"Lone Star," A Small Communication Satellite for Texas

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The Texas State Legislature has passed a bill recommending that the state examine the feasibility of developing a "home grown" satellite for communications applications. Such a satellite would be built in Texas by a state, industry, and university consortium with the intent that it be launched by a Texas built launch vehicle. This paper reports on a quick look design and trade-off study that was performed to determine the characteristics and possible configurations that such a small satellite system would have. The communications link budget is discussed and used to define the communications payload that would be carried by the satellite. Mission constraints and orbital options are considered, as well as launch vehicle performance to the possible working orbits. A baseline spacecraft is presented and the principal subsystems are discussed.

INTRODUCTION AND BACKGROUND

In the spring of 1989, during the 71st session of the Texas State Legislature, the Texas Senate Space Science and Industry Commission recommended that Texas examine and evaluate the development of a Texas communication satellite. State Senator J.E. "Buster" Brown, with supporting testimony from Dr. John Freeman, Professor, Rice University, sponsored Senate Concurrent Resolution (SCR) 23, which the 71st Texas Legislature subsequently enacted. The legislative intent of SCR 23 is to :

- Establish a long-range telecommunications plan for Texas,
- Provide 24-hour communications for the oil industry, hospitals and medical emergency teams, law enforcement, and environmental research,
- Explore applications of technologies such as digital satellite and packet switching,
- Use new technologies to advance the technical capabilities of Texas industries and campuses,
- Provide educational research and unite universities, industries, and state government in a common goal.

SCR 23 directs the Texas Department of Information Resources (DIR) and the State Purchasing and General Services Commission (SPGSC) to perform a feasibility study of the satellite project, the results of which will be presented to the 72nd Legislature in January 1991. Dr. Freeman and the DIR established a study team to implement the satellite design and cost portion of the study. One of us, Mr. Eaker, was asked to lead this study team. Two workshops dealing with the applications and development of a "Lone Star" communications satellite for Texas were held. The workshop participants consisted of representatives from universities, industry, state government, and non-profit institutions.

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At the workshops, the communications needs of the various groups were presented and have subsequently been used as guidelines for the basic configuration of a relatively small, low-cost, satellite that could be launched by one of the commercial launch vehicles now under development by private industry. These workshops provided an opportunity for the various groups to exchange ideas about the pros and cons of various satellite configurations and requirements. Satellites for both geosynchronous and non-geosynchronous orbital periods were discussed. There are well known advantages and disadvantages for satellites in both types of orbits, and as the study progressed, it became apparent that if costs were to be generated in a timely manner, it would be advantageous to choose one type of orbit and to develop a preliminary design and costs for a satellite in that orbit. Therefore, the decision was made to develop a basic design for a geosynchronous satellite with the knowledge that costs for a smaller and lighter non-geosynchronous design would be a subset of the larger design.

GUIDELINES

The Texas legislature presented an exciting challenge when they charged the DIR with the responsibility to devise a long range telecommunication plan that included a "home grown" communications satellite for Texas. The study team took this challenge seriously and recognized that the satellite should reflect a design that could be developed and hopefully launched by a team from Texas universities, industries, and private research groups. As the study progressed, it became apparent that there was interest and capability within these groups to construct and launch a communications satellite.

As the result of workshop discussions, several key guidelines were established that served as focal points for the preliminary design presented here. These guidelines were:

Payload Mass - It was recognized that launch costs are a function of the payload mass. Therefore, it was determined that the satellite must be as small and light as possible if the program were to have any hope of succeeding. It was also important to attempt to use one of the launch vehicles under development by the Texas space industry.

Type of Orbit - Numerous orbits were reviewed and some of the pros and cons associated with them are presented in a later section. Utilizing a geosynchronous orbit allowed the design to progress in a straightforward manner and was expected to result in a maximum cost for a single satellite system.

Operating Frequency - The operating frequency for the baseline configuration became a choice between the C, K_u, and K_a bands. The K_a band is noted for its susceptibility to rainfade and atmospheric attenuation, which could be large if the assigned geosynchronous slot placed the satellite low on the horizon. The basic choice between the C and K_u bands was largely reduced to mass considerations. The K_u band was chosen since it utilizes smaller and lighter antennas for the spacecraft and Earth systems.

Transponder Bandwidth - During the workshops it was noted that there were telecommunication requirements that spanned the range from the very low through the very high frequencies. The low frequency spectrum included applications such as remote monitoring of rivers and creeks to detect flooding, while the high frequency use was for two-way video for education and emergency medical activities. As specific needs were identified, it became apparent that the satellite transponders should be designed to handle a broad range of data types including analog, digital, and high and low definition video. As a baseline, it was decided to make the bandwidth of each transponder a multiple of 36 MHz, which is frequently used for communication satellites. It was noted that the basic 36 MHz bandwidth could be used in many ways. For example, 8 standard television signals could be accommodated in 36 MHz, since a standard TV signal utilizes a video bandwidth of 4.2 MHz. Or the 36 MHz could accommodate 24 channels of T1 (1.5 MBPS) digitized voice data without data compression - as many as 90 channels with compression. Another configuration could place as many as 6000 channels of analog voice data over the 36 MHz.

Number of Transponders - The choice of the number of transponders was largely determined by the need to keep the satellite mass as small as possible to accommodate the small class of expendable launch vehicles that could be used for this program. It was also noted that for geosynchronous operation the Federal Communication Commission (FCC) requires, at least informally, that for a given orbital position the satellite utilize all of the 500 MHz bandwidth available for that location. This understanding, plus the desire for low mass, shaped the decision to limit the number of transponders to 6--three to operate horizontally polarized and three to operate vertically polarized using one basic antenna reflector for both polarities. This configuration would allow each transponder to have a bandwidth of 144 MHz (4 X 36 MHz) for a total of 432 MHz and also provide for 68 MHz of baseband data for satellite health and safety or other low bit rate information. The 144 MHz bandwidth could also be implemented by a solid-state amplifier, which is desirable due to its low mass.

Orbital Lifetime - After a preliminary investigation into the launch capabilities of the Texas vehicles, which were first being considered, it was obvious that satellite mass must be minimized if these vehicles were to be used. Therefore, to minimize mass it was decided to limit the amount of station keeping propellant in favor of maximizing the number of transponders. This trade-off resulted in a mission lifetime of 7 years. Typical geosynchronous satellites have expected lifetimes of 10-12 years; therefore, a 7 year lifetime may be too short to be cost effective when considering typical launch costs. A trade-off study between launch cost using small expendable launch vehicles, development costs, and satellite replacement costs will be performed once the launch system is known.

