the speed of sound to the square root of the absolute temperature of a gaseous medium when the ratio of specific heats and the molecular weight are constant.

An aircraft device is described which involves the use of two booms placed several inches apart and projecting ahead of the vehicle.[7] These booms are in the form of slender half-wedges with the inside surface aligned parallel to the oncoming stream in order to achieve a very weak shock and a minimum of rotational flow between the transmitter and receiver booms. As is illustrated in Fig. 9, the longer boom would carry the emitter and the shorter boom the receiver. This device is a possible alternative to the total temperature tube, but it would require considerable development.

Techniques utilizing closely spaced resonators (such as the Sonotherm[8]) are not feasible aboard a hypersonic vehicle because at high altitudes the high power required heats the air in the confined space, and the boundary-layer thickness is on the order of the plate spacing.

Resistance Thermometer

The Russian's Central Aeronautical Division has had repeated success up to 75 km, using tungsten wire resistance elements mounted on a MR-1 rocket.[9] The temperature of the free atmosphere is calculated from the temperature of the resistance wire, which is placed longitudinally in the flow-around current of air. Error with this method increases with altitude. The mean-square error at an altitude of 40 km is about 5°, at 50 km it amounts to 10°, and at 75 km it is 20°.

Further studies are in order on the use of resistance wire elements. They are versatile in that they can be mounted or wound into many configurations. The dissipation constant of these elements could be high, time-constant low, and excellent radiation characteristics also could be achieved by the proper selection of wire and coatings. The reproducibility from one element to another could be better, both from an electrical and physical viewpoint, than thermistors, probably making the use of individual calibration curves unnecessary.

Vortex Thermometer

The use of the Hilsch effect resulted in the development of the axial flow vortex thermometer, which is currently employed on meteorological aircraft. However, the recovery factor of the device begins to increase from zero at relatively low altitudes, e.g., 15 km, and it currently has a time constant of 10 sec. Its Mach number limitation does not permit its use on rockets during ascent.

Calculation

The one method of determining free-air temperature which appears to work at all altitudes is to calculate it from some parameter that is amenable to direct measurement, such as density. If the temperature, gravitational field, and the mean molecular weight of the atmosphere are assumed to be constant over the small height intervals from h_1 to h_2, then temperature can be calculated from the following well-known equation:

$$T_2 = \frac{\rho_1 \bar{M}_2}{\rho_2 \bar{M}_1}\left[T_1 - \frac{\bar{M}_1}{\rho_1 R}\int_{h_1}^{h_2} \rho\, g\, dh\right] \tag{2}$$

Pressure then can be calculated from the temperature profile by means of the perfect gas law up to about 90 km and from the molecular scale temperature above this altitude.

An initial temperature must be available for the lowest level in order to start the integration. This could come from a recent balloon sounding or a 10 mb upper air chart. The value of T_2 would be sensitive to errors in the initial values of temperature, density, and density profile. As the step-by-step integration proceeds upward, it becomes relatively insensitive to error in the local density profile.[5] The altitude of the vehicle should be available from the radar track. The acceleration of gravity and the molecular weight of air as a function of altitude are readily available.[2]

Natural Radiation

The only method for measuring the free-air temperature remotely which appeared to warrant investigation was the infrared radiation intensity method. This technique is under study at Barnes Engineering Company.[10] In certain bands where a gas is strongly absorbent, the emission of the gas will be equal to that of a blackbody at the same temperature as the gas. Since the blackbody radiation can be directly related to a specific temperature, the gas temperature can be ascertained by measuring the thermal emission in an absorption band.

A major problem with this technique will be temperature gradients in the boundary layer and shock waves. A problem that is unique to the present application is contamination of the atmosphere by entrainment of the rocket engine exhaust products. Carbon dioxide from the exhaust probably would not be in equilibrium with the free atmosphere.

Wind

The only feasible location for a wind instrument appears to be the escape rocket tip, or the vehicle nose tip if no escape rocket is used. Since wind-induced angle of attack is a very critical flight parameter, considerable time and effort were given to investigating wind measurement techniques. Our research led us to the conclusion that the two best ways of measuring the wind were either with a fixed conical differential pressure tube or a modified spin-pressure modulation technique.

Fixed Cone

Actually, the differential pressure tube, in the form of the F-16 Q-Ball shown in Fig. 8, presently is being used to measure angle of attack by NASA. NASA-Marshall has collected a large quantity of wind-tunnel calibration data on the Q-Ball and the most expedient course would be to use it for both high-altitude angle-of-attack and wind measurements.

Recent experimental work done by F. Swalley of NASA-Langley indicates that a conical probe is capable of measuring flow angularity at high Reynolds numbers and speeds where the shock is attached to within $\frac{1}{3}$°.[11] The tests were conducted in a 3-in.-diam helium blowdown tunnel at Langley Research Center.

