VOLUME II

TECHNICAL REPORT

SYSTEM FOR UPPER ATMOSPHERIC SOUNDING (SUAS)

MARCH 1969

PREPARED UNDER CONTRACT NO. NAS1-7911 BY

BOOZ·ALLEN APPLIED RESEARCH Inc.

FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

LANGLEY RESEARCH CENTER

~~FOR U.S. GOVERNMENT AGENCIES ONLY~~

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SYSTEM FOR UPPER ATMOSPHERIC SOUNDING (SUAS)

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FOREWORD

This study was conducted for the Langley Research Center of the National Aeronautics and Space Administration by Booz, Allen Applied Research Inc., Bethesda, Maryland under Contract NAS1-7911. Mr. T.P. Wright, Jr., of the Flight Vehicles and Systems Division, was the LRC Technical Representative of the Contracting Officer. The study was initiated on February 1, 1968 and completed on 31 January 1969.

The study was under the cognizance of Mr. C.F. Riley, Jr., Vice President, of Booz, Allen Applied Research Inc. Mr. W.E. Flowers, Research Director, was the Program Manager. Principal BAARINC staff contributors were Messrs. William E. Brockman, J. Frank Coneybear, Harry L. Crumpacker II, John L. Hain, and David W. Weiss. During the course of the study, Mr. Frederick F. Fischbach, of the High Altitude Research Laboratory, University of Michigan, Mr. G. Harry Stine, a private consultant, and Dr. N. Engler, University of Dayton Research Institute, were engaged as consultants.

Reports produced as a result of this study are:

Vol. I - Summary Report
Vol. II - Technical Report
Vol. III - Conceptual Design
Vol. IV - Technology Development Plan
Vol. V - Program Development Plan

Volume I is an overview of the project listing results and conclusions.

Volume II is the complete report on the project containing all of the technical analysis.

Volume III is the conceptual design which details the recommended sounding system.

Volume IV is the Technology Development which is an orderly description of the remaining technical problems that need to be resolved prior to system procurement.

Volume V is the Program Development Plan which is an overall plan for the implementation of the system for Upper Atmospheric Sounding.

I. MISSION REQUIREMENTS AND SPECIFICATIONS

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The overall mission requirements determine the characteristics of the sounding system. The envelope of possible launch configurations is set by the mission specifications for the launch system, some of which are derivable directly from overall mission requirements, some of which arise from the evolving characteristics of other subsystems, in particular, the payload.

Requirements and specifications are presented in this chapter, first in the form of a general description, and then by discussions and listings of specifications and constraints.

1. GENERAL DESCRIPTION

The synoptic sounding system, as conceived, will operate on a worldwide basis from about 1974 to 1984. The objectives of the principal payload are to measure temperature, density (or) pressure, and wind vector velocity. It is expected that the net payload weight will be 5 to 10 pounds. Occasionally, ozonesondes may be launched by the vehicle used for the prime measurements although remote sensing may prove more effective.

An analysis to determine the number of launch sites, their locations and rate of launches from each site was not included in the scope in this study. However, in order to provide a baseline for sizing and costing this system, somewhat artificial quantities were agreed upon after discussion with potential users and with the LRC. These are:

- 100 sites, collocated with the approximate number of rawinsonde sites existing in the Northern Hemisphere
- 100 launches per site per year.
- 10-year operating lifetime.

On a worldwide basis, many of the sites will be located in foreign countries and manned by foreign nationals. Some locations will be remote and personnel available may be trained to a low performance level relative to U. S. range personnel. At the same time, launch locations near populous areas may also be desired. The system will be civilian controlled and operated, including the launch vehicle logistics and data handling.

2. PERFORMANCE AND OPERATIONAL SPECIFICATIONS

Based on the above general description, a set of mission specifications has been prepared. It is subdivided into two areas — performance and operational — as shown in Tables I-1 and I-2. The

Table I-1
Performance Specifications for Launch Systems

Item	Specification
Maximum Altitude	The nominal, maximum altitude shall be 130 km.
Net Payload Weight	The range of net payload weight to be considered will be from 5 to 10 lbs. with a nominal value of 7 lbs. established.
Trajectory .	The velocity and position trajectory of the vehicle will be essentially the same for each launch
In-Flight Stability	The vehicle must be stable, either aerodynamically, or spin stabilized, during the portion of flight when the payload is attached to the vehicle
Altitude Dispersion	No specification currently established
Impact Dispersion	No specification currently established
Off-Design Operation	No specifications have been established for the effect of payload or maximum altitude requirement changes during operational use of the system.

Table I-2
Operational Specifications for Launch Systems*

Item	Specification
Time Frame	
Operational Date	The system shall become operational during the period 1971 to 1977. 1974 has been established as a nominal date
Length of Service	The system will be operational for approximately ten years
Site Considerations	
Number of Sites	Approximately 100 launch sites will be established
Worldwide Site Location	Sites will be located, on land, including foreign territory. The sites may be remote and/or easily accessible.
Site Mobility	Site locations will be fixed for the length of service of the system
Other, On-site Launch Operations	Rawinsonde sites will be collocated with launch sites
Site Area Requirement	The land area required for launch should be a minimum, subject to the other mission specifications (particularly the mass hazard specifications)

*These specifications were established to permit system design and sizing. LRC will not be involved in system operations, only in the development of the system.

Table I-2 (Continued)

Item	Specification
Measurements	
Parameters	The payload sensors will measure temperature, pressure (or density), and wind vector velocity. Ozone measurements will be made as a separate payload on the same launch vehicle
Altitudes for Measurements	Measurements will be made in situ essentially continuously from 30 to 100 km
Measurement Philosophy	Measurements will be made via devices placed in situ at the altitude being measured
Launch Timing	
Number of Launches Per Year	Approximately 100 probes will be launched per year at each site (2 per week, nominal)
Time Between Launch	Occasional, closely-spaced launches should be possible at each site (interval of 10 to 30 minutes)
Launch Synchronization	Launch should be possible from all of the worldwide sites at essentially the same time (± 30 minutes)
Safety	
Falling Mass Hazard	The launch system shall not pose a falling mass hazard

Table I-2 (Continued)

Item	Specification
Safety (cont.)	
Airborne Mass Hazard	Unspecified (will be considered in hardware design, and can be resolved through operational consideration)
Directional Range Aiming	The launch system will be traversable as launch range geometry so dictates
Misfire	It shall be possible to terminate launch operations during launch preparations and remove the vehicle from the launcher, if necessary
Storage and Handling Hazard	The launch system shall not pose a hazard from fire, explosion, radiation, or toxicity, such that it cannot be handled, stored, and shipped by civilian, commercial methods
Loading and Arming Hazard	The launch system will be developed such that during loading and arming operations, a personnel error, a deficiency or inadequacy of design, or subsystem/component failure will probably not result in fatal personnel injury or substantial damage to the system
Launch Hazard	The launch system will be developed such that during launch operations, a personnel error, a deficiency or inadequacy of design, or subsystem/component failure will probably not result in fatal personnel injury or substantial damage to the system (other than the vehicle)

Table I-2 (Continued)

Item	Specification
Reliability & Maintainability	
Reliability	The launch system reliability will be such that the probability of attaining desired altitude will be between 90 and 99 percent (nominal value of 95 percent)
Maintainability	The launch system will be designed for simple, go-no, go pre-launch checkout, for a minimum of on-site vehicle and ground support equipment maintenance and checkout
Staffing	
Crew Size	Total on-site staff requirements will be between two and five men
Personnel Training	The system will be capable of operation by relatively untrained personnel of a technician level, including foreign nationals
Operation	The system will be compatible with worldwide vivilian operation and logistics
Other	
Shelf Life	The launch vehicle (system) will have a shelf life of at least three years
Export Category	The launch system will be non-military hardware such that it can be operated and controlled by foreign civilian nationals.

performance specifications indicate the basic launch vehicle characteristics that are required. For example, the specification on trajectory states the necessity of measuring at essentially the same location on each launch. Many of these specifications are derived from the contractual Statement of Work.

Such specifications are affected by expected variations in measurements. For example, it has been postulated that there are gravity waves at 100 km level with horizontal amplitudes ranging from 100 to 1000 km.and vertical changes taking place in distances ranging from 1 to 10 km (Reference 1). On this basis, trajectory differences in the horizontal on the order of 10 km might be acceptable, whereas 100 km would not.

Operational specifications were derived from the worldwide nature of the system. The falling mass hazard requirement is established to indicate that (1) either sufficient launch range land must be acquired or (2) the system must be frangible and consumable in such a way that people and property are not endangered by firing. It is recognized that it is not practical to remove the airborne mass hazard completely.

Operational practice will require the close coordination with possible air traffic in the firing corridors to eliminate chances of interference between launch operations and possible air traffic.

Operational sites will be established in a worldwide network although current provisions do not include establishment of shipboard launch sites. It is desired, however, that the operational problems related to water launch be relatively minimal so that these sites can be included in the future.

Worldwide use of the system by relatively untrained personnel indicates that new concepts in launch reliability are required. It is suggested that industrial reliability concepts of continuous operation for years with reduced tolerances and little maintenance is more appropriate to this system than conventional aerospace practice of high tolerances, redundancy, and high maintenance costs.

3. ENVIRONMENTAL CONSTRAINTS

The environmental constraints established herein are derived primarily from the requirement of being able to launch each of the 100 probes at essentially the same time. The goal is to remove inclement weather as a deterrent to launch (though recognizing that under extreme weather conditions, timely launch is improbable).

Requirements are that the system can be launched in rain, high winds, and foggy conditions with very little slippage due to weather. From a design point of view, the launch wind requirement can be important in some systems, as will be discussed in a later section.

Environmental constraints also include requirements on the storability of the system. The underlying philosophy for storage is that the vehicles and payloads will probably be stored on site for some period of time before launching. Also, the launch sites will be widely dispersed in terms of climatic environments, necessitating a wide range of storage environment capabilities.

Environmental constraints are listed in Table I-3.

II. SENSOR ANALYSIS

1. INTRODUCTION AND REQUIREMENTS

The objective of the overall study being carried out by Booz, Allen Applied Research was to produce a conceptual design for a sounding system which would be low-cost, highly reliable, simple, and synoptic. Within that system, the sensors are the basic, data acquisition subsystem, and must have similar requirements. Further, the sensors, together with the overall mission requirements, set the boundaries from within which suitable sounding systems must be chosen. Thus, the sensors are, at the same time, constrained as to approaches and very important to the overall system.

The sensors must provide measurements of wind speed and direction, temperature, and pressure or density as a function of altitude from 30 kilometers to approximately 100 kilometers. Consideration was given to the feasibility of taking measurements during the ascent as well as the descent stage of the flight. Initially, limited study was made of the feasibility of measuring water vapor and ozone content; later this was changed to ozone only.

Table I-3
Environmental Constraints on Launch Systems

Item	Specification
Rainfall	The launch system should be operable in rainfall up to 4 inches per hour. Exposure to that rain intensity for up to 30 minutes prior to launch should not degrade performance.
Winds	The system should be capable of holding, ready for launch in 53 knot winds. Launch (wind weighted) should be possible in 35 knot surface winds.
Visibility	The system should be operable with 100 foot ceilings and 1/10 mile surface visibility.
Temperatures	The launch system should be storable (up to 6 months) and operable in temperatures from -40°C to +50°C.
Humidity	The launch system should be storable (up to 6 months) and operable in humidities up to 95 percent.
Salt Spray	The launch system storage (up to 6 months) or operation should be unaffected by salt sprays up to 20 percent (by weight) solution.
Shelf Life	Three years in a controlled temperature, humidity environment.

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The desired accuracies of measurement are:

- Temperature $\pm$ 2°C rms at 30 kilometers, and not exceeding $\pm$ 5°C rms at 100 kilometers.
- Wind velocity $\pm$ 5 percent rms, vector error.

Pressure and density accuracies should be consistent with temperature accuracies. No detailed tuning or calibration is to take place at the site.

The fact that sensors are part of a payload which, in turn, is part of the overall system, affects sensor choice. For example, sensor choices can be affected by the requirement that tracking and data acquisition equipment be relatively low in cost, reliable, and semi-permanently located. Similarly, if a sensor requires costly ground support equipment, the sensor cost cannot be divorced from the cost of the support equipment.

2. CHARACTERISTICS OF THE ENVIRONMENT

The major requirements on the sensors are to indicate the magnitudes of density (or pressure), temperature, and wind velocities. (References 2, 3, 4, and 5)

Pressures and densities decrease markedly over the altitude range. For example, the pressure decreases by 2 orders of magnitude between ground level and 30 kilometers, and decreases by more than 4 orders of magnitude between 30 and 100 kilometers.

Winds and temperatures do not change so grossly over this altitude range. The maximum mean wind is about 125 knots. Minimum and maximum mean temperatures extend from about 165°K to about 325°K. At the highest altitudes, however, the atmosphere is so diffuse that temperatures do not have the same implications in terms of effects as they do at lower altitudes.

(1) Some Pressure/Density Characteristics

Based on the preceding discussion, it can easily be seen why conventional ground-level barometers will not work at high altitudes, and why many sensors which will work over some range of altitudes will not cover the complete 30-100 kilometer range. Similarly, many density effects also drop off very rapidly with increasing altitude.

Table II-1 shows changes in density and pressure with altitude, as well as the change in certain associated parameters.

Table II-1
Densities and Pressures and Some Resulting Characteristics

	Parameters						
Altitude	Density	Pressure	Buoyancy Effect per Cubic Meter		Force on a 10 Cm (4 in) Dia. Disk		Mean Free Path
Km	g/m^3	Millibars	g	Lb	Dynes	Lb	Cm
0	1.2×10^3	1×10^3	1.1×10^3	2.4×10^0	7.9×10^7	1.77×10^2	6.6×10^{-6}
30	1.8×10^1	1.1×10^1	1.8×10^1	4×10^{-2}	8.6×10^5	1.9×10^0	4.5×10^{-4}
40	4×10^0	3×10^0	4×10^0	8.8×10^{-3}	2.4×10^5	5.2×10^{-1}	2.0×10^{-3}
60	2.2×10^{-1}	2×10^{-1}	2.2×10^{-1}	4.8×10^{-4}	1.6×10^4	3.5×10^{-2}	2.7×10^{-2}
80	2×10^{-2}	1×10^{-2}	2×10^{-2}	4.4×10^{-5}	7.9×10^2	1.8×10^{-3}	4.1×10^{-1}
100	5×10^{-4}	3×10^{-4}	5×10^{-4}	5×10^{-4}	2.4×10^1	5.2×10^{-5}	1.6×10^1

For example, it lists the force which could be exerted on a 10-cm disk such as might be used in a large aneroid-type instrument.

(2) Temperature Characteristics

As already noted, the lowest value for the coldest mean temperature is about 165°K. This is found between altitudes of 80 and 90 km. The highest value for the warmest mean is about 325°K, at ground level.

Of more interest, however, are the lapse rates (negative of the rate of change in temperature with altitude) and the lapse rate reversals. These can be clearly seen in Figure II-1 which shows a composite standard temperature versus altitude curve.

Temperatures also have very interesting horizontal gradients. It is expected, however, that these will be derived from multi-shot soundings, rather than being pertinent to the sensor requirements of individual vehicles.

(3) Winds

As noted, the value of the maximum mean wind is about 120 knots. (See Figure II-2) However, a wind sensor is

FIGURE II-1
Temperature versus Altitude

(Composite Standard - All Geographic Locations and Seasons)

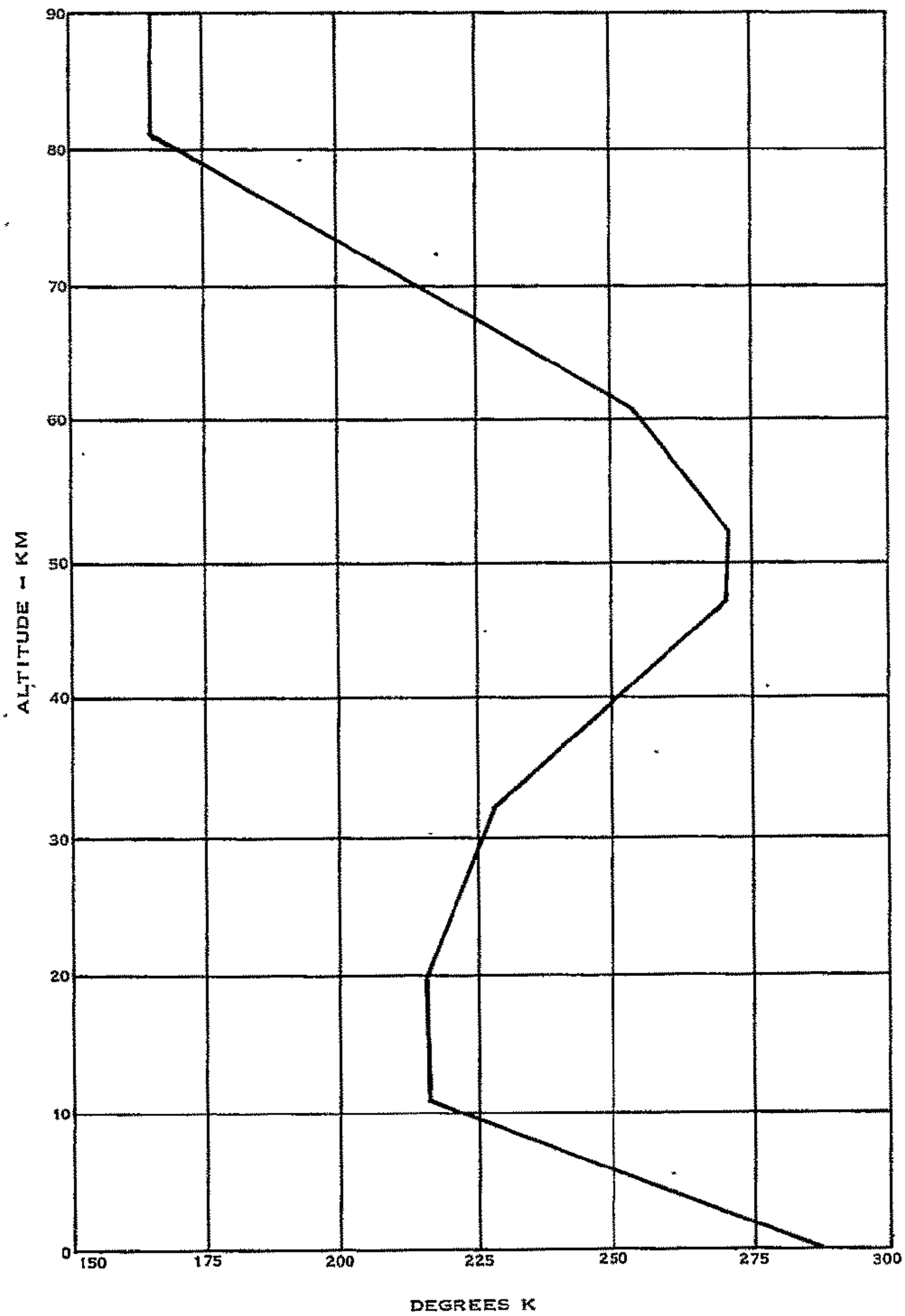
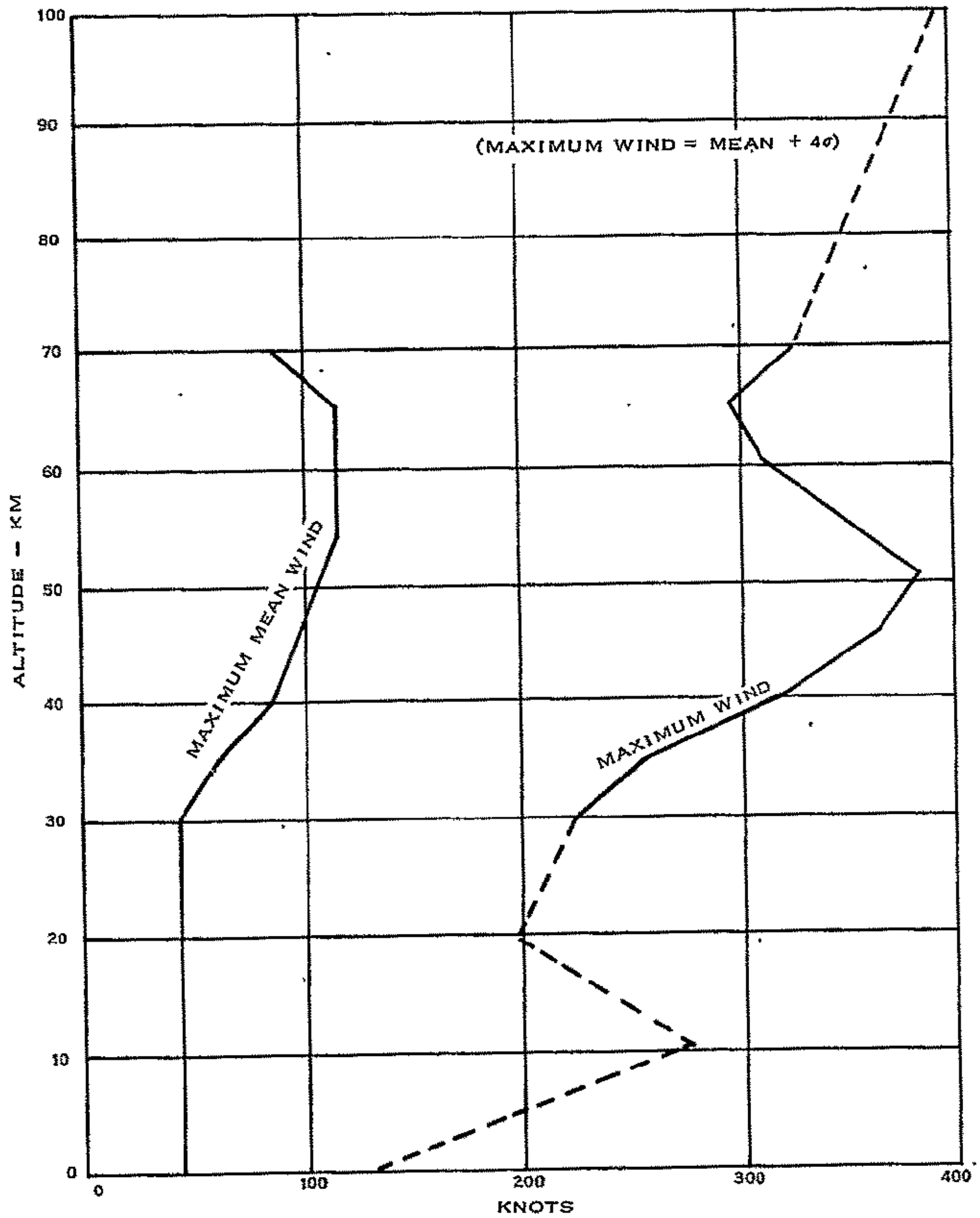


FIGURE II-2
Wind versus Altitude

(Composite Standard - All Geographic Locations and Seasons)



obviously useless if and when it is flown in a wind too violent for sensing. Therefore, a maximum expected wind speed line is also shown.

(4) Gravity Waves

In addition to the mean atmospheric structure and variations discussed above, dynamic oscillations occur and their measurement promises to be an important goal of future sounding systems. These oscillations are known as gravity waves. They are caused by the gravitational restoring force applied to large-scale perturbations (diurnal thermal tides) and small scale perturbations originating from cyclonic disturbances, fronts, jet streams, winds over mountains and planetary Rossby waves. (References 6 and 7)

These gravity waves of horizontal character can and will propagate upward depending upon the wind field above. In the winter, waves originating from fixed sources usually reach the thermosphere, but are not particularly energetic. In the summer, stratosphere wind reversals tend to nullify all such waves. The planetary Rossby waves, which are energetic, can propagate upwards to the thermosphere in winter when a breakdown of the westerly flow around the pole occurs. This action has been observed by several experimenters and is accompanied by heating and turbulence in the lower thermosphere. This propagation accounts for major energy interchange in the vertical structure and is a key, not well understood, to atmospheric circulation.

Potentially, even more important is the suggested mechanism that thermospheric turbulence is responsible for carrying ionizable components of the E-region down to the D-region and creating a temporary, large increase in electron density. The simultaneous effects of sudden thermospheric warming, wavelike turbulence and D-region radio wave anomalous absorption has been documented recently. If the interrelationship of the ionosphere and neutral stratosphere and mesosphere is thus (or similarly) established, a tremendously increased interest will be focused on the results of the present sounding system study.

Because of the short time period and small scale of gravity waves (excepting tides), they have not been properly observed or understood. Only with a very cost-effective system can we afford to make sufficiently dense observations in space and time to give an adequate description of this important phenomenon.

3. THE EVALUATION OF PRESENT AND PROPOSED SENSING TECHNIQUES WITH RESPECT TO THEIR APPLICABILITY

All known sensing techniques which have been developed, are being developed, or are proposed, and which have potential for sensing winds, temperature, density, or pressure in a substantial portion of the 30-100 km region have been considered in the evaluation. Included also are two suggested methods which have not been formally proposed but which analysis indicated most logically to be considered for possible feasibility studies.

All techniques were evaluated on the basis of their present state of development, on the basis of their expected state of development in 1975 if the currently funded or proposed programs are continued, and on the basis of their potential state of development in 1975 if the recommendations of this study are implemented. Techniques for the measurement of water vapor and ozone are not included. Due largely to the low accuracy required, these constituents will probably be measured remotely from satellites by 1974. If not, the required frequency of observations will be low enough to permit a separate package to be developed and flown.

(1) Techniques Evaluated

Winds

- Passive Sphere
- Grenades
- Parachute and Ballute
- Chaff
- Tri-Methyl Aluminum Trails
- Meteor Trails
- Speed of Sound
- *Ion Wind Techniques

Density

- Active Inflated Sphere
- Active Solid Sphere
- Passive Sphere
- Spinning Wire
- Meteor Trails
- Molecular Fluorescence Densitometer
- Beta Ray Densitometer
- Laser Backscatter
- Speed of Sound

Pressure

- Pitot & Pitot Static Gage
- Cryogenic Crystal
- Solid State Pressure Transducer

Temperature

- Grenades
- Thermistor
- Molecular Fluorescence Densitometer
- Speed of Sound
- *Active Inflated Sphere
- *Active Solid Sphere
- *Passive Sphere
- *Beta Ray Densitometer
- *Spinning Wire
- *Meteor Trails
- *Laser Backscatter
- *Pitot & P. Static Gage
- Cryogenic Crystal

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*Derived from another parameter.

(2) Techniques Rejected

1. Grenades

This is currently a valid technique which is costly from every standpoint. Its merit is primarily the lack of competitive temperature measurements in the 65-95 km altitude range. It is well-developed and no dramatic breakthroughs are anticipated. (References 8, 9, and 10)

Its drawbacks, in addition to cost, are its susceptibility to vertical winds in the upper altitude range, and the fact that only average layer temperatures can be deduced, thus preventing good vertical resolution.

2. Vapor Trails

The visual observation of smoke and vapor trails is a valid present technique for winds and is particularly valuable for fine vertical resolution. (References 11, 12, and 13) Such trails are, however, restricted to certain times of day and proper seeing conditions. As such, the technique is not capable of synopticity.

3. Active Inflated Sphere

The accelerometer-instrumented, inflated sphere was developed for the express purpose of extending the falling sphere technique to higher altitudes. (References 14 and 15) As such, the heart of the instrument (a system of three single-axis accelerometers), was chosen to work at low accelerations and does not exhibit large dynamic range capabilities. The experimenter claims to have measured 140 km densities by this method, and acknowledges that the design is particularized for altitudes above 100 km.

This technique is costly because of its construction characteristics and because of the requirements for ejection and inflation, as well as the excess rocket altitude needed to give measurable drag deceleration. The extension of this method to cover the 30 to 100 km range is considered unattractive due to an ultimate cost which is too high.

4. Active Solid Sphere

This falling sphere technique requires a very sensitive accelerometer due to the unfavorable mass/area ratio. This pluse the high cost of the sphere itself, ejection, excess altitude requirement, and inability to accurately obtain data at the highest altitudes is the cause of rejection as a synoptic tool. (References 1 and 16).

5. Beta-ray Densitometer

This technique uses a radioactive source which permits electrons to be forward-scattered by air nuclei to a collector ring of Geiger-Muller counters. (Direct transit between source and counters is prevented by mounting the source next to a small, properly shaped lead shield.) (References 17 and 18)

The accuracy of the density inferred from scattering measurement is not presently acceptable, though improvements in theory and additional testing may overcome this problem. The instrument as presently conceived is ultimately expensive as a synoptic tool and the carrying of radioactive materials, even in small amounts, is not acceptable worldwide.

6. Alpha Particle Densitometers

Other techniques are possible for measuring gas densities by means of radioactivity. One involves measuring the energy degradation in energy alphas from a monoenergetic alpha source. Another involves determining a count rate for alphas from a source with a broad spread in energy. Both techniques use alphas because of their high interaction rate. Again, the instrument does not seem acceptable on a worldwide basis because of the radioactivity involved.

7. Solid State Pressure Transducer

A current effort is underway by Research Triangle Institute to develop a sensitive pressure transducer for atmospheric pressure measurement. A design objective is the capability of measurement from sea level to 10^{-5} or 10^{-6} atmospheres (approximately to 100 km).

The transducer is characterized by a piezo function effect. The unique feature of the transducer is the silicon needle sensor--a p.n. function fabricated in the apex of a silicon needle.

The present effort has indicated the high precision required to machine and assemble these components of the sensor, its temperature sensitivity, and its lack of repeatability at the lower pressure ranges.

These limitations may be corrected by additional effort, however, the technique is considered to be very early in its research stage and a review of the developments do not make it a realistic candidate for an operational system in the time frame being considered.

8. Cryogenic Crystal

This technique uses an extremely cold crystal exposed, within a tube, to incoming ambient air. The air condenses on the crystal and changes its characteristic frequency. (References 19 and 20) Ultimately, such a technique may be useful for very high altitudes. Because of the many technological and theoretical problems involved, it is not believed that this method can be used for this application.

9. Electric Discharge Techniques

A number of techniques can be listed under this category, including several quite useful for laboratory measurements. They include the cold cathode-type ionization gage, the Alphatron, the hot filament (Bayard-Alpert) gage and the Rofe arc discharge gage. All of these discharge techniques depend upon the fact that the current between plates or between a filament and a plate is a function of the amount of ionization produced which is a function, in turn, of air density. (References 21 and 22)

An example of the cold cathode technique is the ion pressure gage built for ALSEP. (References 23 and 24) In this type, ionization is produced by a high voltage field (4,500 v.d.c.). A high magnetic field (magnetron) is used to lengthen the electron path and consequent ionization. This configuration is not expected to be practical for altitudes less than about 80 km and the cost, even in quantity, is expected to be greater than $1,000 per unit.

A different approach to increasing ionization is to use a radioactive source as with the Alphatron. (Reference 22) This device does have adequate altitude range capability. The Alphatron is judged to be too expensive and would be rejected in any case because of the radioactivity involved.

Ionization can also be supplied by means of a hot filament as with the Bayard-Alpert type gages. In order to prevent burn-up of the hot filament, its operation is restricted to altitudes greater than 90 km. (References 21 and 22) It is also expected to be expensive because of associated electronics.

Another approach is the arc discharge, as proposed by Rofe of Australia. (Reference 25) Complete details of this instrument have been requested but not yet received. It is believed that this gage is very sensitive to the amount of natural ionization existing at the higher altitudes.

One problem common to all such gages is that of mounting it such a way as to sense truly ambient conditions. Corrections can, of course, be made; however, such corrections imply the possibility of error.

10. Parachute and Ballute

The tracking of parachutes has proved valid as a wind technique in the 50 km region and below. The parachute techniques are limited with respect to altitude range because the fall rate cannot be optimized over large ranges of density. Ballutes and other nonaerodynamic deployment devices are successful in extending the useful altitudes upwards, but have little effect on the altitude range. Because of their adaptability and low cost, they remain candidates for a system employing a combination of sensors. (References 26, 27, and 28)

11. Speed of Sound Devices

It has been proposed that a measurement of speed of sound in-situ be made by a device carried aloft and having both source and sink. Grenade techniques presently used have the sink located on the ground. A transmitter and receiver operation would measure the transit time of an acoustic wave between them. Analysis shows that such a device would probably have weight and volume dimensions far too large for consideration if measurements at less than 1 torr were attempted.

A device was testing several years ago which operated on a principle of acoustic loading the sound receiver by generating a standing wave between source and sink. By varying the frequency, which alters the sound speed and allowing close spacing between transmitter and receiver, the acoustic loading on the receiver was found maximized at the anti-resonant frequency of the receiver. By measuring the receiver output versus frequency and knowing the spacing, the sound speed was deduced. The transmitter and receiver were on the order of 1-1/2 inch diameter and spacing was 3/2 wavelength at about 28 KHz.

A feasibility study was completed in 1962 on the interference device described. The study showed, with the equipment then utilized, that errors due to aerodynamic effects of wind, acoustic noise due to wind, energy losses, phase velocity effects due to wind, thermal effects of the boundary layer, radiation, and expansion, could all be eliminated or minimized by proper design. The study concluded that the device was feasible to 65 km. Extrapolation of the basic quantities to higher altitudes indicate that the size and power requirements become unmanageable. Follow-on development for an operational system was not undertaken.

While this and other speed of sound devices appear accurate below 70 km, their basic complexity makes them uneconomical in competition with other systems, such as thermistors, and, therefore, of little interest. Above 70 km, the generation of measurable acoustic energy requires too much power for applicability to small rocket or gun-launched vehicles.

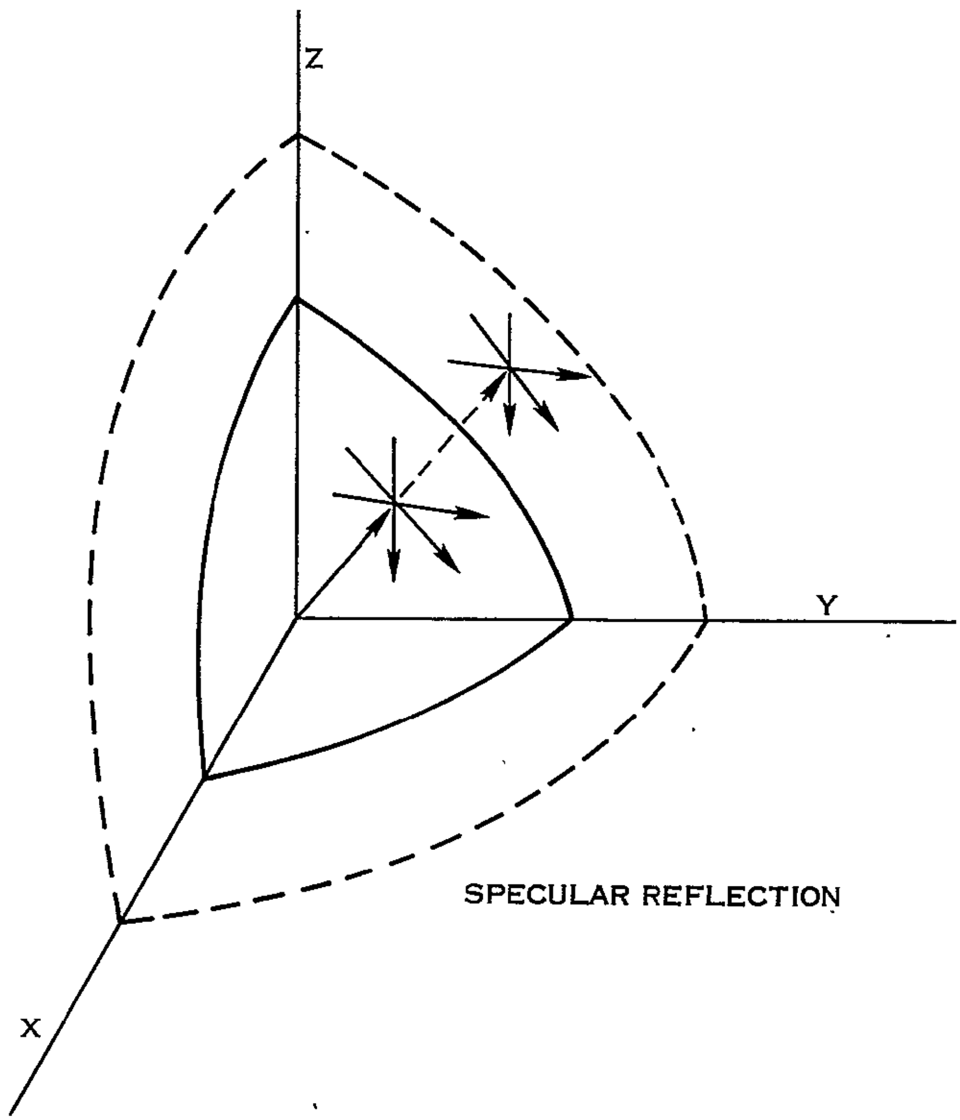
Several ground-based systems were investigated and are discussed below.

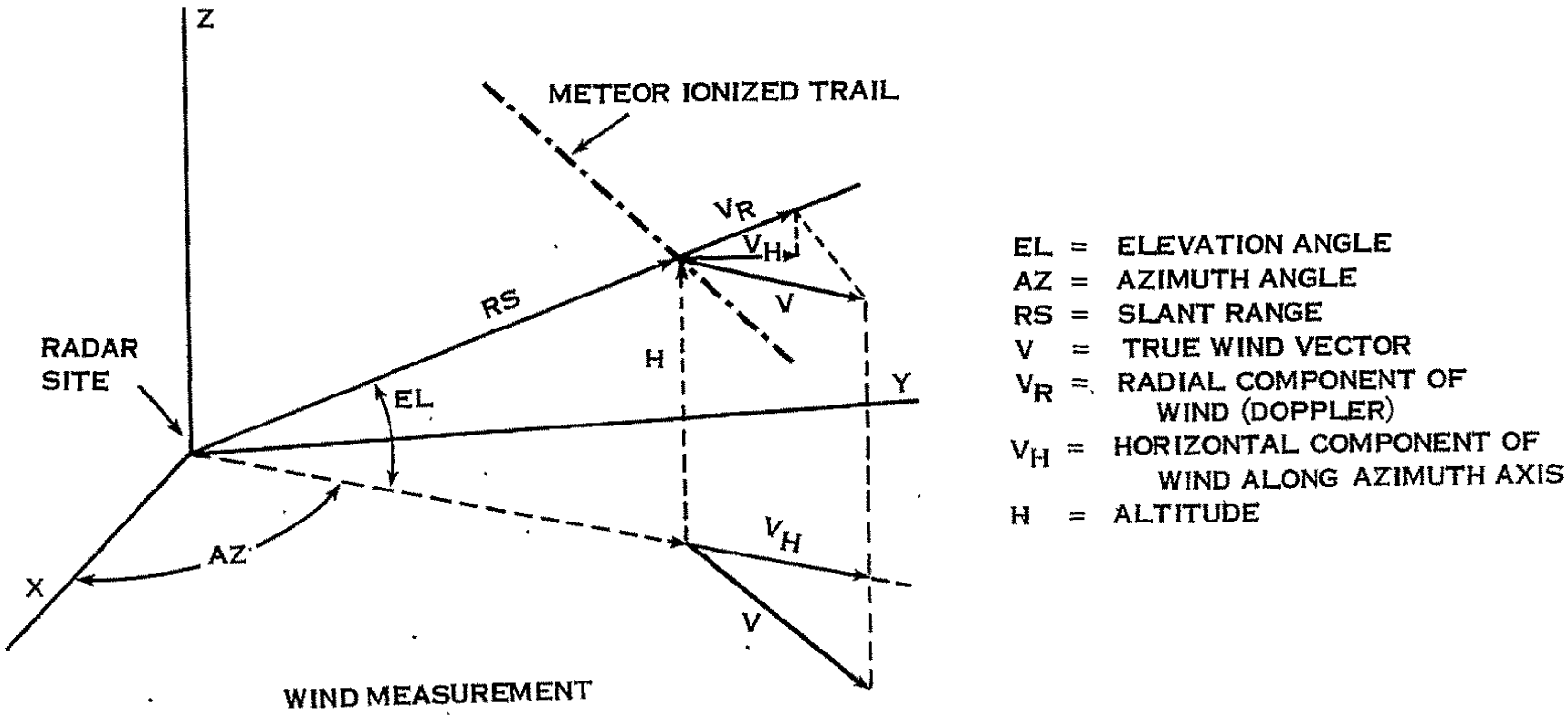
12. Meteor Trail Radar

When a meteor enters the earth's atmosphere, a trail of ionization is produced that may be 10 to 40 km long at an altitude of 80 to 120 km. Under certain conditions, this ionized trail will scatter electromagnetic radiation with sufficient intensity to provide detectable signals. The most favorable case for reception occurs when specular (i. e., mirror) reflection is possible within the geometry imposed by the trail and receiver/transmitter antenna orientation. If the receiver and transmitter are collocated, this will occur when the trail is perpendicular to the radiated beam, in the same fashion as ordinary radar detection. Of the large number of meteors entering the volume under radar observation, only a small portion will at some point on the trail have the proper orientation for a strong reflection (Figure II-3). However, the number of occurrences is still sufficient for measurements of some upper atmosphere parameters (200 to 20,000 per day depending on equipment used). (References 29 through 33).

Wind velocity is measurable as a result of the Doppler shift related to the motion of the trail. This Doppler shift corresponds to the radial velocity of the trail relative to the observing site. Thus, the measured velocity, V_r, is the component of the wind vector along the axis of the radar beam as shown in Figure II-4. To convert this radial velocity to a horizontal velocity, assumptions must be made about the vertical component. Normally, it is assumed that the vertical component averages to zero over the time period during which an average radial velocity has been determined (typically, 1/2 hours to 24 hours). Then, the horizontal velocity V_h, is just V_r multiplied by the cosine of the radar's elevation angle. However, we still only have the component of the wind in the direction of the ground track of the radar. To convert this component to the true wind direction, it is necessary for a single station to make measurements on two orthogonally located meteors, and take the sum of the squares of the two measurements.

FIGURE II-3
Meteor Trail Radar





'IGURE II-4. Meteor Trail Radar

A variety of standard radar techniques have been applied to meteor measurements including pulse/Doppler, continuous wave, and pseudo-random noise coding. Because the reflection coefficient and maximum altitude for detection are inversely dependent on frequency, the experimental work has been done at the lowest frequencies consistent with directional antennas and low manmade interference levels. Most commonly used are the frequencies between 25 to 75 MHz.

Several stations in current (or recent) operation are listed in Table II-2. All of these stations are capable to some extent of producing wind data, but are primarily tools of broader research programs. In each case, data reduction involves highly-skilled personnel for preprocessing before automatic data reduction, although the Havana station does use some pattern recognition techniques to reduce this preprocessing. None meet the normal criteria of an operational sensor for day-to-day use.

At first examination, radar echoes from meteor trails appear an ideal solution to a problem with only limited alternative solutions. This is because the primary sensor is available at no charge and is continuously in the region under consideration. However, the complexity of the ground equipment and sophistication required for operation more than outweight the advantages of a free sensor. If winds are to be measured in stratas of 5 to 10 km and averaged over periods of 15 minutes to an hour, high-powered transmitters, elaborate antennae systems, and remote multiple receiver sites linked with real-time data processing are required. The equipment cost to provide coverage from 80 to 100 km synoptically, with respect to a 15-minute time span, would exceed $750,000 per site, and require development of equipment and techniques more suited to operational measurements. The usefulness of such a system would still be questionable, since vertical movements are ignored and measurements are averaged over a 5-10 km strata. Overall accuracy would be in the range of 10 to 50 percent. Thus, the technique does not currently provide sufficient cost benefit ratio to warrant incorporation.

Table II-2
Stations in Current Operation

Organization	Location	Type	Freq.	Rate/Hour	Height Accuracy	Cost
AFCRL	Lexington/ Dartmouth	Pulse/ Doppler	36.8 Mhz. 73.6	5	±5 km.	$100K
Stanford U. (AF)	Stanford, Calif.	Coherent random noise coded pulse Doppler	30/14	20	±2.5	$50K
Smithsonian (AF)	Havana, Ill.	Coherent multistation pulse/Doppler	40.92	500	±3	$1,000K
C.N.E.T.	Garchy, France	CW interference	30	10-20	±3	$150K
U. of Adelaide	Adelaide, Australia	CW or pulse/ Doppler multi-station	26.8 27.5	30	±2	$200K
Sheffield U.	Yorkshire, England	coherent pulse/ Doppler-building multi-station	25	20	? (calibrated by decay rates)	$30K

13. Ion Wind Techniques

Ionospheric investigators have developed several methods for tracking ionospheric motions in conjunction with other measurements such as electron density. (References 35 and 36). In the lower ionosphere such motions are fairly well related to the neutral winds. However, because of the expense of these methods in general, their limit to the upper portion of the altitude range and the competition afforded by Meteor Trail Radar (discussed previously), these techniques were rejected from further consideration.

14. Laser Backscatter

This method is valid and inexpensive on a per measurement basis, (Reference 37) The inference of density from backscatter is not sufficiently accurate at present, but may be improved in the next few years. While this method is one which probably will have considerable future importance, it requires clear skies with little or no cloud cover to operate and is therefore not suitable as a synoptic technique.

(3) Techniques Found Acceptable

1. Falling Spheres

The falling spheres included in systems studied in this report are made of half mil mylar and are filled with isopentane. Sphere sizes range from less than a meter in diameter to several meters. Spheres are carried in the vehicle collapsed and are inflated at ejection. They are generally carried packed within a cylindrical container having a length to diameter ratio consistent with the type of vehicle.

For the system under consideration, the sphere descent begins at an altitude above which data is desired in order for the sphere to attain sufficient velocity for drag to affect the sphere and to provide sufficient density for integration for temperature data calculations. (References 38, 39, 40, and 41).

All of the techniques involving falling spheres use the net deceleration of a freely falling sphere to indicate drag, and from drag, infer atmospheric density. The differences in techniques result from different approaches to the problem of measuring deceleration.

There are basically three approaches, which are:

- Tracking by radar (passive sphere)

- The use of a receiving/transmitting transponder (transponder sphere), and

- The use of accelerometers and a radio transmitter (active sphere).

For our somewhat specialized requirements, we have studied applications of the passive sphere, but not the active sphere. In addition, we have analyzed the use of a transponder sphere.

In general, the passive sphere characteristics are consistent with the descriptions already given for falling spheres. In addition, the mylar surface is aluminized or an internal corner reflector is incorporated in order to provide a better radar target.

The development of the sphere itself has benefited from a number of years of study and has probably progressed about as far as possible. Packing and ejection schemes for specific requirements will require additional development. There is a need for across-the-board research on applicable drag coefficients. The chief requirement, however, is for extensive radar development. This will receive additional discussion in a later section.

Some typical value of pertinent characteristics for passive spheres of today are listed below.

Diameter	.66 - 2 meters
Weight of sphere (including filler and filler container)	50 - 515 grams
Payload weight (including casing and ejector)	1360 - 2000 grams
Payload volume	18 - 90 cubic inches
Resistance to g loading	Some spheres have been gun-boosted
Storability	Adequate
Cost	Varies. In $1,000 range for one-at-a-time research items.