COMMUNICATIONS LINK

System Variables

To keep the overall size and weight of the "Lone Star" Satellite within the capability of a Texas launch vehicle and the cost within acceptable bounds, every effort must be made to minimize the size, weight, power consumption and complexity of the transponder system while retaining the capability to provide the desired communications access and reliability. A preliminary study was conducted to define the transponder and associated communications system parameters based on these criteria. Minimizing the transponder power output requirements was given high priority since this not only affects the size, weight, and complexity of the transmitter portion of the transponder but is also the primary factor that sets the battery weight and capacity, the solar panel size and weight and the heat dissipation requirements. For purposes of this discussion, the communications link for the Lone Star Satellite includes three major sub-systems:

1) The Satellite transponder and associated antenna;
2) The down-link ground station (receiver and antenna);
3) The up-link ground station (transmitter and antenna).

The major variable factors that define the communications capabilities and the sub-system requirements of the satellite system for any application (i.e., voice, video, computer data) in any modulation format (analog - FM, AM, SSB, etc.) or digital include the following:

1) Satellite:
   a) Frequency
   b) Antenna Size
   c) Transponder Output Power
   d) Transponder Receiver Noise Temperature
   e) System Bandwidth
   f) System input signal-to-noise requirements

2) Down Link Ground Station:
   a) Frequency
   b) Antenna Size
   c) Receiver System Noise Temperature
   d) Receiver Bandwidth
   e) Signal-to-Noise, C/N, requirements

3) Up-Link Ground Station:
   a) Frequency
   b) Antenna Size
   c) Transmitter Power
   d) Transmitted Bandwidth

Path Loss

An additional consideration is the path loss between the satellite and the ground stations. This signal attenuation factor includes the loss in free space plus additional loss due to atmospheric (particularly rain) attenuation, aiming errors, polarization errors, etc. The free space path loss between isotropic antennas is a function of the path length and the frequency and may be expressed as:

$$L_b = 92.5 + 20 \log f + 20 \log d \qquad (1)$$

where L_b = free space path loss in dB
f = frequency in GHz
d = path length, km

For geosynchronous orbit the path length, d, is somewhat dependent on the latitude of the ground station and on the difference in longitude between the satellite and the ground station but has a nominal value on the order of 35,000 km. Using this value:

$$L_b = 183.6 + 20 \log f \qquad (2)$$

For C-band (4.2 GHz) and for K_u-band (12.2 GHz), the free space downlink path loss is:

$$L_4 = 196.1 \text{ dB @ } 4.2 \text{ GHz} \qquad (3)$$

$$L_{12} = 205.3 \text{ dB @ } 12.2 \text{ GHz} \qquad (4)$$

Signal-to-Noise

The microwave power radiated by the satellite transponder must be sufficient to insure adequate carrier signal-to-noise ratio (C/N) at the ground receiver for the worst conditions under which communications must be maintained. The power received is given by:

$$P_r = P_t - L_b + G_t + G_r - L_a \qquad (5)$$

where P_r = power delivered to receiver input, dBW
P_t = power transmitted, dBW
L_b = path loss, dB
G_t = transmitter antenna gain, dB
G_r = receiver antenna gain, dB
L_a = Addition path or feed line loss, dB
dBw = power relative to one-watt expressed in dB

The received power, P_r, must be sufficient to overcome the receiver noise and provides an adequate carrier to noise (C/N) ratio. The C/N may be expressed as:

$$C/N = P_r/kTB \qquad (6)$$

where k = Boltzmann's Constant = 1.38×10^{-23}
T = system temperature, °K
B = system bandwidth

which may be written in dB as:

$$C/N = EIRP + G_r - L_b - 10 \text{ Log } T - 10 \text{ Log } B + 228.6 \quad (7)$$

where

$$EIRP = P_t + G_t = \text{effective isotropic radiated power} \quad (8)$$

Equation (7) for C-band and K_u-band reduces to:

$$(C/N_4) = EIRP + G_r - 10 \log T - 10 \log B + 32.5 - L_a$$

$$= C/N @ 4 \text{ GHz in dB} \quad (9)$$

$$(C/N_{12}) = EIRP + G_r - 10 \log T - 10 \log B + 23.3 - L_a$$

$$= C/N @ 12 \text{ Ghz in dB} \quad (10)$$

The minimum required C/N is dependent upon the modulation mode and the desired signal-to-noise ratio of the detected signal from the receiver. For FM systems, C/N = 8 dB is the minimum and 9 dB is typical for acceptable communications or viewing (TV) quality.

Satellite Antenna

It is generally desirable for the satellite antenna gain, G_i, to be the maximum allowable to provide the highest possible EIRP for a given transponder power, P_r. However, there are two important limits on the allowable antenna gain. One limit is set by the maximum size of the antenna structure that can be accommodated on the satellite or within the launch package. For lower frequencies, especially, this may be a dominant consideration. A second limit is set by the antenna pattern spot size (footprint) on the earth's surface that is required by the intended usage. The "Lone Star" Satellite is primarily intended to provide communications within the State of Texas and can use a smaller spot size and higher gain antenna than would be possible if worldwide or CONUS coverage were required. It is desirable to take advantage of this possibility of a smaller allowable spot diameter to minimize the transponder power, the up-link transmitter power, and the size of the ground station antenna for both up-link and down-link. The gain and the beamwidth of an antenna are both functions of the effective size (area) in terms of the wavelength at the operating frequency. The gain of an antenna is given by

$$G_i = \eta \, 4\pi \, A/\lambda^2 \quad (11)$$

where G_i = power gain referenced to isotropic
η = efficiency
A = effective area
λ = wavelength

For a parabolic antenna as commonly used for satellite communications, both on the ground as well as on the satellite, an efficiency factor $\eta = 0.55$ is typical. For this case, the antenna beamwidth, θ, between half power points is approximately:

$$\theta = 70 \, \lambda/D \text{ degrees} \quad (12)$$

where D = diameter of antenna.

To cover the entire state of Texas, a minimum spot diameter of 750 miles is needed. This can be provided from a satellite in geosynchronous orbit by an antenna having a beamwidth of 1.9°. However, if this is the half-power beam width, the signal would be down by 3dB at the edges compared to that in the middle of the state. To provide more uniform signal intensity over the state and to provide some allowance for satellite aiming error, an antenna having a half-power beam width on the order of 3.5 - 4.0 degrees is recommended. To produce such beamwidth, the antenna diameter for C-band and K_u-band would be as follows:

C-Band: $\lambda = 7.5$ cm
3.5° Antenna Diameter = 150 cm (59-in.)
4.0° Antenna Diameter = 131 cm (51.7-in.)