The F-16 Q-Ball is, in fact, a spherical segment and does not meet the pointed cone and attached shock requirements of the foregoing theory. However, through utilizing aerodynamic calibration factors determined from extensive wind-tunnel tests, an empirical relation between differential pressure and the angle of pitch and yaw has been established.[6, 12] Knowing these angles and the vector velocity of the launch vehicle, it is possible to calculate the speed and direction of the wind which is causing the pitch and yaw. In actual practice, the computation will be rather complex since the vehicle constantly is pitching and yawing in response to the wind and steering orders. However, it should be possible to devise a computer program that can take each of these factors into account while it is processing the differential dynamic pressure data.

Spin Pressure Modulation

The only successful attempt at directly measuring winds in the mesosphere and above which we have encountered is the

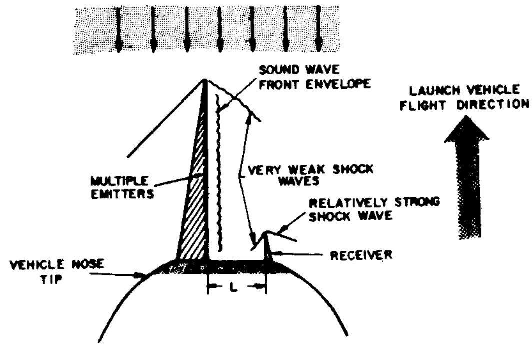

Fig. 9 Acoustic meter mounting configuration.

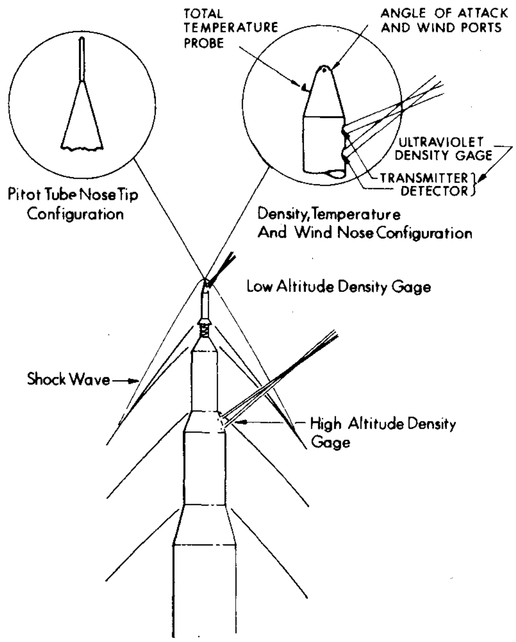

Fig. 10 Onboard measurement techniques.

flight of Ainsworth, Fox, and LaGow,[5] at Fort Churchill during the IGY. They used an Aerobee equipped with a pitot-static tube similar to that depicted in Fig. 7, and utilized the cyclical pressure measured at the static port in a method called spin pressure modulation.

In the present case, it is necessary to make some changes in the Ainsworth technique, which requires a large angle of attack. We suggest two cylinders mounted orthogonally, each of which would rotate about its own axis. Each of these rotating cylinders would contain a static pressure port that would be monitored. The reading from the pressure port when no wind is blowing is due to the impact of the air caused by the upward motion of the vehicle. If a wind were blowing, the resultant vector formed by the booster velocity and the wind velocity would cause the pressure peak to shift to one side. A measurement of this shift and the vehicle velocity could be used to determine the direction and magnitude of the wind.

Conclusions and Recommendations

As a result of our investigation, we have concluded that the most promising technique is remote measurement. However, the only property that is readily amenable to onboard remote measurement is density. Consequently, we have formulated recommendations for immersion-type measurements for the other properties. If forward locations for the immersion-type instruments cannot be made available, then free-flight or dropsonde techniques must be adopted. Therefore, we have formulated recommendations for dropsonde techniques also.

Onboard Instrument Recommendations

Based on our study of large liquid-fueled launch vehicle requirements, we make the following recommendations for flight test instrumentation (see Fig. 10).

Density: either an ultraviolet-induced fluorescent or a differential solar absorption technique. In each case, the instrumentation is located aboard the launch vehicle and the actual measurement is made in the undisturbed air outside the shock layer.

Pressure: a pitot-static tube mounted on the launch vehicle tip. A cylindrical pitot tube is preferred because it is less sensitive to error due to angle of attack. However, a stagnation pressure port in the tip of a conical nose could be utilized if it is well calibrated for angle-of-attack effects.

Temperature: a total temperature probe mounted on the launch vehicle nose. The total temperature is converted to the ambient value by computation using the measured pressure data and shock-expansion theory.

Wind: monitor the differential dynamic pressure across a set of pitch and yaw ports located in a conical nose tip such as the F-16 Q-Ball.