Some typical values of pertinent characteristics for the postulated passive sphere to be used are:

Diameter of sphere	1 meter
Weight of sphere (including filler and filler container)	130 grams
Payload weight (including casing and ejector)	1360 grams (3 pounds)
Payload volume	40 cubic inches
Resistance to g loading	Hardened loads can be routinely gun-boosted
Storability	Adequate
Cost (10^5 lots)	$85

Falling spheres, carried aloft in rockets, have been used hundreds of times since 1952 to measure density, temperature and winds in the upper atmosphere. The fundamental equation of the experiment is the familiar one of aerodynamic drag:

$$F_D = m\ a_D = 1/2\ \rho\ V^2\ C_D A$$

where

F_D = drag forces

(Reference 9)

m = sphere mass

a_D = drag acceleration

ρ = ambient density

V = sphere velocity

C_D = coefficient drag

A = sphere cross-sectional area.

Value for the C_D as a function of Mach number and Reynolds number are established from ground measurements carried out in wind tunnels and ballistic tunnels. (Reference 38) Having measured density as a function al altitude, the equations of state and of hydrostatic pressure are combined to permit the calculation of temperature.

$$T_z = \frac{1}{\rho_z}\left[\frac{M}{R}\int_{z_o}^{z} -\ \rho g dz + \rho_o\ T_o\right]$$

where

T_z = ambient temperature

z = altitude

z_o = starting altitude

M = gram molecular weight, known from other measurements

R = universal gas constant

g = acceleration of gravity

ρ_o = ambient density at z_o

T_o = ambient temperature at z_o

Typically the integration of density proceeds downward from the starting altitude z_o which is the altitude of the highest valid density data. The arbitrary choice of ρ_o and T_o at this point may introduce an error in the calculated temperature which, however, decreases and becomes negligible by comparison with other errors at a point about 15 km below the starting altitude.

Horizontal wind velocities can be determined from ground-tracking sphere motions. The winds can be computed from the equations of motion, or more simply but less accurately, be taken as equal to the projections of the sphere velocities. Vertical winds are neglected. Some work has been done on the effects of vertical winds and on the small effect of neglecting them. Also, it has been shown that vertical winds might be measured by observing simultaneously two spheres of different mass-to-area ratios. The method, however, has not been fully developed. The upper limit of wind data is dictated by a combination of radar tracking errors and the sphere fall rate. An improved system may be capable of extending the upper limit to considerably above the present commonally accepted 70 kilometers level.

Insofar as the sphere itself is concerned, operations are quite simple, consisting of merely ejection and inflation. Obtaining rates of deceleration, however, requires quite careful measurements using a very sophisticated radar system. Whereas the sphere is a very low-cost item, an adequate radar is apt to be a very high-cost item.

Extensive experience has been attained in the fabrication and use of spheres. There is little expectation of any major gains, with regard to the sphere itself. Better drag coefficients are a possibility and research leading to them is desirable. More experience in the use of radar in sphere experiments is not expected to be particularly helpful. Directed development, aided by the normal improvement in the state of the art, will be needed

The use of an aluminized mylar corner reflector within the sphere surface would increase the radar target, leading possibly to a somewhat cheaper radar. It would, however, increase the weight and cost of the sphere. On balance, it seemed better in decreasing the requirements on ground-support to go a good deal further and consider the use of a transponder sphere, discussed in the next section.

Vertical wind effects have been cited as a disadvantage in the use of spheres. However, vertical winds have little effect on density data above 73 km. Below 73 km, the effect is subject to considerable smoothing and is unimportant in most cases. In any case, vertical winds cannot be measured by a single sphere technique. As compared with other sensors, spheres should be less affected at high altitudes than grenades (not a system considered in this report). Pitot systems would be adversely affected during the low-speed portion of their flight.

There is little operational experience in the use of spheres as payloads for gun-boosted vehicles. There seems to be no inherent reason, however, that spheres and related equipment could not be hardened and used routinely.

Based on a ten-year storage experience with magnetic tapes, and the six years experience of the Echo satellite in space, the storability characteristics of mylar should be adequate.

2. Passive Sphere with Transponder

Transponder sphere characteristics are consistent with the description already given for falling spheres in general, except that, in addition, a transponder is attached to the mylar surface and some type of antenna is required.

A transponder sphere system is a new, proposed technique so that experience with the system as a whole is lacking; however, a great deal of the experience obtained with passive spheres is relevant. Particularly, this includes general experience in drag effects, packing and ejection.

If it is not obtained as part of the on-going research effort on passive spheres, there will be a need for across-the-board research on applicable drag coefficients. Packing and ejection schemes for specific requirements will require additional development. Finally, there will be a requirement for the development of tracking equipment with characteristics specifically aligned to the requirements of the transponder sphere. This will receive additional discussion in a later section.

Some typical values of pertinent characteristics for the postulated transponder sphere to be used are:

Diameter of sphere	2 meters
Weight of sphere (including filler, filler container and transponder)	960 grams
Payload weight (including casing and ejector)	2,725 grams (6 pounds)
Payload volume	120 cubic inches
Resistance to g loading	Hardened loads can be routinely gun-boosted

Storability	Adequate (but see discussion)
Cost (10^5 lots)	\$535

As with the passive sphere, the transponder sphere should have the capability of measuring density from 30 km to an altitude in excess of the 100 km required. It should have the capability of measuring winds from 30 to 70 kilometers, or a little less. It also is an indirect sensor.

Ejection and inflation should be nearly as simple for the transponder sphere as for the passive sphere. Activating the transponder presents an additional problem but not one which is inherently and extremely difficult. As compared with the passive sphere, the transponder sphere permits somewhat lower cost ground equipment at the expense of a larger, more complicated, heavier payload.

It would be desirable to have the same weight-to-area ratio for the transponder sphere as is used for a passive sphere. This is not possible. (The mylar skin area and weight increase with D^2 as does the frontal area, and the weight of the isopentane increases with D^3.) The diameter at which the weight-to-area ratio is a minimum could be found by solving the equation

$$\frac{D}{D_o} = \left(\frac{2T}{I_o}\right)^{-3}$$

where

T = Transponder weight

I_o = Weight of isopentane in a sphere diameter D.

In fact, a practical, near-optimum size of 2 meters is assumed.

Most of the fabrication, storage, and operational problems added by the use of a transponder involve connections between sphere, transponder, and antenna. The problems can be more subtle than is obvious initially, but they are technological problems which are solvable.

Locating the transponder on the inside and outside surface of the sphere have both been suggested. The external location has obvious advantages for maintenance, pre-flight adjustments, and battery replacements. It does leave unresolved doubts as to possible effects on drag coefficients. Some authorities have considered that locating the transponder within the sphere would be very difficult because it affects the integrity of the sphere. The routinely-fired Australian 2-meter spheres, however have a hole approximately 3 inches in diameter at each pole, one of which removable and replaced by an isopentane canister before firing.

If it is undesirable to mount the transponder externally or to make provisions for last minute attachment, a dry-packed, liquid electrolyte battery could be used with a diaphragm which would be ruptured mechanically or magnetically from outside the sphere. This arrangement provides no means for maintenance or adjustment.

The antenna could be a metallized surface on the inside of the sphere. Another approach would be to use short wire dipoles. The wire could protrude through the skin and remain sealed through ejection, inflation, and operation.

With a mass of one pound fixed to the surface of a 2-meter sphere, attempted orientation and severe undamped pendulation should occur at high altitudes. This should have little or no effect on the drag coefficient. When passing through the transition to subsonic flight at about 73 km, one would expect a minor effect on drag coefficient, but as drag coefficients near Mach 1 are rather unreliable anyway, this will not degrade system performance. In the subsonic regime, no effect is anticipated.

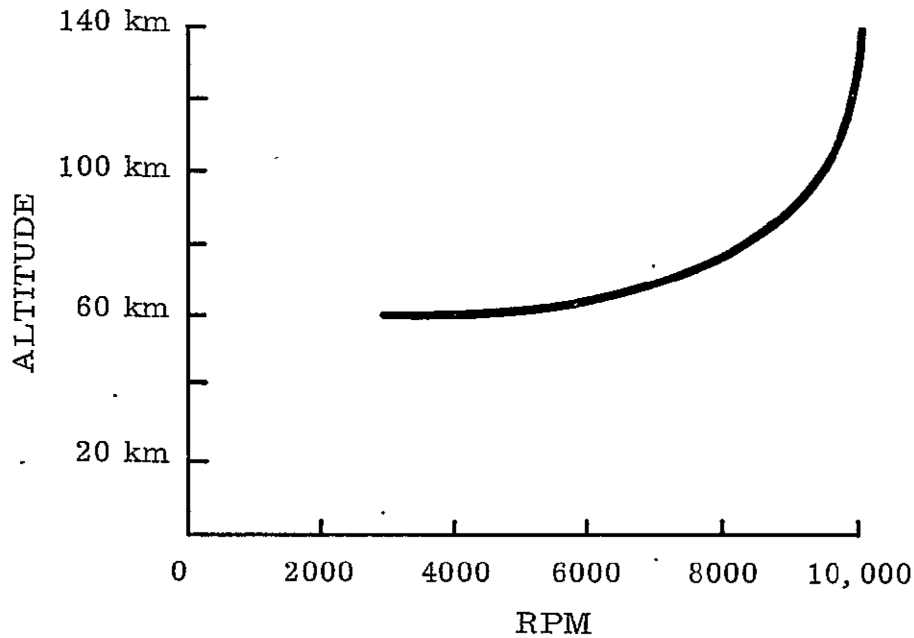
Below 70 km, the difference of pressures inflating the sphere gradually diminish until the sphere collapses usually at 30-40 kilometers. The acceleration in this regime is about 1 g. The sphere will suffer distortion sufficiently severe to alter the drag coefficient for only a few kilometers above collapse. This distortion is not considered to be a problem.

3. Spinning Wire Densitometer (SWD)

The Spinning Wire Densitometer, like the falling sphere, measures the effect of drag in order to infer density. The instrument consists essentially of a cylindrical center body, containing a radio transmitter and two radially aligned wires attached to opposite sides of the center body. It is spun up to a high rpm before being released from the vehicle. Any number of devices can be used to produce the spin; the original concept involved a pyrotechnic motor. (A current version uses a series wound DC motor.)

Originally, the wires were to double as antenna and high-drag components. (The modulated signal generated by the rotating antenna was to have provided a direct indication of rpm.) As it turned out, the signal/noise ratio of available 1600 Mhz transmitters was found to be insufficient. The center portion must now include some means of measuring rotation, as by sensing a changing light intensity or cutting the earth's magnetic field with a magnetometer coil. (The magnetometer approach would provide a 24-hour-per-day capability.)

The present payload consists of two portions: first, a spin-up device—DC series would motor and its associated battery and second, the center body consisting of a battery, a transmitter and either a photo-optical sensor or magnetometer device, and the thin wires (5-7 mil diameter) wrapped around the center body. At an altitude of approximately 80 kilometers, the nose cone is separated and the DC motor is energized to spin-up the device to approximately 10,000 rpm which is achieved at about 120-130 kilometers apogee. At that time, the motor-battery portion separates from the remainder of the payload and the spinning device begins free-fall trajectory. Laboratory research indicates that the spin rate will decay approximately as shown below:



The sensing device will measure the period of rotation, thus providing an indication of rpm at the end of the spin-up phases for an initial point in the sequence of slow-down due to drag.

The results described above represent research and development which has been going on for several years. Several flight tests have been flown; results have not been obtained due to telemetry and tracking failure.

The pertinent characteristics of the present model of spinning wire densitometer are listed below:

Length and diameter of central cylindrical body	5-6 in. x 1.5 in. diameter
Length and diameter of wires	14 in. x 5-7 mil diameter
Spin rate measurement	Sun sensor

Spin-up device and rpm	DC motor to 10, 000 rpm
Payload weight	≈6 lb.
Payload volume	≈6 cubic inches
Cost (lots 10^5)	≈$300

The sensor system has a good cost potential. The payload itself should be inexpensive ($200 to $300). Because of the method of determining drag (i. e., angular deceleration), an expensive high-precision radar is not required. However, the omission of an expensive radar is based on the assumption that trajectories can be made repeatable.

As compared with falling sphere methods, it should be easier to measure decelerations because the techniques for measuring periods of rotation can be made (theoretically) very accurate. However, because of the differences in linear speeds near the tip and near the base of the spinning wires, interpretation is apt to be more difficult. Further, while experimental verification of spin-down rate should be possible, the inclusion of effects due to the fall of the sensor through the atmosphere, would be extremely difficult to analyze.

The possibility of precession, due to winds and misalignment at separation and the effect of this on accuracies, have not been determined.

4. Molecular Fluorescence Densitometer (MFD)

This atmospheric density measuring device is characterized by making measurement of density as contrasted to those devices making measurement of the effects of the atmosphere on measuring devices; i. e., falling spheres, spinning wire densitometer. A description of the device and the principles of operation are described in reference 42. (Also references 43, 44, and 45.)

The system has been flown; however, the experiment was only partially successfully due to telemetry failure. One important result of the experiment was an indication that sufficient secondary emission of electrons took place so that the flight package retained a neutral charge (rather than going highly positive, due to the emission of electrons into space, leaving the vehicle with a net positive charge).

The immediately scheduled flights for this prototype device are:

- Two flights scheduled for late in 1968, sponsored by J. F. Morrissey of AFCRL
- One flight scheduled for January 1969, sponsored by W. Vaughn and R. Smith for NASA/Marshall Space Flight Center with the University of Michigan.

The present device has the following physical characteristics:

- Weight - approximately 20 pounds
- Volume - 6.3 inches diameter x 23 inches long
- Acceleration - 100 g's or less
- Storage characteristics - (1) approximately four weeks without standby electric power (electric power for ion pump to maintain vacuum for electron gun); (2) approximately six months with electric power (110 volts).
- Cost - approximately $20,000 each (prototypes).

A presented system is estimated to have the following characteristics:

- Weight - approximately 7 pounds

- Volume - 4 inches diameter x 20 inches long

- Acceleration - 100 g's or less

- Components - (1) substitution of a logarithmic amplifier for range switching of the photomultiplier amplifier to reduce complexity of the electronics; (2) use of a magnetically deflected electron beam to allow axial alignment of electron gun and associated focusing and accelerating electronics to reduce diameter; (3) elimination of cryogenic pumps (for the electron gun), if the system remains sealed (i.e., not operational) below 60 km.

- Cost - not likely to go below $1,500 each, as a result of cost reduction program and large production quantities.

Major cost reductions are dependent upon a low cost photomultiplier tube, logarithmic amplifier, 20 kilovolt power supply, storage batteries (28v, approximately 4 amps 1/2 hour).

In order for this system to operate synoptically, a means to prevent sunlight from exciting the photomultiplier tube must be provided—a sun sensor and a mechanism (either a electromechanical shutter or electronic switching of the photomultiplier).

This sensing system has a common complexity with the pitot system, i.e., the device must be evacuated and the vacuum maintained in the electron gun compartment below 10^{-5} torr (120 km) before flight.

The device itself is sensitive, both in the physical construction, i.e., filaments associated with the electron gun and the photomultiplier tube, and the precision of the measurements required, i.e., photomultiplier current. Alignment—optically and mechanically—must be precise. The potential requirement for a collection of the electron beam current may be difficult to achieve.

In addition, the composition of the gas being measured will affect the measurement; a question exists as to the applicability of this technique in the ionosphere.

5. Pitot Sensor System

The conventional pitot technique uses the difference in static and dynamic pressure experienced by a probe moving through a fluid to compute the relative velocity between probe and fluid. The pitot technique discussed here uses a pitot system to measure the dynamic pressure with an on-board gage. The probe velocity is measured with a ground-based radar and computes a static pressure trace as the probe passes through the atmosphere in its flight.

For sounding systems, in general, radioactive ionization gages, thermistor gages, and hot wire ionization gages are all possibilities for measuring chamber density or pressure. Additional instrumentation is also needed to measure air temperature within the tube and monitor the flight operational aspects.

For the measurement of atmospheric structure, the radioactive ionization gage has been flown frequently over the last decade. This gage, when accompanied by proper supporting instruments, has made accurate atmospheric structure measurements over the entire 30 to 100 km region. However, due to the necessity of carrying a radioactive device of long half-life handling, storage and political requirements of a worldwide system cannot be met and the device is not applicable to the present study. A thermistor gage which measures chamber density is available for use with pitot sensor sounding rockets, but as with all thermistor devices, its accuray is degraded at low pressures by errors caused by thermal conduction and radiation. Despite the higher-than-ambient pressures in the chamber, the pitot system with thermistor appears to have small potential above 70 km. (Below that altitude, less complex and comparatively less costly systems can be used.) A thermistor gage, therefore, is not proposed for use with the pitot system for this application. (References 46 through 50.

The one pitot device which appears to have the potential for development and which would permit an economical and accurate method for density measurement between 70 and 115 km (or more) is the hot-filament ionization gage. With this gage, the ram-compressed ambient gas is ionized by an electrically heated filament precisely located with respect to a cylindrical (or other configuration) electrode. The ion-current obtainable between filament and electrode is a function of the density in the chamber. A typical ion-current is 5×10^{-10} amp for 5×10^{-5} torr which is essentially linear to 10^{-5} amp for 0.5 torr. A high quality amplifier is used to convert this small current to a telemeter input. Because of the wide dynamic range inherent, switching is required. A tracker and a telemetry receiver on the ground provides the data for recording.

Although ionization gages have been used for many years, the hot-filament gage is a fairly recent development for the purpose of increasing sensitivity and altitude. It has been under flight development for two years and several have been flight tested. Most testing, however, has been in conjunction with a radioactive ionization gage for comparison and in-flight calibration. Considerable development remains to be done. The present development work is not aligned with the requirement of the synoptic system of this study, but mainly for maximum precision at higher altitudes.

Unlike the thermistor-pitot sensor, the pitot with a hot-filament gage is severely limited to minimum altitudes, due to filament burnout at high pressures. Seventy kilometers appears to be an absolute lower limit. The instrument is basically fragile and gun-launch is considered impossible.

Because of the use of different gages and flight operations, it is difficult to list "typical" characteristics. One set of characteristics, based on one version of a University of Michigan pitot system using a radioactive ionization gage, is given below.

Length - 6 feet

- Diameter - 3.5 inches diameter forward
  7 inches diameter aft

- Weight - 60 pounds

- Primary Gage - "Densatron" radioactive tritium ionization gage

- Other gages - Wall temperature thermistor (for gas temperature in chamber); Solar aspect system (measures sun angle to provide rocket angle of attack needed for data reduction)

- Tracking - DOVAP with on-board transponder

- Operational Considerations - Useful data acquirable only on up-leg portion of flight

- Resistance to g loading - Not suitable for use with guns

- Storability - Adequate

- Cost - as a single item produced by a research group and containing unseparable R&D costs, about $40,000.

Some typical values of pertinent characteristics for the postulated pitot system using a hot-wire filament gage are:

- Length - 33 inches

- Diameter - 4.5 inches maximum

- Weight - 16 pounds

- Primary Gage - Hot-filament gage

- Other Gages - Wall temperature sensor (type not yet specified)

- Tracking - DOVAP type with on-board transponder

- Operational Considerations - Useful data acquirable only on up-leg portion of flight

- Resistance to g loading - Not suitable for use with guns

- Storability - Adequate

- Cost (lots of 10^5) - \$1,500.

When working backwards from data, in order to establish atmospheric structure, many complications appear which degrade the precision of the results from pitot sensors. To infer the ambient pressure or density, it is necessary to know the velocity of the pitot gage relative to the longitudinal axis of the instrument. This requires fairly precise rocket tracking. It also requires fairly precise rocket attitude information and, if the trajectory angle is not near the vertical, fairly precise wind knowledge. Thermal effects within the gage must be taken into account and the position of the filament (for hot-filament ionization gages) must be precise.

The pitot system may well be inherently expensive. In any case, there has been little effort toward cost reduction, As a result, the system is, in fact, quite expensive in its present form, and extensive development would be needed to reduce costs if the pitot sensors were selected for the synoptic system of this study.

Future developments which might be effective in cost reduction are:

- Use a transponder with single commutated channel for tracking and telemetry instead of the present multi-channel telemeter

- Eliminate the use of an attitude sensing device (magnetometers or lunar/solar sensor) and trust to a prior knowledge of rocket attitude

- Use a low-quality amplifier rather than the best available

- Convert to a solid-state range switching.

It is noteworthy that all but the last item could lead to a degradation of data which might be substantial, yet the cost cuts are essential if the hot-filament gage version of the pitot sensor is to be cost-competitive. Thus, while the costs projected in the cost analysis presented elsewhere are realistic, there is skepticism on the part of many pitot gage investigators that the device which could be built for such costs would be sufficiently accurate to meet the study requirements.

Also of present concern are the requirements for pre-launch checkout. A vacuum system prevents filament destructions and a calibration of some type is required. Storage must be in a controlled environment and protection on the launcher in the form of an ejectable cap must be provided. A cost penalty in the form of excess rocket capability must also be paid, since apogee must be above 145 km for valid density measurements at 115 km (which are required for temperature at 100 km). All of these factors combine to make the cost versus accuracy picture crucial to the technique's applicability.

6. Thermistor and Parachute

The thermistor and parachute system is a widely used method of measuring both winds and temperatures in the upper atmosphere. (References 51 and 52) Temperatures are measured by a bead-type of thermistor, usually of .01-inch diameter and are telemetered to a ground-based receiver-recorder. The electrical properties (resistance) of the thermistor vary logarithmically with temperature. It is this variation of the thermistor properties which is recorded and compared with a calibration of the device to obtain temperatures. The bead is coated and shielded to reduce the influence of solar radiation. It is also thermally insulated from its support.

Wind velocity is measured from the motions of the parachute and thermistor during its descent. The measurements are made either by ground-based radar or by a telemetry receiver-transmitter, when a transponder is included in the launched payload. When radar is used, the parachute is aluminized to act as a reflector. Azimuth, elevation and range are recorded by conventional radar techniques. When the transponder and telemetry are used, the azimuth and elevation are recorded from the antenna-theodolite and range is determined from the transponder information.

The parachute is sized so that it provides a significant drag on the device. The payload, then, moves freely with the horizontal winds. Although the device can be ejected either during ascent or at peak altitude, it is commonly ejected at or near apogee. In most cases, a small explosive charge separates the device from the nose cone. The parachute is then inflated by ram air.

The thermistor has benefited from several years of study. Significant reductions in size and weight as well as increases in performance have occurred. Some typical, current values of pertinent characteristics of the thermistor and parachute are listed below:

- Parachute diameter (deployed) - 7-13 feet
- Parachute volume (packed) - 15-100 cubic inches
- Thermistor payload system volume (with radio and batteries) - 22-135 cubic inches
- Thermistor payload system weight (with radio and batteries) - 0.76-4.5 pounds
- Resistance to g loading - currently about 250 g's, appears possible to harden to $50,000 g's
- Storability - requires some environmental control
- Cost - varies \$100 - \$1,000 in small lots.

The values of pertinent characteristics for the thermistor and parachute system using a dart vehicle which is used in this study are:

- Parachute diameter - 8 feet
- Payload weight (including ejector, staves, piston, etc.) - 3 pounds
- Payload volume - 50 cubic inches
- Weight of parachute and thermistor - 1.1 pounds
- Resistance to g loading - 30,000 to 50,000 g's
- Storability - some environmental control
- Cost (10^5 lots) - $40.

No significant changes in these characteristics are anticipated. An anticipated change in telemetry frequency will tend to require an increase in size and weight. It is expected that a reasonable development program will be successful in reducing the size and weight to about current value by 1975.

The major remaining problems appear to be the influence of aerodynamic heating on the temperatures which are sensed and the accuracy of wind data which is derived from the device. These problems are discussed in the next section.

The parachute and thermistor are a well-developed combination of sensors which are capable of measuring temperature and winds from sea level to a maximum of 70 km. The payloads, including shielding, sensing devices, transmitter, and transponders, have undergone extensive development and, thus, are available in several sized packages for different vehicles. Ejection of the payload can be accomplished simply and the device appears amenable to hardening for gun launch. The device can be sealed for storage environments compatible with rocket or gun projectiles and can be relatively low-cost. On the other hand, little work has been done to data to accommodate the transmitter or transponder to the anticipated changes in the assigned meteorological frequencies. The total effect on payload size and weight is somewhat hard to determine, but with a reasonable development effort, will probably be negligible by 1975.

The major problems which affect the thermistor and parachute are related to the accuracy of the data which they provide. The estimated accuracy of wind measurements is about ± 10 mps at 70 km and ± 5 mps below 60 km (after correction for the fall from 70 km). The wind measurement accuracy is related to the fall rate; a high fall rate implies low accuracy. Since the fall rate is inversely proportioned to density, and since density decreases with altitude, the accuracy will tend to deteriorate as altitude increases. Extension to higher altitude requires a larger parachute or increased sensor sophistication. Currently, either technique rapidly reaches the point of diminishing returns; significant altitude increases do not appear feasible.

The same altitude limitation also applies to the thermistor. This device requires sufficient density to allow convection to predominate over radiation or conduction in the effect on the thermistor. Present thermistor results between 55 and 65 km are being questioned and a definitive study has established 70 km as an ultimate limit. (Reference 50)

The density also affects aerodynamic heating through its influence on fall rate. Aerodynamic heating is currently considered to be the main source of inaccuracy for the device. This problem does not appear amenable to simple corrections because of wide variations from launch-to-launch in the temperature and density in the upper atmosphere. This variation has been estimated to be greater than 30 percent between flights.

The current estimated temperature accuracy is about ± 2 percent below 60 km with appropriate corrections. Because of the combination of the usual rate of fall (greater than 100 mps) above 50 km and the thermistor time constant, the device cannot be used to obtain data on small-scale temperature variations.

7. Chaff

Chaff is any form of material which can fall comparatively slowly through the atmosphere, spread out into a relatively large "cloud" in order to provide a better radar target. There is also a necessity for packing or compressing the chaff into a small package for delivery to the desired altitude.

The forms of chaff which have been used include:

- Short lengths of small diameter wire
- Nylon chaff (.008 inches diameter)
- Glass chaff (.001 inches diameter).

At the higher altitude ranges being investigated, the mass/area ratio of the chaff should be lower. If a multiple ejection scheme is used, different configurations (chaff diameter) would be required at the different altitudes.

Chaff has been used as a payload with both rockets and gun-boosted vehicles. Although chaff was relatively popular about 5 years ago, its use has fallen off in this country. At the COSPAR XI Plenary Session in May 1968, the Russians reported considerable success in obtaining wind measurements with glass chaff at altitudes up to 90 km. They used multiple packets of 600 grams each in order to overcome the disadvantage of rapid dispersion which results in a poor radar target. They also used a continuous spiral scan observing at the same time all of chaff clouds simultaneously. (References 52, 53, and 54)

Some typical characteristics of the Russian operations are summarized below:

- Material - aluminized glass fiber
- Fiber diameter - 45 microns (.002 inches)

- Fiber length - 5 cm (1.97 inches)
- Weight per packet - 600 grams plus pyrotechnic, timing, and ejection devices
- Total payload weight for 3 packets - 1800 grams plus pyrotechnic, timing, and ejection devices
- Type of scanning - spiral.

Some typical values of pertinent characteristics for the postulated chaff payload to be used are:

- Material - aluminized glass fiber
- Fiber diameter - 45 mk (.002 inches minimum)
- Fiber length - 5 cm
- Weight per packet - 454 grams including pyrotechnic, timing, and ejection devices
- Total payload weight for 5 packets - 2,270 grams (5 pounds)
- Type of scanning - continuous scanning using phased-array radar.

For altitudes above 70 km, the only feasible wind measurement techniques are chaff, meteor trails, smoke/vapor trails, and possibly spheres. Smoke and vapor trails are not synoptic. Meteor trails give relatively long-time and volume average wind and require a totally different type of radar than that which might be required for sounding system payload. While 90 km seems to be the limit for present chaff applications, it is believed that the maximum altitude can be pushed to 100 km.

Chaff-derived wind data is ordinarily presented without stated accuracy limits. This appears to be due to the impossibility of "calibrating" chaff and the inability to define what portion of the cloud is being tracked. At lower altitudes, chaff falls faster than parachutes; therefore, chaff is preferable at lower altitudes.

4. PAYLOAD CANDIDATES

The sensors described previously have been grouped into six payload systems, whose launch vehicle requirements are similar. These six candidates are shown in Table II-3. Candidate 1 is the Passive Sphere with the minimum weight and volume requirement of any payload system. The altitude requirements for this candidate system are approximately 140 kilometers and it is not launch acceleration-limited. Candidate 2 is the Transponder Sphere which has the altitude requirements, no acceleration limits, and approximately twice the weight of the Passive Sphere. Candidate 3 is the Passive Sphere plus one chaff package to measure winds above 70 kilometers. It is approximately the same weight and volume as Candidate 2. Candidate 4, incorporating the Spinning Wire Densitometer, weighs approximately 15 pounds, has a volume of 245 cubic inches and is not acceleration-limited. Candidate 5 is of the same weight, but 435 cubic inches volume, is acceleration-limited. Candidate 4 requires a minimum altitude of 120 kilometers, while

Table II-3
Basic Payload Candidates

No.	Minimum Diameter (Inches)	Description	Altitude (km)	Weight (lbs)	Volume (in^3)	Acceleration-Limit
1	1.5	Passive Sphere	125-140	3	40	None
2	2.5	Transponding Sphere	125-140	6	120	None
3	1.5	Passive Sphere & Chaff	125-140	7	90	None
4	4.0	Spinning Wire (SWD) Thermistor & Chaff	120	15	245	None
5	4.0	MFD, Thermistor Chaff	100	16	435	100-150 g's
6	4.5	Pitot, Thermistor & Chaff	120-145	25	660	100-150 g's

5 does not require altitudes in excess of 100 km. Payload Candidate 6, with the pitot probe, is the heaviest (25 pounds) and has the greatest volume (660 inches). It is acceleration-limited to approximately 150 g's and has a minimum altitude requirement of 120 kilometers.

III. LAUNCH VEHICLE ANALYSIS

III. LAUNCH VEHICLE ANALYSIS

1. THE CONCEPTS EXAMINED

The numerous launch systems which exist represent, to a great extent, the results of the different approaches to meeting the problems of altitude, low-level drag, wind-at-launch, and, of course, cost. For example, rockets have been carried aloft by balloons and airplanes to minimize the altitude requirement. Thrust versus time rocket firing patterns have been specially tailored to minimize drag effects. Gun-boosted vehicles and rocket-launched "darts" have been designed with small frontal areas to minimize drag. Wind-at-launch constraints may make it desirable to examine boost concepts (such as catapults) which might, otherwise, be of less interest, and may affect the launcher design and the thrust pattern of concepts picked for further study.

A number of concepts with some apparent potential were identified and examined in a systematic manner. First, in order to minimize the probability of omitting possible candidates, propulsion methods were categorized according to energy source and according to conversion type (as rocket, gun, gun-boosted, etc.). Then the resulting launch concepts were measured against the applicable mission specifications, item by item, until the launch concept was either rejected, or determined as suitable for detailed optimization and comparison.

Launch concepts representing the many launch systems are shown and defined in Table III-1. Note that many systems based on these concepts are, or have been, used for sounding work.

2. LAUNCH CONCEPTS REJECTED

Balloon and airplane-borne rockets were rejected on the basis of complexity and cost. Gun-boosted ramjets were rejected on the basis of development status and probable complexity. Catapult performance is not commensurate with size, weight, complexity and cost. The direct use of satellites is not consistent with the 30-100 km altitude range; satellite-borne probes would be too costly a solution.

(1) Balloon-Borne Rockets

1. Performance Characteristics

An example of this concept might be a rocket-borne aloft, suspended from a 30 to 50-foot diameter, polyfilm,

Table III-1
Launch Concepts

CONCEPT	DEFINITION
Rejected	
Balloon-Borne Rocket	A single or multi-stage rocket which is initially lifted to an intermediate altitude by a gas-filled balloon.
Airplane-Borne Rocket	A single or multi-stage rocket which is initially lifted to an intermediate altitude by a conventional airplane.
Gun-Boosted Ramjet	An airbreathing, ramjet powered, payload carrying projectile which receives its initial momentum from a gun.
Catapult-Boosted Rocket	A single or multi-stage rocket which is initially accelerated by a catapult device.
Satellite and Satellite-Borne Probes	A satellite used either as an in situ (low altitude) device or to carry re-entering probes.
Require Further Consideration	
Single-Stage Rocket	A single-stage, solid or liquid rocket with either single or dual thrust level (Includes dart-type vehicles).
Multi-Stage Rocket	A rocket with more than one thrusting stage.
Gun-Boosted Flight Vehicle	A payload carrying projectile which is accelerated to sufficient velocity by a gun to attain the desired altitude.
Gun-Boosted Rocket	A single or multi-stage rocket which is initially accelerated by a gun.

helium-filled balloon. At an altitude of about 20,000 to 30,000 feet, the rocket is fired upward through the balloon. Balloon-launched rockets have been used in several experiments to launch small payload-carrying rockets. (Reference 55)

2. Advantages and Disadvantages

This approach decreases the size of the rocket required by decreasing the amount of altitude to be covered, and also by reducing drag losses (since the balloon rises above the denser part of the atmosphere).

Balloon-launched rockets have two major drawbacks. The first is a stability problem—the rocket tends to swing (oscillate) under the balloon. The second problem is the distance that the balloon drifts during its ascent to 30,000 feet.

In a typical wind profile, the balloon will drift more than 50 km during ascent (30 meters/sec ascent velocity). In order to control the launch so that essentially the same trajectory (or at least, apogee) occurs for each firing, an elaborate control system would have to be carried on the balloon.

3. Conclusions

The requirement for an elaborate control system outweighs any advantages. The concept has, therefore, been rejected.

(2) Airplane-Borne Rockets

1. Performance Characteristics

In the aircraft-borne rocket concept, the rocket is carried to an intermediate altitude by a conventional aircraft (Reference 56). Launch from the aircraft is usually accomplished during a pull-up or climb. In the current application, the rocket could, for example, be carried by a light airplane to an altitude of 15,000 to 25,000 feet.

2. Advantages and Disadvantages

As in the case of the balloon, the energy imparted to the launch vehicle by the aircraft allows the use of a smaller, less expensive rocket. However, aircraft operations are severely restricted by adverse weather. In order to meet the visibility and rainfall specifications, sophisticated electronic equipment is required on the aircraft and at the airport, for take-offs and landings. Such electronic equipment is now undergoing development, but its operation and maintenance is not compatible with the personnel specifications of this study.

3. Conclusions

This concept has been rejected due to the relative complexities and cost associated with its operation.

(3) Gun-Boosted Ramjet

1. Performance Characteristics

In this concept, a gun is used initially to accelerate a ramjet-powered vehicle. The ramjet is then ignited and propels the projectile to a velocity sufficient to reach 100 km. The fuel is usually assumed to be a liquid, although this is not necessarily true. As with a conventional rocket, the vehicle could move essentially vertically through the atmosphere. The ramjet might be wrapped around a center body which could contain both fuel and payload; tail-fins could be placed on the wrap-around shroud.

2. Advantages and Disadvantages

The specific impulse of a JP-4 fueled ramjet is 5 to 10 times that of a conventional rocket (Reference 57) On this basis, the amount of fuel required would be less than for a rocket and no oxidizer is carried. The vehicle fuel cost, as well as tankage weight, of this concept would be significantly less than those of a rocket.

On the other hand, a ramjet is a poor accelerator at Mach numbers less than about 1.5. It would, therefore, have to be accelerated to about that velocity by a gun or rocket. The incoming air flow changes rapidly as the altitude and velocity change, requiring corresponding changes in the fuel flow rate by factors of between 1.2 and 3 (Reference 58), requiring some sort of control system such as proportional valving (Reference 64). In any case, the system as a whole is not developed although some development was accomplished in the early 1960's.

3. Conclusions

This concept has been rejected because it is doubtful that the system can be state of the art by 1974 and because of the attendant probability of a complex, high cost system not compatible with mission specifications.

(4) Catapult-Boosted Rockets

1. Performance Characteristics

A catapult is a device which uses some type of energy to accelerate one of its components, which, in turn, pulls or pushes the vehicle to be boosted. Hydraulic, explosive, and steam catapults have all been used for boosting aircraft to relatively low velocities using fairly low accelerations.

Catapults were examined as part of an effort to maintain completeness of coverage. For the same reason gas and even spring actuated catapults were examined briefly. The possibility that the wind-at-launch specification might imply significant velocities prior to launch release was also a factor in examining catapults.

2. Advantages and Disadvantages

Catapults might be considered safer than the direct use of chemical explosives in guns and launch tubes.

Catapults require components, such as valves, actuators, accumulators, etc., which lead to size, weight, complexity and cost. The requirements on catapult design of a high velocity boost are excessive.

3. Conclusions

This concept has been rejected due to size, weight, complexity and cost not commensurate with performance.

(5) Satellite and Satellite-Borne Probes

1. Performance Characteristics

Direct measurements by sensors on satellites is not practical considering the 30-100 km. altitude requirements for making measurements.

A remaining possibility would be the use of probes fired downward from orbiting satellites.

2. Advantages and Disadvantages

Satellites offer the potential of serving a number of data acquisition sites, while requiring only a single launch site. Such a single launch site could alter the personnel requirements derived from the implicitly assumed requirement of multiple launch sites. The fact that probe trajectories would not normally be vertical does not contraindicate the concept; consider the use of "GHOST" balloons.

Each satellite would need to carry a number of probes. Even so, the specification that "Launch should be possible from all worldwide sites at essentially the same time" would require the existence of a number of satellites in orbit at the same time.

3. Conclusions

This concept has been rejected on the basis that it would be excessively expensive.

The previous analysis grouped the acceptable launch vehicle candidates into the following four categories:

- Rockets (single and multiple stages)
- Rocket-boosted darts
- Gun-boosted rockets
- Gun-launched projectiles.

This grouping is ordered according to maximum acceleration levels encountered. The design capabilities; limitations and estimated costs of these vehicle families are presented in this section of the report.

The sensors described in Chapter II are grouped into six basic payload packages, each of which has its unique launch vehicle requirements. Launch vehicle candidates compatible with each payload have been sized. The falling mass hazard, which is potentially so important for a worldwide synoptic sounding system, is discussed from a launch vehicle standpoint. The payloads which present a significant falling mass hazard, already have provision for parachutes to reduce the impact velocity.

A brief vehicle cost summary is presented with a detailed system cost analysis given in Chapter VI. Finally, conclusions with respect to launch vehicle payload compatibility are given in the last section.

(6) Single-Stage Rockets

Single-stage rockets are in use in a variety of sounding systems. Several types can be identified, representing, to a considerable extent, different approaches to the problem of minimizing the requirements for extra propellant (and cost) in overcoming low-level, aerodynamic drag.

The first type uses relatively slow-burning propellant (or low-fuel flow) and, after burnout, remains attached to the payload until apogee. (The slow burn keeps the velocity and drag relatively low through the dense atmosphere in the early portion of the trajectory.) This type of exemplified by the ARCAS vehicle. A further refinement in this approach is the dual-thrust, single-stage rocket, a somewhat newer type. (Reference 59) Basically, it is similar to the slow-burning rocket, except that two thrust levels are provided by a single rocket motor.

The third type of rocket uses fast-burning propellant, but the flight vehicle separates from the rocket case at burnout and coasts to apogee, unhindered by the relatively high-drag rocket case. (The flight vehicle is much smaller in frontal area than the rocket, and shaped to minimize drag.) This type of system is exemplified by the "Dart," systems such as the Loki-Dart and Judi-Dart. (Reference 60).

A number of studies of the single-stage concept for various sounding rocket missions are available. (References 61 through 67.) These studies consider solid propellants, pre-packaged liquid propellants, and hybrids (either liquid fuel or liquid oxidizer). These studies also illustrate several of the major advantages of the single-stage concept:

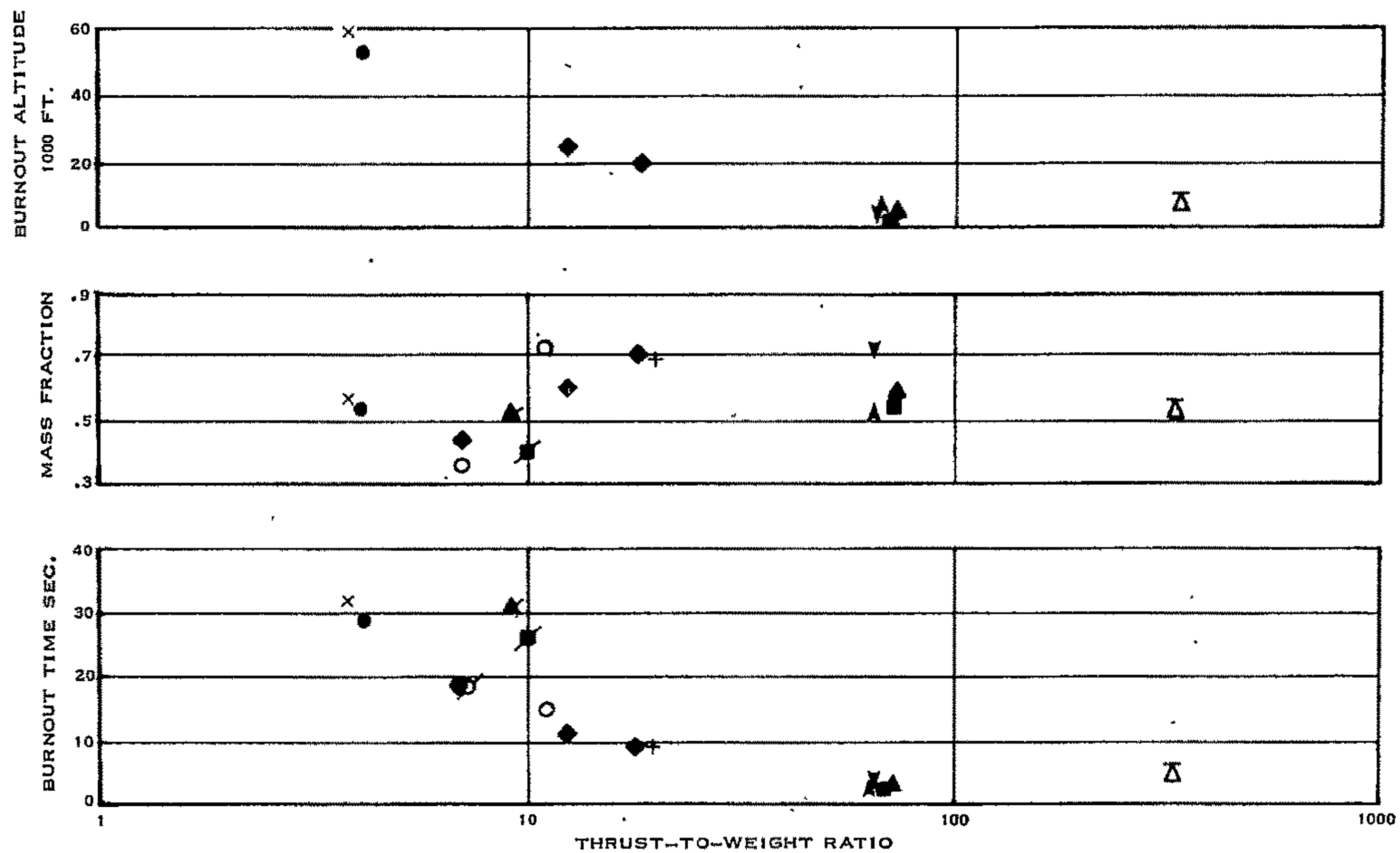
- Significant design and operational experience
- Single, package launch vehicle
- Relatively simple launch operations.

Experience in the design and operation of rockets as part of instrumented sounding systems dates to the late 1940's. Currently, the state of the art has progressed to the point where the entire launch vehicle can be prepared at the factory.

As far as the launch vehicle is concerned, field operations need not consist of much more than placing the vehicle on the launcher, making relatively simple electrical circuit tests, and firing.

Typical characteristics of a number of single-stage sounding rockets are presented in Figure III-1. The figure presents variations of burnout altitude, burn time, and mass fraction as a function of initial thrust-to-weight ratio. The data indicate that normal lift-off thrust-to-weight ratios for current single-stage rockets which separate near apogee are on the order of 5 to 10. Burn times are on the order of 20 to 30 seconds. Despite the possibility of higher mass fractions, the typical value for vehicles such as this fall in the range between .5 and .7. Initial weights of 100 to 300 pounds may be expected, depending on the choice of propellant. Burnout altitudes are on the order of 30,000 to 50,000 feet.

Data for a number of Dart-type vehicles are also presented in Figure III-1. The thrust-to-weight ratio of these vehicles is on the order of 100-150. Burn times are 1 to 3 seconds and burnout occurs between 5,000 and 10,000 feet. Weights and mass fraction are about the same as those for the



NOTE: Each symbol represents the characteristics of a current Single-Stage Vehicle

FIGURE III-1. Effect of Thrust-to-Weight Ratio on Some Single-Stage Sounding Rocket Design Parameters type of vehicle which separates near apogee. Typical data for a dual-thrust, single-stage rocket are also included in Figure III-1. (These data were obtained from Reference 59.) The characteristics are very similar to those for the single-thrust level rockets.

Significant design differences exist between the single-stage rocket which separates near apogee and the Dart vehicles. The aerodynamic heating and structural loads on the Dart vehicle are significantly greater than those on the lower thrust-to-weight ratio vehicle, because the Dart achieves high velocity at low altitude. On the other hand, the short range of the booster rocket for the Dart causes it to fall near the launcher, thereby reducing the problems of falling mass dispersion.

A rough comparison of the cost of several types of single-stage sounding rockets is indicated in Table III-2, derived from several sources. For this table, the cost of payload, including instrumentation, was removed, so that comparisons can be made on the basis of propulsion costs alone.

Typically, the cost per launch of sounding rockets which achieve a 65 km altitude with a nominal 5 to 15 pound payload

Table III-2
Typical Cost per Launch of Single-Stage Sounding Rockets

Concept	Cost Per Vehicle	Basis	Reference
Loki-Dart	$ 500	100 units	62
Liquid - Separation at Apogee	$ 755*	5000 units	67
Solid - Separation at Apogee	$ 500-700	1000 units	59
Solid - Dual Thrust	$1300	1000 units	59
Solid - Separation at Burnout	$1320	1000 units	59
ARCAS	$1850	1000 units	62

* Rocket at conceptual design stage as of this report.

is on the order of $1,200 to $2,000 per launch. Note that the Judi-Dart carries a maximum of only 6 pounds, gross (Dart and payload) to 60 km.

The cost of $755 for a liquid rocket is within a range suggested by TRW for a rocket now under study which uses a new controllable thrust motor. A rocket for a comparable mission has not yet been completely designed but the motor has undergone proof-of-principle firing tests. (Reference 67)

Reliability information on typical single-stage vehicles of both the Dart and separation near apogee designs are included in Table III-3 (Reference 59). The reliability data indicates that the most common cause of failure is the lack of correct expulsion of the payload when apogee is achieved. Reasons for this failure include both manufacturing problems—payload mating—as well as expeller design problems.