K_u-Band: $\lambda = 2.5$ cm
3.5° Antenna Diameter = 50 cm (19.7-in)
4.0° Antenna Diameter = 44 cm (17.2-in)

The gain (over isotropic) of these antennas, 2200 (33.4dB) for 3.5° beamwidth and 1688 (32.3dB) for 4° beamwidth is the same for each frequency. This is calculated from

$$G_i = (27000)/(\Theta)^2 \qquad (13)$$

G_i = power gain over isotropic

Down-Link Ground Station

To meet the needs of many users over the State of Texas, a large number of down-link ground stations are expected to be needed to receive and make use of the information and programming passing through the satellite. While some of these will also require up-link capability to originate and transmit information through the satellite, the needs of many will be satisfied with receive-only capability. This capability is particularly expected to be the case for schools making use of educational TV programming. Because of the large number, every effort should be made to minimize the cost of these ground stations consistent with maintaining adequate quality of reception and operational reliability. One way of achieving minimum cost is to make maximum use of the antennas and receivers that are available for home satellite TV reception. Current systems now being manufactured are of excellent quality and because of the high production volume, are available at very low cost compared to the commercial and industrial systems. Again, because of the savings resulting from high production volume, the 10-feet (nominal) diameter antenna most commonly used with the home satellite TV receivers is available at lower costs than many smaller antennas and the quality is adequate for years of reliable service. These antennas could also be used to advantage in a full scale ground station having up-link, down-link and analog or digital transmission modes for voice, computer data, video and other communications. Because of the economy that can be realized, the use of such (10-feet diameter) antennas for full bandwidth TV reception is considered as an initial criterion in setting the other system requirements. For communications requiring smaller bandwidth, the size of the ground station antenna can be proportionately reduced. A small antenna is particularly important for mobile communications and the expected performance and requirements for these, and other, applications are also addressed in this study.

The requirements for reception of the full bandwidth FM modulated TV type video are the most demanding of all the services that are expected to be carried over the satellite. Each of these TV channels has a nominal available bandwidth of 36 MHz and are spaced at 40 MHz intervals. Using this bandwidth, along with a ground antenna 10-feet in diameter and state-of-the-art performance figures for other pertinent elements in the system, the EIRP requirements can be determined for the satellite using equations (9) and (10). The variables for 4 and 12 GHz used in the calculations are as follows:

	4 GHz	12 GHz
G_r	39.5 dB	49.1 dB
B	36 MHz	36 MHz
T	75°K	120°K
L_a	1.0 dB	3.0 dB

The attenuation term L_a includes 0.5 dB for feed loss plus 0.5 dB at 4 GHz and at 2.5 dB at 12 GHz, respectively, for rain attenuation. For high antenna elevation angles, this rain attenuation allowance is sufficient for 99.8% path reliability except in Southeast Texas where it would be about 99.5%. Using these parameters, the satellite EIRP required for 9.0 dB C/N, is calculated from equations (9) and (10) and the satellite transponder power, P_t, is determined from (8) based on a 3.5° antenna beamwidth (Gain = 33.4 dBi):

4 GHz: EIRP = 32.4 dBw
P_t = -1.0 dBw = 0.8 Watts

12 GHz: EIRP = 36.0 dBw
P_t = 2.6 dBw = 1.80 Watts

It should be noted that this EIRP and P_t (satellite transponder power output) are somewhat lower than is common for K_u band satellites and is based on taking full advantage of state-of-the-art low noise amplifiers, which have only recently become available, and a somewhat larger ground station antenna. For wider band transponders, the indicated EIRP and P_t should be available for each 36 MHz increment of bandwidth, i.e., if the bandwidth is 54 MHz, then the transponder power, P_t should be 1.5 times that for 36 MHz and the EIRP should be 1.8 dB greater. Similarly, when the full transponder bandwidth is not utilized, such as would be the case for multiple voice channels, digital data channels, etc., a ground station of lower antenna gain (smaller antenna) and higher noise temperature can be used to produce an adequate C/N ratio. The G/T ratio is frequently used as a figure of merit to indicate the ground system antenna gain and system noise temperature ratio needed to achieve an acceptable C/N ratio. Typical values along with the antenna gain and size for 4 and 12 GHz are tabulated below for several different transponder bandwidth utilization figures. It is assumed that the total transponder power, P_t, is available for the bandwidth being utilized and that the EIRP and ground station noise temperatures and C/N are as previously indicated for a 36 MHz channel.

TABLE 1
G/T & ANTENNA SIZE SUMMARY

Bandwidth Utilized	4 GHz G/T Min	4 GHz Antenna Dia.	12 GHz G/T Min	12 GHz Antenna Dia.
36 MHz	20.7	10.0 ft	28.3	10.0 ft
18 MHz	17.7	7.0 ft	25.3	7.0 ft
10 MHz	15.1	5.3 ft	22.7	5.3 ft
3.6 MHz	10.7	3.2 ft	18.3	3.2 ft
1.8 MHz	7.7	2.2 ft	15.3	2.2 ft
360 kHz	0.7	1.0 ft	8.3	1.0 ft
36 kHz	- 9.3	9.5 dBi	-1.7	19.1 dBi
18 kHz	-12.3	6.5 dBi	-4.7	16.1 dBi
12 kHz	-14.1	4.7 dBi	-6.5	14.3 dBi
3.6 kHz	-19.3	-0.5 dBi	-11.7	9.1 dBi

The data in Table 1 indicates the potential of very small down-link antennas at the earth station for moderate bandwidth data and voice communications to fixed and mobile units. For C-band a simple antenna having hemispherical directivity would be adequate for voice and narrowband data. Such an antenna could be fixed on the vehicle and would not require tracking to account for vehicle heading, tilt, or the roadway grade. The antenna would, however, need to be sufficiently restricted in field-of-view at low elevation angles to minimize pick-up of the thermal radiation from the earth to maintain the specified receiver system noise temperataure. A K_u band dish or flat plate (microstrip array) antenna several inches across would be adequate for voice at moderate data rate communications, but coarse tracking of the satellite will be needed to correct for vehicular orientation.

MISSION DESIGN AND ORBITAL OPTIONS

Mission design considerations for "Lone Star" begin with identifying the orbits that may be best suited for the intended communications applications. The final orbit selection shapes the overall program from both a technical and a cost standpoint. The orbit selection will drive launch vehicle selection, satellite mass, communication capabilities, the satellite's subsystem's designs, as well as legal and political issues associated with communication satellites. For global or large area communications applications the geostationary orbit (GSO) has especially significant advantages that make it a highly desirable place to put a communications satellite. There are, however, some significant disadvantages to the GSO. One of these disadvantages is that GSO space is a limited global resource; a resource which is in demand by not just individual continents, but by individual countries (and now states?) as well. Therefore, the GSO region of space is regulated by international treaty, and obtaining a usable GSO slot may be difficult, if not impossible. The increased demand for regional communication capabilities coupled with the limited accessibility to GSO drove the consideration of four different orbital options for analysis and study. These options are: 1) the traditional GSO, 2) the Sun-synchronous Twelve hour Equatorial Orbit (STEO), 3) the Molniya Orbit, and 4) the Low Earth Orbit (LEO). All of these orbits have been considered, and some used, for past communications satellite programs. The orbits, depicted in Figure 1, all have unique advantages, as well as their unique disadvantages.