Dropsonde Instrument Recommendations

Dropsondes would enable one to completely escape the perturbed environment of the launch vehicle and to achieve a near-perfect isentropic flow field. Suitable instruments are available from current sounding rocket programs. Their characteristics are well known and no detailed description is warranted here. A possible exception is the risesonde, which is the standard University of Michigan active falling cylinder.[13] In the present application, since the launch vehicle is still accelerating, it is ejected so that it rises along the trajectory for a while before it falls.

The disadvantage of free-flight techniques is that the test range is cluttered with data acquisition units since each instrument must be monitored separately. Also, the soundings are made downrange of the actual trajectory and only the upper portion of the sounding is made in the vicinity of the vehicle.

Our recommendations are portrayed in Fig. 11. We visualize that these devices would be carried in an interstage compartment and jettisoned at appropriate intervals. A single parachute-stabilized device could be dropped at 60 to 70 km for a single sounding somewhat remote from the actual launch vehicle flight path, or a number could be dropped in sequence to give data closer to the actual flight path of the vehicle. A similar procedure could be followed for the free-fall devices above 70 km.

Density below 100 km can be measured by the falling sphere technique. Above 65 km, an active-type sphere is used, and below 65 km, a passive-type is used. In the free-molecule flow domain above 85 km, a risesonde is recommended.

Pressure can be measured at parachute altitudes by a descent-type hypsometer. Above 85 km, it can be measured by the risesonde. Temperature below 70 km can be measured by a thermistor carried aboard a dropsonde; above that altitude, it must be calculated from a density profile. Wind can

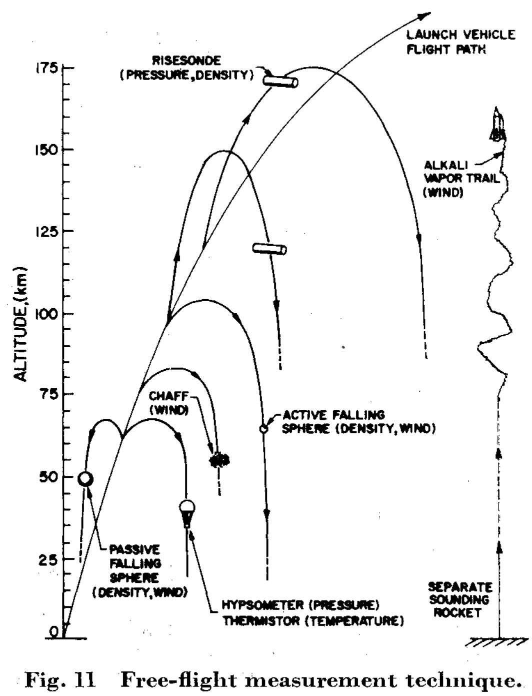

Fig. 11 Free-flight measurement technique.

be measured by tracking any of the freely falling devices. An alternative would be to launch a sounding rocket simultaneously with the launch vehicle. Since it will rise much faster than the launch vehicle, an alkali vapor ejected from it would be a good measure of the wind actually existing at the time of the launch vehicle ascent.

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Feasibility of Large Launch Vehicle Recovery

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Recovery of stages of large two-stage launch vehicles is examined for two design approaches: 1) the basic expandable stage is adapted for recovery by adding recovery equipment and strengthening the basic vehicle for repeated launches and recoveries, or 2) the basic vehicle is designed for recovery in an optimum manner by shaping the vehicle to achieve aerodynamic stability for the re-entry and terminal descent phase. Aerodynamic loads are calculated to be ≤ 15 g's for either stage. Thermal protection for the first stage is required for small thin-walled structures only; large structures absorb the aerodynamic heating without exceeding allowable temperatures. All exposed second-stage structure requires thermal protection. Terminal deceleration is best accomplished by a combination of parachutes and rocket decelerators. Water alighting loads are estimated to be 10 g's if the rocket decelerators bring the recovered stage to a hovering condition over the water. An economic estimate indicates that use of first-stage recovery in a program consisting of 100 launches could reduce direct operating cost by 33% and over-all program cost by 20%.

Nomenclature

a = acceleration, ft/sec^2
A, B = const (in equations for rocket motor weights)
C_D = vehicle drag coefficient
F = rocket decelerator thrust, lb
g = earth's gravitational acceleration
h_s = enthalpy of air stream at stagnation point, Btu/lb
h_w = enthalpy of air at wall surface, Btu/lb
h_{300} = enthalpy of air at 300°C, Btu/lb
I_s = specific impulse, sec
N = deceleration, $N \equiv (F/W) - 1$, g
K = const (in equations for water alighting loads)
p = pressure, psf
$\dot{q}_c$ = stagnation-point convective heat-transfer rate, Btu/ft^2-sec
R_n = nose radius of curvature, ft
t = time from retrorocket ignition, sec
V = instantaneous velocity, fps
V_0 = initial velocity, fps
W = vehicle weight, lb
W_r = retrorocket weight, lb

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