The characteristics of some solid and liquid rocket propellants which might be used in some of the rockets described are shown in Table III-4 (Reference 59). The data are representative, considering both the mission and specification that the vehicle must be exportable. (Advanced, classified

Table III-3
Typical Reliability Characteristics of Single-Stage Rockets

Concept	Function			
	Stage Firing	Aerodynamic Surfaces	Payload Separation from Motor	Payload Explusion from Nosecone
Separation Near Apogee (Solid)	.967	.985	.995	.950
Separation at Burnout (Solid)	.967	.970	.995	.943
Dual Thrust Level (Solid)	.960	.985	.995	.950

Table III-4
Typical Propellant Characteristics

Propellant	Density (lbs/ in.^{3})	Delivered Specific Impulse (sec)	Burn Rate (in./sec.)
Arcite 402	0.0640	232	0.40*
Arcite 368	0.0620	237	2.60*
Arcite 373	0.0640	242	1.90*
TRX-G415	0.0624	240	0.277
TRX-H609	0.0636	236	0.340
TP-G3014A	0.0625	235	0.270
TP-H1001	0.0638	237	0.320
ANP-2864HG	0.0637	243.7	0.365
ANP-2862JM	0.0635	243.2	0.28
ANP-2803HG	0.0635	251	0.31
ANP-2716 HL	0.0619	241	0.285
ANP-2805HY	0.625	240	0.34
DDP-80	0.0644	250	1.00
CY1	0.0635	247	0.55
EJC	0.0654	254.5	0.62
EFR	0.0658	252.5	0.78
PFG_s	0.0682	-----	.60
RDS-501	0.0630	247	0.33
RDS-502	0.0650	247	0.40
RDS-504	0.0630	245	0.65
RDS-505	0.0666	248	0.32
LPC-547	0.0628	244	0.87
LPC-549	0.0636	247	0.32
LPC-1003A	0.0616	246	0.86
LPC-1005A	0.0628	250	0.47
LPC-1008A	0.0670	255	0.40
$CLF_3 + N_2H_4$	0.0442	251	N/A
$N_2O_4 + N_2H_4$	0.0432	249	N/A
RFNA + N_2H_4	0.0442	240	N/A
N_2O_4 + UDMH	0.0435	250	N/A

*Silver-wire imbedded in grain
Sea-Level conditions and chamber pressure = 1000 psia

propellants cannot be considered since they are not, in general, exportable.) Typically, delivered specific impulses might range from 220 to 280 seconds, propellant densities from 0.06 to 0.07 pounds per cubic inch. Burn rates might vary from 0.26 to 2.60 inches per second for solids.

The wind-at-launch specification is of more interest for the single-stage rocket that for any of the other candidates since it could have the most effect on the single-stage rocket. For example, a wind blowing down or up range could change the angle of the rocket and cause an increase or decrease in apogee altitude.

The various analyses of the effect of wind on different vehicles differ in their results. An analysis of several designs indicates the need for a vehicle velocity of 500 to 600 feet/second before leaving the guidance of the launcher (Reference 59). Such a requirement could have serious effects on launch vehicle design. Other analyses of the effect of wind on the Tomahawk could lead to the conclusion that zero-length guidance might be quite adequate (References 68 and 69.)

Some of the differences could be due to different assumptions on wind profiles above the ground (Reference 5), some to differences in vehicle characteristics. Setting the intial boost characteristics (travel on, and velocity off the launcher) must proceed as part of the process of setting an optimum rocket design for the specific mission.

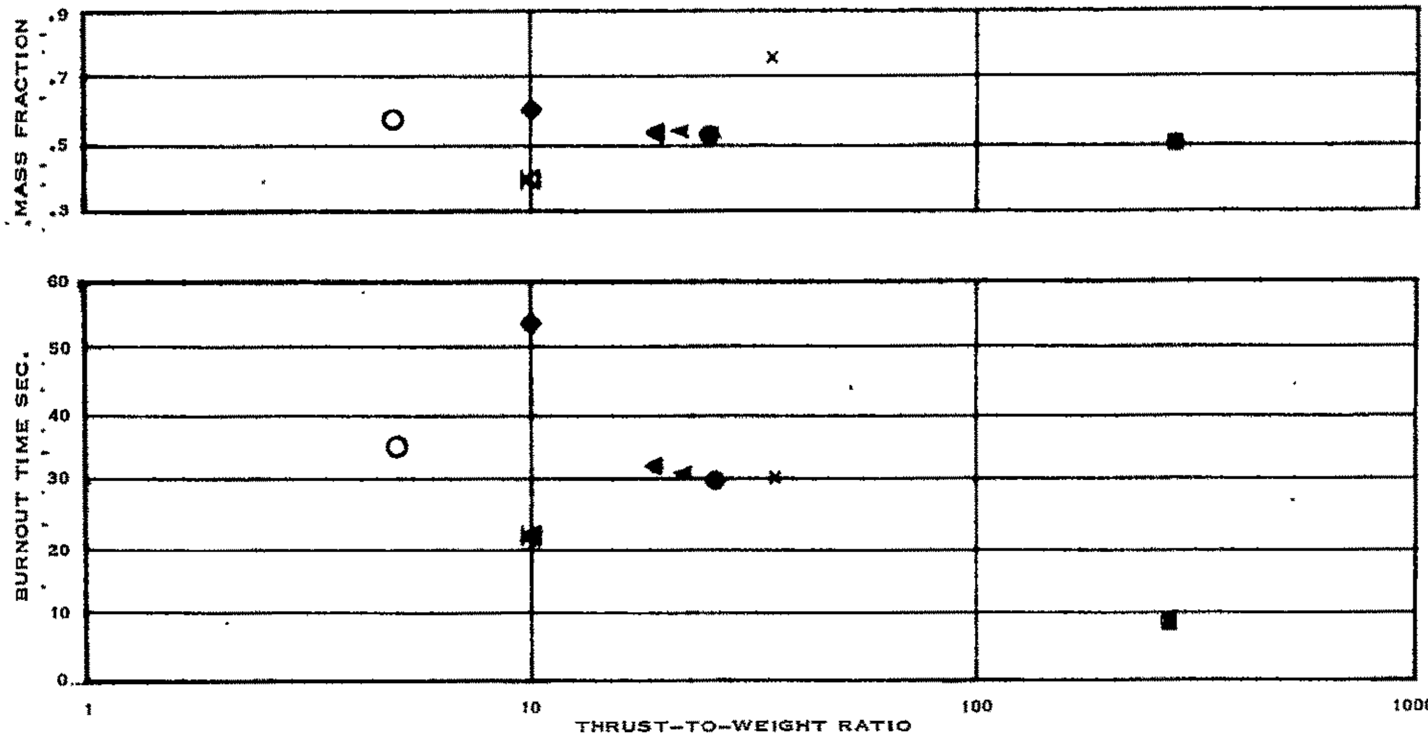
In conclusion, the single-stage sounding rocket system offers a number of advantages and must be considered a candidate, including all the variations of launch concept and type of propellant.

(7) Multi-Stage Rockets

As with single-stage rockets, a number of multi-stage rockets are in use for sounding systems. Typically, these systems contain two stages, with each stage having a different thrust level. Functionally, the first motor provides initial acceleration to the vehicle. After separation of the first stage, the second (or sustainer stage) continues to supply thrust and acceleration until the desired velocity is achieved. The types of propellants which are used include (as with single-stage rockets) solids, hybrids, pre-packaged liquids, and solid-liquid combinations.

A number of current studies (References 59, 63, 64, and 66) have been conducted for sounding systems of this type. The advantage of the multi-stage rocket, as compared with most single-stage rockets, is that its thrust versus time firing pattern can be made more efficient. As with single-stage systems, there is an extensive background of experience, and the rockets are not particularly dangerous or difficult to handle. For handling and shipping, a one-package, or at most, a two-package approach can usually be used.

Typical design data for two-stage sounding rockets are presented in Figure III-2 in which the second stage thrust-to-weight ratio, total burn time, and total mass fraction, are presented as functions of initial thrust-to-weight ratio. Typically, the two-stage rockets appear to fall in classes where the first stage has a 20 to 1 thrust-to-weight ratio and the second stage has a 3 to 1 thrust-to-weight ratio or where the initial thrust-to-weight ratio is about 10 to 1 and final thrust-to-weight ratio is about 3 to 1. Burn times for each stage are generally shorter than for the single-stage rocket but the total of the burn times for the two stages is



NOTE: Each symbol represents the characteristics of a current Two-Stage Vehicle.

FIGURE III-2. Effect of Initial Stage Thrust-to-Weight Ratio on Some Two-Stage Design Parameters usually longer. Weights of approximately 100 to 300 pounds can be expected with total mass fractions of about 0.5 to 0.7. Again, burnout altitudes are 30,000 to 50,000 feet.

The costs of a number of current and proposed two-stage sounding rockets are $1,460 per launch, based on 1,000 units (Reference 59). As in the case of the single-stage systems, typical costs are about $1,200 to $2,000 per launch for comparable missions.

Reliability data (Reference 59) for some typical vehicles are:

- Firing, either stage - .964
- Aerodynamic surfaces - .970
- Payload separation (from rocket) - .995
- Payload expulsion (from nose cone) - .950.

For this concept the most common cause of failure is payload expulsion.

As in the case of single-stage rockets, wind conditions could pose design constraints on the two-stage vehicle. The first stage, however, can be designed to provide the initial thrust. Drag, heating, and aerodynamic loads need not be severe if the first stage burns for 20 to 30 seconds at a thrust-to-weight ratio of from 3 to 7.

Clearly, the two-stage rocket concept can be designed to meet the wind requirement and offers several advantages as a launch vehicle.

(8) Gun-Launched Payload

High-performance guns are currently in use to launch meteorological probes on an experimental basis (Reference 70). In applications of this launch concept, a gun with a bore of 5 inches or greater accelerates a finned projectile to the velocity necessary to reach the desired altitude. The gun is smooth bored to prevent excessive spin and its attendant lateral accelerations. The projectile is much smaller in diameter than the bore to reduce aerodynamic drag. Sabots are used between the projectile and the gun to prevent gas leakage. These fall away after firing in the near vicinity of the launch site.

Currently, the systems consist of military hardware which has been modified for a meteorological application. Five-, seven-, eight-, and sixteen-inch guns are currently in use.

Typically, the barrel is extended by a distance equal to about 75 calibers to attain the desired performance. A number of studies (References 70 through 73) have been made of the concept.

The gun concept requires careful consideration, both because it is important by itself, and because there have been misunderstandings and conflicting claims as to its advantages and disadvantages. Several characteristics of guns which are not widely realized are:

- Payloads tend to be less weight-limited than volume-limited. (This makes it difficult to generate parametric comparisons of guns and rockets.)
- The extremely high accelerations experienced by the payloads are no real handicap for telemetry and impose only moderate difficulties for some types of sensors. (Some sensors, however, will be intrinsically unsuitable.)
- There is appreciable bore wear per shot, except for the 5-inch gun, for which methods have been implemented to reduce the problem.

Some of the gun's advantages and disadvantages are discussed below.

Advantages	Disadvantages
Guns are much more consistent and accurate in their trajectory.	They have a certain lack of flexibility both as to operations and as to payload. (Scheduled sequency of loads is now required, for example.)
Because of their resulting, very small impact area (about 3 miles diameter for the 7-inch gun) they are much safer than rockets when fired near inhabited areas.	Because of their extremely loud noise at firing, their apparent danger and actual nuisance value are much higher than for rockets. (A great deal could probably be done to muffle this noise.)
They seem to have a present cost advantage of a couple of hundred dollars per shot.	This advantage may not last. In any case, it requires a certain minimum number of shots per gun if the higher capital cost is to be amortized. It is not true when compared to Darts.
High wind environments have little effect on the projectile (on the order of 0.3 miles in a 3-mile dispersion region).	The present use of powder in bags can be affected by temperature variations and rain.

Payload and altitude performance characteristics of some typical gun systems are presented in Figure III-3. (References 70, 72, and 76) Curves are presented for 5-, 7-, 8-, and 16-inch guns. The 5-inch gun is not capable of carrying a 10-pound payload to 100 km. The 16-inch gun, on the other hand, has a capacity far in excess of the requirements. The 7- and 8-inch guns, then, are the candidates for this mission.

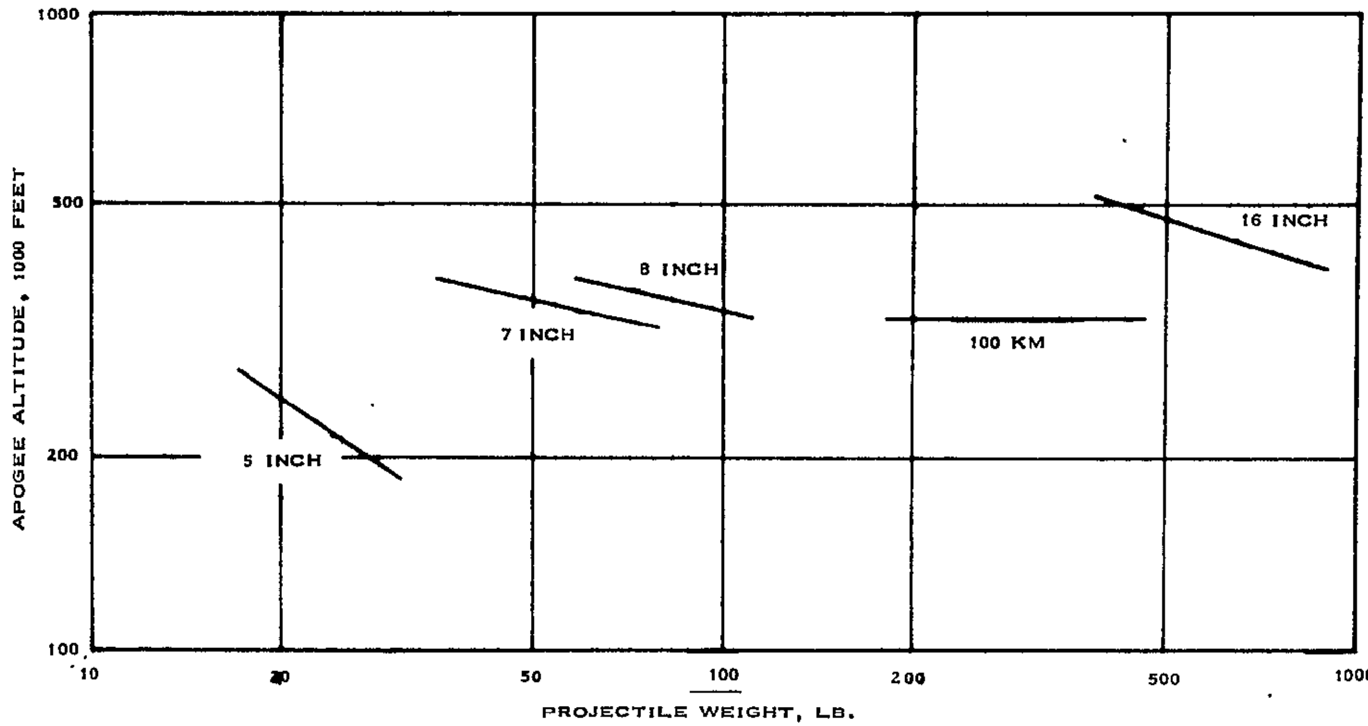


FIGURE III-3. Effect of Projectile Weight on Apogee Altitude for Typical Gun Systems

Figure III-4 presents the cost per launch of a 7-inch gun probe as a function of number of launches. The data for this curve were obtained from Reference 70, and includes a $64, 000 of initial costs, $10, 200 of other costs requiring amortization, and $550 for projectile and powder (but not including sensors or telemetry).

Based on an estimated total of 1, 000 launches (over 10 years) from each site, the estimated cost per launch is $600 to $700. Compared with current rocket launch costs, this range of launch costs represents a significant savings, except when compared with Darts, or the projected costs of the to-be-developed TRW liquid fueled rocket concept.

The dispersion of the gun-launched probes is quite low relative to typical rocket vehicles. As indicated in Figure III-5, the hits can be contained within a three to four mile diameter circle, as opposed to 20 or 30 miles for a typical rocket vehicle. Wind has a negligible effect (.3 miles) on the gun vehicle's trajectory. (Reference 70)

Operationally, several steps are involved in launch. The primary one involves insertion of the projectile into the barrel

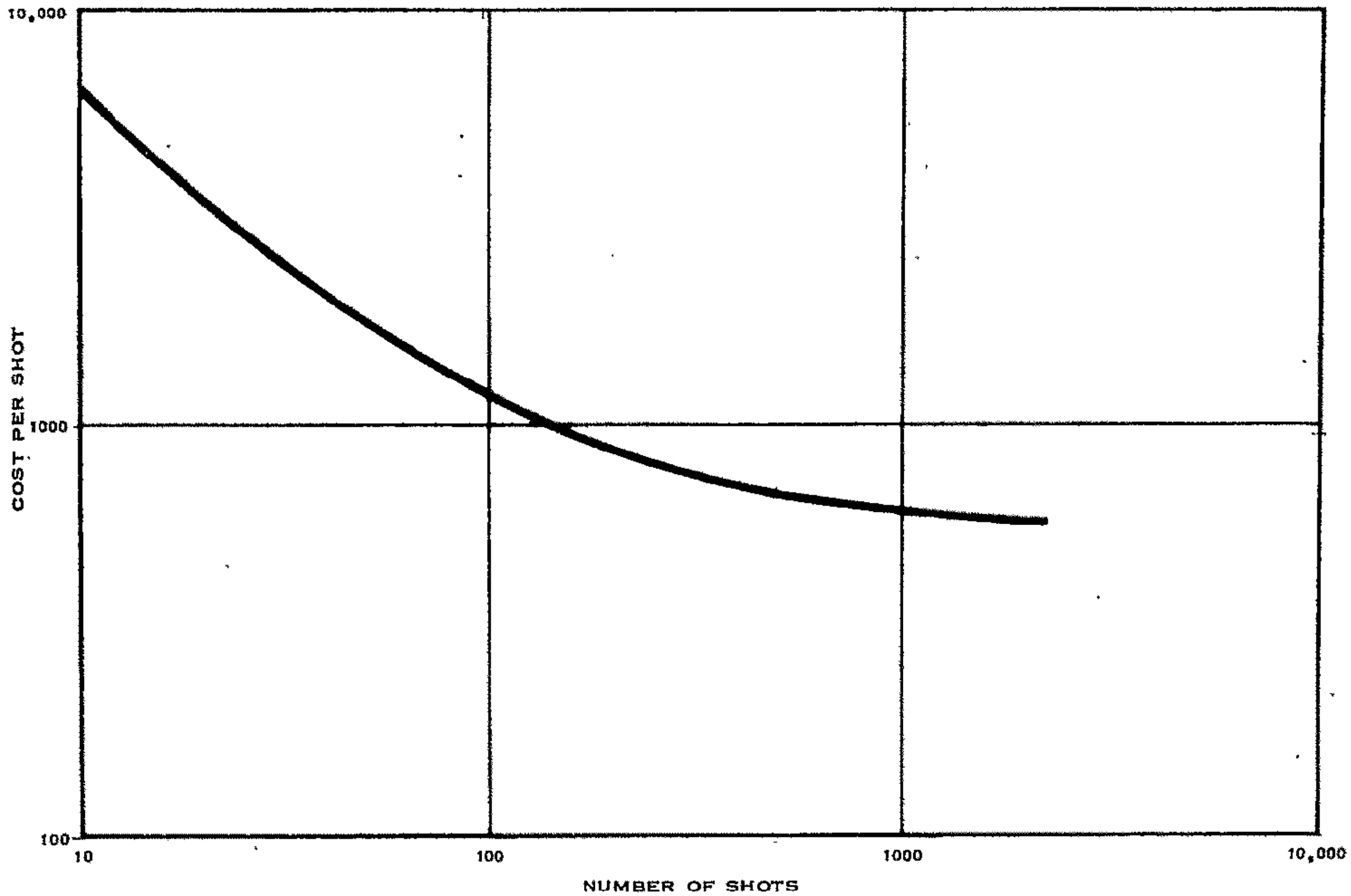


FIGURE III-4. Cost per Round of Firing 7-inch Guns As a Function of Total Number of Rounds Fired (Not Including Cost of Sensors or Telemetry)

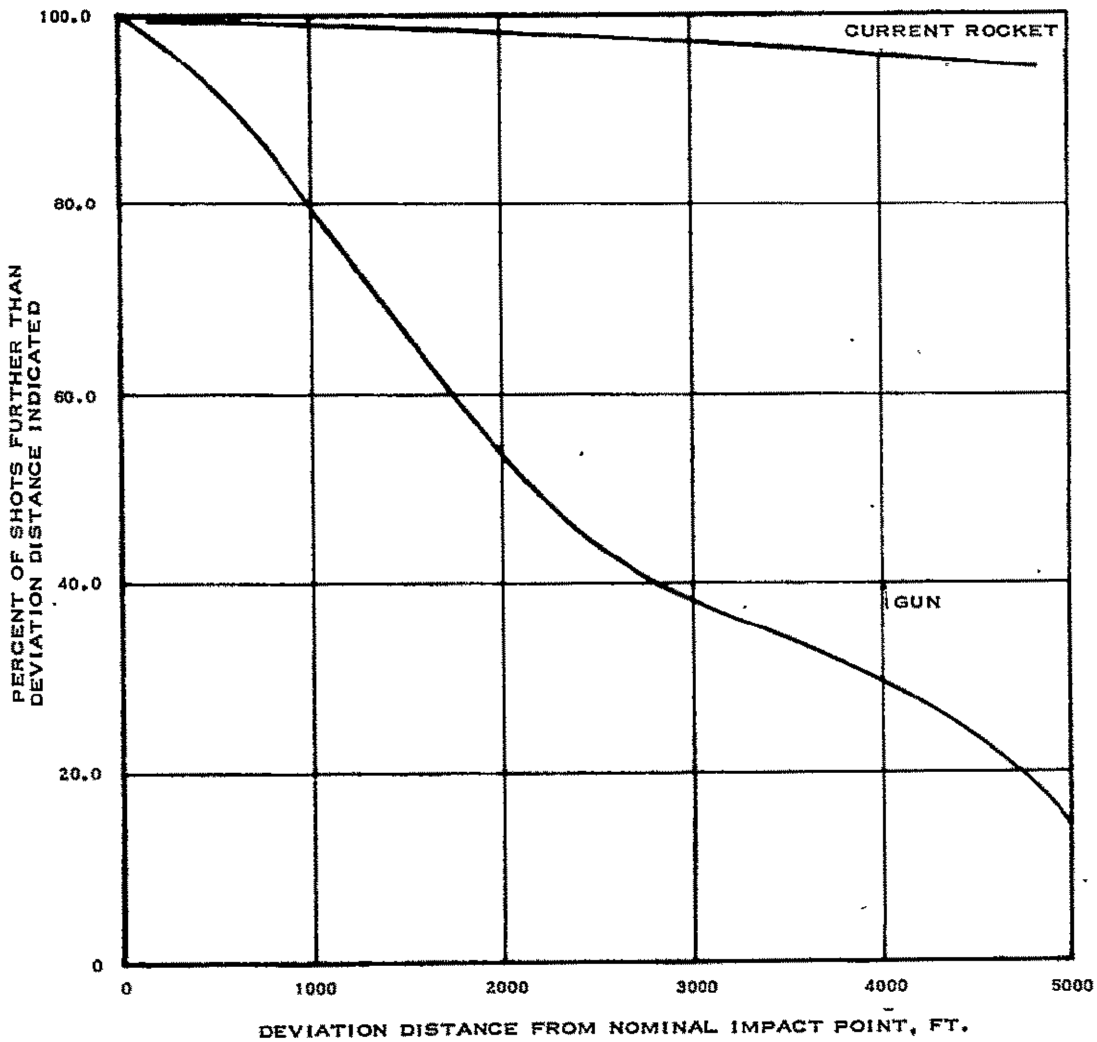


FIGURE III-5. Dispersion of Impact Point of Gun-Launched Vehicles with a powered ram. Loading is not particularly difficult, but does require the assistance of a crane, on site, to lift the loading device in current applications (Reference 74).

The firing reliability of the gun-launch concept is satisfactory. However, two specific problems remain. The first of these is a functional problem with the pyrotechnic timer—payload separation currently occurs at $\pm$ 40 seconds of desired time. The second, somewhat more serious problem, involves barrel erosion. Currently, the 7-inch barrel must be rebored after 30 shots. Techniques which solved the same problem with the 5-inch gun have not been successful with the 7-inch gun.

A number of factors influence the erosion problem; they primarily appear to revolve around the high pressures and temperatures that occur during firing. In designing for this mission, it may be advisable to use a larger bore diameter and lower pressures. Finally it appears that the propellant normally contained in silk bags is very sensitive to changes in ambient temperature or to humidity.

In their current configurations, the guns' major payload related restriction is the available volume and diameter. For the 7-inch gun, the payload volume is about 60 cubic inches and the diameter is about two to three inches. By way of reference, a typical rocket's payload volume is 370 cubic inches with a 4.25 inch diameter (Reference 75). The current volume and diameter restrictions may be amenable to design improvements.

Even in the current state of development, the gun launch concept has a number of apparent advantages. The current problems appear amenable to design or development solutions and a cost advantage is currently apparent. For these reasons, the gun concept continues to be a candidate system.

(9) Gun-Launched Rockets

The use of chemical explosive charges to accelerate sounding rockets has been widely accepted. Basically, two methods of employing the concept have been advanced. The first of these, typified by the ARCAS system (Reference 75) amounts to the addition of a small charge to a relatively simple launcher. The charge is ignited at essentially the same time as the rocket and provides an additional initial impulse to the vehicle. Nominal launch velocities of 200 to 300 feet per second are currently achieved. The method may be applied to the entire spectrum of propellants and stage combinations.

The second method, typified by the HARP Project (References 70, 71 and 76), involves the addition of a rocket motor to what is essentially an artillery projectile. The rocket motor is ignited after the projectile leaves the launcher. Muzzle velocities of 1,000 to 3,000 feet per second is typical. In addition to the 16-inch HARP guns, rocket boost is being widely applied to a variety of projectiles (References 73 and 76). Usually, with this method, the rocket is a solid propellant, single-stage device because of the high accelerations and the lack of need for multiple stages.

Some performance characteristics of gun-launched rockets are presented in Figure III-6. The figure indicates the variation of change, weight, maximum pressure, and peak acceleration with muzzle velocity for an 80-pound projectile in a 7-inch gun. The projectile weight was picked as being typical of the required projectile weight (References 67 and 76). Pressure and acceleration tend to increase logarithmically with muzzle velocity.

A major advantage of gun-launched rocket concept is that a significant muzzle velocity can be achieved with relatively low accelerations of a vehicle, as compared with a gun

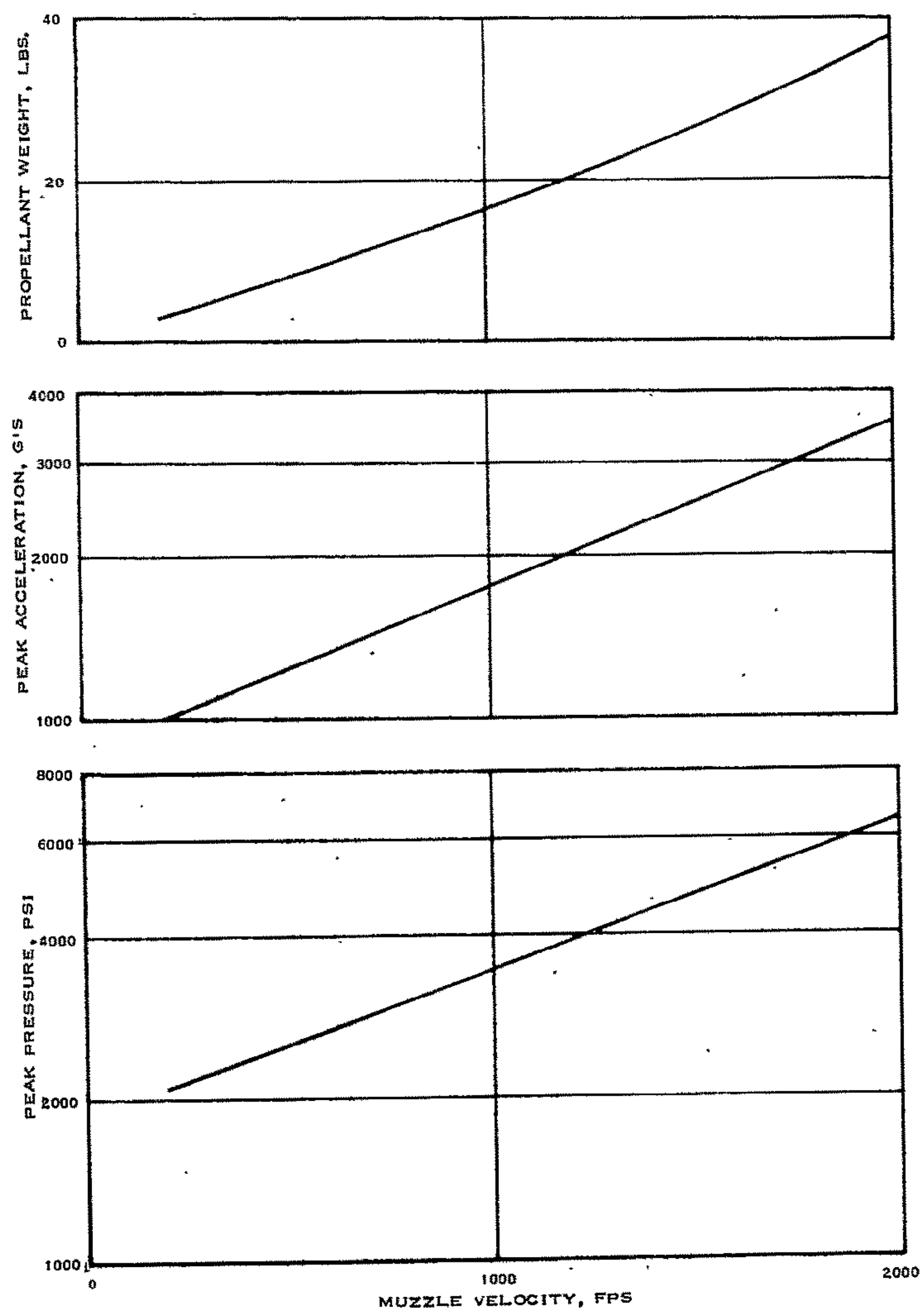


FIGURE III-6. Effect of Muzzle Velocity on Some Interior Ballistics Parameters projectile (on the order of 600 - 6,000 g's), which can attain 100 km and is not particularly sensitive to surface wind conditions. The costs of the ARCAS-type (low-velocity boost) system are essentially the same as for a single-stage rocket. Some difference in development cost might be expected, but these are probably not significant.

The artillery concept undoubtedly will be more expensive than the pure gun-launch payload due to the addition of the rocke which has been hardened for high launch accelerations. Cost estimates for the application of the artillery boost method to this mission have not yet been made due to the early state of development of the concept.

The dispersion of gun-launched rocket is, in all cases, greater than that of a pure gun due to small misalignments of the rocket motor, fins, etc. (Reference 76). However, the concept is capable of mitigating the wind effects on the trajectory (References 59, 70, and 76).

The primary reliability consideration with this concept is the structural integrity and ignition of the first stage of the rocket. The problem is not acute in the ARCAS-type system, in which the rocket and charge fire simultaneously. However, the accelerations imposed by the artillery boost concept pose a difficult design problem for the rocket motor and igniter. Solutions are available (References 67, 70, and 76), but careful development will probably be necessary.

Lockheed Propulsion Company has done considerable development work on full caliber gun-launched solid-fueled rockets using an external liquid jacket and a liquid-filled interior to withstand the high launch accelerations. The exterior liquid encased in a frangible container serves as an efficient obturator and eliminates the need for a tight-fitting sabot. Gun barrel wear is also significantly reduced. The primary advantage, however, of this concept is to permit higher mass fraction rockets to be utilized. This advantage can be substantial for large vehicles, but is lost for the small vehicles under consideration. Thus, the weight comparison presented previously showed little difference in the gross weights of the rocket and the gun-launched rocket. The primary features, then, of the gun-launched rocket are high launch velocity, minimum dispersion and moderately high (500-5,000 g's) accelerations.

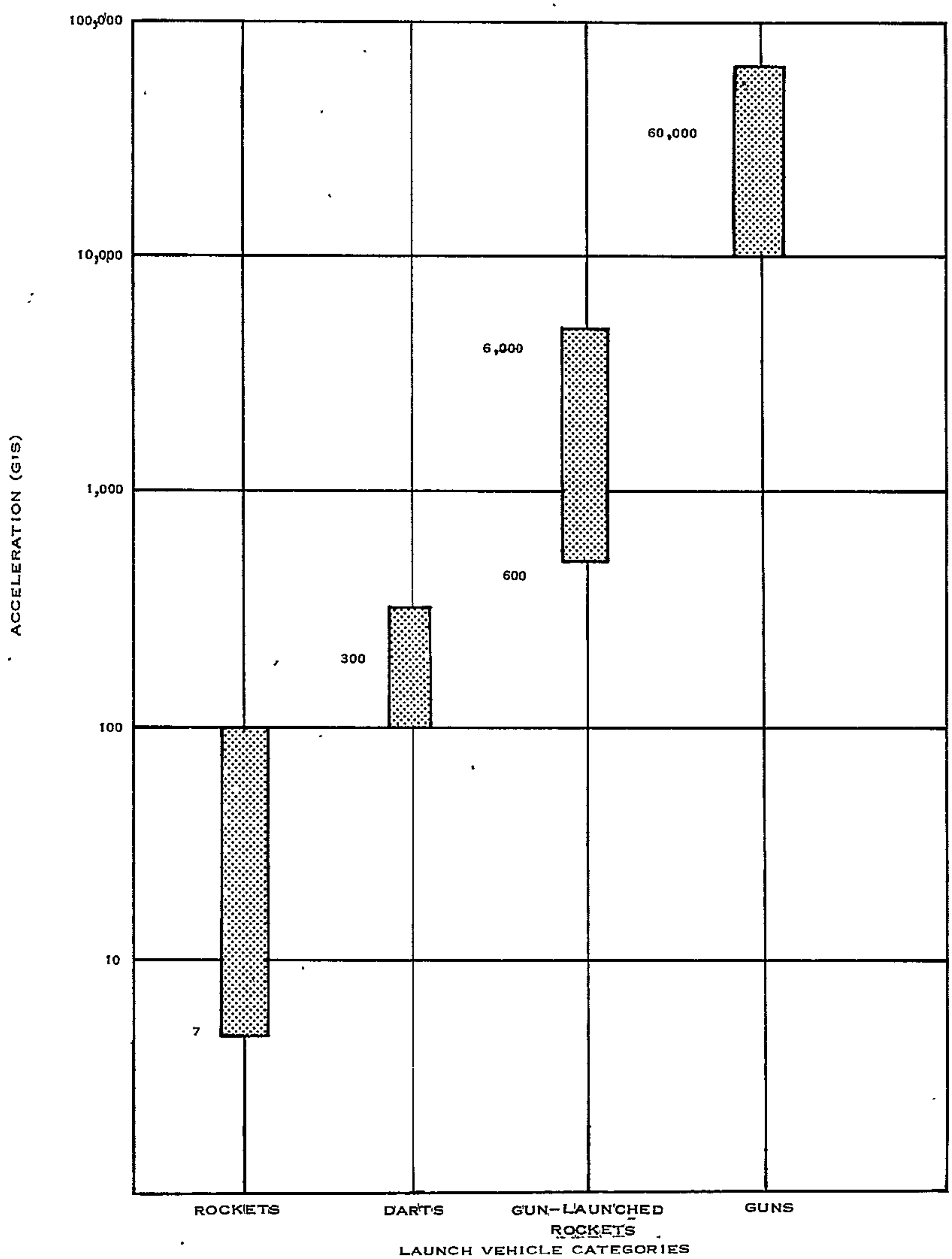
3. LAUNCH VEHICLE CHARACTERISTICS

(1) Acceleration Levels

Figure III-7 shows the range of accelerations expected to be encountered by each of the launch vehicle candidates. The pure rocket vehicle with constant thrust typically has a low-launch acceleration (5-10 g's) and a burning time of 20-30 seconds to achieve maximum altitude. Maximum acceleration occurs at motor burnout. Typical values of vehicle mass ratio are 3 to 4, yielding a burnout acceleration of 3 to 4 g. If the payload is small enough to permit its being packaged in a nose cone significantly smaller in diameter than the booster, then a dart vehicle becomes advantageous to minimize the total drag velocity losses. The booster burns for several seconds (typically 2-4) and then separates (at 5,000 - 10,000 feet) allowing the low-drag, high ballistic coefficient dart to coast to apogee. Acceleration levels are substantially higher than the pure rocket, ranging from 100 to 300 g's for dart designs to achieve altitudes of 75 to 140 kilometers.

Figure III-8 shows the general bounds of available payload volume as a function of vehicle acceleration level. The ordinate

FIGURE III-7
Acceleration Characteristics of Launch Vehicle Categories



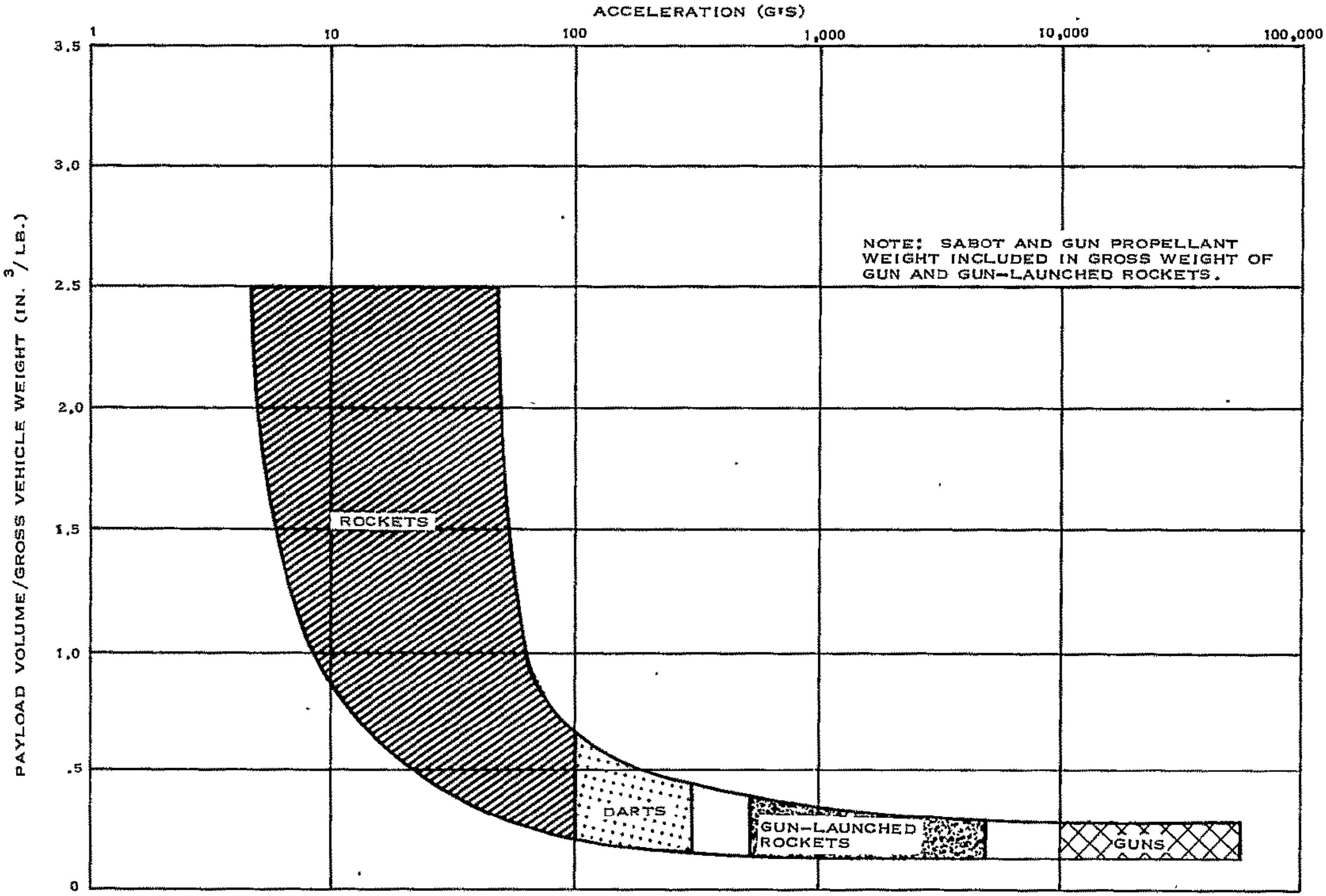


FIGURE III-8
Payload Volume per Unit Vehicle Weight

is given as payload volume (in.^{3}) per pound of vehicle gross weight. Rocket systems provide up to 2.50 in.^{3} per pound of vehicle gross weight while gun-launched projectiles only provide about .25 in.^{3} per pound of gross weight. Note, however, that for consistency in comparison, the gross weight of the gun projectile includes the weight of the sabot and propellant charge, neither of which is a part of the flight vehicle. Both the gun-launched rocket and dart vehicles are also substantially volume-limited—the dart because its diameter must be smaller than that of the booster, and the gun-launched rocket for the same reasons as the projectile. Thus, if payload volume requirements are large, the rocket can be expected to show a considerable gross weight advantage.

The gun-launched rocket uses a gun barrel to contribute from 1,000 to 3,000 feet/second velocity to a rocket vehicle, allowing the rocket flight vehicle weight to be substantially reduced. However, because the velocity supplied by the gun during the short time the vehicle is in the gun barrel, resulting acceleration is considerably higher than the rocket/dart vehicle and ranges from about 600 to 6,000 g's, depending upon the fraction of the total velocity which is supplied by the gun. Finally, the pure gun system must impart the total required velocity to its projectile while in the gun barrel and accelerations of 10,000 to 60,000 g's can be encountered.

(2) Payload Weight and Volume Capabilities

1. Rockets

The payload capability of pure rocket vehicles can be described by the payload α defined as follows:

$$\alpha = \frac{\text{Rocket gross weight}}{\text{Payload net weight}}$$

This payload characteristic for the pure rocket vehicles as a function of altitude as shown in Figure III-9 ranges between 15 and 20 for the altitude regions of concern for this study. Therefore, the minimum payload weight of 3 pounds would require a rocket vehicle of approximatel; 50 to 60 pounds while the largest payload of 25 pounds would require a rocket vehicle in the 400 to 500 pound gross weight class.

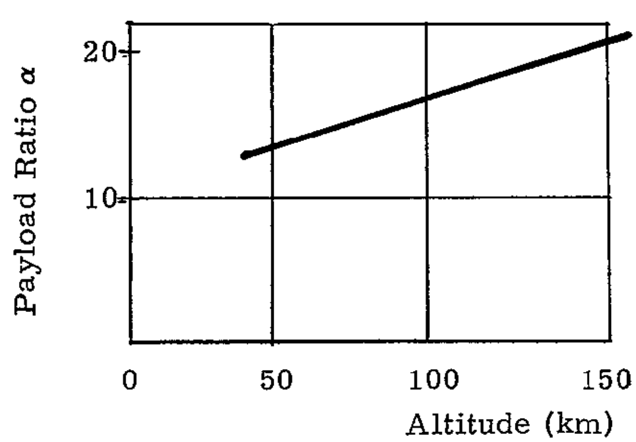


FIGURE III-9
Rocket Payload Ratio as a Function of Altitude

2. Darts

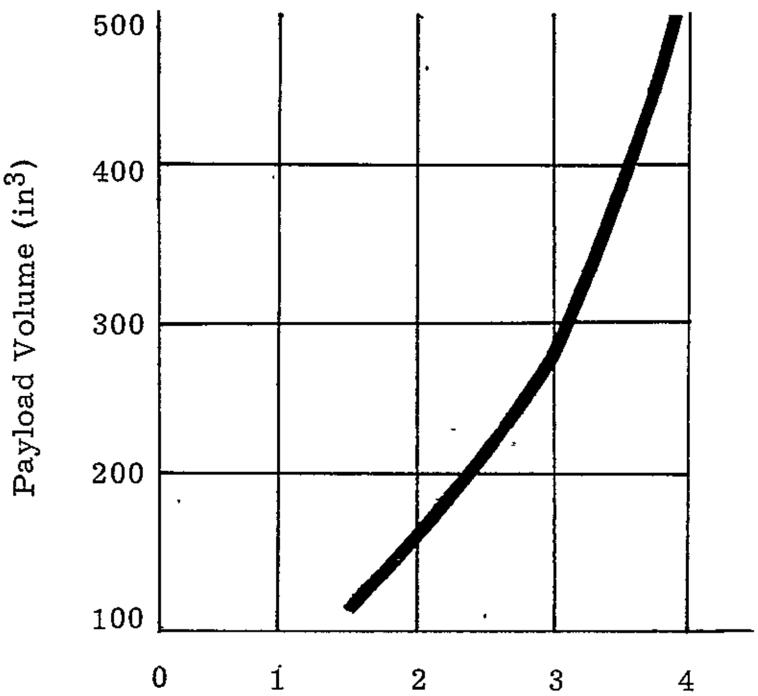
The dart vehicle is volume-limited because of the desire to minimize the dart diameter and the atmospheric drag losses. Darts currently in use for chaff and balloon payloads range in diameter from 1-7/16 inches to 2 inches. Assuming that the darts will maintain a constant length to diameter ratio, the payload volume available is proportional to the cube of the diameter—the constant of proportionality being about 7-1/2 to 8. Thus, the payload volume available as a function of dart diameter is approximately as shown in Figure III-10. Because it is necessary that darts have a high ballistic coefficient (or weight to drag ratio) they also have a high density and generally require ballast in addition to the payload. Thus, the payload weight is not a limiting factor in the selection of the minimum dart diameter.

3. Gun-Boosted Rocket

The gun-boosted rocket can also be described by a payload ratio which is considerably lower than the corresponding payload ratio for the pure rocket vehicle because several thousand feet-per-second of velocity is provided by the propellant in the gun barrel.

The approximate payload ratio characteristic of the gun-launched rocket as a function of altitude is shown in Figure III-11. This payload ratio is approximately 4 to 5 in the altitude region of concern. Thus the maximum payload weight of 25 pounds would require a rocket flight vehicle weight of approximately 100 to 125 pounds. Such a vehicle could be designed for a 5-inch or a 7-inch gun depending upon the payload volume requirements. In smaller gun diameters, payload volume available would be quite limited and consideration for this study has been primarily limited to the 7-inch gun.

FIGURE III-10
Dart Payload Volume Available



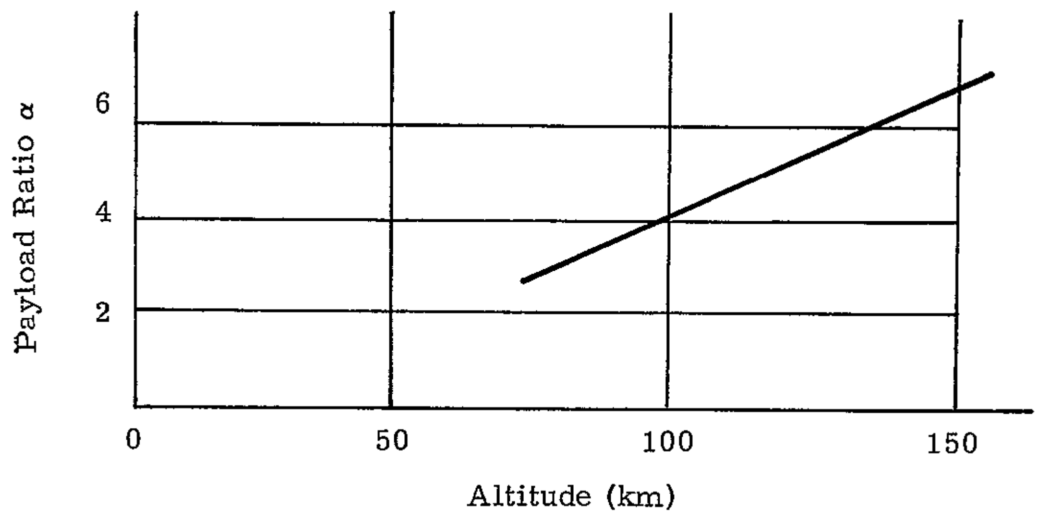


FIGURE III-11
Gun-boosted Rocket Payload Ratio as a Function of Altitude

4. Gun-Launched Projectile

To achieve the altitudes of interest, any gun-launched projectile must be a sub-caliber projectile with a sabot. Only in this way can sufficient acceleration be developed in the gun barrel to achieve the required muzzle velocities, and at the same time, keep the ballistic coefficient of the projectile within satisfactory limits. Thus the same volume limitations of the rocket-boosted dart are inherent in the gun-launched projectile. The volume ratio, α_d, appears to be between 1.0 and 1.1 pounds per cubic inch for sub-caliber projectiles launched from a 7-inch gun. This is relatively independent of altitude between 100 and 150 kilometers. The maximum payload volume available in a 7-inch gun projectile designed to reach 130 to 140 kilometers is 60 cubic inches, and with a payload ratio of approximately 1.1 pounds per cubic inch, this projectile weighs about 65 pounds.