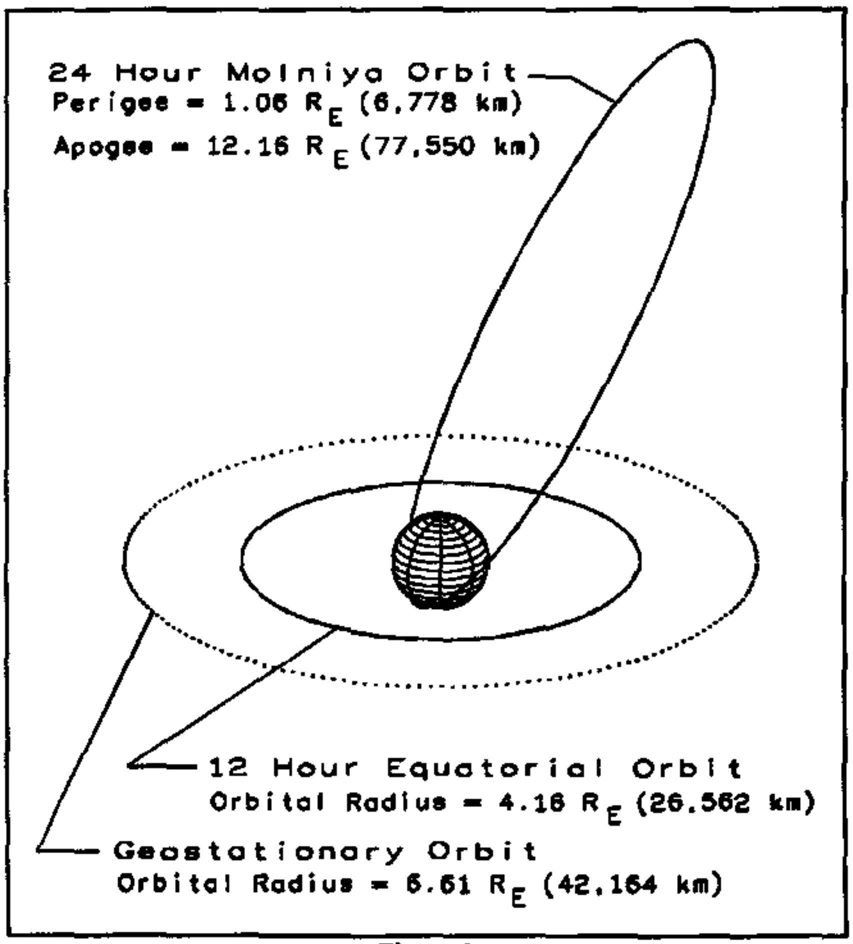

Figure 1
CONFIGURATIONS OF POSSIBLE "LONE STAR" ORBITS

GSO: Geostationary Orbit

The Geostationary Orbit (GSO) is unique in that a satellite in such an orbit will appear fixed in one overhead position relative to a ground based observer. This allows continuous 24 hour coverage and simplifies ground and user station's design. This is possible because of the GSO's unique orbital parameters: the orbital inclination is zero (i=0), the orbit is circular; therefore, having an eccentricity of zero (e=0), the orbital radius is 42,164.8 km (22,767.2 n. mi.), and the orbital period is one sidereal day ($23^h56^m04^s$)[1]. Reaching and maintaining a geostationary orbit is, unfortunately, neither an easy nor inexpensive task since the amount of velocity change (ΔV) required, and therefore propellant mass, is considerable. The ascent to GSO involves four unique velocity changes (ΔV) of the payload. The function of these four ΔV's are:

- ΔV_1: Launch into low Earth orbit (LEO), assumed circular with 185 km altitude
- ΔV_2: Injection into the Geostationary Transfer Orbit (GTO), with perigee at 185 km and apogee at 42,164.8 km
- ΔV_3: Orbital inclination change from initial inclination to O° inclination
- ΔV_4: Circularize GTO at apogee to arrive at final Geostationary orbit

Achieving the final geostationary orbit is obtained by performing two of the four ΔV's as discrete steps and then combining the last two into a single burn. The expense associated with each ΔV is the use of mass in the form of propellant. This mass must be summed into the total payload weight calculation that drives launch vehicle selection. Each additional pound of propellant used in just achieving geostationary orbit will decrease the usable payload mass that arrives there. The only velocity change that a mission planner has some control over is ΔV_3. Since this velocity change is required for inclination control, the easiest way to minimize it is to select a launch facility that is as close to the equator as possible. A summary of the ΔV requirements for reaching GSO from a variety of launch sites is provided in Table 2.

TABLE 2
SUMMARY OF ΔV(km/s) NEEDED TO ACHIEVE GEOSTATIONARY ORBIT FROM LEO

Launch Site (Latitude)	ΔV_2	ΔV_3	ΔV_4	ΔV_{Total}
KSC, FL, USA(28.3°)	2.459	.780	1.479	4.718
San Marcos, Kenya(3°)	2.459	.084	1.479	4.022
Kourou, French Guiana(5.2°)	2.459	.222	1.479	4.160
Tanega Shima, Japan(30°)	2.459	.826	1.479	4.764
Baikonur, USSR(45.9°)	2.459	1.245	1.479	5.183

STEO: Sun-synchronous Twelve hour Equatorial Orbit

The second orbital option considered for "Lone Star" is the Sun-synchronous Twelve hour Equatorial Orbit (STEO). The STEO is a circular orbit (eccentricity of zero, e = 0) having an inclination of zero (i = 0°) and an orbital radius of 26,561.5 km. The combination of these orbital parameters will cause a satellite to circle the Earth twice daily, having a period of approximately twelve hours. The ideal period is $P=11^h59^m$, which allows the satellite to orbit slightly more than twice a day and corrects for the Sun's apparent motion of 0.986° per day[2]. Unlike a satellite in GSO however, a satellite in STEO will not appear fixed in the sky relative to a ground based observer. The STEO satellite will instead slowly sweep out an arc across the sky from the western to eastern horizons. This motion will have two major impacts on the design of the communication satellite system. The first of these impacts is that a single satellite will only be above the horizon for communications access for a limited amount of time each day. The second impact affects the pointing capabilities of both the satellite and the ground and users stations.

The ascent to STEO requires four velocity changes that are similar to the four ΔVs used to achieve GSO, they are:

ΔV_1: Injection into a Low Earth Orbit (LEO)
Assumed to be circular with 185 km altitude
ΔV_2: Injection into a STEO transfer orbit
ΔV_3: Adjust orbital inclination to zero
ΔV_4: Circularize orbit at STEO radius

The magnitude of the required ΔVs are tabulated below in Table 3.

TABLE 3
SUMMARY OF ΔV (km/s) NEEDED TO ACHIEVE STEO ORBIT FROM LEO[3]

Launch Site (Latitude)	ΔV_2	ΔV_3	ΔV_4	ΔV_{Total}
KSC, FL, USA(28.3°)	2.076	1.193	1.435	4.704
San Marcos, Kenya(3°)	2.076	.128	1.435	3.639
Kourou, French Guiana(5.2°)	2.076	.304	1.435	3.851
Tanega Shima, Japan(30°)	2.076	1.263	1.435	4.774
Baikonur, USSR(45.9°)	2.076	1.902	1.435	5.413

A comparison between Tables 2 and 3 indicates that reaching STEO can take slightly less ΔV than required to reach GSO (approximately 9.5% less in the best case). The table also indicates that from high latitude launch sites (above 29.3°) the ΔV requirements for reaching GSO are smaller than the ΔV requirements for reaching STEO. Both of these observations, the small and no ΔV savings, are a result of the costly inclination change manuevers that must be performed at lower apogee when trying to reach STEO. For "Lone Star", however, a low latitude launch, whether from KSC or even farther south, should be readily feasible. Accessibility to a low latitude launch site allows the STEO option to provide some savings in ΔV. At first glance it appears that from a ΔV standpoint the small savings realized in obtaining STEO may not offset the added complexities of operating from such an orbit. This observation is not necessarily true, however, when the station keeping ΔV requirements for orbit maintenance are taken into consideration. In reality, the ΔV savings for STEO station keeping are less than for a GSO satellite. The lower ΔV requirements for station keeping allows less propellant to be carried for equivalent mission durations. The propellant mass savings can be used to lighten the whole satellite, or distributed into other systems.

There are some constraints imposed by operating a communication satellite from STEO. Although the satellite in STEO orbits the Earth twice in 24 hours, an observer at a fixed ground station will see the satellite cross his longitudinal meridian only once a day. This observation is due to the dynamics of the ground station's daily rotation about the Earth's center, while the satellite is revolving about the Earth twice in 24 hours. Figure 2 shows how the ground station and satellite positions relative to one another evolve over a 24 hour period. The satellite is sun synchronized so that it will cross the ground station's meridian at local noon. Synchronization is achieved by making the satellite's true period slightly more than a sidereal half day, to allow for the Sun's apparent motion of 0.986° per day. The STEO satellite remains above the horizon of a Texas ground station for approximately eight hours each day. Only during this time, however, is communication via the satellite possible. Viewed from the ground station, the satellite will rise in the west and travel eastward along an arc of constant angle of declination. The satellite will migrate along this arc at a constant rate of approximately ¼° per minute. Unless an omnidirectional communication system is used, this motion of the satellite requires ground and user station antennas to slew about the polar axis at the same ¼° per minute rate in order to maintain pointing at the satellite.