Higher volume payloads could naturally be accommodated by making use of larger guns. Assuming that the projectiles maintain constant ratios of projectile area to gunbore area and length-to-diameter, the payload volume available is proportional to the cube of the bore diameter. Thus, the payload volume available, as a function of gun-bore diameter, is approximately as shown in Figure III-12.

4. DESCRIPTION OF CANDIDATE VEHICLES

Figure III-13 shows a matrix of basic payload-launch vehicle compatibility. Since the Molecular Fluorescence Densitometer and Pitot Sensors are acceleration-limited, they are not compatible with the gun-boosted rocket and gun systems. The 7-inch gun system, because it is extremely volume-limited, is capable of handling only the passive sphere payload. Finally, the gun-launched rocket, because of its relatively high cost is not considered for the first three payloads, where less-expensive gun, dart and rocket candidates are available.

Thus, the first screening of the 24 possible launch vehicle payload combinations reduces the candidates by 10. The remaining 14 have been carried through preliminary sizing and costing studies. Rocket vehicle sizing was done using the payload ratio characteristics discussed in Section 3 and similarity with specific vehicle designs provided by members of the rocket industry. Dart vehicle sizing was accomplished, assuming a constant value of $\frac{W}{C_D A}$ for the darts

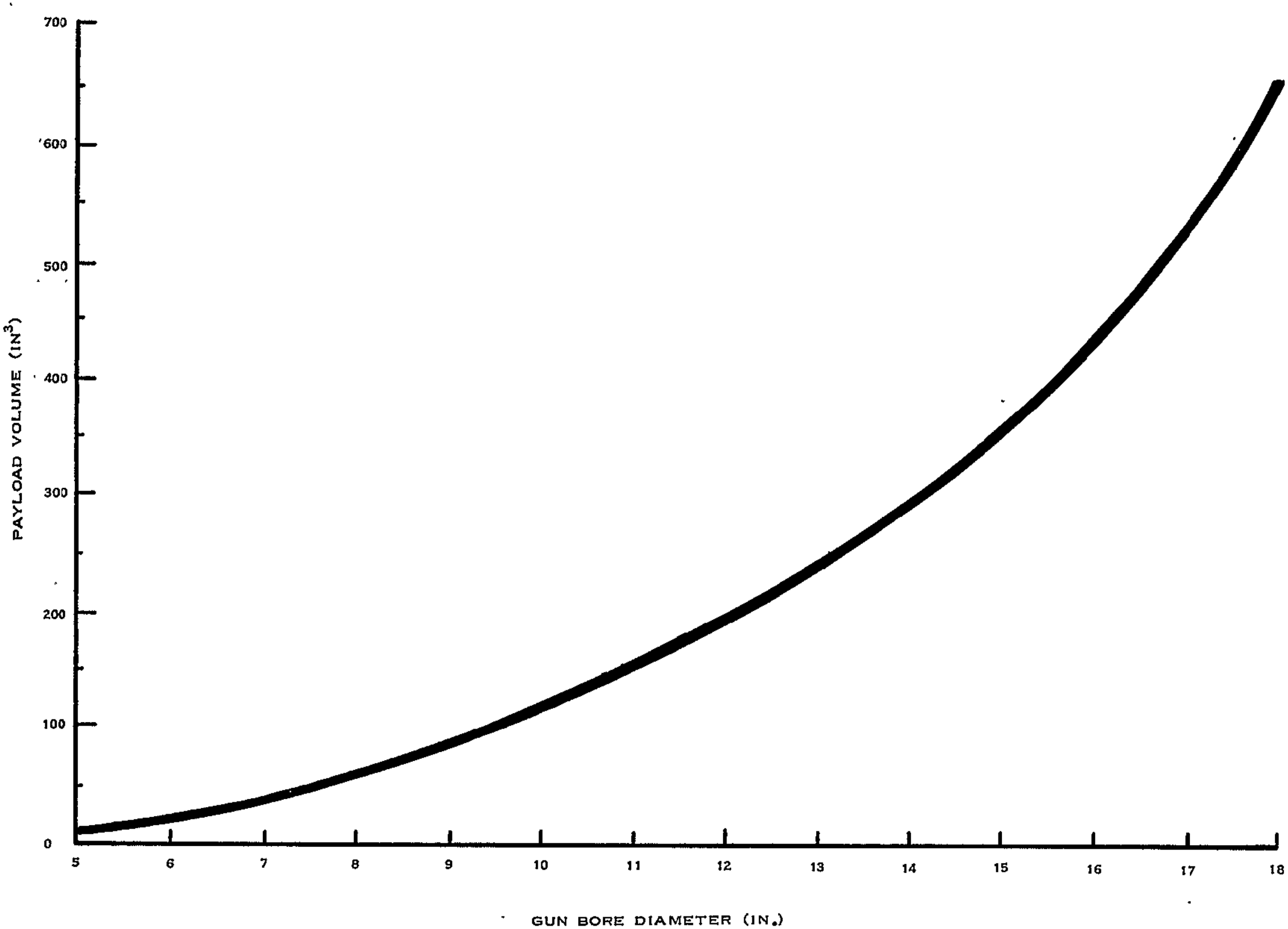


FIGURE III-12
Gun-launched Projectile Payload Volume Available

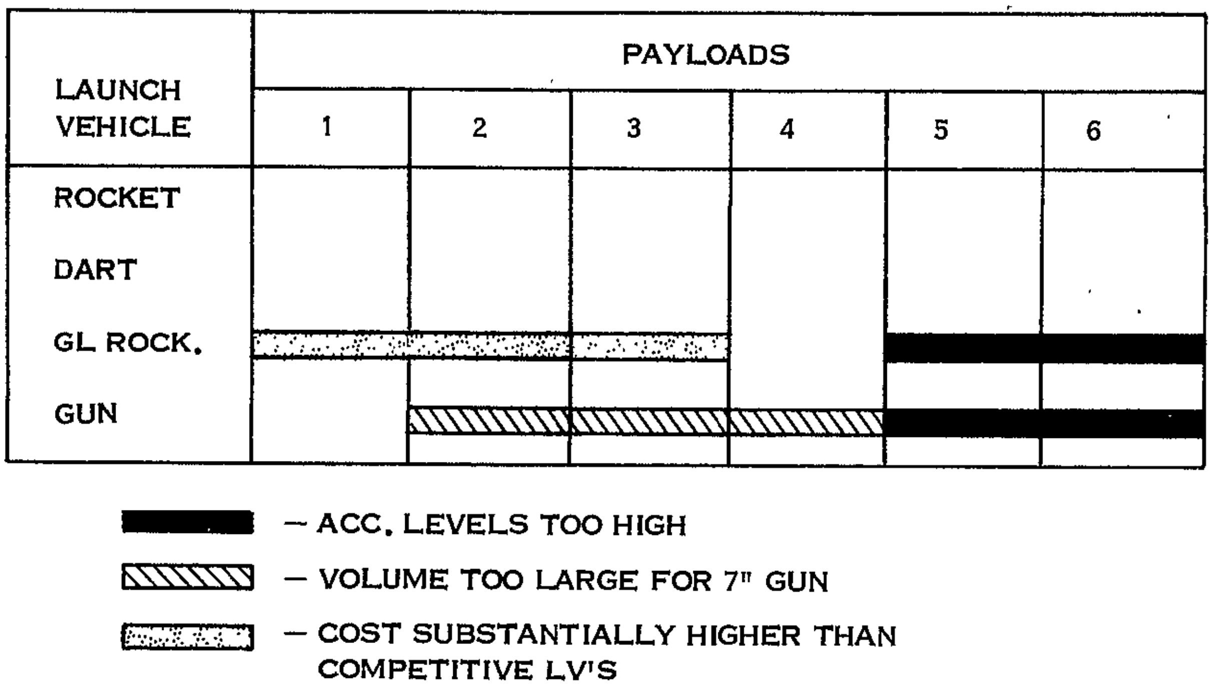


FIGURE III-13. Launch Vehicle-Payload Compatibility of 4,200 lb/ft^2 (the figure for the super Loki-dart). Assuming a booster burnout altitude of 5,000 feet, the dart velocity requirements to reach the prescribed altitude were determined using the following closed form equation relating velocity and altitude of an unpowered vertical probe previously developed (Reference 77).

$$S = \frac{V_o^2 \; e^{-\eta \rho_o}}{2g} + \frac{1}{\beta} \left\{ \eta \rho_o - \frac{(\eta \rho_o)^2}{2} \right\}$$

S = Altitude

V_o = Initial Velocity

ρ_o = Density at initial velocity

β = Density gradient coefficient ($\frac{1}{\beta}$ = 20,000 feet)

$\eta = \frac{C_D A}{W}$

C_D = Drag coefficient (average for dart)

W = Dart weight.

Table III-5 summarizes the size and weight characteristics of the remaining fourteen candidate systems. For payload 1,* the rocket

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* See Table II-3 for payload definitions.

Table III-5
Launch Vehicle—Payload Design Characteristics

	PAYLOAD CANDIDATES					
	Passive Sphere 1	Transponder Sphere 2	Passive Sphere & Chaff 3	SWD 4	MFD 5	Pitot 6
Payload Characteristics						
Weight (lbs)	5	6	8	14.8	15.8	25.5
Volume (in^3)	40	120	90	245	435	660
Diameter (in)	1.5	2.5	1.5	4.0	4.0	4.5
Altitude (km)	130	130	130	120	100	120
<u>ROCKETS</u>						
Total Vehicle weight (lbs)	80-100	120	160	300	250-275	450-500
Motor diameter (in)	4.5	5	5.25	6.75	6.5	8
Vehicle length (in)	96	120	96	160	155	190
<u>DARTS</u>						
Total weight (lbs)	100	320	160	550	475	650
Motor diameter (in)	4.75	6.25	6.75	8.25	7.75	8.75
Dart weight (lbs)	20	45	20	125	125	155
Dart diameter	2	3	2	5	5	5.5
Vehicle length (in)	140	210	140	310	300	340
<u>GL ROCKETS - 7" Gun</u>						
Flight Vehicle weight (lbs)	---	---	---	125	---	---
Rocket diameter (in)	---	---	---	6.5	---	---
Rocket length (in)	---	---	---	80	---	---
Powder weight (lb)	---	---	---	20-40	---	---
<u>GUN - 7" (3" projectile)</u>						
Projectile weight (lbs)	45	---	---	---	---	---
Sabot - weight (lbs)	20	---	---	---	---	---
Powder weight (lbs)	100	---	---	---	---	---
Projectile length (in)	55	---	---	---	---	---

and dart systems are approximately the same size, though it should be noted that the dart system weight is very sensitive to dart diameter. Payload 2 requires a volume of about 120 in.^{3} and Payload 3 about 90 in.^{3}. Since 2 has a 2-1/2 inch minimum diameter, a 3-inch dart would be required, and this in turn leads to a total launch vehicle weight of approximately 230 pounds. Payload 3 might be accommodated in a 2-inch dart with a reduction in vehicle size.

Payloads 4 (spinning wire densitometer) and 5 (molecular fluorescence densitometer) are both the same weight and have a 4-inch diameter requirement. A rocket system to carry this payload is estimated to weight 250-300 pounds, depending upon the altitude required, while the 5-inch dart required results in a dart system weight of 475-550 pounds. Thus, the diameter requirement results in the dart system being substantially heavier than the rocket.

Finally, candidate system 6 has a payload of 4-1/2 inches in diameter which requires a still larger dart resulting in a system weight of about 650 pounds, comparable to a rocket vehicle of 450-500 pounds.

The gun-launched rocket was considered for payload 4 (spinning wire densitometer) since this payload is not acceleration-limited.

Based on preliminary design data furnished by Lockheed Propulsion Company, it appears that a 7-inch caliber rocket vehicle, with a flight weight of about 125 pounds, would be capable of meeting the altitude requirements.

5. DISCUSSION OF THE FALLING MASS HAZARD

(1) Introduction

A problem of major concern for the worldwide synoptic sounding network is that of falling mass hazards associated with the launch vehicle and the payloads. Provision has been made in each of the candidate payloads for a parachute to return any hazardous payload to earth at a velocity of 15-20 ft/sec. It thus remains to determine how best to minimize the hazard for the various launch vehicle candidates. The simplest choice from a vehicle standpoint is to provide a range area for impact of the vehicles. This is the procedure that is used in rocket test ranges today. There are at least 25 such world-wide ranges today which use the Arcas vehicle, most of which fire over water. However, if the grid for a synoptic sounding system were to require several hundred sites it would not be practical to provide the required land or water impact zones to minimize the falling mass risk. Several development programs have been or are currently being carried out to develop alternatives for providing large range areas. These alternatives are discussed below.

(2) Alternatives

Four basic alternatives exist to minimize the falling mass hazard from synoptic sounding vehicles, including the possibility of providing a range impact area. These alternatives are:

- Range Impact
- Parachute Descent
- Frangible Vehicle
- Consumable Vehicle.

Table III-6 indicates the compatibility of these possible solutions with four candidate launch vehicles under study.

Typical impact area requirements for each of the candidate vehicles are shown in Table III-7. It can be seen that, even for the gun-launched projectile, the impact area requirements are probably larger than could be met in many worldwide sites.

Parachute recovery of any of the launch vehicle candidate systems requires considerable additional payload weight and

Table III-6
Compatibility of Launch Vehicles with Methods of Combating the Falling Mass Hazards

	VEHICLE						
	Rocket		Dart		Gun/Rocket		Gun Projectile
	SOLUTION						
	Motor	Nose Cone	Motor	Dart	Motor	Nose Cone	
Range Impact	x	x	x	x	x	x	x
Parachute Descent	x	x	x	x	x	x	x
Frangible Vehicle	x	x	x	x	?	?	
Consumable Vehicle	x		x		?		

Table III-7
Typical Impact Area Requirements

LAUNCH VEHICLE SYSTEM	$3\ \sigma$ DISPERSION* (NM)	IMPACT AREA** REQUIRED (NM^2)
Dart Booster	2-3	25
Dart	10-15	600
Rocket	12-20	1000
Projectile	4-6	100

*Radius
**Based upon acquisition of rectangular land mass.

volume to carry the parachute. Depending upon the payload chosen, a 6 to 20 pound parachute, requiring a packing volume of 260 to 800 cubic inches, would be required to land the combination payload and rocket vehicle at a velocity of one foot per second. The additional weight and volume requirements of the parachute would be reflected directly in the additional size and weight of the launch vehicle required. This additional weight detracts from the design goal of a minimum size launch vehicle for ease of handling.

Several studies have been sponsored by NASA and the Air Force in recent years to develop concepts for frangible and consumable launch vehicles. Several methods of fragmenting vehicles have been identified. The first requires the use of sheet explosive or shaped charges to fragment a paper or fiberglass vehicle case into pieces small enough so that they do not present a falling mass hazard. The technical feasibility of this method has been demonstrated. However, it does introduce an explosive hazard which requires additional evaluation.

A second method of fragmentation incorporates a very, high-strength, but brittle, ceramic developed by Corning Glass Works. A vehicle constructed of this material can be fractured or "diced" by use of an electrically actuated dimple motor. This method, if fabrication costs are not excessive, offers an attractive alternative to the explosive fragmentation method. An additional design concept using a wire wrap is being explored by Rocketdyne, but detailed data was not available.

Several methods for designing consumable cases have been proposed. One, proposed by Rocket Research Inc., uses an aluminum foam as the base for the solid propellant. The feasibility of this system has not yet been demonstrated, but it would offer an attractive alternative to the explosive fragmentation method. A second method utilizes a vehicle structure fabricated from a combustible nitro-cellulose material. The technical feasibility of this method has not as yet been successfully demonstrated.

(3) Discussion

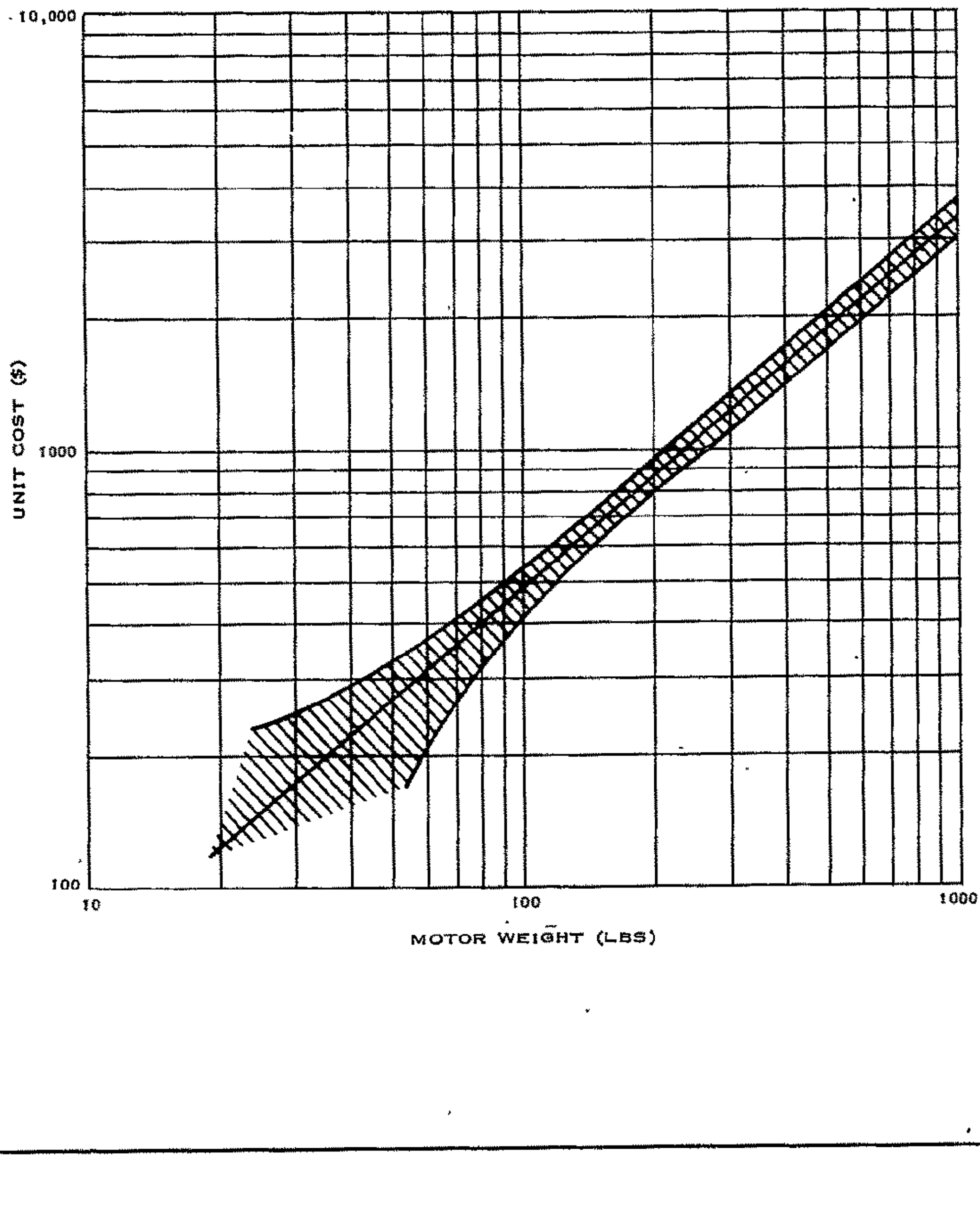
Of the above four proposed alternatives to the falling mass hazard, providing sufficient range impact area is by far the simplest and represents the minimum technological challenge. However, because of the high cost of real estate, it is unlikely that this solution would either be practical or possible. Of the other concepts, the frangible or consumable vehicle cases would be desirable, provided that the vehicle performance penalties are not too great and the cost penalty is acceptable. If neither of these methods should appear to be promising, then the use of a parachute to return the spent rocket case to earth would be required. An additional detailed study of these alternatives would be required before such a decision could be reached.

6. LAUNCH VEHICLE COST ANALYSIS

A detailed systems cost analysis is presented in Chapter VI. The major aspects of launch vehicle costing are summarized here to reflect the primary cost factors involved in the selection of the launch vehicle alone. The final selection of a candidate system (or systems) is dependent on overall cost considerations and trade-offs between flight vehicle and ground systems costs.

Figure III-14 shows the range of estimated costs as a function of weight for all types of rocket motors. This curve encompasses all motors reviewed for this study. It reflects a broad range of costs for the motors less than 100 pounds, where the dart vehicle boosters have shown a substantial cost advantage over longer burning time rocket motors. However, for motors of several hundred pounds up to about 1500 pounds, the spread narrows and projections for the costs of existing vehicles in the 10,000 per year quantities seem to agree quite well regardless of motor type.

FIGURE III-14
Estimate Rocket Motor Costs
in Quantities of 10,000 Per Year



Using Figure III-14 as a basis for motor costs, the rocket and dart vehicles, a range of launch vehicle costs (exclusive of payload) has been established. For the dart vehicles, the cost of the dart body must be added to the motor costs. Costs of the gun projectile are estimated, based on consultation with the Army Ballistic Research Laboratories and projections from current Army quantity procurements of artillery ammunition. Cost estimates for the gun-boosted rockets do not have a firm base. Our estimates are based on discussions with industry proponents of these vehicles, and our best efforts to extrapolate costs from conventional rocket vehicles to the vehicles designed for the high acceleration environment of the gun barrel.

The range of launch vehicle cost estimates is given in Table III-8.

Table III-8

Launch Vehicle Cost Summary

		LAUNCH VEHICLE COSTS		
PAYLOADS	LAUNCH VEHICLES	EXPENDABLE	NON-EXPEND.	TOTAL
1. P. SPHERE	ROCKET	500–600	—	500–600
	DART	350–550	—	350–550
	GUN	270–400	20–70	290–470
2. T. SPHERE	ROCKET	550	—	550
	DART	950–1000	—	950–1000
3. P. SPHERE & CHAFF	ROCKET	600	—	600
	DART	950–1000	—	950–1000
4. SWD	ROCKET	1200	—	900–1100
	DART	1800–2200	—	1800–2200
	GL ROCKET	1000–1800	20–50	1020–1850
5. MFD	ROCKET	900–1100	—	900–1100
	DART	1600–2000	—	1600–2000
6. PITOT	ROCKET	1700–1800	—	1700–1800
	DART	2200–2600	—	2200–2600

7. CONCLUSIONS

The following general conclusions are reached with respect to launch vehicle selection:

- For the passive sphere payload (#1), both the dart and gun-launched systems are cost-competitive. The gun system subjects the payload to higher acceleration and presents a greater falling mass hazard because of its high density.
- For payloads 2 and 3 (transponder sphere and passive sphere plus chaff) the rocket vehicle is lighter in weight than the dart, but a competitive procurement is recommended since dart vehicle costs in the past have been somewhat lower than competitive rocket systems.
- Rocket systems show a clear cost advantage for the three larger payloads. However, the gun-launched rocket has potential for the non-acceleration limited spinning wire densitometer payload, and it is recommended that design and cost studies be sponsored on this vehicle to establish its performance capabilities and cost.
- The falling mass hazard problem may be severe if the proposed launch site density is high (i.e., greater than 100 worldwide sites). Frangible and consumable vehicle concepts presently in development would then have to be adapted to the vehicle designs proposed here.

IV. TRACKING SYSTEM ANALYSIS

IV. TRACKING SYSTEM ANALYSIS

1. INTRODUCTION

Investigation of sensing techniques for atmospheric structure measurements has led to the conclusion that the passive falling sphere device is the lightest and least expensive satisfactory payload which can be foreseen. The attendant support equipment, which includes an expensive high-accuracy tracking radar appears incompatible with a synoptic sounding system. In the past, radars such as the FPS-16, FPQ-6, and TRADEX have been used successfully to track passive spheres, although none was designed for such a purpose. The specifications of sphere-tracking-radar parameters are distinctly different from, although largely compatible with, those for universal missile-tracking. The cost of large missile-tracking radars is well in excess of $1 million and ranges up to $30 million. Yet, not only are they less-than-optimum devices for sphere-tracking, in some regimes they even fail to meet the minimum requirements.

An investigation was conducted to define and cost a passive-sphere tracking radar system using phased-array technology and modern manufacturing and packaging techniques and a CW tracker to be used with a transponding sphere.

2. REQUIREMENTS

The basic tracking problem is to develop a time/position trace of the sphere during its passage through the region of interest. This time/position trace can be converted into a wind vector profile and an air density profile. The air density profile can be further reduced to produce a temperature profile. The primary tracking system design problem is then sphere position accuracy requirements imposed on the tracker by the density/wind vector measurement accuracy goals of the system. System design is directed toward defining a system which will meet these goals.

This tracking situation is dynamic. The time/position trace will consist of discrete measurements of elevation angle, azimuth angle, range or distance of the sphere from the tracker and a rate of change of that distance or range rate. See Figure IV-1. These quantities are designated α, β, R, and $\dot{R}$. α and β are the orthogonal angles which define the line-of-sight to the sphere with respect to the centerline of the field of view of the tracker. R is the distance of the sphere from the tracker and $\dot{R}$ is the rate of change of that distance.

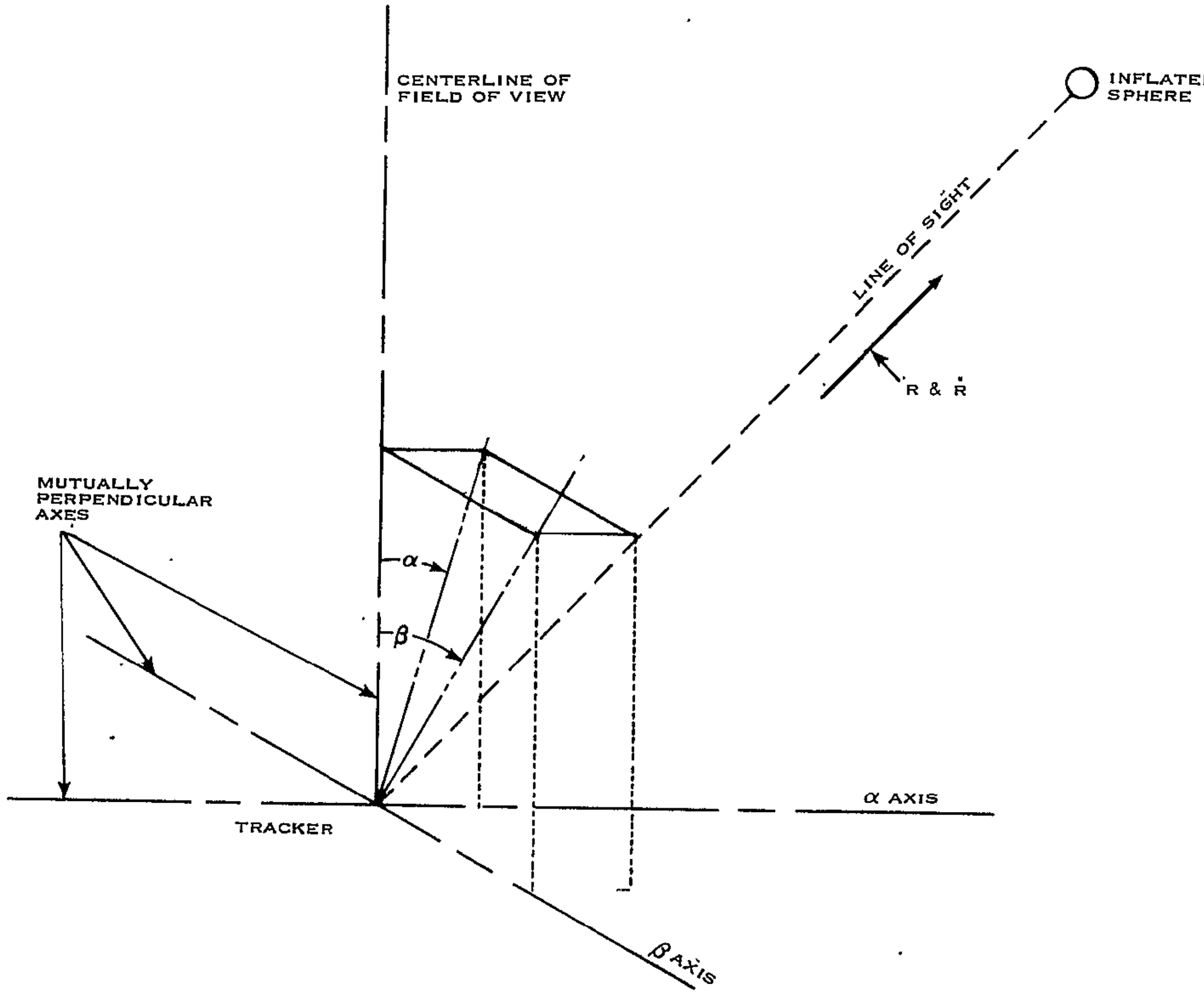


FIGURE IV-1. Geometry of Measured Quantities

Because of the extreme accuracy requirements placed on the system and the differences in achievable accuracy associated with each of the four variables, an absolute statement of requisite accuracy for each variable is not practical. The required overall accuracy is the result of trade offs between variables as well as data reduction techniques. A useful set of accuracy statements has evolved from the analysis of typical trajectories (See Appendix B) and from the results obtained using conventional electromechanical trackers. Electromechanical tracking systems have generally been found inadequate (See Appendix A). The AN/MPS-19 is inadequate for this application. The AN/FPS-16 produces useful results for the ascent leg of the trajectory, however, due to inadequate signal-to-noise ratio at the receiver, its accuracy falls far below requirements when the slant range exceeded 75 kilometers. The AN/FPQ-6 exhibited somewhat better performance. The TRADEX system, however, produces useful data over the entire trajectory. This system has two possible advantages over the other systems; vastly increased power at the transmitter and an integral doppler velocity readout.

Considering the specific flight profiles shown in Figures IV-2 and IV-3 it becomes apparent that measurements of angular motion as seen from the tracker predominates during mid-trajectory, whereas the measurement of radial motion dominates the earlier and later parts of the trajectory. In addition, the absolute velocity of the sphere varies widely from that of the launch vehicle just after inflation, to a relatively low value at apogee, then to supersonic speeds during fall, and finally to a low subsonic speed just prior to collapse. If one were to specify the customary radar parameters and readout accuracies for each of the regimes, the result would be several radars operating at various pulse repetition rates, various pulse widths, and with significantly differing angular resolution capabilities. The solution to this dilemma is a semi-adaptive radar operating under preprogrammed control. The traditional electromechanical tracker uses a multiplicity of special purpose analog subsystems which are not flexible. By way of contrast, many of these functions in a phased-array system are executed by a general-purpose digital computer. The flexibility of the phased-array system is limited by the ingenuity of the programmer.

The parameters to consider include:

- Transmitted power (peak)
- Pulse width

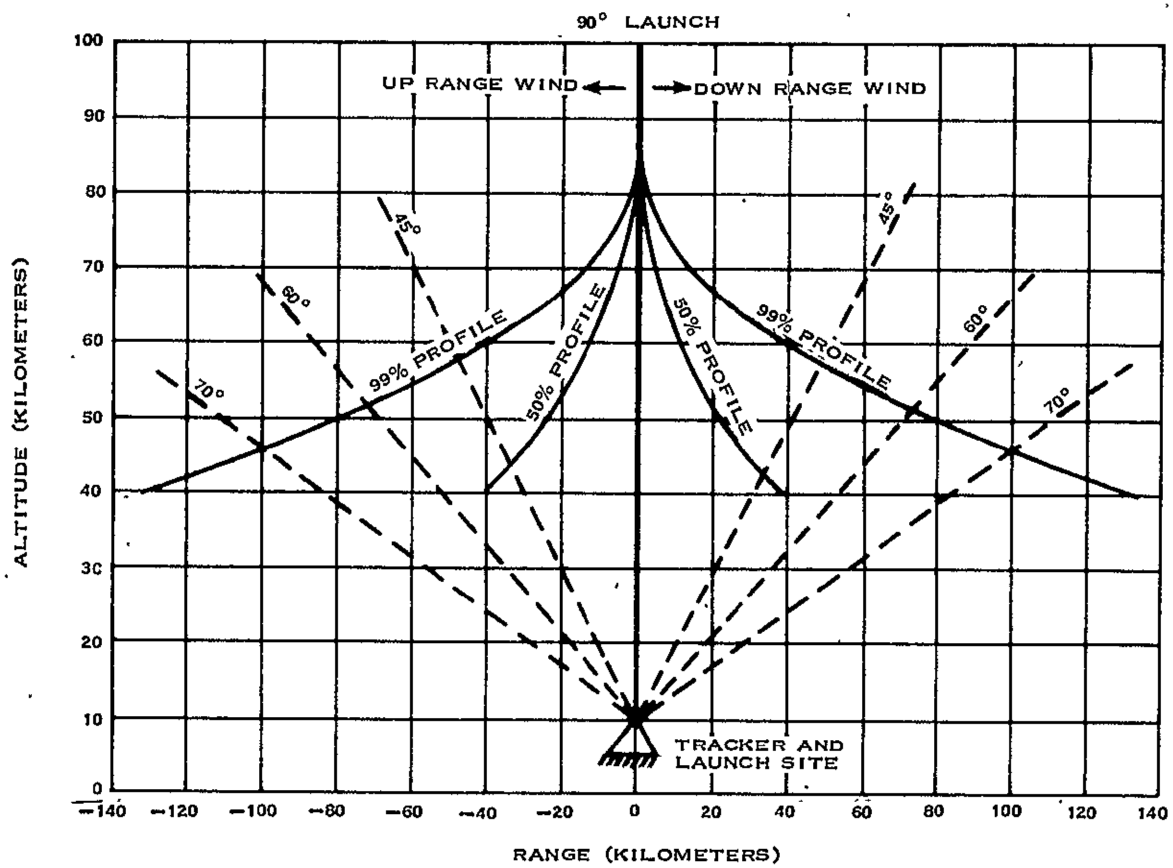


FIGURE IV-2. Flight Profile - Vertical Launch

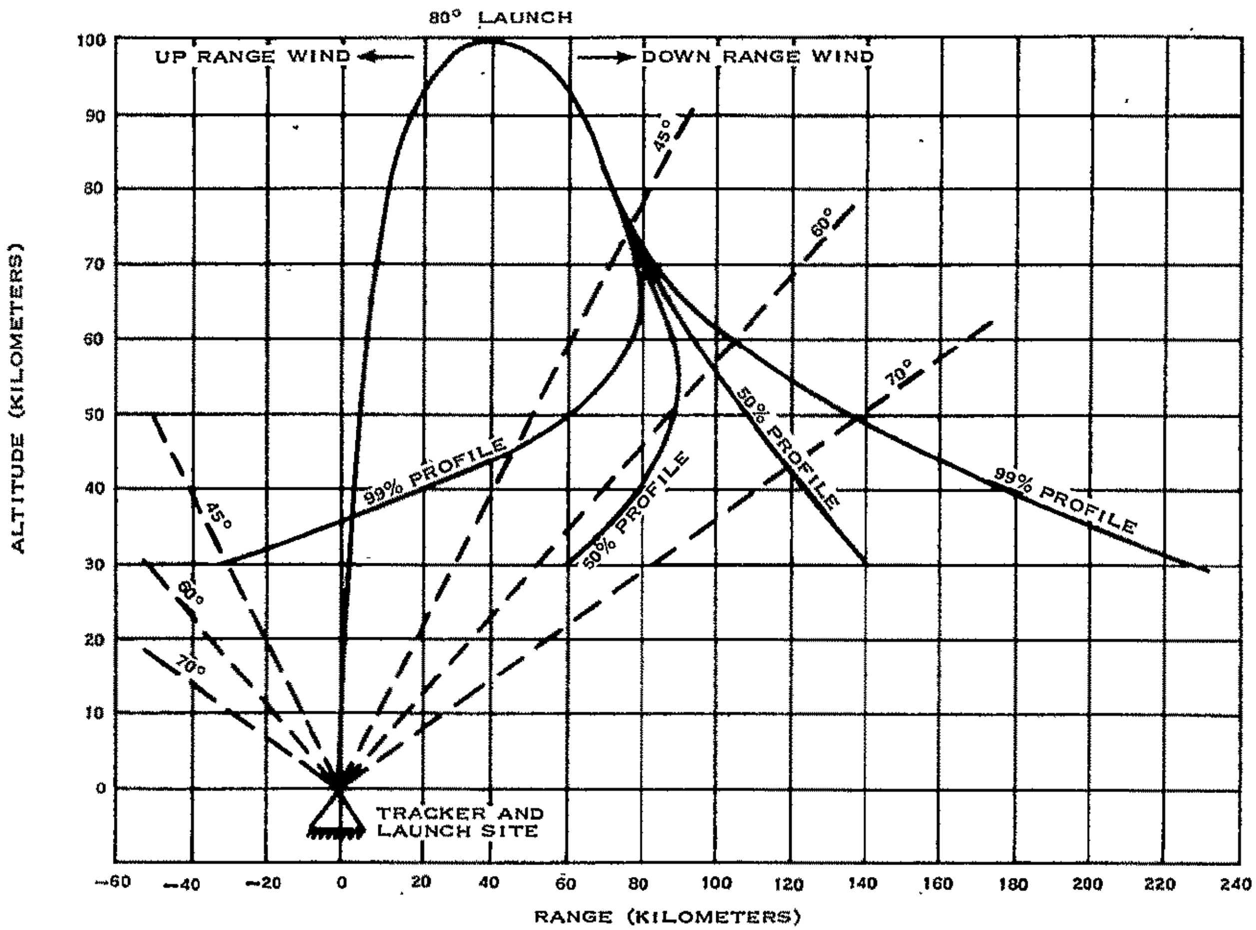


FIGURE IV-3. Flight Profile - Inclined Launch

- Pulse repetition rate
- Beam width
- Beam splitting resolution.

Certain other parameters will exhibit interlocking relations with the above, but are subsidiary considerations. These include:

- Operating frequency
- Antenna size
- Receiver noise figure.

The first requirement is to guarantee an adequate signal-to-noise ratio over the portion of the sphere trajectory which is measured. The classical radar equation in simplified form is:

$$\frac{S}{N} = \frac{P_t G_t G_r \lambda^2 \sigma}{R^4 B F_o L}$$

where

P_t = Power in watts

G_t = Gain of transmitting antenna

G_r = Gain of receiving antenna

λ = Operating wavelength in cm

R = Slant range - nm

σ = Sphere cross section in m^2

B = Receiver bandwidth in Hz

F_o = Receiver noise figure

L = System losses.

Assuming operation at S-band frequencies, the following values may be realized:

λ^2 = 10 db

σ = 0 db

R^4 = 48 to 80 db (from 15 to 100 nm)

B = 63 db (with $B\tau = 1.4$ and $\tau = 1$ microsecond)

F_o = 4 db

L = 10 db.

Thus, for a signal-to-noise ratio of 20 db, the transmitted power, transmitting antenna gain, and receiving antenna gain must total 167 db. If the transmitter power is assumed to be one megawatt, the requisite antenna size (assuming equal area for transmitting and receiving) becomes 12 meters in diameter.

However, there are trade-offs available to the designer. Receiver bandwidth may be materially reduced by increasing the transmitted pulse width. The price is an increase in average transmitter power and a reduction in the accuracy of the range readout. The increase in average power is not a severe problem unless carried to extremes; while the reduction in range accuracy might, for this application, be more than compensated for by the resulting enhancement of pulse doppler processing to obtain radial range rate data. Another possible trade-off lies in the transmitting and receiving antennas. The basic function of the transmitting antenna is that of illuminating the sphere, therefore, its beam shape and beam steering resolution are not critical. By contrast, the resolution and accuracy of the angular readouts are largely dependent upon the beam diameter, beam shape, and beam forming increments of the receiving antenna. Thus, a receiving antenna somewhat larger than the transmitting antenna could be used to maintain the signal-to-noise ratio and at the same time enhance the angular accuracy. The relatively lower cost of the receiving elements, as compared to transmitting elements, makes this concept attractive.

A basic centerline accuracy in angle determination of 0.2 mils is considered adequate. This basic accuracy can be realized over a steering region up to 45 degrees off the centerline along either axis. See Figure IV-4. Adequate performance with degraded accuracy will be realized out to 60 degrees, and tracking capability with further reduction in accuracy, extends to 70 degrees.

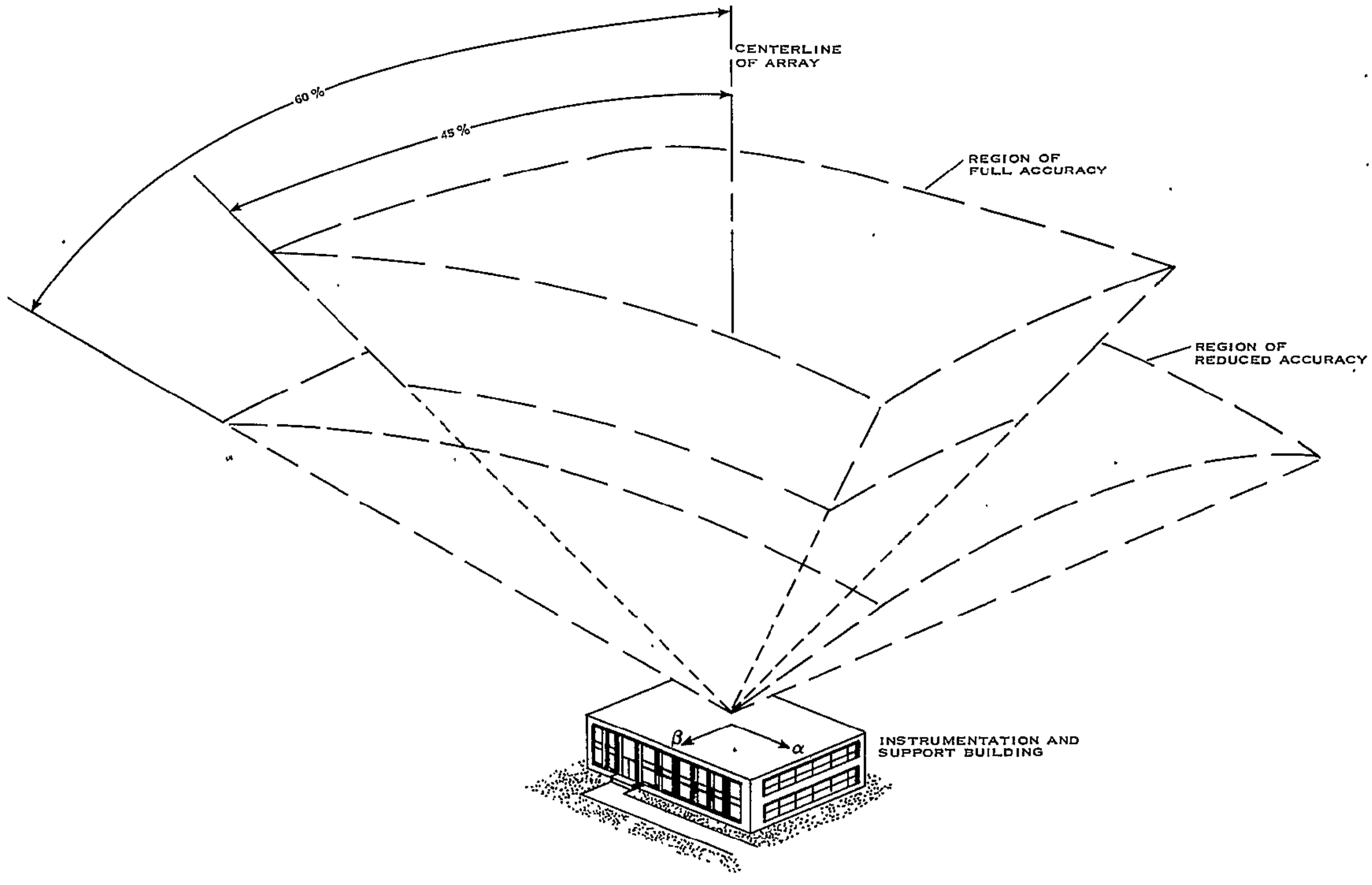


FIGURE IV-4. Tracker Coverage Pattern

Range readout accuracy of 25 yards ia apparently adequate, but may not be required when range rate data is available. This data may enhance the quality of the meteorological data output to the point that it is possible to relax range measurement accuracy.

3. DATA ACQUISITION/TRACKING SYSTEM DESCRIPTION

Figure IV-5 is a function block diagram of the data acquisition/tracking system. The heart of this system is a general-purpose digital computer. The system functions in the following manner. Upon receipt of a lift-off signal from the launcher, the computer will energize the transmit array in such a way as to illuminate a sector to intercept the ascending vehicle at a predetermined altitude. The pulse repetition rate will be such as to permit unambiguous ranging on the vehicle. Simultaneously, a cluster of receiving beams will be formed in the same sector. Automatic angle tracking of the vehicle will be initiated when a predetermined number of sequential receiver pulses cross the 10 db threshold. Deployment of the sphere from the launch vehicle will produce two targets for the trackers, each with markedly different reflective characteristics. A pair of displays will permit the operator to select the one corresponding to the sphere, and range tracking (along with velocity determination) will be initiated. This function might well be accomplished by using the computer to make the decision based on a priori drag data.