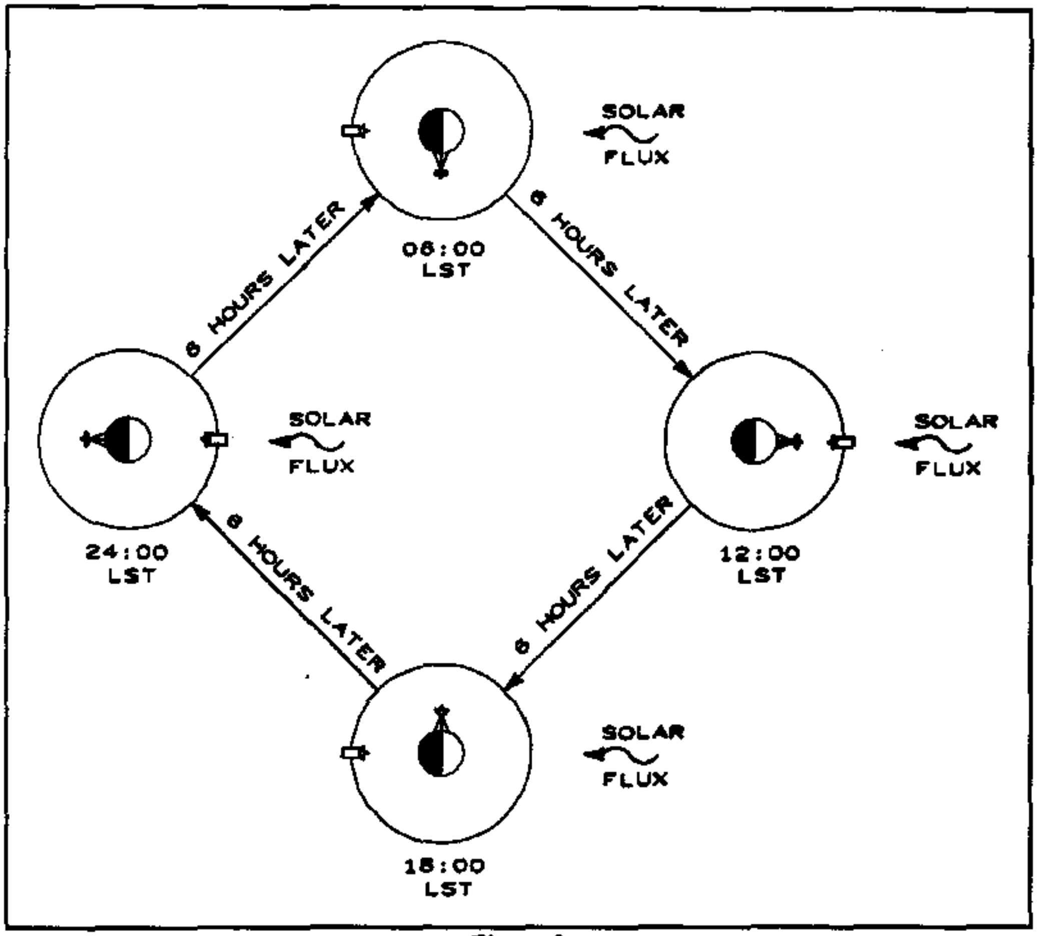

Figure 2
STEO CONFIGURATION OVER 24 HOURS

There are also two significant inconveniences associated with STEO operations: maintaining antenna beam pointing for the communications satellite, and accommodating the variable range between the ground stations and the satellite. Both of these problems are caused by the relative motions of the satellite with respect to the ground stations. When the communication link between the satellite and the ground station is first established, the antenna beam will be pointing to a region near, or on, the Earth's limb. As time progresses, the target region will move closer to the center of the Earth's disk. As more time elapses, the region of interest will migrate across the observed disk until it is on the opposite limb. Shortly thereafter, the communication link is broken as the region of interest completely disappears from sight. The range is greatest when the communication link is first made and when it is finally lost. These two maximums occur when the target region is nearest the Earth's limb. The shortest range occurs at local noon, when the satellite is at the same longitude as the target region. Minimum range corresponds with the target region being nearest the center of the Earth's disk. Unfortunately, the varying range prevents the STEO antenna lock from being maintained by a simple fixed rate rotation of the satellite's antenna pointing vector. Note that this is not a problem for a satellite in GSO because the range is constant and therefore a fixed rate rotation (equal to the 24^h period of the orbit) will keep the antenna pointing vector locked on the region of interest. The pointing problem can be resolved by incorporating an attitude control system into the STEO satellite that allows for variable rate rotation manuevers. The variable range must be accounted for in the communications system design.

Molniya Orbit

The Molniya orbit has become a frequently used non-GSO alternative from which communication satellites have been operated. The orbit's principal advantage is that it can service high latitude regions, which cannot otherwise be serviced by a GSO satellite. The Soviet Union, given its high latitude geography, has made extensive use of the Molniya orbit since the mid 60's[4]. The Molniya orbit is also easier to reach from high latitude launch sites, such as those in the Soviet Union. The Molniya orbit typically has a period of 12 or 24 hours, is highly eccentric, and must have an inclination fixed at 63.4°. The combination of these parameters defines an orbit that allows a satellite to dwell about apogee and therefore remain high above an observer's horizon for a usable portion of each day. A 24 hour Molniya orbit is shown in comparison with GSO and STEO in Figure 1.

To deliver a satellite to a Molniya orbit requires a series of ΔV maneuvers designed to establish proper inclination, perigee, and apogee radii of the orbit. Direct launch into a 63.4° inclination orbit is not possible from either of the two existing U.S. launch facilities. Therefore, an inclination change maneuver must be performed after low Earth orbit insertion to achieve the needed inclination. Additional ΔVs are also needed to establish the proper perigee and apogee altitudes. From a circular low altitude orbit, these orbital changes can be performed with either three or two ΔVs. The two-burn method has been demonstrated to be the most efficient means of achieving Molniya[5]. The nature of these two ΔVs are:

ΔV_1: Injection from LEO into a high apogee transfer orbit
ΔV_2: Raise transfer orbit apogee and perigee to the final desired values, and simultaneously adjust the inclination to 63.4°

The scope of these ΔVs are summarized in Table 4 for various combinations of apogee and perigee radii that will establish the 12 and 24 hour Molniya orbit.

Molniya orbits are functional for communications applications because the satellite spends a significant amount of accessible time above the user's horizon. These satellites spend a majority of their time traversing and ascending the orbit's apogee leg, which is ideally directly overhead of the serviced region. The access time can be up to 8 hours, depending upon the orbits period and the service region's latitude. Much like a STEO satellite, the Molniya satellite will rise, traverse an arc across the sky, and then set during the course of the day. Unlike a STEO satellite, however, the Molniya satellite will not traverse its arc at a constant rate. This rate will decrease as the satellite moves towards apogee, and then increase as it moves away from apogee towards its setting horizon. This variable rate will require that the users station's antennas have the ability to track at different rates.

The nature of the Molniya orbit also makes it a non-synchronous orbit, i.e. its rising, setting, and apogee times as viewed from Earth will vary over the course of the year. Therefore, a single satellite cannot supply constant time of day coverage throughout the course of a year. The communications problems that arise because of this orbit can be resolved by deploying a constellation of satellites in various Molniya orbits. This approach has been successfully used by the Soviet Union to allow communications coverage on a 24-hour year-round basis[2].

TABLE 4
SUMMARY OF ΔV REQUIREMENTS FOR VARIOUS MOLNIYA ORBITS[5]

Period (hr)	Perigee Altitude (km)	ΔV_1 (m/s)	ΔV_2 (m/s)	ΔV_{Total} (m/s)
12	370	2,435	479	2,914
24	370	2,728	385	3,113
12	555	2,423	481	2,904
24	555	2,723	384	3,107
12	740	2,411	487	2,898
24	740	2,718	385	3,103

Molniya satellites must also be able to provide the antenna pointing capabilities necessary to maintain antenna lock. This pointing control requires an attitude control system capable of performing variable-rate attitude maneuvers. The highly variable range between the ground stations and the satellite must also be accounted for in the communications system. This range factor can be quite significant, especially in the case of the 24 hour Molniya orbit with an apogee radius of 77,354 kilometers.

LEO: Low Earth Orbit

The relative ease of access to low Earth orbit (LEO) makes it a desirable place from which to operate any satellite. A variety of launch vehicles can provide direct LEO insertion. This can allow significant mass savings since no transfer-orbit propulsion system is required for the satellite. Communications operations from LEO, however, are constrained by the nature of the orbit. The principal constraint is limited communications access time with the satellite. This constraint is a result of the relatively short orbital periods associated with LEO satellites. Increased altitudes will increase orbital periods but will have a negative impact on a launch vehicle's mass to orbit capabilities. LEO satellites are also not accessible for communication link on every orbital pass. To overcome these constraints several mission options can be considered. Two of the most popular options studied have been the "store and forward" satellite and a constellation of small satellites residing in LEO. Each of these two options can provide some communications capabilities within the orbital constraints imposed by operating from LEO.

On-Orbit Environmental Issues

A satellite spends its operational lifetime in a less than ideal environment. Orbital environmental factors will work to degrade subsystems and cause the consumption of the limited supply of expendables. Radiation exposure, thermal environment, and eclipse cycles all affect a satellite's on-orbit performance over the course of its lifetime. A variety of perturbing forces disturb the satellite's orbit and attitude, requiring that corrective measures be taken. Knowledge of the orbital environment and its affects on space systems is required so that the satellite can be designed to operate and survive the environmental exposure. The nature of the orbital environment varies somewhat from orbit to orbit. Therefore, orbital environments should be compared to assess what impact they will have on the overall mission design and cost. Summarized in Table 5 are several environmental considerations for the four orbital options studied.

TABLE 5
ORBITAL ENVIRONMENTAL FACTORS[2,4,10]

	LEO	12^h MOLNIYA	STEO	GSO
Principal Orbit Perturbing Influence	Earth's oblateness	Earth's Oblateness	Lunar/ Solar	Lunar/ Solar
Annual Station Keeping Reqmts. (m/s/yr)	None Considered	8.33	32	50
Principal Attitude Perturbing Influence	Gravity, Atm. Drag Solar Pressure	Gravity, Atm. Drag Solar Pressure	Solar Pressure	Solar Pressure
Eclipse Season Duration (days/year)	365	365	150	90
Eclipse Duration (min)	Variable, 30+	Variable	58	72

LAUNCH VEHICLE CONSIDERATIONS

The principal cost and design driver associated with a satellite program will typically be the launch vehicle. It is not at all uncommon for launch services to cost more than the satellite itself. Cost estimates for placing payloads into LEO vary within a range of somewhere between \$5,000 to \$15,000 per pound[6]. Rates to higher orbits are of course even greater. Recent information in the press indicates that Intelsat paid approximately \$22,000 per pound to place the Intelsat 6 satellites into GSO using Martin Marietta's Commercial Titan III[7,8]. Given these rates, it becomes obvious why minimizing a satellite's mass is such a necessity.

An attempt is made in Table 6 to summarize the critical parameters of a variety of smaller ELVs that could be used to place a small communication satellite into orbit. The vehicles listed either currently exist, or are in some stage of development. Their capabilities cover a reasonable mass spectrum for the orbits that were considered for "Lone Star." It is recognized that there are a variety of foreign vehicles also available, but these are not considered in the preliminary report. Only a few of the vehicles are, or hope to be, produced in Texas. LTV's Scouts and SSI's proposed Conestogas are the only true Texas launch vehicles in the study.

Payload to Orbit Capabilities

An assessment of mass capabilities for the orbits under consideration was performed using the information in Table 6. All of the vehicles listed have the capability to place some mass into the orbits of interest. This mass could be a reasonable number into a LEO only, or it may be a significant number that could be placed as high as GSO. Since "Lone Star" was conceived as a small satellite, dedicated launch on a larger vehicle - Delta, Atlas, Ariane, and Titan - was not considered. It is possible, however, that a "Lone Star" satellite configured for one of the smaller launch vehicles could be designed or adapted for launch as a secondary payload on one of the larger vehicles at a considerable cost advantage. The dedicated launch vehicle options that remain would use one of the smaller vehicles - Scout I/II, Pegasus, Taurus, or a Conestoga - to place the "Lone Star" satellite into a working orbit. Payload mass estimates for the considered orbits can be made when the data for the smaller ELVs are combined with the parameters for the considered orbits. These estimates are summarized in Table 7.

TABLE 6
LAUNCH VEHICLE PERFORMANCE[9,10,11,12]

Launch Vehicle (Status)	Prime Contractor	Launch Facilities	Estimated Cost	Performance
Scout I(F)	LTV	SMK,KSC,WFF	$10M	480 lbs. to 555 km. circular orbit
Scout II(D)	LTV	?	$15M	990 lbs. to 555 km. circular orbit
Pegasus(F)	OSC	Air launched	$6.3M	950 lbs. to 150 n.mi. circular eq. orbit
Taurus(D)	OSC	KSC, VAFB	$15M	830 lbs. into 28.5° GTO
Conestoga's:				
210-48(D)	SSI	WFF	?	160 lbs. into 37° GTO
310-48(D)	SSI	WFF	?	250 lbs. into 37° GTO
221-48(D)	SSI	WFF	?	550 lbs. into 37° GTO
421-48B(D)	SSI	WFF	?	970 lbs. into 37° GTO

NOTES:
OSC = Orbital Sciences Corp.
SSI = Space Services Inc.
LTV = LTV Corp.
SMK = San Marcos, Kenya
WFF = Wallops Flight Facility
KSC = Kennedy Space Center
VAFB = Vandenburg Air Force Base
F = Flown
D = Development

LEO Payloads. The ELVs considered can place a wide range of payload mass directly into a variety of low Earth orbits. The actual data presented in Table 7 consists of the estimated performance for each ELV to a 400 kilometer altitude circular orbit of various inclinations. The inclination values are based on the lowest inclination orbit possible from currently approved launch sites[6,9,10,11,12]. The mass range available to LEO is considerable and allows for a great deal of flexibility in mission planning for a LEO satellite system. A small "store and forward" satellite could easily be launched as a secondary or piggy-back on any of these vehicles. Alternatively, several small satellites that comprise part of a constellation could be launched simultaneously on any of the vehicles. If one of the larger vehicles is chosen, it might be possible to place all or a significant portion of the constellation into orbit with only one launch.

Molniya Payloads. The Molniya data presented in Table 7 are derived from the given LEO data and is for a 12 hour Molniya orbit with a 400 kilometer perigee altitude. It is important to note that the weight numbers given assume that direct injection into 63.4° inclination LEO is possible. Direct injection may or may not be the case using launch sites that are available today. The significance of this is the ΔV, and therefore mass, savings that are possible since no inclination change would be necessary. If the 63.4° inclination LEO can be achieved, then only one additional propulsion firing would be needed to raise the orbit's apogee and create the Molniya orbit. The weight numbers given also include the dead weight of the spent kick motor used to achieve the high altitude apogee. The true satellite weight is the number in the table minus the dead weight of the kick motor used. With these considerations, the mass range that can be placed into Molniya is considerable.

TABLE 7
ESTIMATED ELV PERFORMANCE (lbs) to CONSIDERED ORBITS[6,9,10,11,12]

Vehicle	LEO	12^h Molniya[1]	STEO[2]	GTO	GSO
Scout I	500[3]	205	120	--	--
Scout II	1,000[3]	415	235	--	--
Pegasus	860[4]	355	205	--	--
Taurus	3,400[5]	1,410	815	1,000[7]	560[8]
Conestoga 210-48	650[6]	270	150	--	--
Conestoga 310-48	1,100[6]	455	260	--	--
Conestoga 421-48B	2,800[6]	1,160	655	1,000[7]	560[8]

NOTES: 1 - 400 km perigee altitude, assumed direct launch into 63.4° LEO, includes PKM dead weight
2 - Estimated values using LEO numbers into a 5° transfer orbit, includes AKM dead weight
3 - 400 km. circular, i=4°
4 - 400 km. circular, i=0°
5 - 400 km. circular, i=28.5°
6 - 400 km. circular, i=37°
7 - 5° GTO achieved from near equatorial launch site
8 - Includes AKM dead weight

STEO Payloads. The STEO data presented in Table 7 are derived from the given LEO data and by assuming that the initial LEO is achieved from a near equatorial launch facility. This assumption appears to be reasonable since the launch vehicle contractors either have, or claim they will have, access to a near equatorial launch facility[13]. These facilities allow for the low inclination transfer orbit to STEO, thereby reducing the magnitude of the costly inclination change maneuver. Since two ΔVs are required to reach STEO from LEO, upper stage propulsion systems must be included in the weights that are injected into the initial LEO. Only two of the launch vehicles considered, Taurus and Conestoga, have an upper stage that can be used for the transfer orbit insertion[11,12]. If any of the other vehicles were used, a perigee kick motor for transfer orbit insertion would have to be carried. An apogee kick motor must be included with the satellite for final STEO insertion regardless of which vehicle is selected. When AKM dead weights are deducted, the available wet satellite weights are 755 and 610 pounds respectively. These are very reasonable weights to work with for a small communications satellite. The available weight coupled with the communications capabilities that can be provided from STEO make it a very attractive non-GSO alternative.