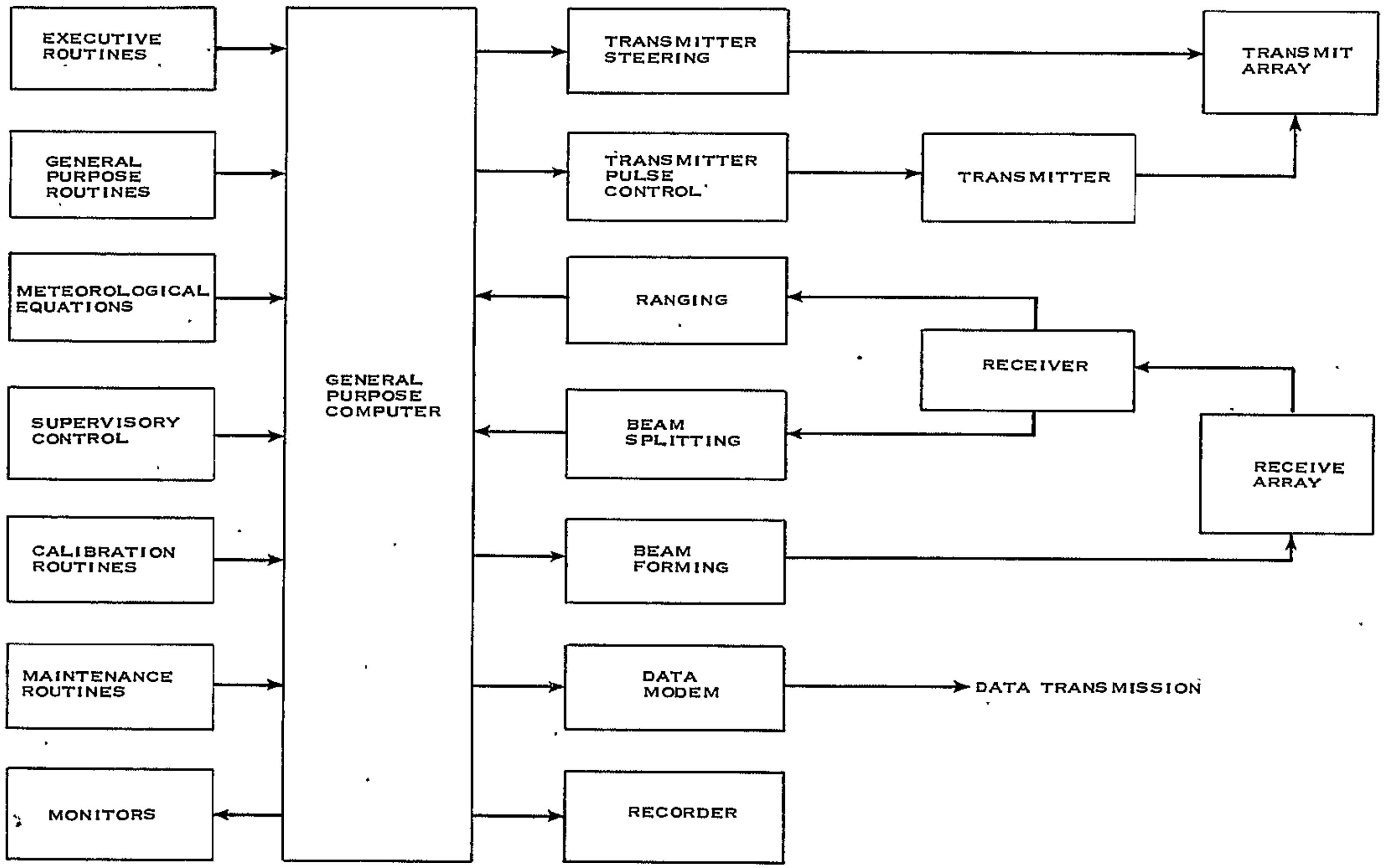


FIGURE IV-5. Tracking and Data Processing Subsystem

The computer will continue to steer the transmitted beam in such a direction as to keep the sphere illuminated. It will simultaneously form a monopulse quad beam at the receiver to ascertain the instantaneous line-of-sight to the sphere. Beam splitting techniques will be used to increase the angular resolution by 20:1. The interval between pulses will be gradually increased as the range increases to ensure nonambiguous readings. As the sphere velocity becomes very low, the redundancy in readings will become considerably in excess of that which is useful in achieving requisite accuracy in density profile and wind vectors, so the pulse spacing may be still further increased.

The tracker will be housed in a one-story building with the planar antenna lying flush on the flat roof. The roof will have sufficient tilt to optimize coverage. The specific angle of tilt will be a function of the launcher's nominal inclination and possibly of prevailing winds. The arrays themselves will be solid-state, integrated-circuit units assembled on automated assembly lines. The transmitter units will include the final power amplifier, the steering elements, and the radiating elements. The receiving units will include the radiating elements, the low-noise receiver front ends, and elements of the beam-forming matrices. The range tracker will be an early-gate/late-gate, solid-state digital tracker.

The circuitry is such as to lend itself to mass production techniques and to complete assembly and checkout in the factory. The on-site activity will be limited to installation, plug-in, and calibration. The design will include full maintenance and calibration routines under computer program control so as to permit operation and maintenance by personnel of minimal skills.

The most critical part of the tracker is the receiving array. It will contain about 10,000 identical elements packaged into convenient modular subassemblies.

4. PERFORMANCE SPECIFICATION

Tracking -	Multiple (1-5) targets tracked simultaneously, acquisition of rocket before separation (20 square meter target at 5 km), all targets within same 20° cone of total coverage.
Accuracy -	$\pm$ 0.1 milliradian angular, $\pm$ 5 meters range (1 values, $\pm$ 20% of boresight, 1 hit, 160 km range, data rate of 1 point per 2 milliseconds, 1 square meter target).
Range -	200 km maximum
Coverage -	60° cone around a fixed boresight
Operation -	All-weather, minimum technical attendance

Characteristics:

S-band, phased-array, one face

High-duty cycle - 40 percent

Pre-packaged for simple on-site installation with minimum checkout and test

Solid-state, modular automatic fault isolation for plug-in maintenance

Built-in, general-purpose digital computer

Initial on-site calibration may require appreciable use of equipment and personnel

Routine calibration fully automated.

5. COST

There are three distinct kinds of costs which will be encountered in the evolution of the tracker. They are:

- Feasibility model
- Development
- Production.

Using current techniques and know-how, a single feasibility model will cost at least $2, 500, 000, and may well run as high as $3, 000, 000. Development costs of a production model could run anywhere up to $20, 000, 000. There are currently various studies under way, particularly at L-band and at S-band, frequencies, which should reduce these development costs. In several cases the ultimate goal of the study is the development of integrated circuit phased-array elements for automated assembly. The break point in the cost versus quantity curve for automated fabrication of integrated circuits is estimated at 40, 000 units. Since a production run of 100 trackers will require about 1, 000, 000 elements, the full cost benefits of automated fabrication may be realized. The best estimate for the total FOB factory price of this tracker in quantities of 100 is $700, 000 apiece. The uncertainty is about plus or minus $200, 000.

6. PHASED-ARRAY SYSTEMS

A phased-array tracker offers two inherent advantages over an electromechanical tracker. Acquisition is far less of a problem since scan patterns may be implemented in microseconds in lieu of seconds, and multiple target tracking is much more readily attained. Both characteristics are useful in the synthesis of a synoptic sounding system. The facile acquisition capability minimizes the need for skilled operating personnel and the need for backup vehicles. The multiple target tracking capability eases the transfer of track from the aerodynamic vehicle to the inflated sphere, thus, further minimizing the need for skilled operators and backup vehicles.

A brief examination of how a phased-array system functions will show how these characteristics arise. They may then be related to. the operational aspects of the sounding system. A simple linear dipole has a relatively broad directional pattern as shown in Figure IV-6. Locating two or more radiating elements close to each other will result in an effective adding of their radiated power in the far field; that is, at some distance from the antennas. If all antennas are driven in phase, the effective beam width is narrowed in proportion to the number of antennas. See Figure IV-7. If, however, these antennas are driven not in phase, but rather with an integral incremental phase shift, the beam will tend to retain its narrowed contour but be reoriented off the axis of the array. A practical method of steering such an array is shown in Figure IV-8. A tapped delay line provides the uniformly incremental phase shifts to be inserted between the transmitter and the antennas. The amount of shift and the size of the angle steered off axis is determined by the steering frequency since the phase shift of a real delay line is a function of frequency. The steering frequency itself is removed from the transmitter frequency by post-delay line mixers, thus permitting narrow band operation. Antenna reciprocity permit comparable beam forming on received signals.

FIGURE IV-6
Simple Dipole

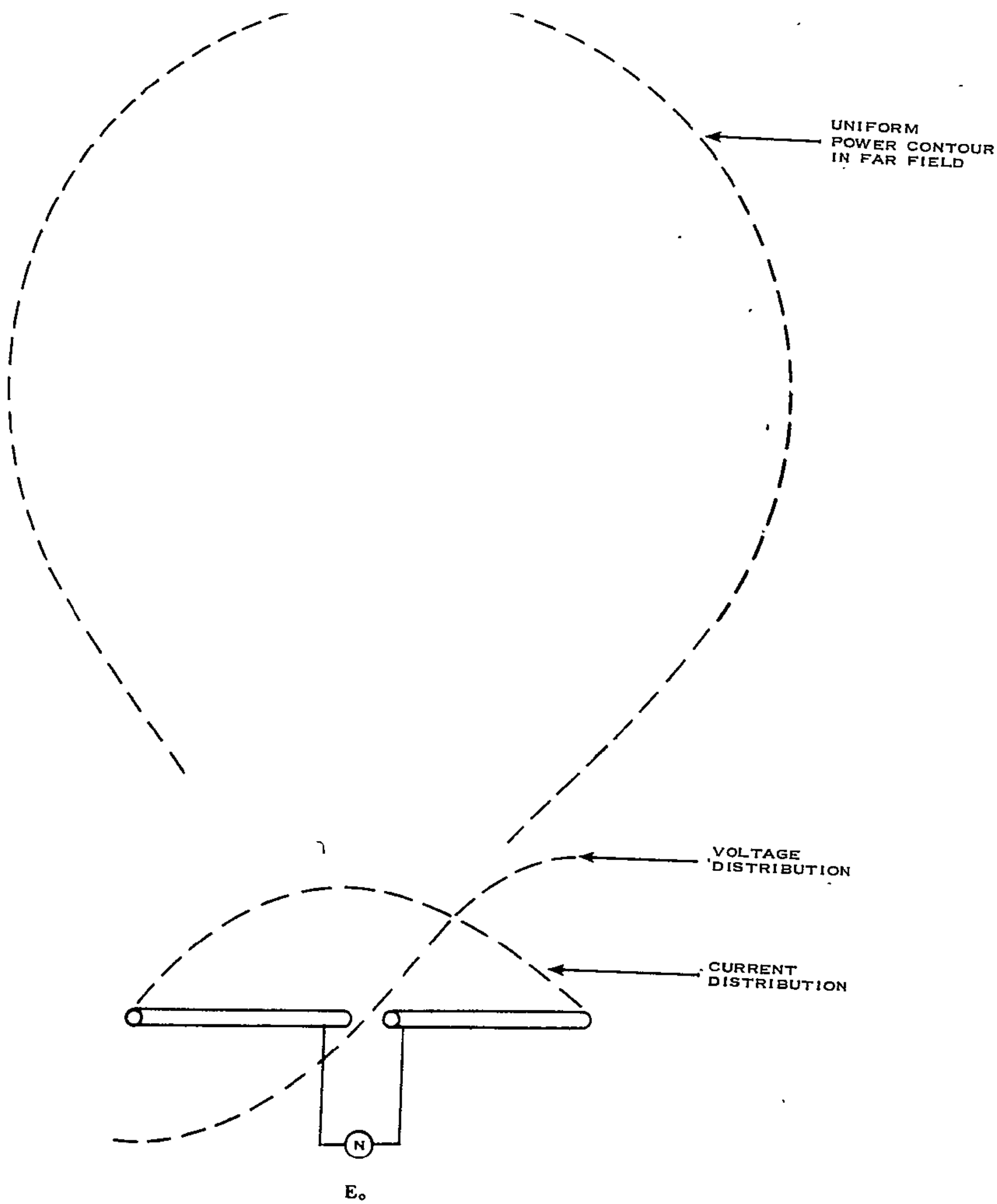


FIGURE IV-7
Phase Shifted Array

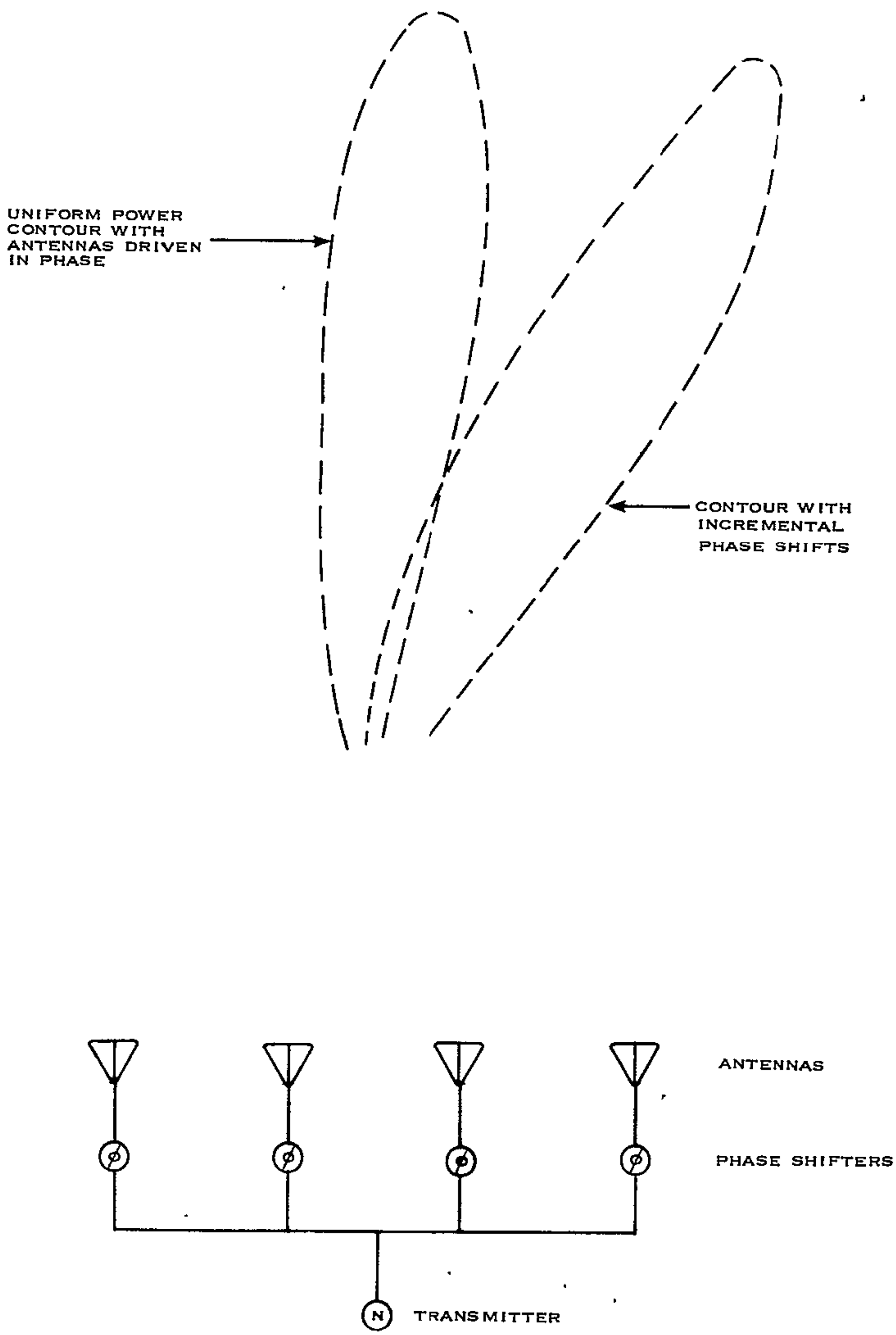
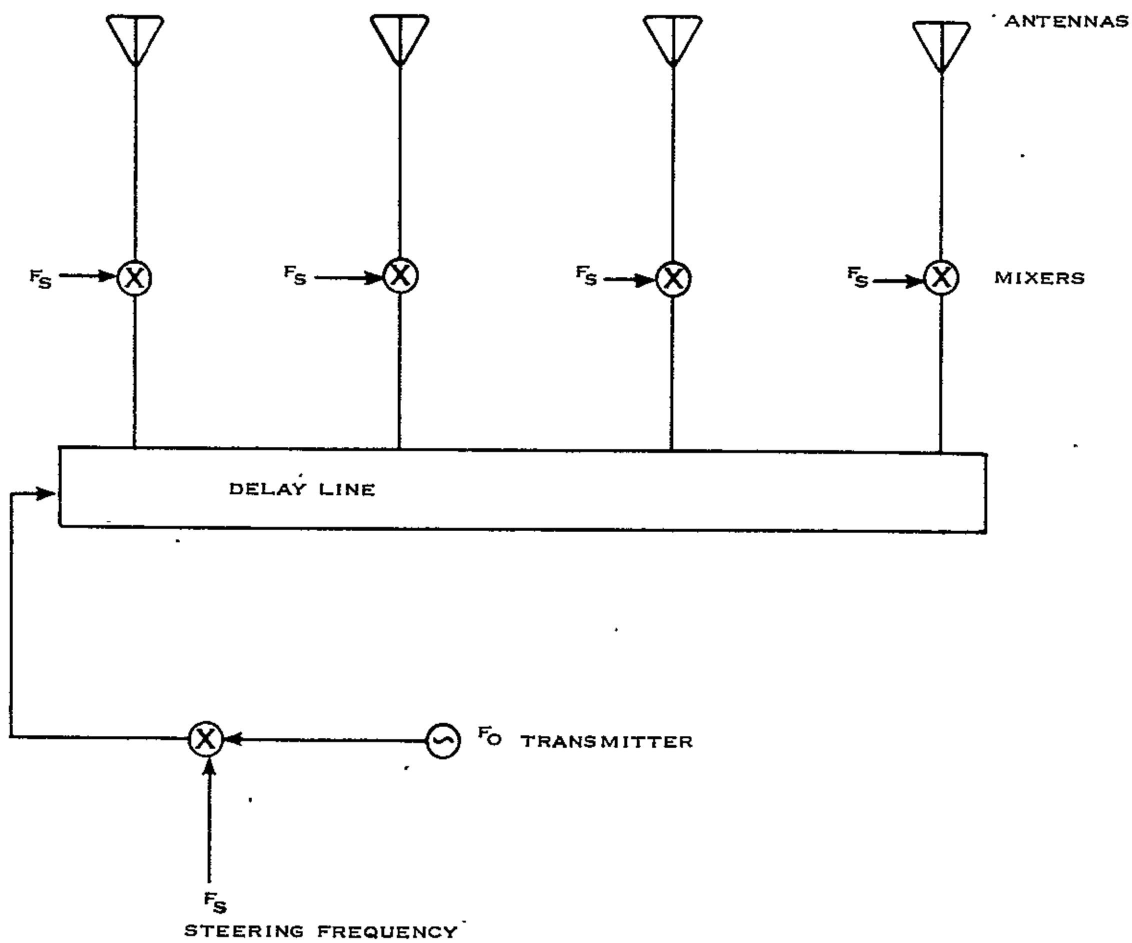


FIGURE IV-8
Typical Steering Technique



Since a stable frequency synthesizer operating under the control of a digital computer can steer transmitted beams to any point within the tracker field of view at a speed limited only by the Input/Output capabilities of the computer, illumination of widely varying sectors can be accomplished at will, reducing the acquisition problem to a trivial one. The fact that two or more such sectors may be illuminated in sequence at a rate which produces the effect of simultaneity permits received beams to be formed on two or more targets with simultaneous readouts. Thus, track of all components of the airborne vehicle may be maintained until positive identification of the sphere is established.

During the tracking operation some degree of adaptive control pulse spacing and real-time prediction and data smoothing will be performed by the computer. After flight termination the recorded tracking data may then be batch processed by the same computer to extract the requisite meteorological data: pressure, temperature, wind velocity, and wind direction. These data may then be formated for facile transmission, and actually transmitted to the control center for synoptic use.

7. ERROR ANALYSIS

Investigations of accuracy in density, temperature and wind measurements, as determined by the ROBIN/FPS-16 System, have been reported in the literature (Reference 78 and 79). This analysis is intended to gain additional insight into the relation between independent error components in the tracking system and the resulting quality of the meteorological data. In particular, the possible enhancement of these data by use of the Doppler measurements to supplement or even supplant the basic range measurement is of interest.

Previous authors estimating the errors in the meteorological parameters using sphere techniques have approached the problem by assuming published radar error estimates. But they have ignored the fact that these are valid only over a defined range of signal-to-noise ratios, and that the sphere's effective cross-section was inadequate to maintain this ratio over the entire trajectory. Therefore, when predicted errors were compared to those obtained experimentally, discrepancies often appeared. Further investigation confirmed the fact that the radar was being used in a manner that invalidated the assumed radar error values. As a result, much of the published meteorological parametric error data is open to question.

The following series of fundamental relations will serve as a guideline in constructing the error analysis.

- The density deviation of the falling sphere technique is a function of:
  - vertical accelerations and velocities
  - gravity
  - drag coefficient.
- Wind deviation is a function of:
  - horizontal accelerations and velocities
  - vertical accelerations and velocities
  - gravity.
- Errors in density and wind can be given as a function of errors in:
  - range
  - elevation angle
  - range rate
  - elevation rate
  - range acceleration
  - elevation acceleration.
- Direct measurement can be made of:
  - range
  - range rate
  - elevation angle.
- Mathematical fitting techniques will be used to derive estimates of:
  - range acceleration
  - elevation rate
  - elevation acceleration.

- Errors in derived parameters can be expanded as functions of errors in measured parameters, if assumptions are made concerning:

  - function of data points
  - fitting functions
  - data measurement frequencies.

- Errors in density and winds can then be given as functions of errors in the measured parameters:

  - range
  - range rate
  - elevation angle.

- By specifying error requirements for:

  - density
  - winds

  and assuming state-of-the-art errors for

  - range
  - range rate

  design requirements for errors in elevation angle can be specified.

In the development of a falling sphere/radar system, the problem by radar error type was identified. Most experimenters, using this sounding technique, have developed the opinion that the FPS-16 radar is often not adequate to produce the required accuracies. This investigation represents an initial attempt to answer the question:

How good does the radar have to be to produce meterological parameters with given error estimates? The advantage of developing a radar system for a specific task is that the designer can specify the radar parameters to be measured, and thus, reduce the number of critical parameters that must be derived from basic measurements.

The falling sphere technique is sensitive to errors in the first and second time derivations of the coordinates, as well as errors in coordinate parameters themselves. In light of this, if the range rate and elevation rates could be measured directly, the errors introduced by the mathematical smoothing technique to produce these parameters could be eliminated. If the errors in the measurements can be kept smaller than the derived value errors, the accuracy of the meterological parameters will be enhanced.

Most falling sphere, data-reduction techniques are based on Cartesian coordinate data derived from spherical radar coordinate data. As a result, the error equations developed for the meteorological parameters are expressed in Cartesian parameters. For the purpose of a radar design exercise, the relationships between the meteorological parameter errors and the radar parameters must be known.

For ease of analysis, the normal two-dimensional balloon flight profile was assumed. This is a reasonable assumption when the sounding rocket is launched in the direction of the prevailing wind from a location directly downwind (or upwind) from the radar site.

The error in a parameter (density or winds) is a function of the errors in range and elevation, and the errors in the first and second derivatives of the range and elevation.

If the first and/or second derivatives are obtained by fitting a function to the data points and taking the derivative, the error can be separated into two parts. One part is the error due to noise in the data, the other due to lack of fit by the function to the physical laws that produced the data points.

The error due to noise is a function of

- fitting length (number of points)
- data frequency
- noise in measured parameter
- method used (polynomial and degree, etc.).

The error due to lack of fit is a function of

- fitting length (number of points)
- data frequency
- numerical characteristics of the function that produced the data points
- methods used (polynomial and degree).

In the trade-off study of radar design only the errors in measured parameters are subject to manipulation. Therefore, only the error due to noise in the measured parameters will be considered here. The error due to lack of fit is properly addressed in the analysis preceding the design of the data processing.

(1) Density Error

The expression for the percent error in density as presented is: (Reference 79)

$$\left(\frac{\sigma_\rho}{\rho}\right)^2 = \left(\frac{2\,\sigma_{W_z}}{\dot{z} - W_z}\right)^2 + \left(\frac{2\,\sigma_{\dot{z}}}{\dot{z} - W_z}\right)^2 + \left(\frac{\sigma_{\ddot{z}}}{\ddot{z} - g}\right)^2 + \left(\frac{\Delta \ddot{z}}{\ddot{z} - g} - \frac{2\,\Delta \dot{z}}{\dot{z} - W_z} - \frac{\Delta C_D}{C_D}\right)$$

where σ_q^2 is the variance of the random error in parameter q

Δq is the bias error in the parameter q

ρ is the density

C_D is the drag coefficient

W_z is the vertical wind

$\dot{z}$ is the vertical velocity

$\ddot{z}$ is the vertical acceleration

g is the acceleration due to gravity.

The vertical winds and their variance can be estimated independently and their contribution to the percent error in density will be included in this analysis.

The bias error is the error due to lack of fit and will not be considered at this time.

The error in density due to the error in drag coefficient is discussed under problems to be solved, but it is not germane to the tracking error problem. Rewriting the equation for percent error in density, neglecting the terms eliminated by the assumptions gives:

$$\left(\frac{\sigma_\rho}{\rho}\right)^2 = \frac{4}{(\dot{z}-W_z)^2}\,\sigma_{\dot{z}}^2 + \frac{1}{(\ddot{z}-g)^2}\,\sigma_{\ddot{z}}^2 + \frac{4}{(\dot{z}-W_z)^2}\,\sigma_{W_z}^2 .$$

(2) Wind Error

The expression for a horizontal wind in the above notation is -

$$W = \dot{x} - \frac{\ddot{x}\,\dot{z}}{\ddot{z}-g} .$$

If the assumption concerning bias error is again made, the variance in the wind, as obtained by the application of the normal error variance technique is

$$\sigma_W^2 = \sigma_{\dot{x}}^2 + \left(\frac{\dot{z}}{\ddot{z}-g}\right)^2 \sigma_{\ddot{x}}^2 + \left(\frac{\ddot{x}}{\ddot{z}-g}\right)^2 \sigma_{\dot{z}}^2 + \left(\frac{\ddot{x}\,\dot{z}}{(\ddot{z}-g)^2}\right)^2 \sigma_{\ddot{z}}^2 .$$

(3) <u>Coordinate Transformation</u>

The normal equations used to transfer Cartesian coordinates to spherical coordinates are:

$$X = R \cos E$$

$$Z = R \sin E$$

where X = down range distance

Z = vertical distance

R = slant range

E = elevation angle between horizontal and slant range.

The first and second time derivatives are

$$\frac{dX}{dt} = \frac{dR}{dt} \cos E - R \frac{dE}{dt} \sin E$$

$$\frac{dZ}{dt} = \frac{dR}{dt} \sin E + R \frac{dE}{dt} \cos E$$

$$\frac{d^2X}{dt^2} = \frac{d^2R}{dt^2} \cos E - 2 \frac{dR}{dt} \frac{dE}{dt} \sin E - R \frac{d^2E}{dt^2} \sin E - R\left(\frac{dE}{dt}\right)^2 \cos E$$

$$\frac{d^2Z}{dt^2} = \frac{d^2R}{dt^2} \sin E + 2 \frac{dR}{dt} \frac{dE}{dt} \cos E + R \frac{d^2E}{dt^2} \cos E - R\left(\frac{dE}{dt}\right)^2 \sin E .$$

Written in more compact notation where

$$\dot{q} = \frac{dq}{dt} \qquad \ddot{q} = \frac{d^2q}{dt^2}$$

the derivatives become

$$\dot{X} = \dot{R}\cos E - R\dot{E}\sin E$$

$$\dot{Z} = \dot{R}\sin E + R\dot{E}\cos E$$

$$\ddot{X} = \ddot{R}\cos E - 2\dot{R}\dot{E}\sin E - R\ddot{E}\sin E - R\dot{E}^2\cos E$$

$$\ddot{Z} = \ddot{R}\sin E + 2\dot{R}\dot{E}\cos E + R\ddot{E}\cos E - R\dot{E}^2\sin E\ .$$

An application of the normal technique to produce error variances equations for the first and second derivative yield

$$\sigma^2_{\dot{X}} = \cos^2 E\,\sigma^2_{\dot{R}} + \left(\dot{R}\sin E + R\dot{E}\cos E\right)^2\sigma^2_E + \dot{E}^2\sin^2 E\,\sigma^2_R + R^2\sin^2 E\,\sigma^2_{\dot{E}}$$

$$\sigma^2_{\dot{Z}} = \sin^2 E\,\sigma^2_{\dot{R}} + \left(\dot{R}\cos E - R\dot{E}\sin E\right)^2\sigma^2_E + \dot{E}^2\cos^2 E\,\sigma^2_R + R^2\cos^2 E\,\sigma^2_{\dot{E}}$$

$$\begin{aligned}\sigma^2_{\ddot{X}} = {} & \left(-\ddot{E}\sin E - \dot{E}^2\cos E\right)^2\sigma^2_R + \\ & \left(\ddot{R}\sin E + 2\dot{R}\dot{E}\cos E + R\ddot{E}\cos E - R\dot{E}^2\sin E\right)^2\sigma^2_E \\ & + \left(-2\dot{R}\sin E - 2R\dot{E}\cos E\right)^2\sigma^2_{\dot{E}} \\ & + 4\dot{E}^2\sin^2 E\,\sigma^2_{\dot{R}} + \cos^2 E\,\sigma^2_{\ddot{R}} + R^2\sin^2 E\,\sigma^2_{\ddot{E}}\end{aligned}$$

$$
\begin{aligned}
\sigma^2_{\ddot{Z}} = {} & \left(\ddot{E}\cos E - \dot{E}^2 \sin E\right)^2 \sigma^2_R \; + \\
& \left(\ddot{R}\cos E - 2\dot{R}\dot{E}\sin E - R\ddot{E}\sin E - R\dot{E}^2\cos E\right)^2 \sigma^2_E \\
& + \left(2\dot{R}\cos E - 2R\dot{E}\sin E\right)^2 \sigma^2_{\dot{E}} \\
& + 4\dot{E}^2\cos^2 E\, \sigma^2_{\dot{R}} + \sin^2 \sigma^2_{\ddot{R}} + R^2\cos^2 E\, \sigma^2_{\ddot{E}} .
\end{aligned}
$$

(4) Simplification of Equations

The coordinate transformation section has produced the necessary, if unwieldy, tools for the expression of the errors in density and winds as functions of spherical parameters.

Further algebraic manipulation is required to complete the expressions of the errors in the radar parameters and errors. To simplify this task, the following factors will be defined in Table IV-1, F Factors.

It should be noted that all of the factors are functions of trajectory parameters and contain no error terms.

Substitution of the factors into the Cartesian derivatives and their error yields

$$\dot{X}^2 = F05 \qquad \ddot{X}^2 = F07$$

$$\dot{Z}^2 = F06 \qquad \ddot{Z}^2 = F08$$

$$(\ddot{Z} - g)^2 = F15 .$$

Table IV-1

F Factors

$$F01 = \sin^2 E$$

$$F02 = \cos^2 E$$

$$F03 = R^2 \cos^2 E$$

$$F04 = R^2 \sin^2 E$$

$$F05 = (\dot{R} \cos E - R\dot{E} \sin E)^2$$

$$F06 = (\dot{R} \sin E + R\dot{E} \cos E)^2$$

$$F07 = (\ddot{R} \cos E - 2\dot{R}\dot{E} \sin E - R\ddot{E} \sin E - R\dot{E}^2 \cos E)^2$$

$$F08 = (\ddot{R} \sin E + 2\dot{R}\dot{E} \cos E + R\ddot{E} \cos E - R\dot{E}^2 \sin E)^2$$

$$F09 = \dot{E}^2 \cos^2 E$$

$$F10 = \dot{E}^2 \sin^2 E$$

$$F11 = (-\ddot{E} \sin E - \dot{E}^2 \cos E)^2$$

$$F12 = (-2\dot{R} \sin E - 2R\dot{E} \cos E)^2$$

$$F13 = (\ddot{E} \cos E - \dot{E}^2 \sin E)^2$$

$$F14 = (2\dot{R} \cos E - 2R\dot{E} \sin E)^2$$

$$F15 = (\ddot{R} \sin E + 2\dot{R}\dot{E} \cos E + R\ddot{E} \cos E - R\dot{E}^2 \sin E - g)^2$$

$$F16 = (\ddot{R} \sin E + R\ddot{E} \cos E - W_z)^2$$

$$\sigma^2_{\dot{X}} = F02\,\sigma^2_{\dot{R}} + F06\,\sigma^2_{E} + F10\,\sigma^2_{R} + F04\,\sigma^2_{\dot{E}}$$

$$\sigma^2_{\dot{Z}} = F01\,\sigma^2_{\dot{R}} + F05\,\sigma^2_{E} + F09\,\sigma^2_{R} + F03\,\sigma^2_{\dot{E}}$$

$$\sigma^2_{\ddot{X}} = 4F10\,\sigma^2_{\dot{R}} + F08\,\sigma^2_{E} + F11\,\sigma^2_{R}$$
$$+ F12\,\sigma^2_{\dot{E}} + F02\,\sigma^2_{\ddot{R}} + F04\,\sigma^2_{\ddot{E}}$$

$$\sigma^2_{\ddot{Z}} = 4F09\,\sigma^2_{\dot{R}} + F07\,\sigma^2_{E} + F13\,\sigma^2_{R}$$
$$+ F14\,\sigma^2_{\dot{E}} + F01\,\sigma^2_{\ddot{R}} + F03\,\sigma^2_{\ddot{E}} \,.$$

Substitution of the Cartesian derivatives and their errors into the density and wind error equations yield

$$\left(\frac{\sigma_\rho}{\rho}\right)^2 = \left(\frac{4F01}{F16} + \frac{4F09}{F15}\right)\sigma^2_{\dot{R}} + \left(\frac{F14}{F15} + \frac{4F03}{F16}\right)\sigma^2_{\dot{E}}$$
$$+ \left(\frac{F01}{F15}\right)\sigma^2_{\ddot{R}} + \left(\frac{F03}{F15}\right)\sigma^2_{\ddot{E}} + \left(\frac{4F09}{F16} + \frac{F13}{F15}\right)\sigma^2_{R}$$
$$+ \left(\frac{4F05}{F16} + \frac{F07}{F15}\right)\sigma^2_{E} + \left(\frac{4}{F16}\right)\sigma^2_{WZ} \,.$$

$$\sigma^2_{W} = \left(F02 + \frac{4F10\,F06}{F15} + \frac{F01\,F07}{F15} + \frac{4F09\,F06\,F07}{F15^2}\right)\sigma^2_{\dot{R}}$$
$$+ \left(F04 + \frac{F12\,F06}{F15} + \frac{F03\,F07}{F15} + \frac{F14\,F06\,F07}{F15^2}\right)\sigma^2_{\dot{E}}$$
$$+ \left(\frac{F02\,F06}{F15} + \frac{F01\,F06\,F07}{F15^2}\right)\sigma^2_{\ddot{R}}$$
$$+ \left(\frac{F04\,F06}{F15} + \frac{F03\,F06\,F07}{F15^2}\right)\sigma^2_{\ddot{E}}$$

$$+\left(F10+\frac{F11F06}{F15}+\frac{F09F07}{F15}+\frac{F13F06F07}{F15^2}\right)\sigma_R^2$$

$$+\left(F06+\frac{F08F06}{F15}+\frac{F05F07}{F15}+\frac{F07F06F07}{F15^2}\right)\sigma_E^2$$

It again becomes obvious that further simplification is necessary to continue manipulation. The following factors are defined in Table IV-2, G Factors. It should be noted that these factors contain only trajectory parameters and do not contain error terms.

Table IV-2
G Factors

$$G1 = \frac{4F01}{F16} + \frac{4F09}{F15} \qquad G2 = \frac{4F03}{F16} + \frac{F14}{F15}$$

$$G3 = \frac{F01}{F15} \qquad G4 = \frac{F03}{F15}$$

$$G5 = \frac{4F09}{F16} + \frac{F13}{F15} \qquad G6 = \frac{4F05}{F16} + \frac{F07}{F15}$$

$$G7 = F02 + \frac{4F10F06}{F15} + \frac{F01F07}{F15} + \frac{4F09F06F07}{F15^2}$$

$$G8 = F04 + \frac{F12F06}{F15} + \frac{F03F07}{F15} + \frac{F14F06F07}{F15^2}$$

$$G9 = \frac{F02F06}{F15} + \frac{F01F06F07}{F15^2}$$

$$G10 = \frac{F04F06}{F15} + \frac{F03F06F07}{F15^2}$$

$$G11 = F10 + \frac{F11F06}{F15} + \frac{F09F07}{F15} + \frac{F13F06F07}{F15^2}$$

$$G12 = F06 + \frac{F08F06}{F15} + \frac{F05F07}{F15} + \frac{F07F06F07}{F15^2}$$

$$G0 = \frac{4}{F16}$$

(5) Typical Balloon Trajectories

The estimation of errors in the meteorological parameters are dependent on values of the balloon trajectory parameters. A two-dimensional trajectory program was written and operated on an SDS 940 computer. The output was a table of the spatial distances, velocities, and accelerations at even 10,000 meter altitude increments extracted from the total trajectory computed at a time increment of 1 second. The program was operated from a remote terminal in a time-sharing system so that the input parameters could be varied quickly and their effect on the trajectory be seen immediately.

The necessary input parameters to the program were:

- The 1965 Standard Atmosphere - temperature, and density
- A Robin drag table (Reference 78)
- Wind profiles used previously in vehicle dispersion calculations
- Balloon characteristics and trajectory starting data.

Five trajectories were computer, one each for the -99%, -50%, 0, +50%, and +99% wind profiles.* All other

* Minus (-), Plus (+) refers to up and down range direction.

input parameters such as apogee, balloon area/mass ratio, drag table, standard atmosphere were held constant.

The purpose of five trajectories was to determine how sensitive to variations in balloon flights the final meteorological errors are.

(6) Data Frequency

The number of data points produced by the tracking device is a function of the slant range to the sphere. Table IV-3 lists a typical altitude-range profile.

The pulse period in sec is given by this expression

$$\text{Pulse period} = 100 + 2R\,(12.5)$$

where R is the slant range in nautical miles. The Repetition Rate is the inverse of the Pulse Period and is also given in the table. The minimum Repetition rate is about 600/second so a nominal value of 500 points/second will be used in the remaining analysis.

(7) Computation of Measured Radar Parameter Errors

The errors in the measured parameters in the radar system can be given as

$$\sigma_R = \frac{\tau}{2\sqrt{S/N}}\,, \quad \sigma_E = \frac{8}{\sqrt{S/N}}\,, \quad \sigma_{\dot{R}} = \frac{25{,}000}{\tau\cdot\sqrt{S/N}}$$

Table IV-3

Radar Repetition Rate

Altitude Km	Typical Range Meters	Pulse Period	Rep. Rate/ Second
130	136015	1029.7 μsec	971
120	146307	1100.0	909
110	148560	1115.4	897
100	149476	1121.7	892
90	149640	1122.8	891
80	149925	1124.7	889
70	155249	1161.1	861
60	162024	1207.4	828
50	184263	1359.5	736
40	208910	1527.9	654
30	224524	1634.7	612

where τ is the pulse length in μ sec and $\sqrt{s/n}$ is the signal to noise ratio in energy units. For a phased-array radar the s/n for the range and elevation are given by

$$s/n_{db} = 116 - 40 \log_{10} (\text{Range meters}/1852)$$

and for range rate

$$s/n_{db} = 130 - 40 \log_{10} (\text{Range meters}/1852)$$

a 40 μ sec pulse length was used for computations.

(8) Mathematical Fitting Techniques

It is a standard procedure to fit a low order polynomial to the data points to obtain the velocities and accelerations. In general the two methods that yield a second derivative are a polynomial of at least the second degree and two successive first derivatives of either finite differences or low order polynomials.

It can be shown that the errors in velocities and accelerations are smaller when a quadratic polynomial is used than when two successive linear polynomials are used to estimate acceleration. Therefore, in succeeding analysis, a quadratic polynomial will be fitted to the data points, and the velocity will be estimated from the first derivative, while the acceleration will be estimated from the second derivative.

The error in the first derivative as a function of the error in the parameter is given by

$$\sigma_{\dot{q}}^2 = \frac{12}{N(N+1)(N+2)\Delta t^2}\sigma_q^2 = D1\,\sigma_q^2$$

the error in the second derivative is given by

$$\sigma_{\ddot{q}}^2 = \frac{720}{(N-1)N(N+1)(N+2)(N+3)\Delta t^4}\sigma_q^2 = D2\,\sigma_q^2$$

(9) Selection of Radar Parameters to be Measured

The basic radar parameters that can be measured are elevation angle and slant range. In addition, use can be made of the doppler shift phenomena to measure the range rate which is the first time derivative of range. There are, therefore, two methods of obtaining estimates of range rate; by direct measurement, or by mathematical derivative of the range.

The ratio of the smoothed error to the measured error is only a function of the smoothing interval, n. The point at which it is better (lower error) to smooth than to measure can be determined from this ratio. Using the computation of a Repetition Rate of 500 points/second and the knowledge of the approximate smoothing interval for elevation angle it can be shown that for this problem, less error is made by smoothing R to get $\dot{R}$ than if $\dot{R}$ were measured directly by the doppler shift.

(10) Vertical Wind Effect on Density Error

The error in density was previously derived and shown to be dependent on both the error in Vertical Wind (W_z) and the magnitude of Vertical Winds. Numerous experimenters have both measured and inferred the magnitude of Vertical Winds. Unfortunately for our analysis, these estimates have a very large variation, as well as almost undetermined confidence levels (error estimates).

Rather than use a single estimate for either the Vertical Wind or its error, a matrix of ranges of values were used for vertical winds. The error in the values were estimated as percentages of the winds rather than a constant amount.

The errors in the meteorological parameters were then computed for each element in the Wind/Wind Error matrix. The results confirm the intuition that the magnitude was not as important as the estimate of the error. (See Appendix B).

(11) Final Development of Error Model

The final result of the preceeding section on Simplification of Equation yielded expressions for the error density and error in horizontal winds as functions of the variances of : range, range rate, range acceleration, elevation angle, elevation rate, elevation acceleration, and Vertical Winds, as well as values dependent on the trajectory, smoothing interval, and Vertical Winds.

A preceding section indicated that only the range and elevation angle should be measured with the rates and acceleration being derived mathematically. The section on Mathematical Fitting Techniques contains the expression for the errors in rates and accelerations when these quantities are derived. The model can then be modified by substituting for the variances of $\dot{R}$, $\ddot{R}$, $\dot{E}$, and $\ddot{E}$. The model now becomes:

$$\left(\frac{\sigma_\rho}{\rho}\right)^2 = G1 \cdot D1\,\sigma_R^2 + G2 \cdot D1\,\sigma_E^2 + G3 \cdot D2\,\sigma_R^2 + G4 \cdot D2\,\sigma_E^2 + G5\,\sigma_R^2 + G6\,\sigma_E^2 + G0\,\sigma_{W_Z}^2$$

$$\sigma_W^2 = G7 \cdot D1\,\sigma_R^2 + G8 \cdot D1\,\sigma_E^2 + G9 \cdot D2\,\sigma_R^2 + G10 \cdot D2\,\sigma_E^2 + G11\,\sigma_R^2 + G12\,\sigma_E^2$$

and can be simplified to be functions of the measured parameters, E and R.

The final model, the one used in the parametric study, then becomes

$$\left(\frac{\sigma_\rho}{\rho}\right)^2 = H1\,\sigma_R^2 + H2\,\sigma_E^2 + G0\,\sigma_{W_Z}^2$$

$$\sigma_W^2 = H4\,\sigma_R^2 + H5\,\sigma_E^2$$

where the H factors are functions of the G factors, and D1 and D2, the smoothing factors.

For the parametric study, all quantities except the smoothing interval were specified for the density error expression. The smoothing interval necessary to balance the equation was then computed. The same interval was then used to estimate the error in the horizontal wind. The number of data points, and therefore, the time was multiplied by the fall velocity to determine the spatial interval over which the estimate of the meteorological parameter was made.

8. THE CONTINUOUS WAVE TRACKER DESCRIPTION

The presence of a transponder on the sphere would permit two significant changes to be made in the fundamental structure of the system. First, the system could operate in a fairly simple continuous wave (CW) mode, thus avoiding having to produce high-powered, short-duration pulses (and, subsequently, process return pulses). Second, an on-board transmitter means that propagation power losses are proportional to the square of the distance rather than to its fourth power signal. The overall block diagram is shown in Figure IV-9.

For simplicity of operation, an interferometer would be chosen for measuring the two angles defining the line of position to the lofted package. This may be implemented with five receiving antennas, three on each of two orthogonal baselines (one antenna being common to both baselines). An offset transmitting antenna is commonly employed. The length of the baseline (the distance between outer antennas along the baseline) is determined by the number of wavelengths necessary to achieve the required angular accuracy. The third is located to permit resolution of the ambiguous angle determined from the first two. Both the receiving antenna configuration and the offset transmitting antenna will introduce errors due to parallax which can be computed and offset.

The transmitted signal is continuous wave, modulated with a series of tones. The double slant range is directly proportional to the phase shift on each frequency and is determined by direct phase

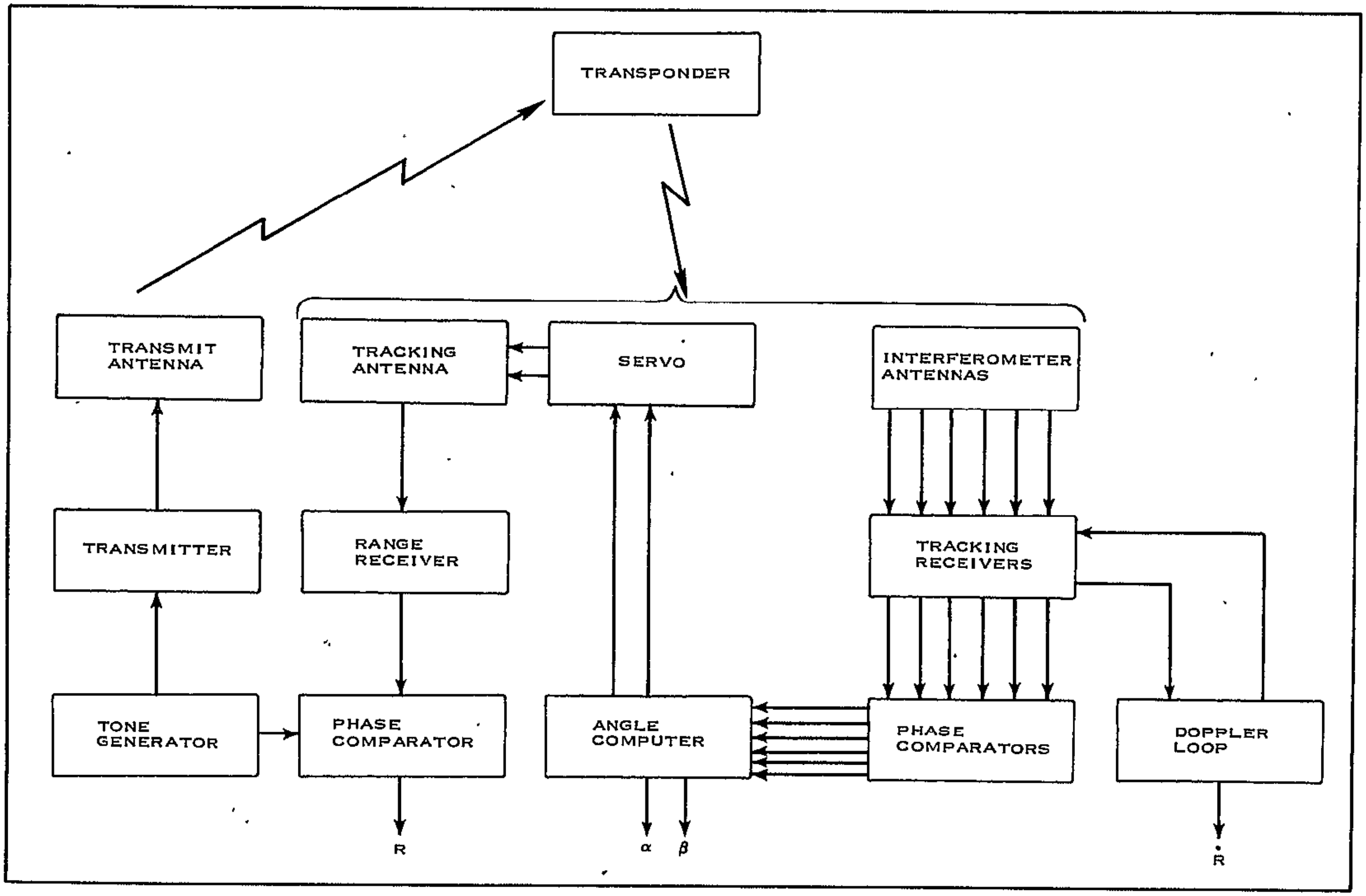


FIGURE IV-9. Block Diagram—Continuous Wave Tracker comparision of transmitted and received tones. The highest frequency used is dictated by the accuracy with which the phase shift may be measured to obtain the required range accuracy. The lower frequencies are used to resolve the range ambiguities.

The transponder in the lofted package must perform three functions:

- Receive the ranging tones on the up-link
- Maintain a constant phase shift on these tones
- Transmit the ranging tones on the down-link

Since two carrier frequencies are involved, the transponder must originate one of them. There are three general possibilities:

- A free-running oscillator at the transmitter frequency
- A frequency offset oscillator
- A frequency multiplier

The specific configuration will be dictated in part by the location of authorized channels in the frequency spectrum. Several current systems use up-links around 400 MHz and by selecting a down-link in the 1600 MHz telemetry band which is an exact fourth harmonic of the up-link may use a frequency quadrupler. In order to reduce propagation anomolies which vary with the inverse square of frequency, it would be desirable to select an up-link in the L-Band, and its second harmonic in the S-Band as the down-link.

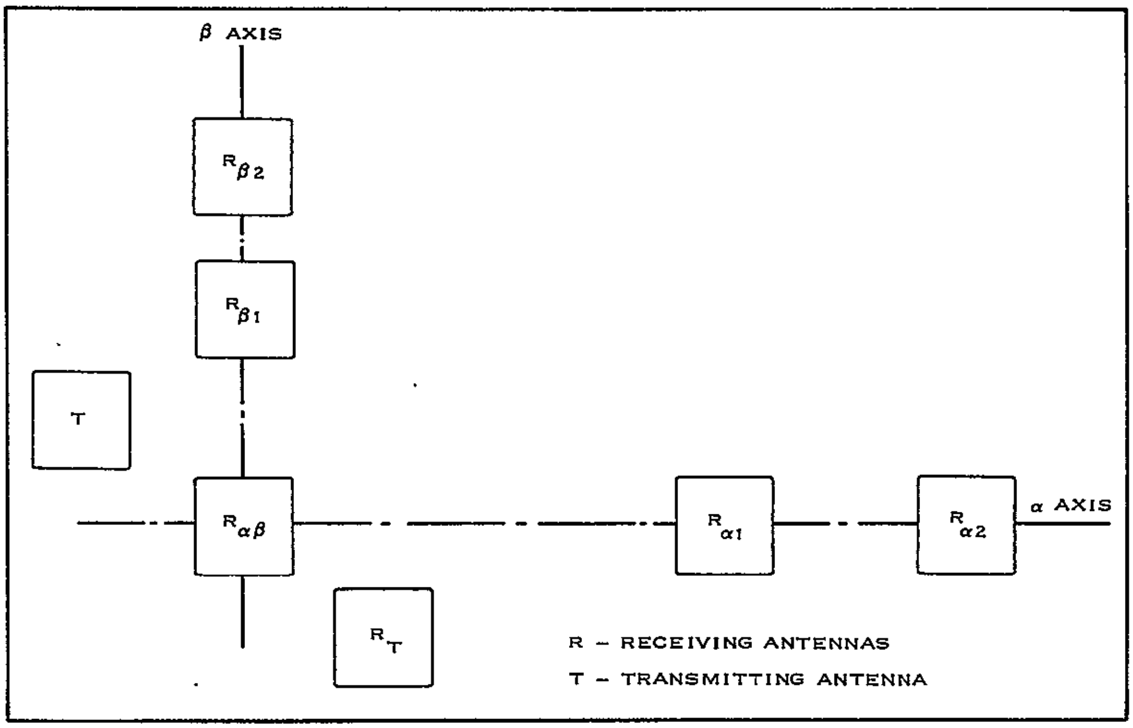
The antenna field layout is shown in Figure IV-10. The overall block diagram is shown in more detail in the ground station block diagram, Figure IV-11 and the transponder block diagram, Figure IV-12.

The accuracy requirements on this tracker are identical to those on the previously described phased-array. However, the cost in ground equipment to achieve these accuracies is significantly reduced by the following considerations:

- Transmitter power is provided continuously so that peak power equals average power.
- Pulse shaping networks are not required.
- Antenna-array steering subsystem is not required.
- No target identification is required at the receiver.
- Doppler measurement is continuous, not sampled.
- Receiver bandwidths may be much narrower.

One offsetting complication does exist. If high-gain antennas are used in ground station, they must be steered and angular accuracy is degraded due to shifting of phase centers on the baseline. If wide angle antennas are used, significant increases in power on both up- and down-links are required. A reasonable compromise is possible by using a relatively powerful ground station transmitter with a broad

FIGURE IV-10
Antenna Field Layout



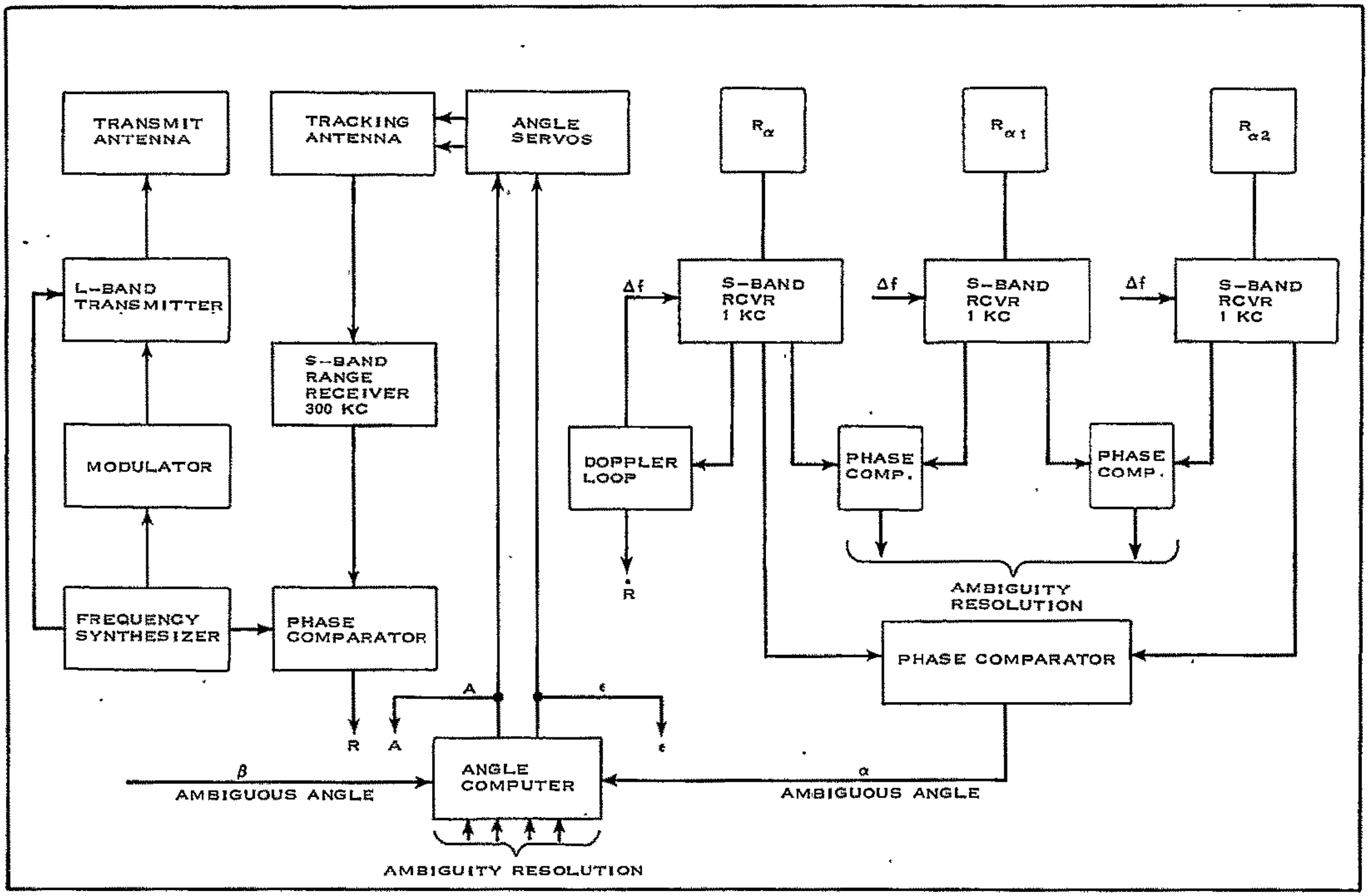
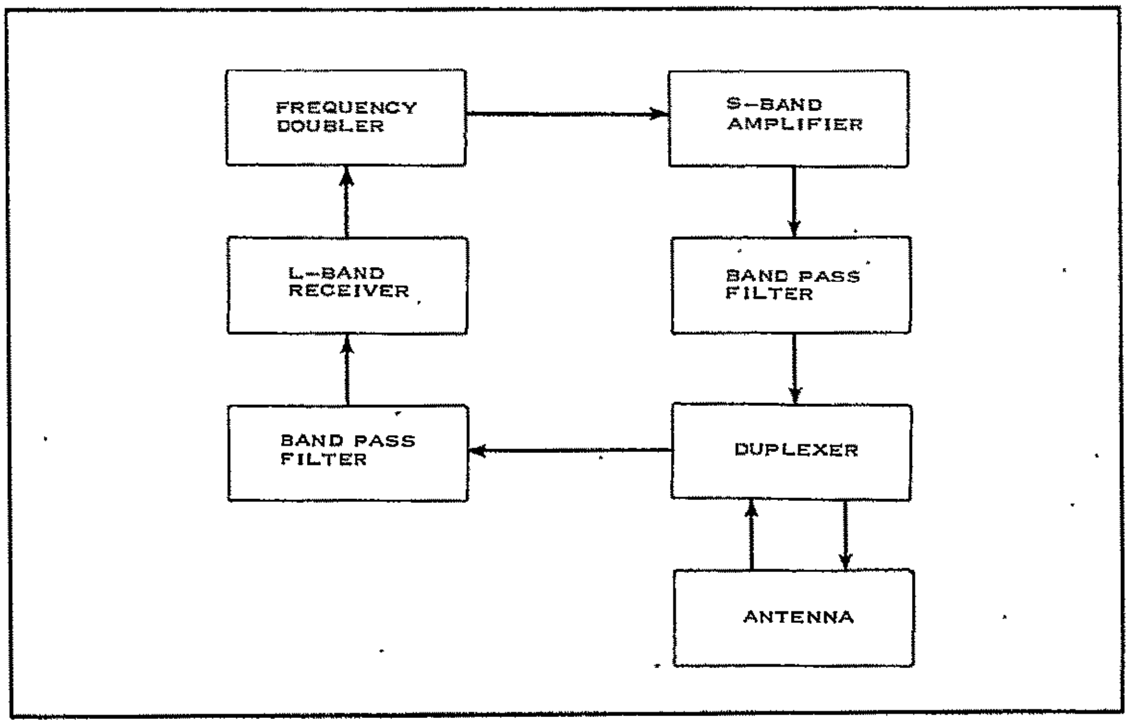


FIGURE IV-11. Block Diagram--Ground Station

FIGURE IV-12
Block Diagram--Transponder



beam antenna to provide 20 db S/N at the lofted receiver, and by using a composite antenna system for the ground receivers. Since the ranging tones require a 300 KHz bandwidth, a narrow-beam, high-gain antenna is required and a tracking antenna is unavoidable. However, a 1 KHz bandwidth tracking receiver is adequate to pass the carrier when the receiver center frequency follows Doppler shift. Thus, the dilemma on received signal-to-noise is resolved by slaving the range antenna to the angle resolver, and the interferometer receivers to the Doppler loop.

9. COST TRADE-OFF

As shown in the preceding section, there is a complete family of solutions to the tracker/data processing requirement. Appendix A shows the results of allowing the density error to assume a series of limits. Appendix B is a set of parametric studies designed to show the relative system sensitivity to variations in selected parameters and their limits. From a cost consideration, the allowable angular error is the critical factor, but the cost versus accuracy curve is far from linear. The fundamental accuracy limit is a direct function of the beamwidth and an inverse function of the signal-to-noise power ratio. For reliable, predictable data, the latter must be held above some discrete limit, but the former is subject to more complex manipulation. There are at least four ways in which significant cost reductions may be realized with some concurrent degradation in angular accuracy:

- Parasitic elements in the array
- Power splitting to feed subarrays
- Fewer elements in the array
- Lower precision in element alignment.

All of these can, to some extent, be compensated by:

- Redundancy in raw data
- Adaptive smoothing routines
- More sophisticated calibration routines
- More specific delineation of the density versus altitude curve requirements.

With a given block of data the apparent accuracy of the calculated density can be raised to successively higher levels but with a concurrent degradation of the accuracy of the concomitant altitude. Therefore, a more detailed knowledge of the significant altitude increments is in order.

Of the four identified methods of reducing array costs, lower precision in element alignment is perhaps the most difficult to compensate since it would require the use of fixed camera data with its consequent expensive processing. In those cases where the misalignment is not stable with time, calibration would tend to be ineffective. The other three methods all result in some degradation of the beam shape, but are amenable not only to compensation by calibration, but, if properly designed, to subsequent upgrading of raw data quality by hardware additions. Use of parasitic elements is an attractive method of holding down costs, since the cost of an active element is 100 times that of a dummy and later substitution of an active one is readily accommodated. Power splitting can be accomplished in several ways, some of which need not degrade accuracy, but those which are most effective in reducing costs will lead to some degradation in beam shape and in uniformity of steering increments. Some compensation by calibration is again possible. The most attractive method would be to design for the full array using parasitic elements and/or power splitting, but initially implementing only a selected segment of the array. Operation would then be possible with degraded accuracy, and the quality of the array upgraded as additional funds became available.

The funding requirements are based on a tracker/data processor having a range uncertainty of 5 meters, an angular uncertainty of 0.2 milliradians, and the data processing required for calibration, operation and data manipulation. Initial implementation with 25% of the array modules would result in a roughly comparable degradation in accuracy which would be fully recoverable at a later date. The cost saving would not, however, be proportional.

V. CANDIDATE SOUNDING SYSTEMS

V. CANDIDATE SOUNDING SYSTEMS

The previous chapters have provided discussions of each of the various subsystems (sensors, launch vehicles and telemetry/tracking) that constitute the major portion of a sounding system. An analysis in Chapter III - Launch Vehicle Analysis, advances the rationale for various combinations of payloads and launch vehicles.

This chapter describes six candidate sounding systems made up of six payloads and three launch vehicle combinations and the attendant group equipment, i.e., telemetry/tracking system that have received detailed cost analysis.

These candidate systems are:

(1) A passive 1-meter sphere, a gun projectile launch vehicle (7-inch gun bore, 3-inch subcaliber projectile), a phased-array tracking radar. (See Figure V-1)

The projectile containing the sphere and inflation/ejection mechanism will be launched with sufficient velocity in order that the sphere, which is inflated/ejected at 85 kilometers on the up-leg of the flight, will coast to an apogee of 120-130 kilometers. The sphere will be tracked from 85 kilometers to apogee

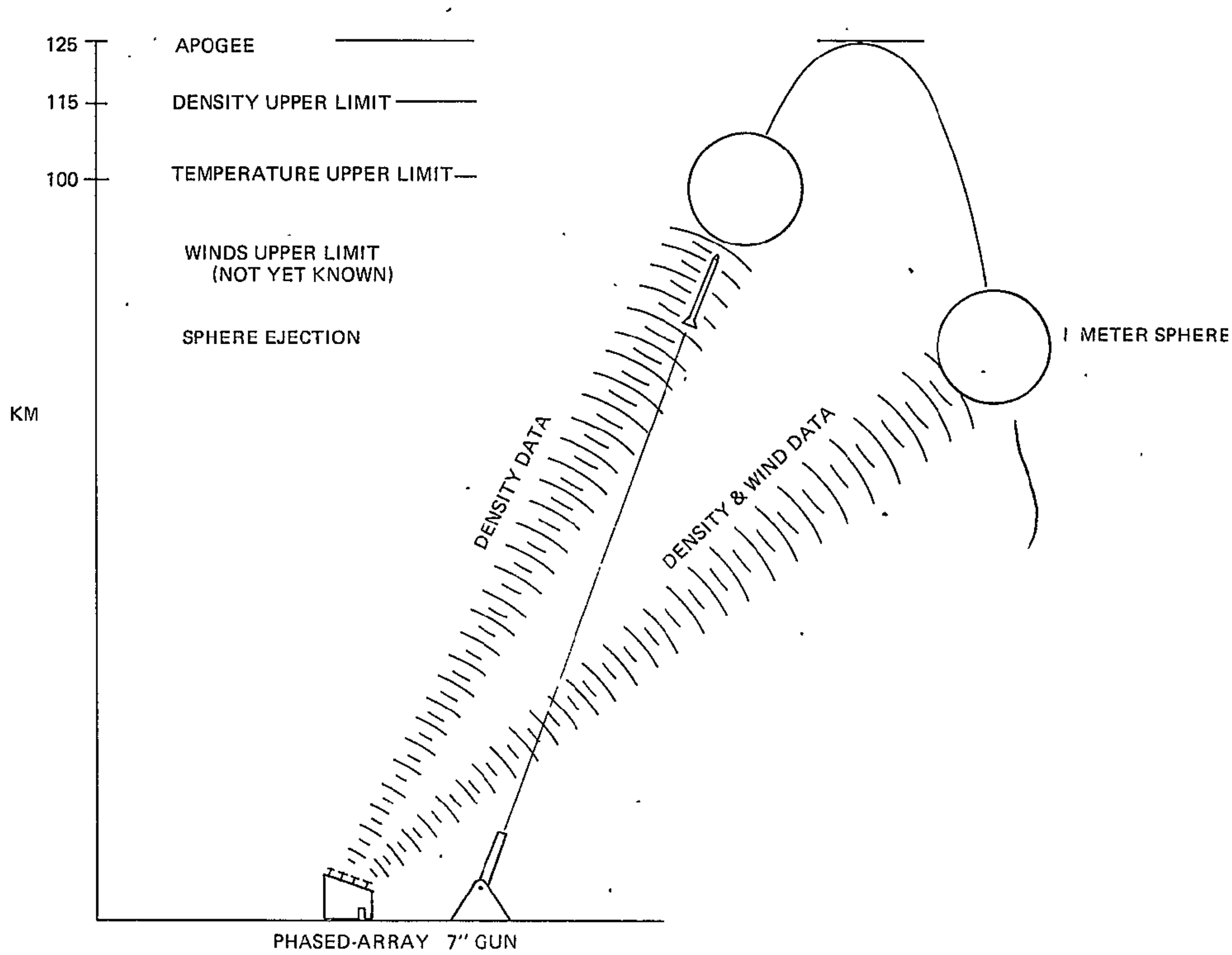


FIGURE V-1. Passive Sphere System

and on the down-leg to collapse near 30 kilometers. The tracking data will provide density/pressure and temperature data over nearly all of the 100-30 kilometer range. Wind data will be available for the range of approximately 85 to 30 kilometers.

(2) A 2-meter passive sphere with a transponder, a rocket-boosted-dart vehicle, an interferometer-type tracking system. (Figure V-2)

This candidate system differs from (1) in that a transponder is available in the sphere to reduce the requirements of the tracking subsystem and (2) the launch vehicle is changed from a gun to a rocket-boosted dart.

(3) A passive sphere, one canister of chaff, a rocket vehicle, a phased-array tracking radar. (Figure V-3)

The sphere provides the same capability as in candidate (1). The chaff provides wind measurement in the 80-90 kilometer region using the same radar as to track both the chaff and the sphere.

(4) A Spinning Wire Densitometer (SWD), a thermistor/parachute and chaff, a rocket-launch vehicle, a phased-array radar and two telemetry ground stations. (Figure V-4)

This alternative uses the SWD to make density measurements from approximately 100 km down to 50 km on its free-fall

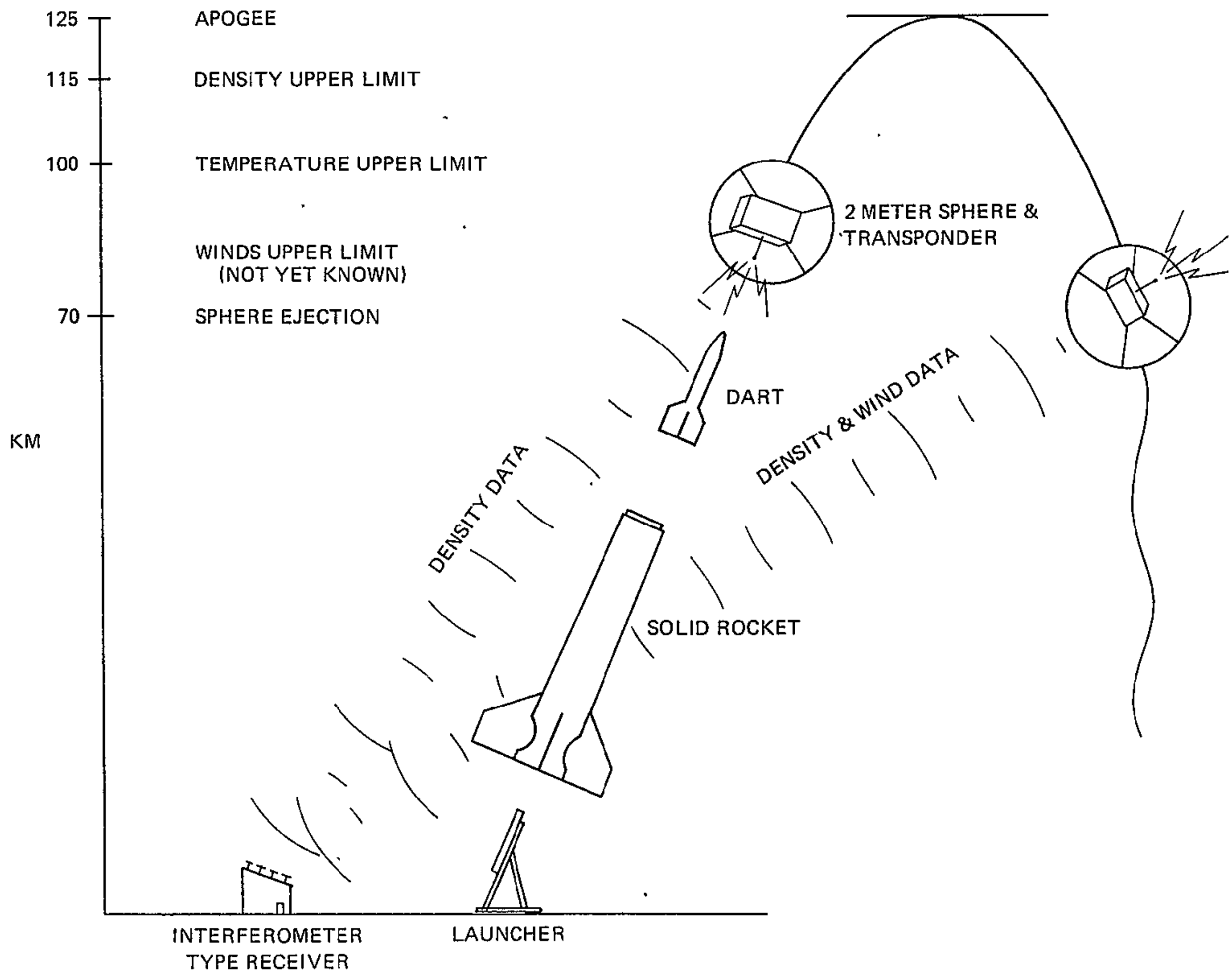


FIGURE V-2. Transponder Sphere System

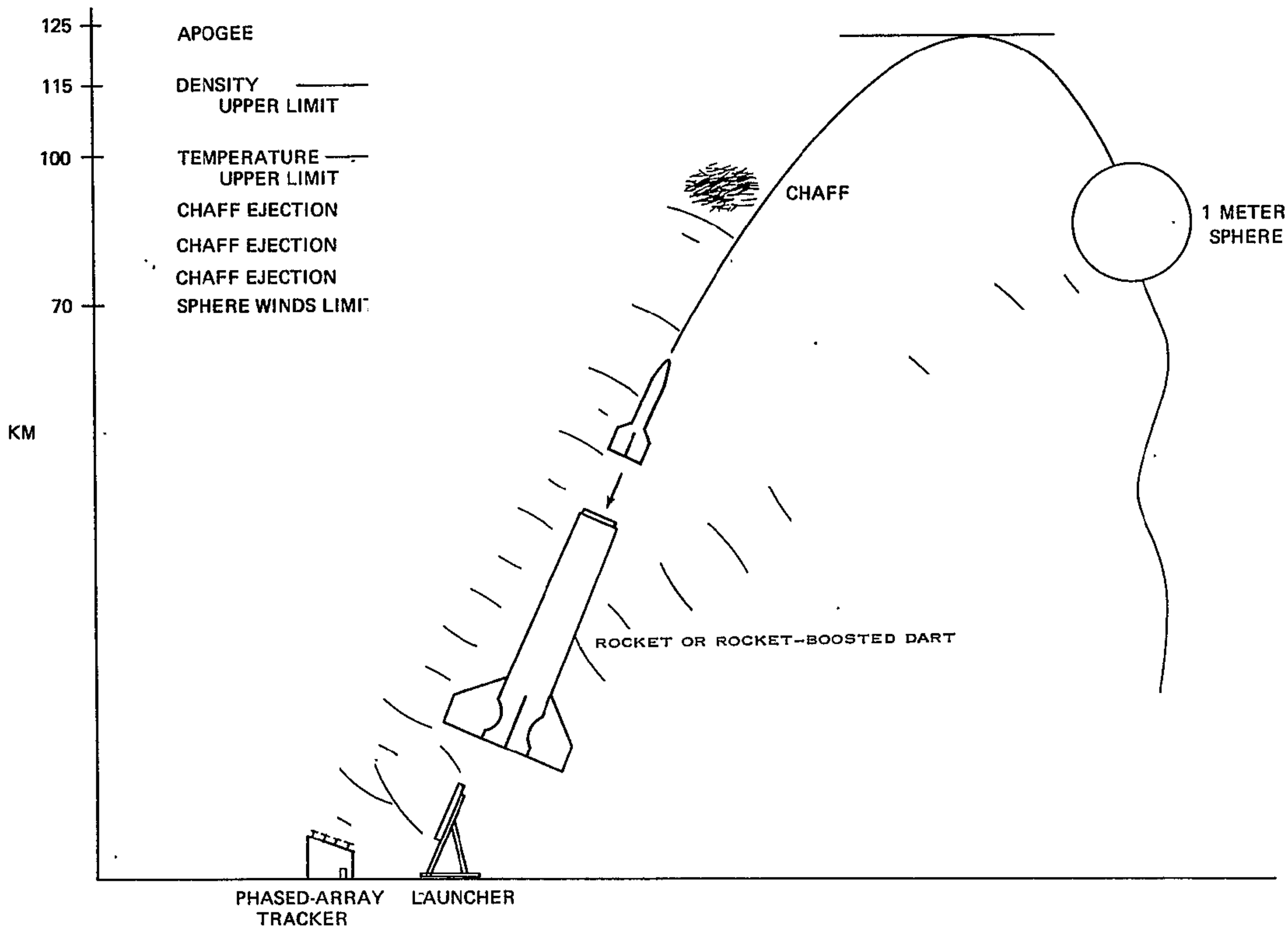


FIGURE V-3. Sphere & Chaff System

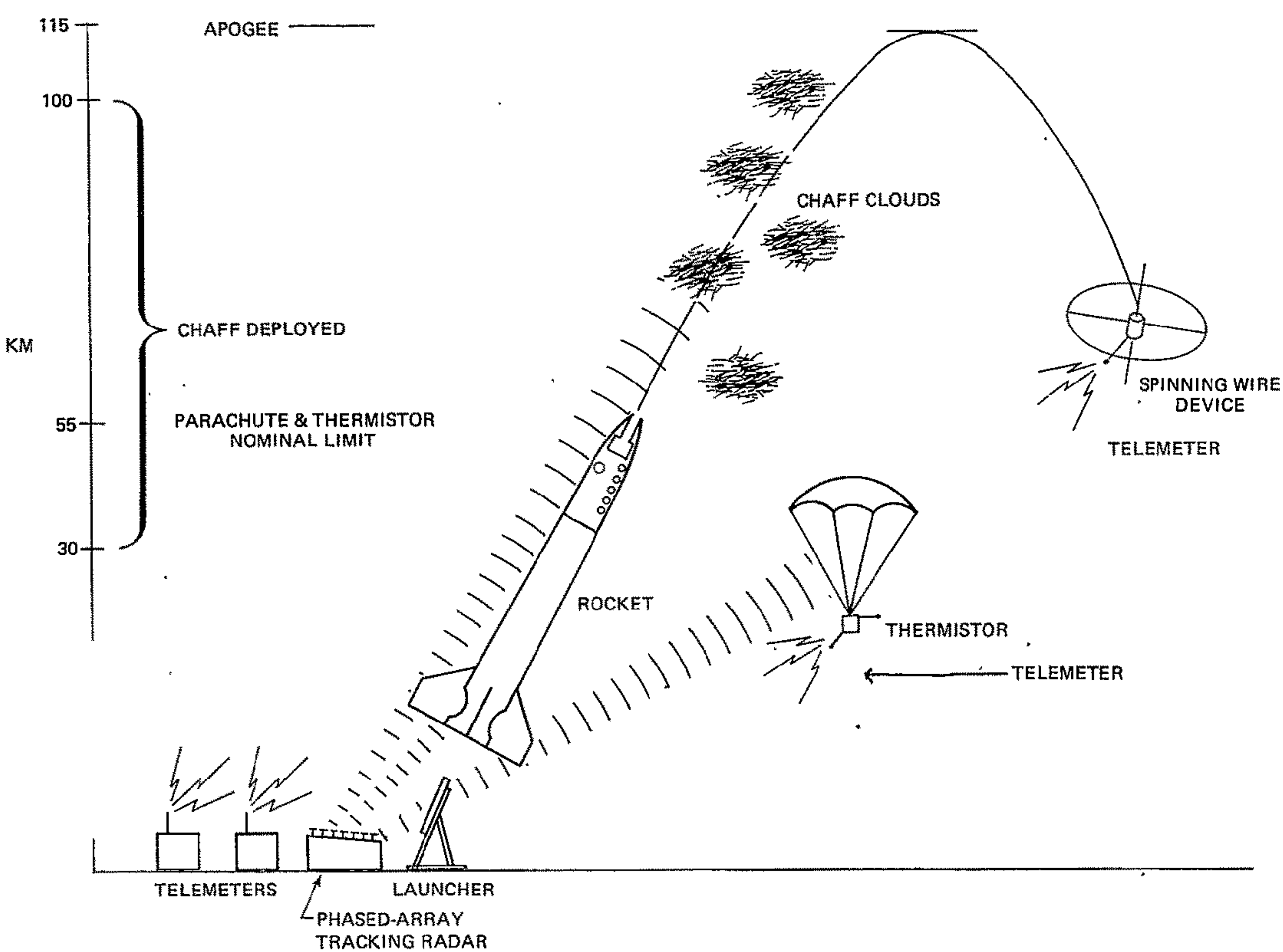


FIGURE V-4. Spinning Wire, Chaff, Thermistor/Parachute System descent to earth. Skin tracking of the SWD takes place on the upward leg to apogee and the down-leg altitude versus time profile will be determined empirically; therefore, radar tracking during descent is not required. The thermistor/parachute is ejected at approximately 60 km to measure/telemeter temperature data on its retarded descent to earth. The chaff is used over the entire range (30-90 km) as in alternative (4).

(5) <u>A molecular fluorescence densitometer (MFD), a thermistor/parachute, chaff, a rocket-launch vehicle, a phased-array tracking radar and two telemetry ground stations.</u> (Figure V-5)

In this alternative, the MFD becomes operational at about 60 km altitude on the ascent and makes density measurements to above 100 km. The parachute/thermistor device is ejected at this same 60 km altitude and makes temperature measurements and telemeters the results on its descent to earth (wind measurements are <u>not</u> made on the parachute/thermistor package). These two sensors then cover the range (30-100 km) for density/pressure and temperature data. Winds are obtained over the 30 to 90 km range by releasing chaff at five altitudes (nominally 30, 45, 60, 75, and 90 km) on the ascent leg and tracking the chaff cluster with the phased-array radar.

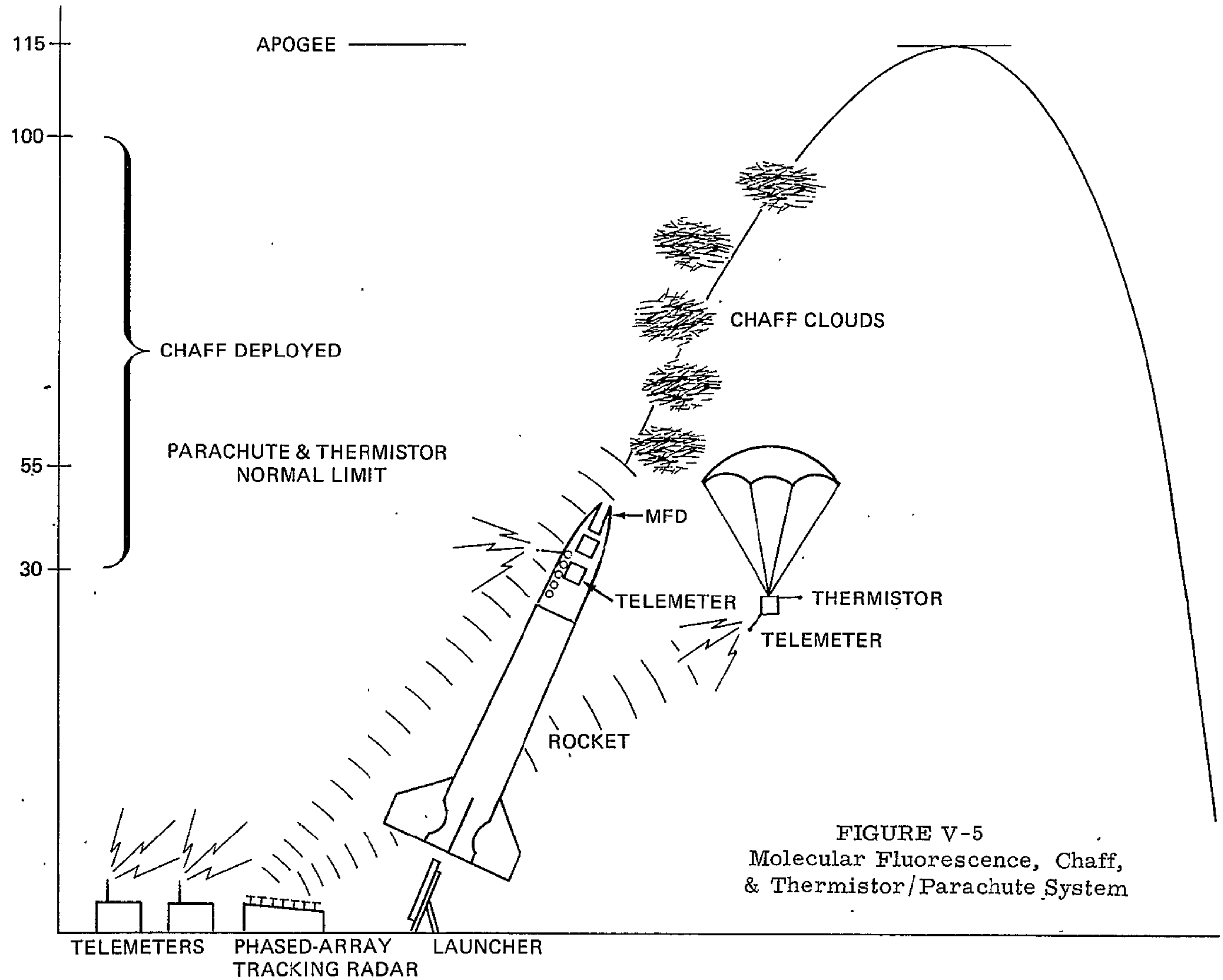


FIGURE V-5
Molecular Fluorescence, Chaff,
& Thermistor/Parachute System

The radar also skin tracks the thermistor's parachute for altitude information and skin tracks the payload on the up-leg for altitude for the MFD measurements.

(6) A pitot system, a thermistor/parachute and chaff, a rocket-launch vehicle, a phased-array tracking radar and two telemetry ground stations. (Figure V-6)

In this alternative, the pitot system makes density (pressure) measurements on the ascent-leg from approximately 70 to greater than 100 km. The thermistor/parachute is ejected at about 70 km and telemeters temperature data on its retarded descent to earth. Chaff is used over the entire range for wind measurement as in alternative (4). As in alternative (4), the altitude of the thermistor/parachute and pitot system (during its up-leg sensing) are determined by the phased-array radar.

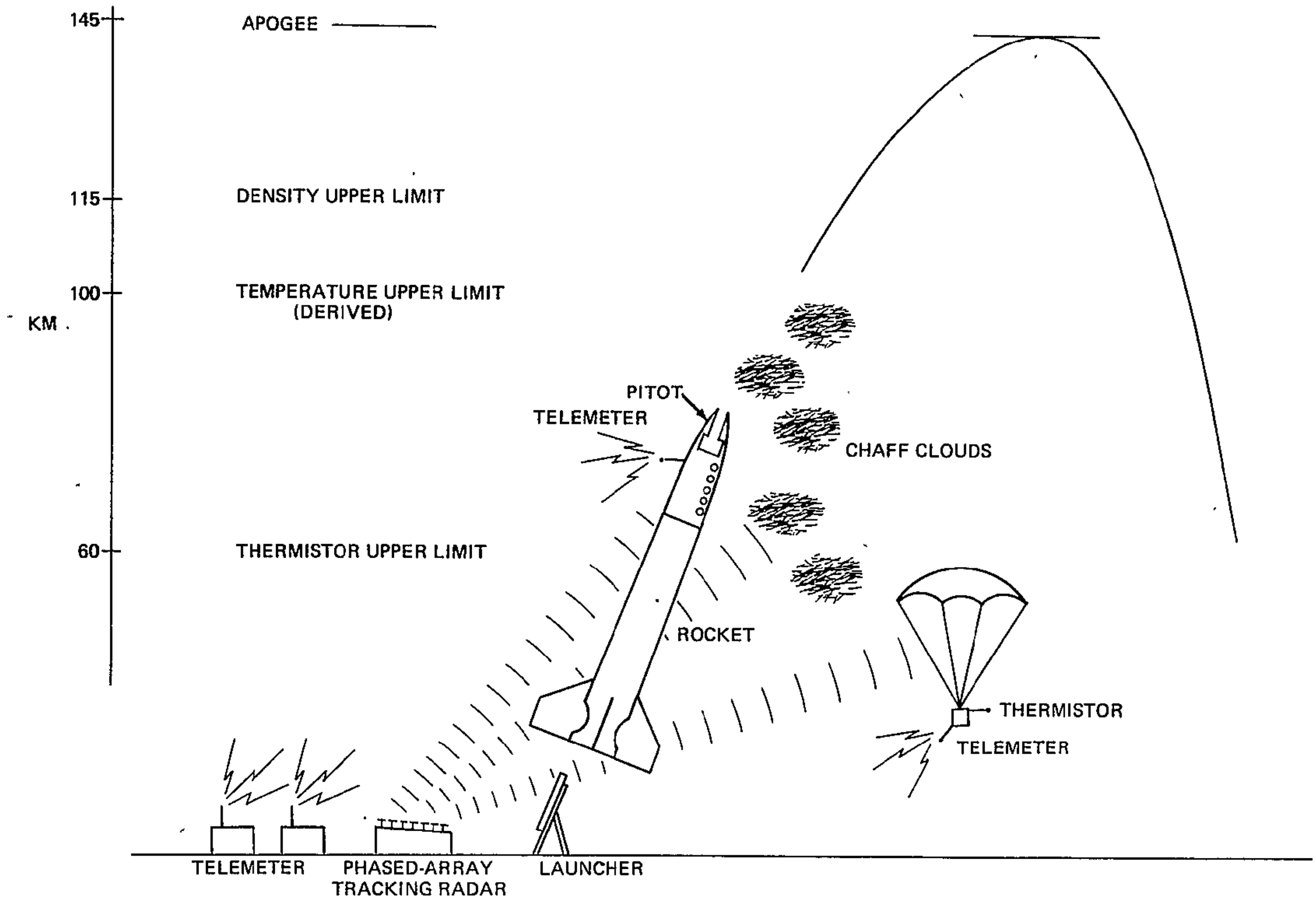


FIGURE V-6. Pitot, Thermistor/Parachute Chaff System

VI. COST ANALYSIS

VI. COST ANALYSIS

1. COST ANALYSIS

The purpose of this analysis is to provide cost data for each of the six sounding systems discussed previously in order to give visibility to differences in costs between techniques. These cost differences will be used in a trade-off analysis to assist in identifying the optimum system(s) in terms of cost and performance.

Figure VI-1 depicts a typical organization for the meteorological sounding system. This diagram shows field installations below the dotted line and administrative and support facilities above the line. The alternatives being considered are all variations in sounding techniques at the field installation level. The administration and support requirements are assumed to be the same for all options. The cost analysis, therefore, addresses only below-the-line (or field installation) costs. The above-the-line costs are not relative to the trade-off analysis and have not been included.

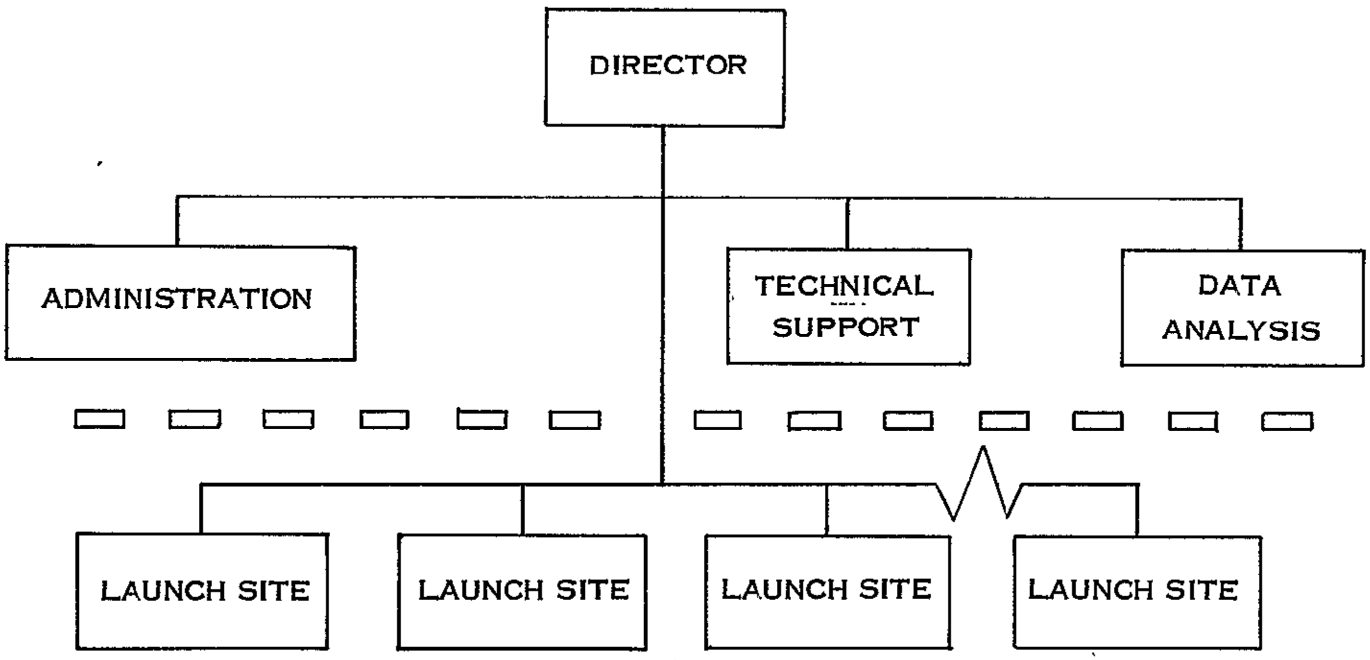


FIGURE VI-1. Meteorological Sounding System

2. METHOD OF ANALYSIS

Each sounding system option has been analyzed from the standpoint of three categories: R&D Costs, Acquisition (or Investment) Cost and Annual Operating Costs. All costs which are traceable to the field installation level are covered under one of these categories. Costs have been computed based on a program of orderly development and procurement and 10 years of system operations.

R&D Costs are based on the current state of the art. They are incurred during FY1, FY2, and FY3 of the program. This recognizes the present level of technological development, upon which the postulated development programs will rest. R&D costs have been developed for the Tracking systems, Launch systems, and Payload systems. These cost estimates are based on several assumptions.

(1) NASA will develop a single complete meteorological sounding system including all of the elements of a cohesive, integrated system. This is opposed to a piecemeal development of a payload system and/or a separately funded launch system. Piecemeal development will result in a higher R&D cost and a less effective overall system.

(2) R&D work is assumed to be accomplished prior to the construction and activation of the operational system. This is essential if NASA is to design the most effective overall system and to effect smooth system implementation. The acquisition schedule, Figure VI-2, assumes an orderly and planned system installation and activation.

Acquisition Costs are those expenditures necessary to build and equip the launch sites. These costs have been subdivided into two categories: Common costs, or those costs which are common to all operations, such as land, site development, etc., and Non-common costs, which are costs peculiar to a particular option. Acquisition costs are incurred over a 2-1/2 year period from FY4 through the second quarter of FY6.

Annual operating costs (or Recurring) are those costs incurred in the normal operations of the system and include such items as personnel and administration expenses, additional supplies of launch vehicles and payloads. These costs begin in FY4 and continue throughout the life of the program.

Figure VI-3 shows the cost model which has been used to establish the methodology for developing cost estimates for the candidate sounding systems. The model is constructed to provide R&D costs,

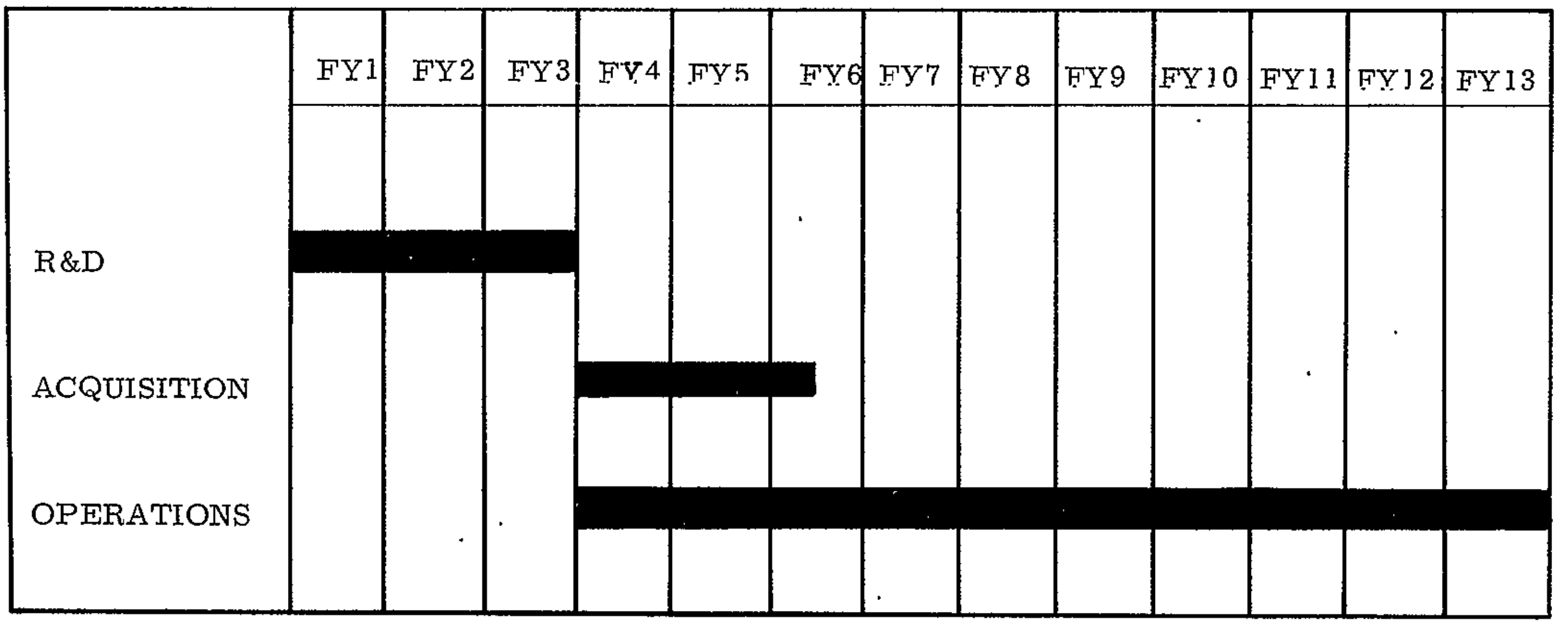
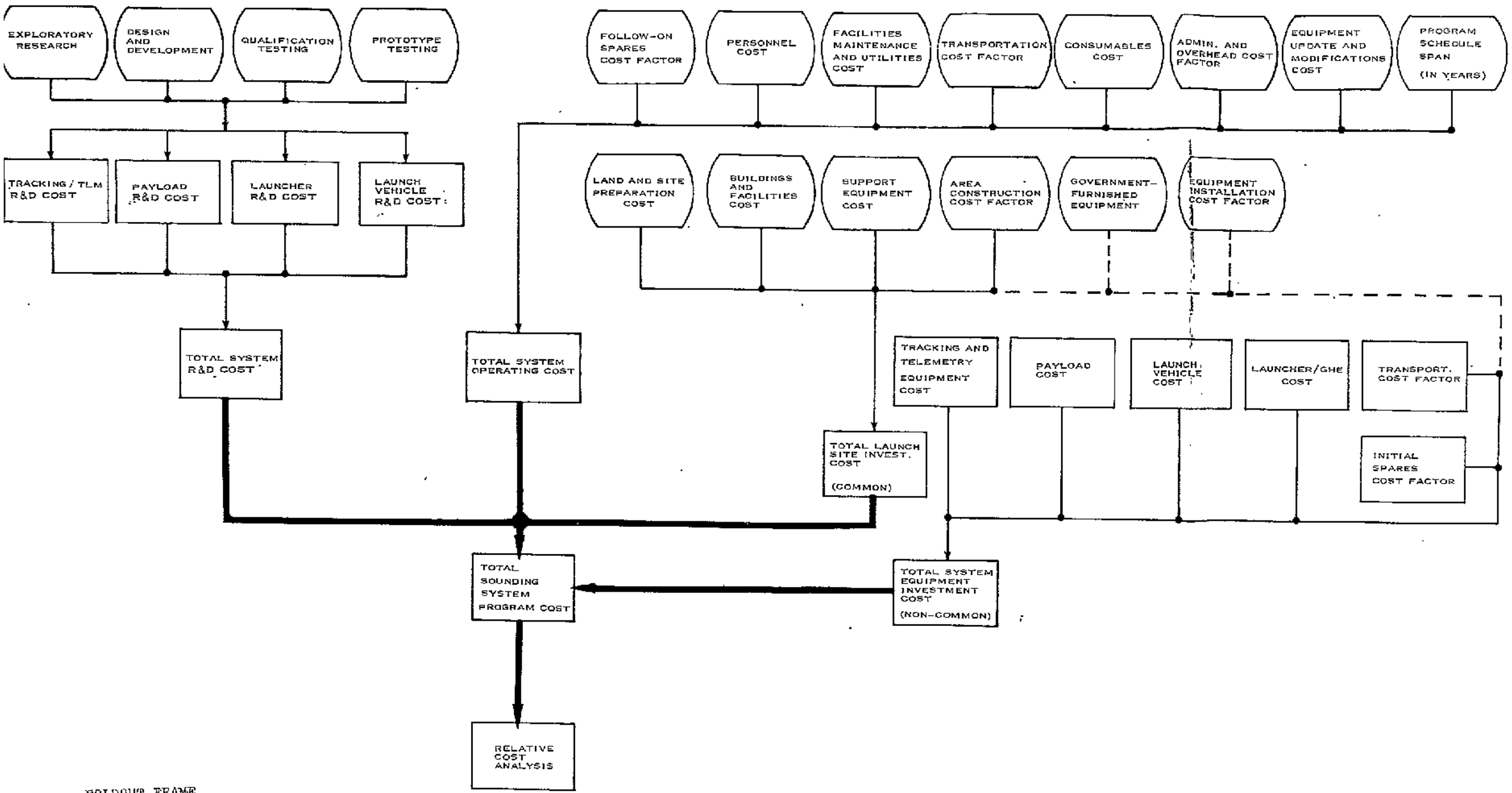


FIGURE VI-2. Meteorological Sounding System Program Schedule

FIGURE VI-3
Meteorological Sounding Systems
Cost Model



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VI-6 A

investment costs, and operating costs. These three cost categories are then summarized to present the total program funding requirements for each sounding system. In addition, they are used in relative cost analyses which give an insight into the amortization of investment costs and the significant cost-benefits of each system.

Total program costs are computed for each option by summing R&D costs, total acquisition costs, and the total operating expense incurred over the life of the program.

3. COST ANALYSIS SUMMARY

Figure VI-4 is a cost comparison of the total program costs of each option over a 13-year program period—three years of R&D and 10 years of operation. The gun-launched/passive sphere system (Option 1) has the lowest estimated program costs of $279 million. The transponder sphere/boosted dart system (Option 2) and the passive sphere/chaff/rocket system (Option 3) have program costs of approximately $359 million and $303 million, respectively. Option 4 which uses chaff, spinning wire densitometer, a thermistor/parachute and a rocket has a program cost of approximately $438 million. Option 5 which replaces the spinning wire densitometer with the molecular fluorescence device has a program cost of approximately $541 million.

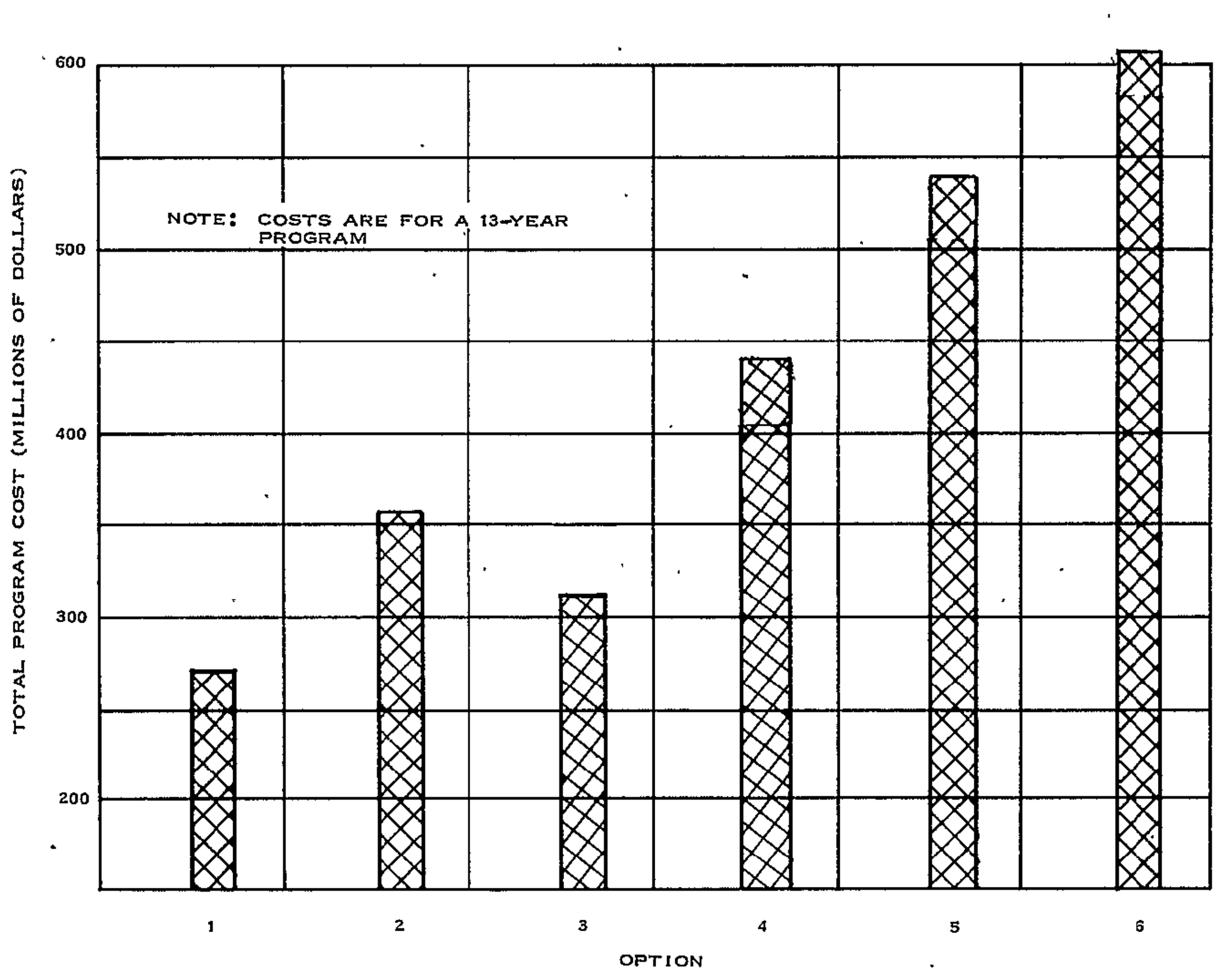

FIGURE VI-4. Cost Comparison Total Program Costs by Option

Option 6 is the same as Option 4 except the spinning wire densitometer is replaced with a pitot sensor. Option 6 has a program cost of approximately $618 million.

Using these costs as a basis, the cost of a single sounding using Option 1 (with R&D and acquisition amortized over 100,000 launchings) over the 13-year program is $2,788 as compared to the per shot cost of Option 6 of $6,177. Table VI-1 is a summary of amortized costs which shows the cost-per-sounding over the full 13-year program, and the cost-per-sounding based on steady-state operations. The latter cost is applicable to a fully amortized system, and it represents full system steady-state operating costs divided by 10,000 launchings per year. These costs are shown graphically in Figure VI-5.

Table VI-2 is a summary of the estimated unit costs for launch vehicles and payloads by option.

Table VI-1
Cost Per Sounding

OPTION*	Launch Costs Based on Annual Operations ($)	Launch Costs Based on Total Program Costs ($)**
1	1,483	2,788
2	2,545	3,586
3	1,835	3,031
4	3,084	4,384
5	4,114	5,407
6	4,887	6,177

*See Chapter V for definition of options
**Includes total R&D, Acquisition and Operating costs for a 13-year program

FIGURE VI-5
Cost Comparisons for
Amortizing 100,000 Soundings

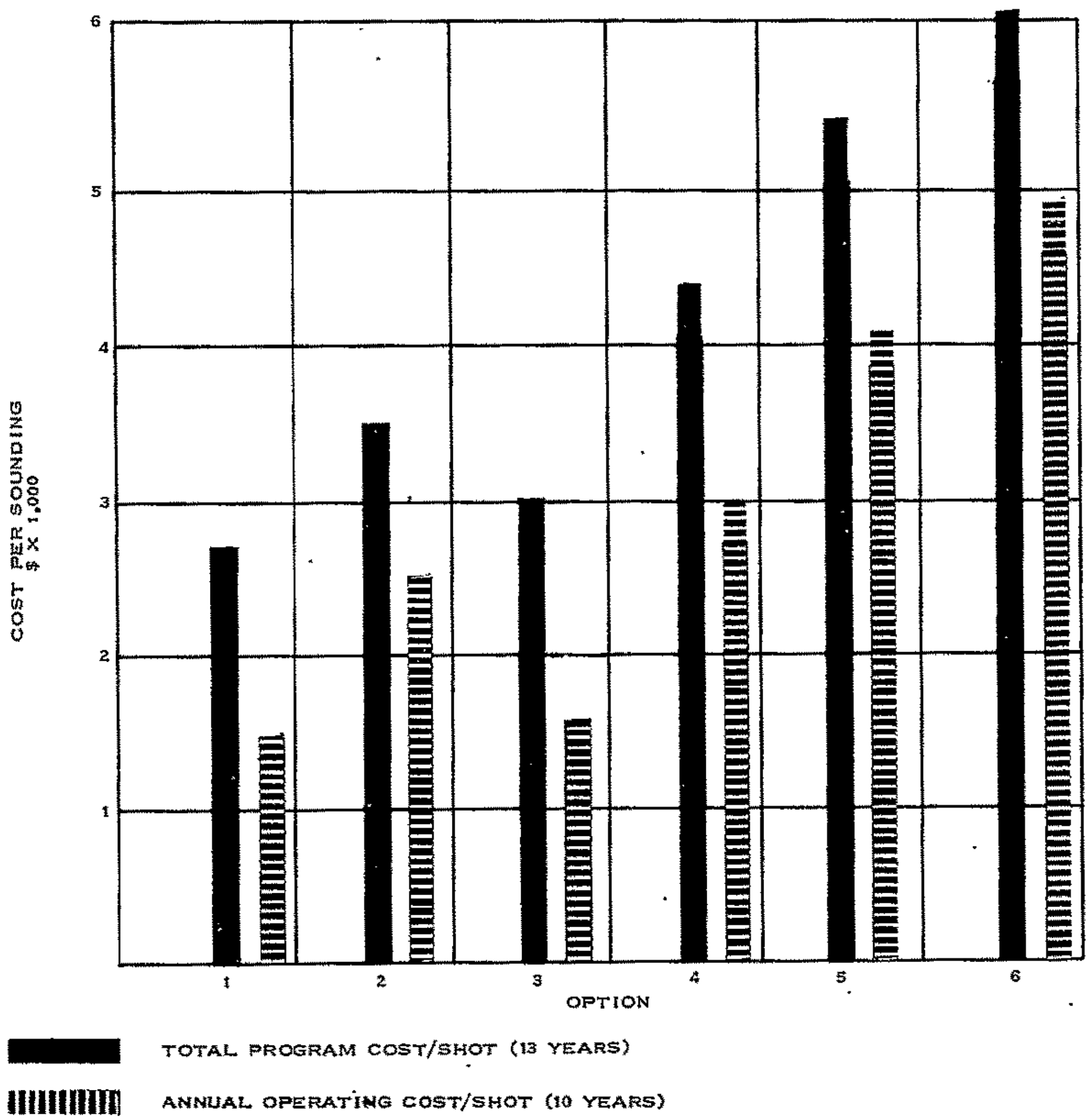

Table VI-2

Unit Costs for Launch Vehicle and the Payload by Option ($)

ITEM	OPTION*					
	1	2	3	4	5	6
Launch Vehicles	347	995	612	1,224	1,020	1,785
Payload	87	530	170	755	1,979	1,979
Total Expendable Cost/Launch	434	1,525	782	1,979	2,999	3,764

A review of Table VI-2 shows that there are large differences in launch and payload costs. The payloads for Options 5 and 6 cost approximately 20 times more than the payload for Option 1. In terms of annual costs, at the rate of 10,000 soundings per year, Options 5 and 6 have a payload incremental cost difference of $18.9 million above Option 1. In terms of launch vehicles, the annual incremental cost difference above Option 1 is approximately $8.8 million for Option 5, and $6.8 million for Option 6. With each launch vehicle/payload differential of $100, the annual cost difference is $1,000,000. The total expendable row of Table VI-2 shows the absolute minimum cost which can be associated with a sounding for each option. It will be observed that Option 6 is approximately five times more expensive than Option 3 and nine times more expensive than Option 1 from

*See Chapter V for definition of options

the standpoint of consumables per shot. A review of the table shows that these costs come closer together as ground equipment, R&D and site operations are amortized over each shot. Passive sensors (Table VI-3) require pulsed radars, which have relatively high R&D costs when compared to the active devices (such as the transponder sphere, Option 2).

Table VI-3
Tracking System R&D and Units Cost Per Option

ITEM	OPTION (x $1,000)*					
	1	2	3	4	5	6
Tracking System R&D	22,300	8,000	22,300	22,300	22,300	22,300
Tracking System Unit Costs	700	700	700	800	800	800

*See Chapter V for definition of options

Figure VI-6 is a graphic presentation of estimated annual operating costs by option.

Tables VI-4 through VI-9 are the detailed developments of total program costs for each option, showing estimated annual R&D, acquisition, and operating expenses.

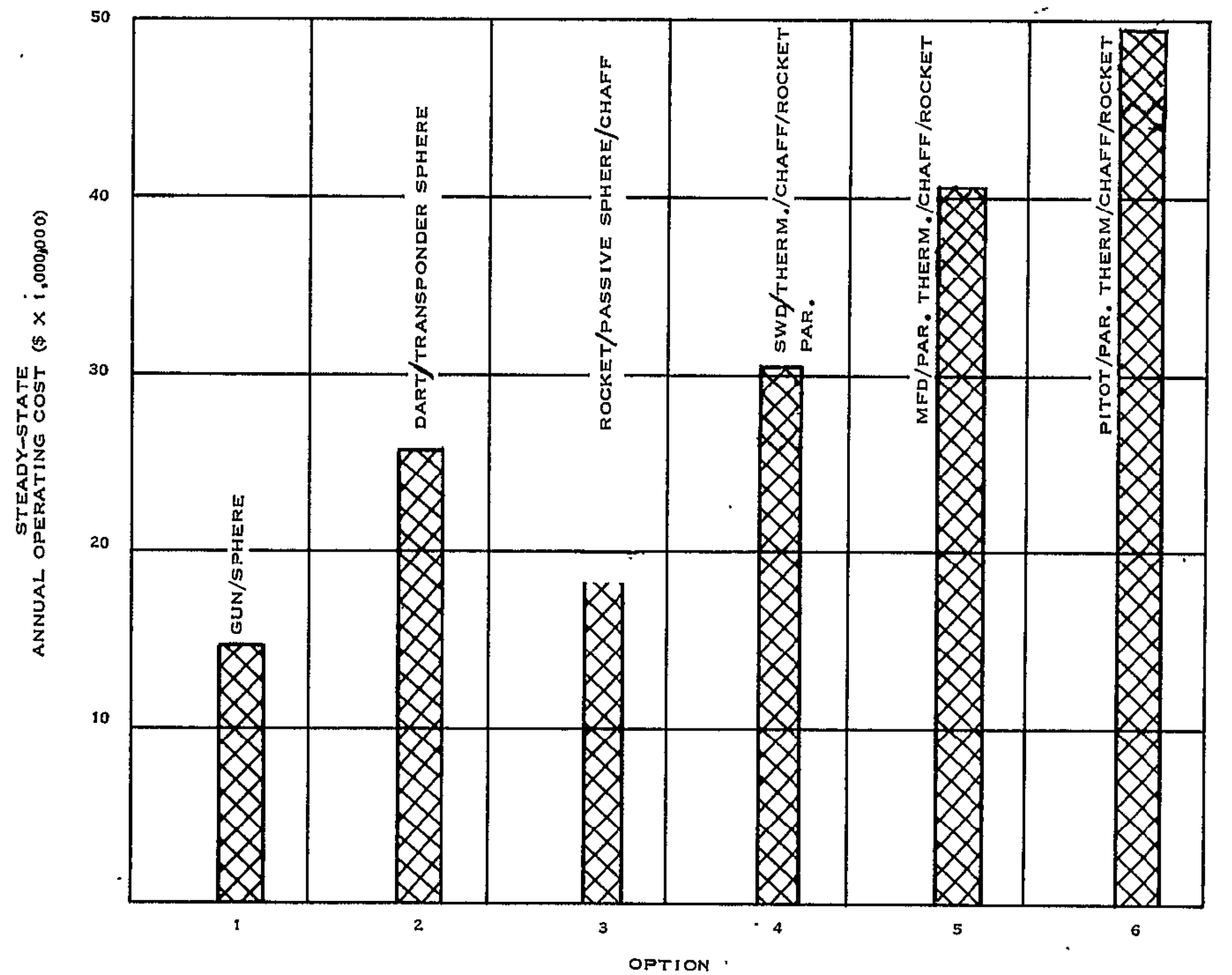

FIGURE VI-6. Cost Comparison Annual Operating Costs

Table VI-4
Preliminary Cost Estimates by Option
Gun/Sphere (Option 1)

COST ELEMENT	THIRTEEN–YEAR PROGRAM COST ($ X 1000)												
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
R&D	9,872	9,872	4,933	--	--	--	--	--	--	--	--	--	--
Acquisition	--	--	--	48,393	47,212	23,606	--	--	--	--	--	--	--
Operations	--	--	--	5,897	10,663	14,534	14,832	14,832	14,832	14,832	14,832	14,832	14,832
Annual Total	9,872	9,872	4,933	54,290	57,875	38,140	14,832	14,832	14,832	14,832	14,832	14,832	14,832
Cum. Total	9,872	19,744	24,677	78,967	136,842	174,982	189,814	204,646	219,478	234,310	249,142	263,974	278,806

Table VI-5
Preliminary Cost Estimates by Option
Dart/Transponder Sphere (Option 2)

COST ELEMENT	THIRTEEN–YEAR PROGRAM COST ($ X 1000)												
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
R& D	4,104	4,104	2,055	--	--	--	--	--	--	--	--	--	--
Acquisition	--	--	--	48,520	47,336	23,668	--	--	--	--	--	--	--
Operations	--	--	--	8,334	17,464	24,881	25,453	25,453	25,453	25,453	25,453	25,453	25,453
Annual Total	4,104	4,104	2,055	56,854	64,800	48,549	25,453	25,453	25,453	25,453	25,453	25,453	25,453
Cum. Total	4,104	8,208	10,263	67,117	131,917	180,466	205,919	231,372	256,825	282,278	307,731	333,184	358,637

Table VI-6
Preliminary Cost Estimates by Option
Rocket/Passive Sphere/Chaff (Option 3)

COST ELEMENT	THIRTEEN-YEAR PROGRAM COST ($ X 1000)												
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
R&D	9,567	9,567	4,784	--	--	--	--	--	--	--	--	--	--
Acquisition	--	--	--	45,883	44,764	22,382	--	--	--	--	--	--	--
Operations	--	--	--	6,802	12,960	17,963	18,347	18,347	18,347	18,347	18,347	18,347	18,347
Annual Total	9,567	9,567	4,784	52,685	57,724	40,345	18,347	18,347	18,347	18,347	18,347	18,347	18,347
Cum. Total	9,567	19,134	23,918	76,603	134,327	174,672	193,019	211,366	229,713	248,060	266,407	284,754	303,101

Table VI-7
Preliminary Cost Estimates by Option
Spinning Wire Densitometer/Thermistor/
Chaff/Rocket/Parachute (Option 4)

COST ELEMENT	THIRTEEN–YEAR PROGRAM COST ($ X 1000)												
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
R&D'	9,648	9,648	4,822	--	--	--	--	--	--	--	--	--	--
Acquisition	--	--	--	55,506	54,152	27,076	--	--	--	--	--	--	--
Operations	--	--	--	10,314	21,260	30,153	30,837	30,837	30,837	30,837	30,387	30,387	30,387
Annual Total	9,648	9,648	4,822	65,820	75,412	57,229	30,837	30,837	30,837	30,837	30,837	30,837	30,837
Cum. Total	9,648	19,296	24,118	89,938	165,350	222,579	253,416	284,253	315,090	345,927	376,764	407,601	438,438

Table VI-8
Preliminary Cost Estimates by Option
Molecular Fluorescence Densitometer/Parachute/
Thermistor/Chaff/Rocket (Option 5)

COST ELEMENT	THIRTEEN-YEAR PROGRAM COST ($ X 1000)												
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
R&D	9,820	9,820	4,908	--	--	--	--	--	--	--	--	--	--
Acquisition	--	--	--	59,688	58,232	29,116	--	--	--	--	--	--	--
Operations	--	--	--	12,966	27,992	40,200	41,139	41,139	41,139	41,139	41,139	41,139	41,139
Annual Total	9,820	9,820	4,908	72,654	86,224	69,316	41,139	41,139	41,139	41,139	41,139	41,139	41,139
Cum. Total	9,820	19,640	24,548	97,202	183,426	252,742	293,881	335,020	376,159	417,298	458,437	499,576	540,715

Table VI-9
Preliminary Cost Estimates by Option
Pitot/Parachute/Thermistor/
Chaff/Rocket (Option 6)

COST ELEMENT	THIRTEEN-YEAR PROGRAM COST ($ X 1000)												
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
R&D	10,048	10,048	5,022	--	--	--	--	--	--	--	--	--	--
Acquisition	--	--	--	62,825	61,292	30,646	--	--	--	--	--	--	--
Operations	--	--	--	14,955	33,041	47,735	48,866	48,866	48,866	48,866	48,866	48,866	48,866
Annual Total	10,048	10,048	5,022	77,780	94,333	78,381	48,866	48,866	48,866	48,866	48,866	48,866	48,866
Cum. Total	10,048	20,096	25,118	102,898	197,231	275,612	324,478	373,344	422,210	471,076	519,942	568,808	617,674

4. R&D COST ANALYSIS

Research and development (R&D) costs will be incurred for all of the candidate sounding systems. The magnitudes of these costs, as presented in this section, are based upon the following criteria.

- It is the intent of NASA to develop a meteorological sounding "system" which, as a system, has been fully developed and tested prior to the delivery of operational equipment and launch site hardware. Accordingly, the program schedule and cost estimates reflect the time and money necessary to meet this objective. Sunk costs have not been considered for any option.

- An engineering analysis has been performed to correlate the current technological status of system hardware with the estimated degree of advancement that must be achieved. The costs that will be incurred in the development of an effective sounding system from this baseline has been calculated for each option.

In order to present a comprehensive picture of R&D requirements, each affected end-item is subsequently discussed in terms of its schedule, cost elements, and costing logic. Tables VI-11 through VI-13 reflect this information for all major system equipment associated with each option.

R&D costs are summarized in Table VI-10. These cost factors reflect the total estimated R&D investment required to bring the individual subsystems to the production level of development. In the

Table VI-10
R&D Cost Development
(R&D Costs x $1,000)

OPTION*	LAUNCH SYSTEM	TLM AND TRACKING	PAYLOAD	TOTAL PER OPTION
1	2,277	21,500	900	24,677
2	1,413	8,000	850	10,263
3	1,413	21,500	1,005	23,918
4	1,413	21,500	2,205	25,118
5	1,413	21,500	1,635	24,548
6	1,413	21,500	2,205	25,118

*See Chapter V for definition of options

analyses of program costs, these costs are expensed at a straight-line rate over ten quarters during a 3-year period, from FY1 through the first two quarters of FY3. The last two quarters of FY3 are available to set up production, so that the first operational sites can be completed at the close of the first quarter of FY4.

Table VI-11 is a breakout of estimated R&D costs for the tracking systems. Table VI-12 is a cost breakout for Payloads systems by Option. Launch system R&D costs are summarized in Table VI-13 which is a detailed development of Rocket and Gun systems R&D costs. It is reasoned that R&D costs for all systems using rocket motors is the $1.413 million shown in Table VI-7. Thus, this cost is used for launch system R&D for Options 2, 3, 4, 5, and 6.

5. ACQUISITION COSTS

Acquisition costs are based on a total system requirement of 101 launch sites (100 operational sites plus one training site). Each site is assumed to launch 100 soundings per year.

Table VI-11
Tracking System
R&D Costs

PULSED RADAR SYSTEM (Option 1,3,4,5,6)	COST ($ x 1, 000)
Development	10, 000 ± 1, 000
Preliminary Design	1, 500 ± 200
Prototype Fabrication	4, 000 ± 300
Prototype Evaluation	1, 000 ± 200
Final Design	1, 000 ± 200
Production Engineering	4, 000 ± 300
Total	21, 500 ± 2, 200

CW RADAR SYSTEM (Option 2)	COST ($ x 1, 000)
Development	3, 000 ± 400
Preliminary Design	1, 000 ± 200
Prototype Fabrication	2, 000 ± 200
Prototype Evaluation	1, 000 ± 200
Final Design	1, 000 ± 200
Total	8, 000 ± 1, 200

Table VI-12
Payload R&D Costs

OPTION*	DESCRIPTION	COST ($1,000)	
1	Passive Sphere (Gun)		900
2	Transponder Sphere (Dart)		850
3	Passive Sphere (Rocket)	650	1,005
	Chaff	355	
4	Chaff	355	2,205
	Spinning Wire	1,750	
	Thermistor	100	
5	Chaff	355	1,635
	Molecular Fluorescence	1,180	
	Thermistor	100	
6	Chaff	355	2,205
	Pitot	1,750	
	Thermistor	100	

*See Chapter V for definition of options

Table VI-13
Launch System
R&D Costs

GUN SYSTEM DEVELOPMENT	COST (x $1,000)		
Preliminary Design	126		
Prototype Fabrication	375		
Test Firings	20		
Design	126		
Production Engineering	80		
Payload Integration	18		
Handling Equipment	62		
Subtotal		807	
Documentation		18	
Test and Checkout Equipment		202	
Subtotal			1,027
Contingencies - 50%			500
Company Expenses - 50% (G&A, P & OH)			750
Total R&D Cost			2,277

ROCKET SYSTEM DEVELOPMENT	COST (x $1,000)		
Preliminary Design (Motor)	72		
Prototype Fabrication	50		
Test Firings	60		
Design	40		
Production Engineering	64		
Payload Integration	12		
Subtotal		298	
Launcher and Handling		100	
Documentation		18	
Test and Checkout Equipment		212	
Subtotal			628
Contingencies - 50%			314
Company Expenses - 50% (G&A, P & OH)			471
Total R&D Costs			1,413

Acquisition costs have been computed on a turn-key basis. That is, the acquisition cost for an individual site not only includes the land, construction and equipment costs, but also a complete stock of spares and a 1-year supply of launch vehicles and payloads. The site acquisition cost covers all of the elements of a fully stocked site, requiring only a crew to put it into operation. As soon as a launch site becomes operational, the consumables (launch vehicles, payloads, spares, etc.) become an operating expense and are carried in the annual operating cost estimate.

Table VI-14, Launch Site Acquisition Schedule, shows all of the elements of system acquisition and displays the number of each required by quarters over the acquisition period. A review of Table VI-14 reveals that construction is planned at the rate of 11 sites for the first quarter and 10 sites each quarter thereafter. The final site will become operational at the close of the second quarter of FY6.

Method of Analysis

For the purpose of this analysis, acquisition costs have been broken into two categories; those which are common to all options and those which are peculiar to a particular option. The common costs, such as land and site development costs, are shown in Table VI-15 (common item acquisition cost factors).

Table VI-14
Launch Site Acquisition Schedule

	FY4				FY5				FY6	
	1	2	3	4	5	6	7	8	9	10
LAND (ACRES)	110	100	100	100	100	100	100	100	100	100
SITE PREP & UTILITIES (ACRES)	110	100	100	100	100	100	100	100	100	100
CONSTRUCTION										
BUILDINGS (SF)	15,400	14,000	14,000	14,000	14,000	14,000	14,000	14,000	14,000	14,000
LAUNCH PAD (EACH)	11	10	10	10	10	10	10	10	10	10
ROADS & PARKING (MI)	11	10	10	10	10	10	10	10	10	10
SECURITY FENCE (LINEAR FEET)	29,700	27,000	27,000	27,000	27,000	27,000	27,000	27,000	27,000	27,000
EQUIPMENT										
LAUNCH VEHICLE (EACH)	1,100	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000
DATA ACQUISITION	11	10	10	10	10	10	10	10	10	10
TRACKING (EACH)	11	10	10	10	10	10	10	10	10	10
LAUNCHER (EACH)	11	10	10	10	10	10	10	10	10	10
PAYLOAD (EACH)	1,100	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000
SUPPORT										
MISCELLANEOUS (EACH)	11	10	10	10	10	10	10	10	10	10
SHOP EQUIPMENT (EACH)	11	10	10	10	10	10	10	10	10	10
INITIAL SPARES (EACH)	11	10	10	10	10	10	10	10	10	10
ENGINEERING SERVICES	—	—	—	—	—	—	—	—	—	—
DESIGN (PER SITE)				(SEE NOTE 2)						
SIOH (PER SITE)				(SEE NOTE 3)						
ESCALATION FACTOR 7.5 PERCENT (5 PERCENT/YEAR FOR 1-1/2 YEARS)	—	—	—	—	—	—	—	—	—	—
CUMULATIVE SITES OPERATIONAL	11	21	31	41	51	61	71	81	91	101

NOTES

(1) FIRST QUARTER INCLUDES ONE TRAINING SITE
(2) DESIGN EQUALS 6 PERCENT OF SITE PREPARATION AND UTILITIES COST PLUS CONSTRUCTION COST.
(3) SIOH EQUALS 8 PERCENT OF SITE PREPARATION AND UTILITIES COST PLUS CONSTRUCTION COST.
(4) SITE ACREAGE ASSUMES SAME LAND REQUIREMENTS FOR FIRING LIQUID OR SOLID-PROPELLANT VEHICLES.

Table VI-15
System Acquisition Cost Factors
Common Items
(Ref Appendix A)

ITEM	UNIT	COST	REQ'D/ SITE
Land Purchase	acre	$ 2,500	10
Site Preparation and Utilities	each	40,000	---
Construction			
Buildings	sq. ft.	25	1,400
Launch Pad	each	1,200	---
Roads and Parking	each	7,800	---
Security Fence	lin. ft.	6	2,700
Equipment			
Miscellaneous	each	3,500	---
Shop	each	60,000	---
Engineering Services			
Design (6%)(1)	each	6,000	---
Construction Supr. (8%)(2)	each	8,000	---
Escalation(3)	each	15,200	---

(1) 6% of Site Preparation, Utilities and Construction.
(2) 8% of Site Preparation, Utilities and Construction.
(3) 7.5% of Total Acquisition Cost (1968 Dollars).

This table lists unit costs for each common line item. Table VI-16 is a tabulation of common item costs, which are computed by applying the appropriate cost factor from Table VI-15 to the quantity requirements shown in Table VI-14. For example, 110 acres of land are required in the first quarter of FY4 (Table VI-14) at a cost of $2,500 per acre (Table VI-15), which represents a dollar value of $275,000 (Table VI-16).

Table VI-17 is a tabulation of noncommon acquisition items. Table VI-18 is the development of launch system unit costs for Table VI-17. Tables VI-19 through VI-25 are total acquisition costs by option. Line item acquisition costs are computed by multiplying the number of items required (Table VI-14) by the appropriate cost factor (Table VI-15); common item costs are entered from Table VI-16.

Tables VI-18 and VI-19 show the addition of transportation and installation to the payload and launch system costs.

6. ANNUAL OPERATING COSTS

Each launch site will incur annual expenses for services and material required to conduct launch operations during the program time period. Some of these costs, such as personnel, will remain

Table VI-16
Launch Site Acquisition Costs
Common Items

ITEM	QUARTERLY ACQUISITION COST* ($1,000)									
	(1973)				(1974)				(1975)	
	1	2	3	4	1	2	3	4	1	2
LAND	275	250	250	250	250	250	250	250	250	250
SITE PREPARATION & UTILITIES	440	400	400	400	400	400	400	400	400	400
CONSTRUCTION										
BUILDINGS	385	350	350	350	350	350	350	350	350	350
LAUNCH PAD	13	12	12	12	12	12	12	12	12	12
ROADS PARKING	86	78	78	78	78	78	78	78	78	78
SECURITY FENCE	178	162	162	162	162	162	162	162	162	162
EQUIPMENT (SUPPORT)										
MISCELLANEOUS	39	35	35	35	35	35	35	35	35	35
SHOP	660	600	600	600	600	600	600	600	600	600
ENGINEERING SERVICES										
DESIGN – 6 PERCENT	66	60	60	60	60	60	60	60	60	60
CONSTRUCTION SUPER-VISION – 8 PERCENT	88	80	80	80	80	80	80	80	80	80
ESCALATION – 7.5 PERCENT										
5 PERCENT/FOR 1-1/2 YEARS	167	152	152	152	152	152	152	152	152	152
TOTAL QUARTERLY COST	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179

COMPUTED AS PRODUCT OF NUMBER REQUIRED (TABLE VI-14) AND COST FACTOR (TABLE VI-15).

Table VI-17
System Acquisition Cost Factors
Noncommon Items

ITEM	OPTION***					
	1	2	3	4	5	6
Launch Vehicle ($ each)	347	995	612	1, 224	1, 020	1, 785
Launcher and Handling Equip ($ each x 1, 000)	114	8	8	8	8	8
Tracking and Telemetry* ($ each x 1, 000)	700	700	700	800	800	800
Payload ($ each)	87	530	170	755	1, 979	1, 979
Support Equipment						
Initial Spares (15% investment)	105	105	105	120	120	120
Special Test & Checkout ($ each set)	**	**	10	10	10	10

* Transportation and Installation Expenses are included in the Primary Cost Estimate and have not been added separately. Military Air Transport is assumed to be primary mode of delivery.

** Special Test and Checkout Equipment associated with rocket systems only.

*** See Chapter V for definition of options

Table VI-18
Launch System Unit Costs

OPTION*	LAUNCH VEHICLE SYSTEM				LAUNCHER & HANDLING EQUIP			
	Item	Unit Cost	Trans-portation	Total	Unit Cost	Trans-portation	Installation	Total
1	Gun	340	7	347	74	2	38	114
2	Dart	975	20	995	5	---	3	8
3	Rocket	600	12	612	5	---	3	8
4	Rocket	1, 200	24	1, 224	5	---	3	8
5	Rocket	1, 000	20	1, 020	5	---	3	8
6	Rocket	1, 750	35	1, 785	5	---	3	8

*See Chapter V for definition of options

Table VI-19
Payload Unit Costs

OPTION*	DESCRIPTION	TOTAL UNIT COST	TRANSPORTATION (2%)	TOTAL
1	Passive Sphere	85	2	87
2	Transponder Sphere	520	10	530
3	Passive Sphere 1 canister of chaff	165	5	170
4	5 canisters of Chaff SWD Thermistor	740	15	755
5	5 canisters of Chaff MFD Thermistor	1,940	39	1,979
6	5 canisters of Chaff Pitot Thermistor	1,940	39	1,979

*See Chapter V for definition of options

Table VI-20
Total Equipment and Facilities Acquisition Cost
Gun/Sphere (Option 1)

ITEM	QUARTERLY ACQUISITION COSTS (THOU. DOLLARS)									
	FY4				FY5				FY6	
	1	2	3	4	1	2	3	4	1	2
Launch Vehicle*	382	347	347	347	347	347	347	347	347	347
Data Acquisition and Tracking Equipment	7,700	7,000	7,000	7,000	7,000	7,000	7,000	7,000	7,000	7,000
Launcher & Groundhandling	1,254	1,140	1,140	1,140	1,140	1,140	1,140	1,140	1,140	1,140
Payload*	96	87	87	87	87	87	87	87	87	87
Support										
Initial Spares*(15% of Data Acq.)	1,155	1,050	1,050	1,050	1,050	1,050	1,050	1,050	1,050	1,050
Special Test & Ckt. Equip.	-	-	-	-	-	-	-	-	-	-
Sub Total	10,587	9,624	9,624	9,624	9,624	9,624	9,624	9,624	9,624	9,624
Total Common Acq. Cost	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179
Grand Total Acquisition Cost	12,984	11,803	11,803	11,803	11,803	11,803	11,803	11,803	11,803	11,803
Cumulative Acquisition Cost	12,984	24,787	36,950	48,393	60,196	71,999	83,802	95,605	107,408	119,211

*These costs appear under operating costs after the second quarter of FY6.

Table VI-21
Total Equipment and Facilities Acquisition Cost
Dart/Transponder Sphere (Option 2)

ITEM	QUARTERLY ACQUISITION COSTS (THOU. DOLLARS)									
	FY4				FY5				FY6	
	1	2	3	4	1	2	3	4	1	2
Launch Vehicle*	1,095	995	995	995	995	995	995	995	995	995
Data Acquisition and Tracking Equipment	7,700	7,000	7,000	7,000	7,000	7,000	7,000	7,000	7,000	7,000
Launcher & Groundhandling	88	80	80	80	80	80	80	80	80	80
Payload*	583	530	530	530	530	530	530	530	530	530
Support										
Initial Spares*(15% of Data Acq.)	1,155	1,050	1,050	1,050	1,050	1,050	1,050	1,050	1,050	1,050
Special Test & Ckt. Equip.	-	-	-	-	-	-	-	-	-	-
Sub Total	10,621	9,655	9,655	9,655	9,655	9,655	9,655	9,655	9,655	9,655
Total Common Acq. Cost	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179
Grand Total Acquisition Cost	13,018	11,834	11,834	11,843	11,834	11,834	11,834	11,834	11,834	11,834
Cumulative Acquisition Cost	13,018	24,852	36,686	48,520	60,354	72,188	84,022	95,856	107,690	119,524

*These costs appear under operating costs after the second quarter of FY5.

Table VI-22
Total Equipment and Facilities Acquisition Cost
Rocket/Passive Sphere/Chaff (Option 3)

ITEM	QUARTERLY ACQUISITION COSTS (THOU. DOLLARS)									
	FY4				FY5				FY6	
	1	2	3	4	1	2	3	4	1	2
Launch Vehicle*	673	612	612	612	612	612	612	612	612	612
Data Acquisition and Tracking Equipment	7,700	7,000	7,000	7,000	7,000	7,000	7,000	7,000	7,000	7,000
Launcher & Groundhandling	88	80	80	80	80	80	80	80	80	80
Payload*	187	170	170	170	170	170	170	170	170	170
Support										
Initial Spares* (15% of Data Acq.)	1,155	1,050	1,050	1,050	1,050	1,050	1,050	1,050	1,050	1,050
Special Test & Ckt. Equip.	110	100	100	100	100	100	100	100	100	100
Sub Total	9,913	9,012	9,012	9,012	9,012	9,012	9,012	9,012	9,012	9,012
Total Common Acquisition Cost	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179
Grand Total Acquisition Cost	12,488	11,191	11,191	11,191	11,191	11,191	11,191	11,191	11,191	11,191
Cumulative Acquisition Cost	12,310	23,501	34,692	45,883	57,074	68,265	79,456	90,647	101,838	113,029

*These costs appear under operating costs after the second quarter of FY6.

Table VI-23
Total Equipment and Facilities Acquisition Cost
Spinning Wire Densitometer/Thermistor/
Chaff/Rocket/Parachute (Option 4)

ITEM	QUARTERLY ACQUISITION COSTS (THOU. DOLLARS)									
	FY4				FY5				FY6	
	1	2	3	4	1	2	3	4	1	2
Launch Vehicle*	1,346	1,224	1,224	1,224	1,224	1,224	1,224	1,224	1,224	1,224
Data Acquisition and Tracking Equipment	8,800	8,000	8,000	8,000	8,000	8,000	8,000	8,000	8,000	8,000
Launcher & Ground Handling	88	80	80	80	80	80	80	80	80	80
Payload*	831	755	755	755	755	755	755	755	755	755
Support										
Initial Spares (15% of Data Acquisition)*	1,320	1,200	1,200	1,200	1,200	1,200	1,200	1,200	1,200	1,200
Special Test & Checkout Equipment	110	100	100	100	100	100	100	100	100	100
SUBTOTAL	12,495	11,359	11,359	11,359	11,359	11,359	11,359	11,359	11,359	11,359
Total Common Acquisition Cost	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179
Grand Total Acquisition Cost	14,892	13,538	13,538	13,538	13,538	13,538	13,538	13,538	13,538	13,538
Cumulative Acquisition Cost	14,892	28,430	41,968	55,506	69,044	82,582	96,120	109,658	123,196	136,734

Table VI-24
Total Equipment and Facilities Acquisition Cost
Molecular Fluorescence Densitometer/Parachute/
Thermistor/Chaff/Rocket (Option 5)

ITEM	QUARTERLY ACQUISITION COSTS (THOU. DOLLARS)									
	FY4				FY5				FY6	
	1	2	3	4	1	2	3	4	1	2
Launch Vehicle*	1,122	1,020	1,020	1,020	1,020	1,020	1,020	1,020	1,020	1,020
Data Acquisition and Tracking Equipment	8,800	8,000	8,000	8,000	8,000	8,000	8,000	8,000	8,000	8,000
Launcher & Ground Handling	88	80	80	80	80	80	80	80	80	80
Payload*	2,177	1,979	1,979	1,979	1,979	1,979	1,979	1,979	1,979	1,979
Support										
Initial Spares (15% of Data Acquisition)*	1,320	1,200	1,200	1,200	1,200	1,200	1,200	1,200	1,200	1,200
Special Test & Checkout Equipment	110	100	100	100	100	100	100	100	100	100
SUBTOTAL	13,617	12,379	12,379	12,379	12,379	12,379	12,379	12,379	12,379	12,379
Total Common Acquisition Cost	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179
Grand Total Acquisition Cost	16,014	14,558	14,558	14,558	14,558	14,558	14,558	14,558	14,558	14,558
Cumulative Acquisition Cost	16,014	30,572	45,130	59,688	74,246	88,804	103,362	117,920	132,478	147,036

*These costs appear under Operating Costs after the Second Quarter of FY6.

Table VI-25
Total Equipment and Facilities Acquisition Cost
Pitot/Parachute/Thermistor/
Chaff/Rocket (Option 6)

ITEM	QUARTERLY ACQUISITION COSTS (THOU. DOLLARS)									
	FY4				FY5				FY6	
	1	2	3	4	1	2	3	4	1	2
Launch Vehicle*	1,964	1,785	1,785	1,785	1,785	1,785	1,785	1,785	1,785	1,785
Data Acquisition and Tracking Equipment	8,800	8,000	8,000	8,000	8,000	8,000	8,000	8,000	8,000	8,000
Launcher & Ground Handling	88	80	80	80	80	80	80	80	80	80
Payload*	2,177	1,979	1,979	1,979	1,979	1,979	1,979	1,979	1,979	1,979
Support										
Initial Spares (15% of Data Acquisition)*	1,320	1,200	1,200	1,200	1,200	1,200	1,200	1,200	1,200	1,200
Special Test & Checkout Equipment	110	100	100	100	100	100	100	100	100	100
SUBTOTAL	14,459	13,144	13,144	13,144	13,144	13,144	13,144	13,144	13,44	13,144
Total Common Acquisition Cost	2,397	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179	2,179
Grand Total Acquisition Cost	16,856	15,323	15,323	15,323	15,323	15,323	15,323	15,323	15,323	15,323
Cumulative Acquisition Cost	16,856	32,179	47,502	62,825	78,148	93,471	108,794	124,117	139,440	154,763

*These costs appear under Operating Costs after the Second Quarter of FY6.

constant with each launch system option, while others, such as launch vehicles, can vary between alternates.

This section presents a summary of operating costs for each launch system option. There are four major cost categories:

- Personnel and administration costs are expenditures required to cover annual personnel salaries and benefits, and the overhead costs related to personnel administration and record-keeping.
- Building maintenance and utilities are those annual costs incurred to provide basic utilities and supplies and equipment for maintaining buildings and grounds. Labor will be provided by the launch crew.
- Consumable costs encompass launch vehicles, payloads, and spare parts. These are annual costs incurred to replenish equipment which is consumed as a result of conducting routine launch operations.
- Update and modification costs are those annual expenditures necessary to perform field modifications on ground equipment, or return it to a depot or manufacturer to have current changes incorporated.

Table VI-26 shows the planned operating schedule during the construction period. Table VI-27 summarizes these operation figures on an annual basis and extends operations over a 10-year period. For example, the personnel and administration column of Table VI-27 shows crew strength to be 26 crews during FY4. This figure is the average of the four quarterly estimates shown in Tables VI-20 through VI-26. The annual operating cost factors are shown in Table VI-28.

Table VI-26
Quarterly Operating Schedule
During Acquisition

ITEM	YEAR									
	FY4				FY5				FY6	
	QUARTER									
	1	2	3	4	1	2	3	4	1	2
Personnel and Administration (crews active)	11	21	31	41	51	61	71	81	91	101
Building Maint. & Utilities (sq. ft. active)	15,400	29,400	43,400	57,400	71,400	85,400	99,400	113,400	127,400	141,400
Consumables										
Launch Vehicles	137.5	262.5	387.5	512.5	637.5	762.5	887.5	1,012.5	1,137.5	1,262.5
Payloads	137.5	262.5	387.5	512.5	637.5	762.5	887.5	1,012.5	1,137.5	1,262.5
Spares (sets)	11	21	31	41	51	61	71	81	91	101

Table VI-27
Annual Operating Schedule

ITEM	YEAR									
	FY1	FY2	FY3	FY4	FY5	FY6	FY7	FY8	FY9	FY10
Personnel and Administrative (crews active-avg)	---	---	---	26	66	98.5	101	101	101	101
Building Maint. & Utilities (sq. ft. active)	---	---	---	36,400	92,400	137,900	141,400	141,400	141;400	141,400
Consumables										
Launch Vehicles	---	---	---	2,600	6,600	9,850	10,100	10,100	10,100	10,100
Payloads	---	---	---	2,600	6,600	9,850	10,100	10,100	10,100	10,100
Spares (set)	---	---	---	26	66	98.5	101	101	101	101

Table VI-28
Annual Operating Cost Factors

ITEM	OPTION					
	1	2	3	4	5	6
*Personnel & Administrative ($ ea. site)	74,190	74,190	74,190	74,190	74,190	74,190
**Building Maintenance & Utilities ($/sq. ft.)	$1.10	$1.10	$1.10	$1.10	$1.10	$1.10
Consumables						
Launch Vehicles	347	995	612	1,224	1,020	1,785
Payloads	87	530	170	755	1,979	1,979
Spares for all options	--	--	--	--	--	--

(1) Initial spares 15% data acquisition and tracking equipment.
Follow-on spares 4% data acquisition and tracking equipment.

*Personnel and Administration Computation

Crew	Grade (GS)	Qty	Salary	Extension
Supervisor	12	1	12,989	12,989
Sr. Technician	11	2	10,945	21,890
Technician	11	2	7,634	15,268
Total Crew Salary				50,147
Eight percent benefits				4,012
TOTAL COMPENSATION				54,159

Total Compensation = 73% of P&A costs.

$$\text{P\&A} = \frac{54,179}{0.73} = 74,190$$

**Based on current costs for new construction covering all ordinary M&US-- HVAC - 15¢/SF, ordinary repairs - 40¢/SF, utilities (gas, water) - 20¢/SF, electricity - 35¢/SF.

Annual operating costs are shown in Tables VI-29 through VI-34. Operating costs are computed by multiplying scheduled quantities by the appropriate cost factors.

Table VI-29
Gun/Sphere (Option 1)

ITEM	ANNUAL OPERATING COSTS (THOU. DOLLARS)									
	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
Personnel & Administration	1,929	4,897	7,308	7,493	7,493	7,493	7,493	7,493	7,493	7,493
Building Main. & U.S.	40	102	152	156	156	156	156	156	156	156
Consumables										
Launch Vehicles	902	2,290	3,417	3,505	3,505	3,505	3,505	3,505	3,505	3,505
Payloads	226	574	856	878	878	878	878	878	878	878
Follow-on Spares (4%)*	2,800	2,800	2,800	2,800	2,800	2,800	2,800	2,800	2,800	2,800
Total Annual Op. Cost	5,897	10,663	14,534	14,832	14,832	14,832	14,832	14,832	14,832	14,832
Cumulative Op. Cost	5,897	16,560	31,094	45,926	60,758	75,590	90,422	105,254	120,086	134,918

*Includes update and modifications costs.

Table VI-30
Dart/Transponder Sphere (Option 2)

ITEM	ANNUAL OPERATING COSTS (THOU. DOLLARS)									
	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
Personnel & Administration	1, 929	4, 897	7, 308	7, 493	7, 493	7, 493	7, 493	7, 493	7, 493	7, 493
Building Main. & U.S.	40	102	152	156	156	156	156	156	156	156
Consumables										
Launch Vehicles	2,587	6,567	9,801	10,050	10,050	10,050	10,050	10,050	10,050	10,050
Payloads	1, 378	3, 498	5, 220	5, 354	5, 354	5, 354	5, 354	5, 354	5, 354	5, 354
Follow-on Spares (4%)*	2, 400	2, 400	2, 400	2, 400	2, 400	2, 400	2, 400	2, 400	2, 400	2, 400
Total Annual Op. Cost	8,334	17,464	24,881	25,453	25,453	25, 251	25,453	25,453	25,453	25,453
Cumulative Op. Cost	8,334	25,798	50,679	76,132	101,585	127,038	152,491	177,944	203,397	228,850

*Includes update and modifications costs.

Table VI-31
Rocket/Passive Sphere/Chaff (Option 3)

ITEM	ANNUAL OPERATING COSTS (THOU. DOLLARS)									
	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
Personnel & Administration	1,929	4,897	7,308	7,493	7,493	7,493	7,493	7,493	7,493	7,493
Building Main & U.S.	40	102	152	156	156	156	156	156	156	156
Consumables										
Launch Vehicles	1,591	4,039	6,028	6,181	6,181	6,181	6,181	6,181	6,181	6,181
Payloads	442	1,122	1,675	1,717	1,717	1,717	1,717	1,717	1,717	1,717
Follow-on Spares (4%)*	2,800	2,800	2,800	2,800	2,800	2,800	2,800	2,800	2,800	2,800
Total Annual Op. Cost	6,802	12,960	17,963	18,347	18,347	18,347	18,347	18,347	18,347	18,347
Cumulative Op. Cost	6,802	19,762	37,725	56,072	74,419	92,766	111,113	129,460	147,807	166,154

*Includes update and modifications costs.

Table VI-32
Spinning Wire Densitometer/Thermistor/
Chaff/Rocket/Parachute (Option 4)

ITEM	ANNUAL OPERATING COSTS (THOU. DOLLARS)									
	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
Personnel & Administration	1,929	4,897	7,308	7,493	7,493	7,493	7,493	7,493	7,493	7,493
Building Main. & U.S.	40	102	152	156	156	156	156	156	156	156
Consumables										
Launch Vehicles	3,182	8,070	8,078	12,056	12,362	12,362	12,362	12,362	12,362	12,362
Payloads	1,963	4,983	7,437	7,626	7,626	7,626	7,626	7,626	7,626	7,626
Follow-on Spares (4%)*	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200
Total Annual Op. Cost	10,314	21,260	30,153	30,837	30,837	30,837	30,837	30,837	30,837	30,837
Cumulative Op. Cost	10,314	31,574	61,727	92,564	123,401	154,238	185,075	215,912	246,749	277,586

*Includes update and modifications costs.

Table VI-33
Molecular Fluorescence Densitometer/Parachute/ Thermistor/Chaff/Rocket (Option 5)

ITEM	ANNUAL OPERATING COSTS (THOU. DOLLARS)									
	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
Personnel & Administration	1,929	4,897	7,308	7,493	7,493	7,493	7,493	7,493	7,493	7,493
Building Main. & U.S.	40	102	152	156	156	156	156	156	156	156
Consumables										
Launch Vehicles	2,652	6,732	10,047	10,302	10,302	10,302	10,302	10,302	10,302	10,302
Payloads	5,145	13,061	19,493	19,988	19,988	19,988	19,988	19,988	19,998	19,998
Follow-on Spares (4%)*	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200
Total Annual Op. Cost	12,966	27,992	40,200	41,139	41,139	41,139	41,139	41,139	41,139	41,139
Cumulative Op. Cost	12,966	40,958	81,158	122,297	163,436	204,575	245,714	286,853	327,992	369,131

*Includes update and modifications costs.

Table VI-34
Pitot/Parachute/Thermistor/
Chaff/Rocket (Option 6)

ITEM	ANNUAL OPERATING COSTS (THOU. DOLLARS)									
	FY4	FY5	FY6	FY7	FY8	FY9	FY10	FY11	FY12	FY13
Personnel & Administration	1,929	4,897	7,308	7,493	7,493	7,493	7,493	7,493	7,493	7,493
Building Main. & U.S.	40	102	152	156	156	156	156	156	156	156
Consumables										
Launch Vehicles	4,641	11,781	17,582	18,029	18,029	18,029	18,029	18,029	18,029	18,029
Payloads	5,145	13,061	19,493	19,988	19,988	19,988	19,988	19,988	19,988	19,988
Follow-on Spares (4%)*	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200
Total Annual Op. Cost	14,955	33,041	47,735	48,866	48,866	48,866	48,866	48,866	48,866	48,866
Cumulative Op. Cost	14,955	47,996	95,731	144,597	193,463	242,329	291,195	340,061	388,927	437,793

*Includes update and modifications costs.

VII. CONCLUSION

VII. CONCLUSION

Six candidate systems have been described and discussed in previous chapters. Their respective capability to obtain synoptic measurements of wind speed and direction, temperature, and pressure (or density) over the range of 30-100 kilometers has been described. This is the prime criteria which enables them to be considered as candidates. The next criteria is the establishment of their compatibility with respect to reasonable development risk and operational constraints (i.e., world-wide use, civilian operation, falling mass hazard insurance, potential operation over ocean areas). The final criterion for candidacy is a system characterized by simplicity and reliability which can be categorized by considerations of cost for research, development, test and evaluation, investment or capital equipment acquisition and by operational — expendable items — spare and replacement equipments, utilities, personnel wages and associated support requirements.

The candidate systems have been described previously. It is convenient to briefly re-identify them here:

(1) Passive sphere, 7-inch gun-launch vehicle, phased-array tracking radar

(2) Passive sphere with transponder, rocket-boosted 3-inch diameter dart, interferometer tracking system

(3) Passive sphere, 1 canister of chaff, rocket or rocket-boosted dart vehicle, phased-array tracking radar

(4) Spinning wire densitometer (SWD), thermistor/parachute, 5 canisters of chaff, rocket-launch vehicle, 2 telemetry ground stations, phased-array tracking radar

(5) Molecular fluorescence densitometer (MFD), thermistor/parachute, 5 canisters of chaff, rocket-launch vehicle, 2 telemetry ground stations, a phased-array tracking rada:

(6) Pitot system, thermistor/parachute, 5 canisters of chaff, rocket-launch vehicle, 2 telemetry ground stations, a phased-array tracking radar.

1. RECOMMENDED SYSTEM

The passive sphere, one canister of chaff, the phased-array tracking radar and the rocket or rocket-boosted dart vehicle constitutes the recommended system.

The following paragraphs summarize the development risks and the operational considerations of the various candidate systems; Chapter VI contains detailed cost analysis of the candidate systems.

The recommended system is not the lowest cost system, however, its capability to measure a greater altitude range of horizontal winds, its lower development risk and greater operational compatibility outweigh the slightly greater costs.

A complete description of this system is contained in previous chapters. Volume III, the Conceptual Design, contains a full system description under a single cover. Volume IV, the Technology Plan, presents the technological problems to be overcome and an orderly plan for their solution prior to the implementation of the system. Volume V, the Program Development Plan, is an orderly plan for the system implementation.

2. DEVELOPMENT RISK

Payload

1. Tracking System

All sounding systems considered have a requirement for a tracking system—either a phased-array radar or an interferometer-type system—which does not exist. Five of the candidate systems (all except the system with the transponder/sphere payload) require essentially the same tracking system, therefore, there is no relative difference in risk/cost between them. The transponder/sphere system requires the development of an interferometer-type tracking system (somewhat like the "single station DOVAP") and its development risk/cost is somewhat less than the other five, a situation which is offset by the necessity to develop a suitable transponder/sphere system.

2. Payload Systems

The passive sphere constitutes two of the candidate payloads and is the minimum risk payload with the exception of the "hardening" requirement for the gun projectile launch concept which is not considered excessive.

The transponder/sphere constitutes the next payload on the risk scale; however, a technological breakthrough is not required. It largely is a matter of a packaging challenge.

The payloads which contain the spinning wire densitometer (SWD), the molecular fluorescence densitometer (MFD) and the pitot system are all high risk concepts. Both the SWD and the MFD are very early in the development cycle, each having "flown" several times with very limited success. The pitot systems, which have been in use for a long time, represent a risk to bring the weight and cost down to acceptable levels without a sacrifice in performance.

3. Launch Systems

None of the launch systems represent a development risk. All three types — 7-inch gun, 3-inch darts boosted by a rocket and the rockets themselves — have a long history of satisfactory performance. The risks on the launch vehicles lie in the projected cost of production.

3. OPERATIONAL CONSIDERATIONS

The attached chart, Figure VII-1, indicates the various degrees of conformance of the candidates against the Falling Mass Hazard situation, surface wind conditions, on-site calibration requirements, handling and shipping considerations, potential for sea-based launch, and versatility-adaptability of the system.

SYSTEM	FALLING MASS HAZARD	SURFACE WIND CONSIDERATION	SEA-BASED LAUNCH COMPATIBILITY	ON-SITE CALIBRATION	HANDLING AND SHIPPING WEIGHT	VERSATILITY & ADAPTABILITY OF LAUNCH SUBSYSTEM
PASSIVE SPHERE, 7" GUN	• LEAST IMPACT AREA • PROJECTILE POSES A POTENTIAL PROBLEM	NO PROBLEM	HIGH INVESTMENT COST PER INSTALLATION	NONE	PROJECTILE & POWDER SEPARATE, • 45 LB PROJECTILE • 120 LBS POWDER	VERY RESTRICTED
TRANSPONDERED SPHERE, DART (ROCKET)	• SMALL IMPACT AREA • TRANSPONDER DISPOSAL IS A PROBLEM	ACCOMMODATED WITH A SHORT RAIL LAUNCHER	LAUNCHER IS SMALL & EASILY INSTALLED	TRANSPONDER CHECK	ABOUT 140 LBS TOTAL	VERY VERSATILE
PASSIVE SPHERE. CHAFF; ROCKET OR ROCKET-BOOSTED DART	• POSSIBLE SMALL IMPACT AREA • LOWEST WEIGHT PER IMPACTING ITEM	ACCOMMODATED WITH A SHORT RAIL LAUNCHER	LAUNCHER IS SMALL & EASILY INSTALLED	NONE	ABOUT 160 LBS TOTAL	VERY VERSATILE
MOLECULAR FLUORESCENCE DENSITOMETER. THERMISTOR PARACHUTE CHAFF, ROCKET-LAUNCHED	• LARGE IMPACT AREA • MEDIUM WEIGHT PER IMPACTING ITEM	ACCOMMODATED WITH DUAL-THRUST ROCKET	MEDIUM SIZED LAUNCHER	EXTENSIVE	ABOUT 300 LBS TOTAL	HIGH
SPINNING WIRE DENSITOMETER THERMISTOR PARACHUTE. CHAFF, ROCKET-LAUNCHED	• LARGE IMPACT AREA • MEDIUM WEIGHT PER IMPACTING ITEM	ACCOMMODATED WITH DUAL-THRUST ROCKET	MEDIUM SIZED LAUNCHER	• SPIN MOTOR • SENSOR • TELEMETRY	ABOUT 260 LBS TOTAL	VERY VERSATILE
PITOT DENSITOMETER THERMISTOR/PARACHUTE CHAFF, ROCKET-LAUNCHED	• LARGE IMPACT AREA • HIGH WEIGHT PER IMPACTING ITEM	ACCOMMODATED WITH DUAL-THRUST ROCKET	LARGE LAUNCHER REQUIRED	EXTENSIVE	450 TO 500 LBS TOTAL	HIGH

FIGURE VII-1. Operational Compatibility

APPENDICES

APPENDIX A

ERROR ANALYSES OF SELECTED ASPECTS OF CURRENT METEOROLOGICAL TRACKING DATA

The current falling sphere techniques use existing instrumentation radars to construct a time/position trace of the falling sphere. The error characteristics of the meteorological data has been inferred from the design specifications of the radars. Unfortunately, some of these data have been acquired with an operating signal-to-noise ratio below that at which the accuracy was defined. Since the limit of accuracy is an inverse function of the S/N (for S/N much greater than one), the quality of these data is not as high as is currently assumed.

Two fundamental questions always arise in considering errors in a system which has evolved through use of equipment designed for other purposes:

- What are the errors in the final data?
- What would these errors be if a specific instrument parameter took on a set of cost-related error limits?

This analysis considers the effect on data errors of varying the angular accuracy which is the most cost-sensitive parameter of the instrument (aside from attaining a workable signal-to-noise ratio). The relations between the meteorological parameter errors and the instrument parameter accuracies have been developed. It should be recalled that the relations are also dependent on the smoothing technique, the estimate of vertical wind, and the sphere trajectory.

The rationale for assuming a quadratic fitting polynomial was developed in the body of this report. The vertical wind and its error were both assumed to the zero. A trajectory was computed based on these initial conditions:

Altitude	140 km
Horizontal Displacement	40 km
Horizontal Velocity	200 meters/sec
Gravitational Acceleration	-9.8 meters/$\overline{\text{sec}}$
Area/Mass Ratio	65.4
Radius of the Earth	6,378,388 meters

The +50% wind profile was used.
The trajectory itself is listed in Table A-1:

Table A-1

Time	Altitude	Horizontal Displacement	Horizontal Velocity	Horizontal Acceleration	Vertical Velocity	Vertical Acceleration	Speed	Drag Coefficient
6	140,000	41,200	200	-.009	-56	-9.367	208	1.600
47	130,000	49,384	199	-.052	-439	-9.286	482	1.600
66	120,000	53,151	197	-.165	-613	-8.920	644	1.600
81	110,000	56,076	191	-.776	-734	-6.485	759	1.600
94	100,000	58,445	167	-3.415	-756	5.918	774	1.600
110	90,000	60,493	82	-5.575	-472	22.554	479	1.600
156	80,000	61,686	11	.055	-112	1.457	112	1.600
236	70,000	63,065	25	.163	-93	.656	93	.500
389	60,000	68,279	41	.075	-49	.147	49	.440
677	50,000	82,110	48	-.036	-26	.042	26	.440
1,240	40,000	104,673	34	-.017	-13	.012	13	.428

Parameter	Units
Time	seconds from apogee
Altitude	meters
Displacement	meters from a vertical axis through launch site
Velocity	meters/second
Acceleration	meters/second^2
Speed	meters/second with respect to surface of earth

The method of analysis was to assume a slant range error of 5 meters and to compute the smoothing interval when the elevation angle error was 0.05 mils and the density error was 2%. The density error limit was then increased to 3% and various values of elevation angle error were tried until the resulting smoothing interval was approximately equal to the 0.05 mils/2% result. The process was repeated for density errors of 4 and 5%. The corresponding wind error was computed at each altitude level, for each combination of elevation angle error and density error. The complete profiles are given in Tables A-2 through A-5.

The following combinations of density error and elevation error yield approximately equal smoothing intervals. The maximum horizontal wind error for each combination is also given.

Elevation σ_E (mils)	Density $\sigma\rho/\rho$ (%)	Wind σ_W (meters/sec)
.05	2	20
.10	3	37
.15	4	54
.20	5	70

Table A-2

σ_E = 0.05 mils $\sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Wind Error (meters/sec)	Smoothing Interval (meters)	Number of Data Points
100	18.9	6432	3734
90	18.9	3191	1768
80	15.2	1787	1197
70	6.1	915	1494
60	3.3	586	1808
50	1.8	322	1869
40	0.8	152	1901
[illegible]	0.4	87	1980

Table A-3

σ_E = 0.05 mils $\sigma\rho/\rho$ = 3%

Altitude (Kilometers)	Wind Error (meters/sec)	Smoothing Interval (meters)	Number of Data Points
100	37.0	6249	3628
90	34.7	3135	1737
80	26.1	1772	1187
70	9.6	919	1500
60	4.5	598	1844
50	2.1	336	1951
40	0.8	164	2046
30	0.3	97	2198

Table A-4

σ_E = 0.15 mils $\sigma\rho/\rho$ = 4%

Altitude (Kilometers)	Wind Error (meters/sec)	Smoothing Interval (meters)	Number of Data Points
100	53.9	6272	3641
90	49.6	3162	1752
80	36.5	1796	1203
70	13.1	935	1527
60	5.9	612	1887
50	2.6	346	2012
40	0.9	170	2128
30	0.3	101	2305

Table A-5

σ_E = 0.2 mils $\sigma\rho/\rho$ = 5%

Altitude (Kilometers)	Wind Error (meters/sec)	Smoothing Interval (meters)	Number of Data Points
100	69.9	6323	3671
90	63.8	3195	1770
80	46.5	1819	1218
70	16.5	949	1549
60	7.3	622	1919
50	3.2	353	2052
40	1.1	174	2177
30	0.4	104	2368

APPENDIX B

SELECTED PARAMETRIC STUDIES

A series of parametric studies were conducted to explore the effects of varying one parameter or its uncertainty over a discrete range while holding others at precisely stated values. These analyses were made to explore the sensitivity of the density measurement to selected uncertainties, which included:

- Elevation angle error
- Altitude error
- Horizontal wind profile
- Vertical winds and their uncertainties.

DENSITY ERROR VS. ELEVATION ANGLE ERROR

In allowing the elevation angle error to assume successively larger values, the only other parameter which was allowed to change as a consequence was the smoothing interval. This has the net effect of increasing the uncertainty as to the altitude at which the computed density was valid and, thus, results in a net uncertainty as to the density profile. The data are listed in Tables B-1 through B-4.

Table B-1

σ_E = .05 mils $\sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1719
40	145	1808
30	85	1936

Table B-2

σ_E = .1 mils $\sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	6852	3978
90	3467	1921
80	1975	1323
70	1032	1685
60	677	2090
50	385	2238
40	191	2384
30	115	2613

Table B-3

$\sigma_E = .2$ mils $\quad \sigma\rho/\rho = 2\%$

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	8932	5186
90	4528	2509
80	2584	1731
70	1352	2208
60	889	2744
50	508	2950
40	254	3180
30	158	3594

Table B-4

$\sigma_E = .5$ mils $\quad \sigma\rho/\rho = 2\%$

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	12,841	7456
90	6514	3610
80	3720	2492
70	1949	3183
60	1283	3960
50	736	4280
40	378	4730
30	251	5708

Constants

$^{\sigma}W_Z = W_Z = 0$ (no vertical wind)

$^{\sigma}R = 2$ meters + 50% wind profile

2% density error

Discussion

Four values of the elevation error estimate were used: (1) .05 mils, (2) .1 mils, (3) .2 mils, and (4) .5 mils.

The purpose is to examine the effect on smoothing interval as lower quality and, therefore, cheaper radars are used.

Results

As expected, a poorer radar requires drastically increased smoothing intervals. In general, as elevation error degrades from .05 mils to .5 mils, the smoothing interval increases by between 200% and 300%. For example, at 90 km altitude, the smoothing interval for 2% density error goes from 2.7 km to 6.5 km when the elevation error is relaxed from .05 mils to .5 mils.

DENSITY ERROR VS. ALTITUDE ERROR

As shown previously, a relaxation in the quality of density data permitted a relaxation in tracker performance. However, for a given tracker, this relaxation affects the requisite smoothing interval and consequently the precision with which the altitude component of the density vs. altitude profile is known. These data are listed in Tables B-5 through B-8.

Constants

$\sigma W_Z = W_Z = 0$ (no vertical wind)

$\sigma_E \geq .05$ mils $\qquad \sigma_R = 2$ meters

+50% wind profile

Discussion

Four values of density error were used: (1) 2%, (2) 3%, (3) 4%, and (4) 5%.

Results

The effect of increasing the smoothing interval is to reduce noise errors. When the density errors are increased, the noise errors are allowed to increase, thus decreasing the smoothing interval. In general, when the density error is

Table B-5

σ_E = .05 mils $\qquad \sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1719
40	145	1808
30	85	1936

Table B-6

σ_E = .05 mils $\qquad \sigma\rho/\rho$ = 3%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	4610	2676
90	2320	1285
80	1314	880
70	684	1116
60	446	1375
50	251	1461
40	123	1533
30	72	1630

Table B-7

σ_E = .05 mils $\qquad \sigma\rho/\rho$ = 4%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	4109	2385
90	2067	1145
80	1171	784
70	609	994
60	398	1226
50	224	1301
40	109	1364
30	64	1445

Table B-8

σ_E = .05 mils $\qquad \sigma\rho/\rho$ = 5%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	3757	2181
90	1891	1047
80	1071	717
70	557	909
60	364	1121
50	205	1190
40	100	1246
30	58	1317

increased by 250% (from 2% to 5%) the smoothing interval is decreased by about 30%, but the change is not linear. The split is approximately:

Density	$2 \longrightarrow 3\%$	Interval - 15% of 2%		15% of 2%
	$3 \longrightarrow 4\%$	Interval - 11% of 3%	or	9% of 2%
	$4 \longrightarrow 5\%$	Interval - 9% of 4%	or	6% of 2%
				30%

Thus, an overall degradation of data results from such a trade-off in favor of altitude precision.

HORIZONTAL WIND PROFILE

The fact that the sonde is rocket-launched means that a significant horizontal velocity component independent of winds exists throughout its descent. The interaction of this component with generalized wind profiles and the tracker geometry produces an asymmetry which was investigated. These data are listed in Tables B-9 through B-13.

Constants

$\sigma W_Z = W_Z = 0$ (no vertical wind)

$\sigma_E \geq .05$ mils $\quad \sigma_R \geq 2$ meters

Density error 2%

Table B-9

σ_E = .05 mils $\quad \sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1541	1033
70	787	1338
60	506	1569
50	262	1522
40	111	1390
30	54	1419

Table B-10

σ_E = .05 mils $\quad \sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1544	1035
70	795	1342
60	517	1596
50	281	1630
40	127	1588
30	63	1508

Table B-11

σ_E = .05 mils $\sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1544	1035
70	803	1311
60	521	1607
50	288	1675
40	136	1698
30	69	1735

Table B-12

σ_E = .05 mils $\sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1719
40	145	1808
30	85	1936

Table B-13

σ_E = .05 mils $\sigma\rho/\rho$ = 2%

Altitude (Kilometers)	Smoothing Interval (meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1549	1037
70	807	1314
60	531	1647
50	315	1831
40	165	2060
30	127	2886

Discussion

Five trajectories were computed with 5 different wind fields: (1) -99% wind profile, (2) -50% wind profile, (3) 0 wind, (4) +50% wind profile, and (5) +99% wind profile.*

The physical effect is that as the % wind profile increases (from -99% to +99%) the sphere is moved farther down range at the lower altitudes.

Results

All effects are below 50 km. The smoothing interval changes from a very small interval to a larger (but still very

*Plus (+) or Minus (-) refer to down range or up range direction.

small) interval. Therefore, a different trajectory produced by varying the wind field does not produce significantly different results.

VERTICAL WINDS

Probably the most serious problem in this system is its inherent inability to distinguish between vertical winds and aberrations in density. To explore the quantitative effects of vertical winds and/or uncertainties in the magnitudes of these winds a range of wind magnitudes, and the smoothing interval computed holding density error to 2% where possible.

Constants

σ_E = .05 mils σ_R = 2 meters

50% wind profile 2% density error

Discussion

The following matrix of values was used for vertical winds and their errors, in meters per second.

W_z

0	3	10	30
σ_{W_z}=0	0	0	0
.1	.5	1	3
.5	1.5	5	15

The purpose is to determine empirically the effects of vertical winds on density errors. A constant vertical wind error was used for all altitudes. This is probably not realistic since vertical winds should vary exponentially with altitude to conserve momentum. As a result combinations of values are comparable for the same altitude, but different altitudes for the same combination should probably not be compared.

A density error of 2% was achieved whenever possible. When the error in vertical wind is large, it is often not possible to achieve 2% density error, since the contribution to density error by vertical wind error is greater than 2%. Whenever this occurs, the contributio to density error by the error in vertical winds is reported in lieu of the total density error. In these cases, it is impossible to compute a smoothing interval, and therefore, a wind error cannot be reported.

Results

The computations verify the intuitive notion that no matter how large the vertical wind is, as long as the value is known with good accuracy (low error), the effect on density error and therefore, smoothing interval is negligible. The problem occurs when there is a large uncertainty (error) in the vertical wind.

When the uncertainty is about 10% of the vertical wind, the result is not too different than when the uncertainty is zero. Also when the uncertainty is 10%, the smoothing interval is about the same, no matter what the vertical wind is. The differences that occur are at the low altitudes where an interval cannot be computed. The larger uncertainty (though percent-wise constant) causes the error contribution to rise much more quickly so that as the uncertainty increases, fewer of the lower altitudes can have density errors of less than 2%.

When the uncertainty is about 50% of the vertical wind, more of the altitudes have density errors greater than 2%. However, for those that have density equal to 2%, and therefore smoothing intervals, the lengths of the intervals are surprisingly constant. In fact, they do not differ greatly from the $W_Z = \sigma W_Z = 0$ case.

When the uncertainty and the vertical wind both get large, the result on density error is disastrous. For example, when $\sigma W_Z = 15$, $W_Z = 30$, $\sigma\rho/\rho = 57.7\%$ at 30 km, (but the same conditions at 90 km give $\sigma\rho/\rho$ of only 3.3%).

The data are listed in Tables B-14 through B-26.

Table B-14

ϵ_E = .05 mils $\sigma\rho/\rho$ = 2%

W_z = 0 σ_{W_z} = 0

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1719
40	145	1808
30	85	1936

Table B-15

σ_E = .05 mils $\sigma\rho/\rho$ = 2%

W_z = 3 m/s σ_{W_z} = 0

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1719
40	144	1805
30	85	1921

Table B-16

$\sigma_E = .05$ mils $\quad \sigma \rho/\rho = 2\%$

$W_z = 10$ m/s $\quad \sigma_{W_z} = 0$

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1719
40	144	1801
30	84	

Table B-17

σ_E = .05 mils $\sigma\rho/\rho$ = 2%

W_z = 30 m/s σ_{W_z} = 0

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1312
60	525	1618
50	296	1718
40	144	1796
30	83	1880

Table B-18

σ_E = .05 mils $\quad$ $\sigma\rho/\rho$ = 2%

$W_z = 0$ $\quad$ σ_{W_z} = 0.1 m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5421	3147
90	2728	1511
80	1547	1036
70	804	1313
60	525	1619
50	297	1724
40	147	1832
30	90	2035

Table B-19

σ_E = .05 mils $\quad \sigma\rho/\rho$ = 2%

W_z = 3 m/s $\quad \sigma_{W_z}$ = 0.5 m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5426	3150
90	2729	1512
80	1549	1037
70	808	1319
60	535	1650
50	319	1856
40		
30		

Table B-20

$\sigma_E = .05$ mils $\quad \sigma\rho/\rho = 2\%$

$W_z = 10$ m/s $\quad \sigma_{W_z} = 1.0$ m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5436	3156
90	2735	1515
80	1552	1039
70	821	1340
60	570	1758
50		
40		
30		

Table B-21

σ_E = .05 mils $\quad \sigma\rho/\rho$ = 2%

W_z = 30 m/s $\quad \sigma_{W_z}$ = 3 m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5553	3224
90	2789	1545
80	1598	1070
70	1106	
60		
50		
40		
30		

Table B-22

$\sigma_E = .05$ mils $\quad \sigma_\rho / \rho = 2\%$

$W_z = 0 \quad \sigma_{W_z} = 0.5$ m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5426	3150
90	2729	1512
80	1549	1037
70	808	1320
60	535	1651
50	321	1868
40		
30		

Table B-23

σ_E = .05 mils $\quad \sigma_\rho / \rho$ = 2%

W_z = 0 $\quad \sigma W_z$ = 5m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5888	3418
90	2935	1626
80	1743	1167
70		
60		
50		
40		
30		

Table B-24

σ_E = .05 mils σ_ρ / ρ = 2%

W_z = 3 m/s σW_z = 1.5 m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5454	3166
90	2744	1520
80	1559	1044
70	848	1385
60	745	
50		
40		
30		

Table B-25

σ_E = .05 mils σ_ρ / ρ = 2%

W_z = 10 m/s σW_z = 5 m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	5874	3410
90	2930	1623
80	1735	1162
70		
60		
50		
40		
30		

Table B-26

$\sigma_E = .05$ mils $\quad \sigma_\rho / \rho = 2\%$

$W_z = 30$ m/s $\quad \sigma W_z = 15$ m/s

Altitude (Kilometers)	Smoothing Interval (Meters)	Number of Data Points
100	none	
90		
80		
70		
60		
50		
40		
30		

REFERENCES

1. L. M. Jones, F. F. Fischbach, and J. W. Peterson, "Seasonal and Latitude Variations in Upper-Air Density," Experimental Results of the U. S. Rocket Program for the I. G. Y. CSAGI 1958.

2. R. A. Minzer and W. S. Ripley, The ARDC Model Atmosphere, Air Force Surveys in Geophysics, No. 68, AFCRL Bedford, 1966.

3. E. F. Campen, Jr., Editor, Handbook of Geophysics, Revised Ed., USAF, Washington, 1964.

4. U. S. Standard Atmosphere, 1962.

5. William L. Weaver, Andrew G. Swanson, and John F. Spurling, Statistical Wind Distribution Data for Use at NASA Wallops Station, NASA TN D-1249, Langley Research Center, Langley Station, Hampton, Virginia, July 1962.

6. Richard S. Lindzen, Data Necessary for the Detection and Description of Tides and Gravity Waves in the Upper Atmosphere, COSPAR's Working Group II, November 1967, Revised, February 1968.

7. Richard S. Lindzen, Editor, Appendices to the Report Entitled Data Necessary for the Detection and Description of Tides and Gravity Waves in the Upper Atmosphere, Prepared for COSPAR Working Group II, February 1968.

8. G. V. Groves, "Theory of the Rocket-Grenade Method of Measuring Temperature, Pressure, Density, and Wind Velocity in the Upper Atmosphere," Royal Society of London, Proc., Ser. A290 (1420) pp. 44-73, February 1966.

9. W. Smith, et al., "Temperature, Pressure Density, and Wind Measurements in the Upper Stratosphere and Mesosphere 1964," NASA TR 245, August 1966.

10. Interview of W. J. Wilkie, Superior Engineering Co., by F. F. Fischbach, Ann Arbor, March 1968.

11. H. Foppl, et al., "Artificial Strontium and Barium Clouds in the Upper Atmosphere," Planetary and Space Science 15 (2), pp. 357-372, February 1967.

12. J. F. Bedinger, "Compendium of Wind Data from the Vapor Trail Technique," GCA Corp. Tech. Rpt. No. 66-7-N, March 1966.

13. J. M. Hoffman, L. S. Nelson, and L. B. Smith, "Yellow-Green Luminosity Accompanying the Injection of Triethylborane into the Upper Atmosphere," Nature, 214 (5084), pp. 158-159, April 8, 1967.

14. G. A. Faucher, J. F. Morrissey, and C. N. Stark, "Falling Sphere Density Measurements," Journal of Geophysical Researcl Vol. 72, No. 1, pp. 299-305, January 1, 1967.

15. Interview of G. A. Foucher, AFCRL, by J. F. Coneybear, BAARINC and F. F. Fischbach, University of Michigan, April 1968.

16. Interview of A. Faire, by F. F. Fischbach, Tokyo, May 1968.

17. H. N. Ballard, B. Sellers, M. Izquierdo, A Parachute-Borne Beta Ray Densitometer, Proceedings of the Third National Conference, Aerospace Meteorology pp. 86-93, New Orleans, May 1968.

18. Interview of H. N. Ballard and M. Izquierdo by J. F. Coneybear, May 1968.

19. Donald A. Wallace, A Cryogenic Crystal Instrument for the Direct Measurement of Atmospheric Density, Proceedings of the Third National Conference, Aerospace Meteorology, May 1968.

20. Interview of Dr. Chuan, Celestial Research Corp., by J. F. Coneybear, New Orleans, May 1968.

21. Andrew Guthrie, Vacuum Technology, New York and London, John Wiley and Sons, Inc., pp. 168-193, 1963.

22. Saul Dushman, Scientific Foundation of Vacuum Technique, New York and London, John Wiley and Sons, Inc., pp. 301-322, 1962.

23. Benjamin M. Elson, "Ion Pressure Gage Built for Alsep," Aviation Week & Space Technology, July 22, 1968.

24. Interview of Paul Sullivan, Norton Research Corp., by J. F. Coneybear, July 1968.

25. Interview of Brian Rofe, Weapons Research Establishment, by F. F. Fischbach, Tokyo, May 1968.

26. M. Kays and R. O. Olsen, "Improved Rocketsonde Parachute-Derived Wind Profiles," U. S. Army Electronics Command, Report ECOM-5086, October 1966.

27. Interview of W. L. Cruickshank by F. F. Fischbach, Aberdeen, Maryland, March 1968.

28. Interview of W. L. Webb by F. F. Fischbach, Tokyo, May 1968.

29. A. A. Barnes, American Meteorological Society Monograph #31, Vol. 9, "Proceedings of Conference on Meteorological Support of Aerospace Vehicles," Miami, May 1956.

30. A. Spizzichino, J. Delcourt, A. Giraud, and I. Revah, Proc. I.E.E.E. 53 1084, (1965).

31. Interview of A. A. Barnes by F. F. Fischbach, Tokyo, May 1968.

32. Interview of W. H. Paulsen, AFCRL by J. F. Coneybear, May 1968.

33. A. Spizzichino and I. Revah, Private Communication, "Wind Measurements from Meteor Trail Observation in the Altitude Range 80 - 110 km."

34. J. B. Gregory, "Radio Wave Reflections from the Mesosphere, (1) Heights of Occurrence," Journal of Geophysical Research, 66, pp. 429-445.

35. G. J. Fraser, "Seasonal Variation of Southern Hemisphere Mid-Latitude Winds at Altitudes of 70-100 km," Journal of Atmospheric and Terrestrial Physics, 1968.

36. C. H. Murphy, G. V. Bull, H. D. Edwards, "Ionospheric Winds Measured by Gun-Launched Projectiles," Journal of Geophysical Research, 71, pp. 4535-4544.

37. Interview of W. H. Paulsen, AFCRL by J. F. Coneybear, May 1968.

38. L. M. Jones, Falling Sphere Method for Upper-Air Density, Temperature and Wind, COSPAR Technique Manual Series, February 1967.

39. L. M. Jones and J. W. Peterson, Falling Sphere Measurements, 30 to 120 km, College of Engineering, University of Michigan 05627-12-S, June 1967.

40. Interview of R. Leviton by F. F. Fischbach and J. F. Coneybear, AFRCL, April 1968.

41. Interview of J. W. Peterson, K. McWaiters, and H. F. Allen by J. L. Hain and F. F. Fischbach, Ann Arbor, June 1968.

42. John B. Wainwright, "Atmospheric N_2 Density Measurements in the 50 km to 130 km Altitude Range Using the Molecular Fluorescence Technique," Celestial Research Corp., S. Pasadena, Calif.

43. Interview of Dr. H. H. deLeeuw, University of Toronto by F. F. Fischbach, Tokyo, May 1968.

44. J. H. deLeeuw and W. E. R. Davis, "Simultaneous Measurement of Temperature and Density by an Electron Beam Luminescence Technique," University of Toronto.

45. John E. Naugle, "Space Applications Programs and Meteorology Flight Program," Statement before the Subcommittee on Space Science and Astronautics, House of Representatives, 1968.

46. J. E. Ainsworth, D. F. Fox, and H. E. LaGow, "Upper-Atmosphere Structure Measurement Made with the Pitot-Static Tube," Journal of Geophysical Research, Vol. 66, No. 10, pp. 3191-3212, October 1961.

47. D. J. Rigali and K. J. Touryan, "Analysis of Pressure Measurements Taken at Altitudes Between 30 and 90 km by Cone-Cylinder Pitot-Static Probes," Sandia Laboratories SC-RR-67-251, June 1967.

48. J. J. Horvath, personal communication to J. F. Coneybear.

49. G. F. Rupert, "Engineering Design of a Pitot-Static Probe Payload," Engineering Report No. 2, University of Michigan, College of Engineering, Space Physics Research Laboratory, April 1967.

50. D. C. Thompson, "The Accuracy of Miniature Based Thermistors in the Measurement of Upper-Air Temperatures," Scientific Report No. 1, Department of Meteorology, Massachusetts Institute of Technology, October 1966, AFCRL-66-773.

51. R. S. Lindzen, "On the Consistency of Thermistor Measurements of Upper Air Temperatures," Journal of Atmospheric Sciences, Vol. 24, pp. 317-318.

52. S. V. Pachomov, "Technique of Obtaining Data about Winds in Mesosphere at Small Meteorological Rockets," XI Plenary Session, COSPAR, Tokyo, May 1968.

53. Interview of W. L. Cruickshank and J. Mester by F. F. Fischbach, Aberdeen, Maryland, March 1968.

54. Interview of W. L. Webb by F. F. Fischbach, El Paso, April 1968.

55. O. Hirao, T. Okamoto; The Rockoon in, "Development of Sounding Rockets in Japan," NASA TT F-87, March 1963.

56. W. J. Bolster, G. C. Googins, "The Design, Development and Testing of a Series of Air-Launched Sounding Rockets," presented at the AIAA Sounding Rocket Vehicle Technology Specialists Conference, Williamsburg, Virginia, February 27-March 1, 1967.

57. William H. Avery, "Beyond the Supersonic Transport," Science and Technology, February 1968.

58. R. L. Weirich, "A Preliminary Feasibility Analysis of Some Launch Concepts for 100 km Sounding Systems," In-house Memorandum, May 27, 1968.

59. D. L. Howley, "A Study to Define a State-of-the-Art Rocket Vehicle Suitable for Synoptic Meteorological Soundings," North American Aviation, Inc., Space and Information Systems Division., SID 65-411-1A, Final Report, Vol. I, April 8, 1965; Revised June 16, 1965.

60. Robert O. Olsen, "The JUDI Meteorological Rocket System," IRIG Document No. 111-64, Meteorological Rocket Network Committee, Meteorological Working Group, February 1965, pp. 191-200.

61. Robert L. Ammons, "The Development of Design Techniques for Single-State Sounding Rockets," Atlantic Research Corp., AFCRL-67-0233, March 15, 1967.

62. R. Frank Atmore, R. Gail Billings, "Final Report for a Study to Guide Research and Development Toward an Operational Meteorological Sounding Rocket System," prepared for National Aeronautics and Space Administration, Office of Space Science and Applications, by Thiokol Chemical Corp., Astro-Met Division, October 1966-1967.

63. Joe H. Brown, Jr., James S. Parks, "Existing Meteorological Rocket Systems," (Supplement to RSIC-181). RSIC-421, U. S. Army Missile Command, Redstone Scientific Information Center, July 15, 1965.

64. Atlantic Research Corporation Data Sheets: Upper Atmosphere Sounding Vehicles, ARCAS, SUPER ARCAS, ARCTURUS, ARCAS Vehicle Launcher, Vehicle Performance Comparison, IRIS, Boosted ARCAS II, Sidewinder-HV ARCAS, Sparrow-HV ARCAS, Booster ARCAS, ARCASONDE 1A, -4, -3, etc.

65. "Final Report on Application of Advanced Solid and Hybrid Motors to Sounding Rockets," prepared for National Aeronautics and Space Administration, Office of Advanced Research and Technology by Space-General Corporation, SGC 928FR-1, July 1966.

66. J. R. Brasfield, "Meteorological and Sounding Rocket State-of-the-Art Study," Meteorological Rocket Program, Vol. I, U. S. Army Missile Command, Redstone Arsenal, RH-TR-65-1, July 1965.

67. Louis V. Hoffman, "Application of Liquid Propellant Engine for Small Sounding Rockets," TRW Systems, Inc., Personal Communication.

68. "Pre-Flight Analysis and Range Documentation Report for the D-Region Tomahawk," prepared for National Aeronautics and Space Administration, Goddard Space Flight Center, by Thiokol Chemical Corporation, Astro-Met Division, August 26, 1967.

69. "Range Safety Data for the Ozonesonde Apache," prepared for National Aeronautics and Space Administration, Goddard Space Flight Center, by Thiokol Chemical Corporation, Astro-Met Division, March 15, 1968.

70. J. B. Edwards, B. P. Kirk, J. R. Temchin, J. N. Carsey, "Survey of Developments in Gun-Launched High Altitude Probes," NPP/RP 66-7, U. S. Naval Propellant Plant, September 12, 1966.

71. G. B. Freeman, F. C. Balskovich, P. J. Douglas, R. E. Weidner, J. L. Burroughs, E. R. Grady, "Gun-Launched Vehicles Cost-Effectiveness Study," prepared for U. S. Army Research Office, by Lockheed Missiles and Space Company, LMSC-688043, September 29, 1967.

72. Dr. Charles H. Murphy, Dr. Gerald V. Bull, "Aerospace Application of Gun-Launched Projectiles and Rockets," Space Research Institute of McGill University, SRI-R-24, February 1968.

73. C. M. Plattner, "Liquid-Filled Cases Stretch Rocket Range," Aviation Week and Space Technology, March 25, 1968.

74. F. F. Fischbach, Booz, Allen Applied Research Inc. Report - Trip to Aberdeen Proving Ground, Maryland, May 1, 1968.

75. Kenneth R. Jenkins, "ARCAS Meteorological Sounding Rocket," IRIG Document No. 11-64, Meteorological Rocket Network Committee, Meteorological Working Group, February 1965, pp. 157-168.

76. S. Wasserman, G. Lattal, J. Smolnik, "Parametric Studies on Use of Boosted Artillery Projectiles for High Altitude Research Probes, Project HARP," Technical Report No. 3147, Picatinny Arsenal, January 1964.

77. "Survey and Evaluation of a 500 km New Sounding Rocket," Booz, Allen Applied Research Inc., prepared for National Aeronautics and Space Administration, Goddard Space Flight Center, November 8, 1965.

78. Nicholas A. Engler, "Development of Methods to Determine Winds, Density, Pressure, and Temperature from the Robin Falling Balloon," AFCRL 65-448, May 1965.

79. James K. Luers, "Estimation of Errors in Density and Temperature Measured by the High Altitude Robin Sphere," University of Dayton Research Institute.