GSO Payloads. Only two of the launch vehicles considered have any real performance to GSO. Taurus and the largest Conestoga have an estimated capability of placing 1,000 pounds into a 5° inclination GTO[13]. The possibility of achieving the low inclination GTO is considered based upon the arguments given above in the STEO discussion. The end result is an estimated 560 pounds delivered into GSO. When the AKM dead weight is deducted from the 560 pounds, the actual beginning-of-life, wet weight for the satellite is estimated to be 515 pounds. There is some hope that this number can be increased once true performance to GTO for these vehicles is known.

CONCEPTUAL "LONE STAR" SATELLITE

To assess the feasibility of the "Lone Star" concept requires identifying the communications capabilities that are achievable from the considered mission approaches. The scope of the communications capabilities is determined by the amount of functional communication payload that is placed into a working orbit. For this communications payload to function, a variety of critical subsystems must also be incorporated into the satellite as support systems. These subsystems are interdependent in operation, yet they all compete for the satellite's limited mass, volume, and power resources. Therefore, the design task is to arrive at a satellite configuration that maximizes the communication payload's capabilities from the selected orbit, minimizes the subsystem's requirements, and meets the mass and volume constraints of the launch vehicle. Presented here is a single conceptual design for a small communications satellite that is intended to operate from a geostationary orbit. The GSO option was selected because it represents a mission profile that is very functional, yet places the greatest constraints on payload mass that can be delivered to orbit.

The conceptual "Lone Star" satellite, in a GSO deployed configuration, is shown in Figure 3. The satellite is equipped with two flat sun tracking solar array panels for power generation and is configured to operate as a three axis stabilized spacecraft. For launch, the satellite is designed to fit within the dynamic envelope of either a

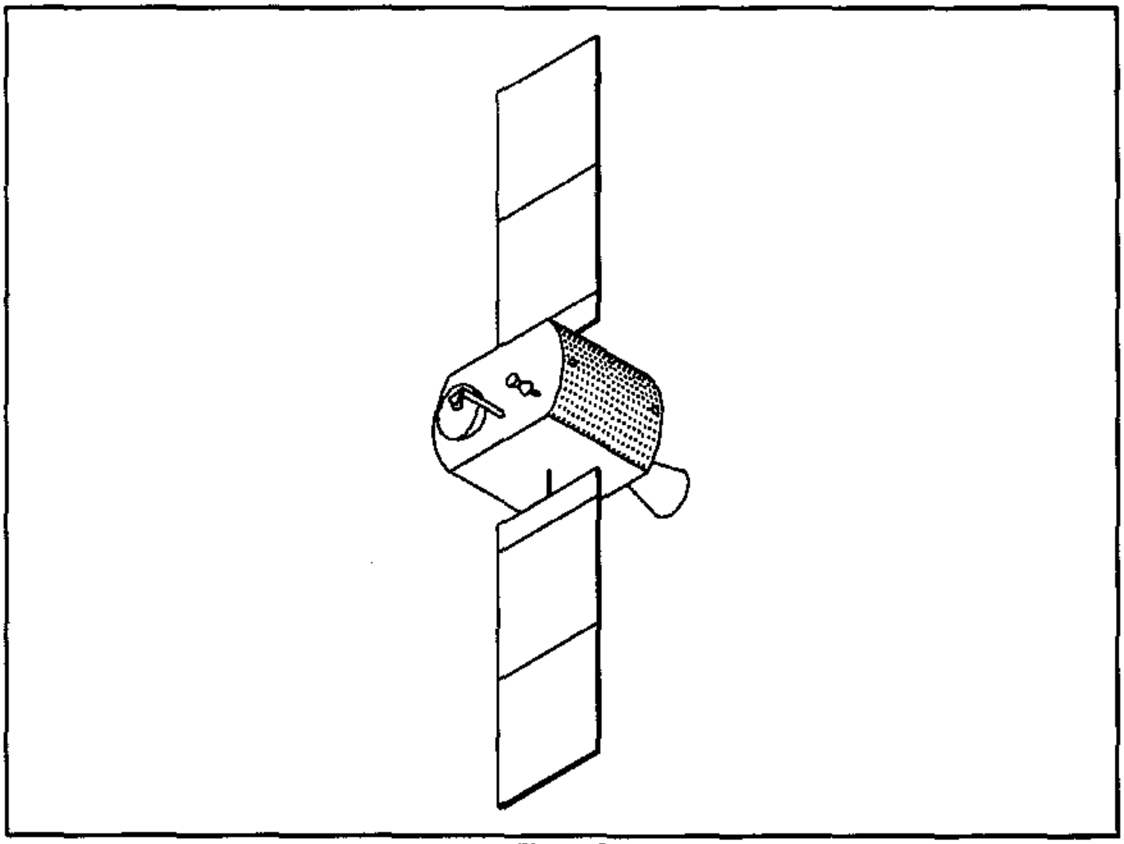

Figure 3
CONCEPTUAL "LONE STAR" SATELLITE

Conestoga 421-48B or a large Taurus payload fairing. The allowable spacecraft mass (including spent AKM) is 560 pounds, which is the maximum anticipated performance of either of the two launch vehicles to GSO (see Table 7). The satellite's communication payload consists of six K_u band transponders, three operating in a vertical polarization and three in a horizontal polarization. The system uses a single 44 cm. diameter narrow beam antenna to provide communications coverage of Texas. The satellite's subsystems and associated expendables are sized to achieve a seven year operational lifetime. The mass and power requirements of the satellite's subsystems are summarized in Table 8. A brief overview of each subsystem is provided below.

TABLE 8
CONCEPTUAL "LONE STAR" SATELLITE MASS & POWER SUMMARY

Subsystems (Quantity)	Mass Total (kg)	Mass Subtotal (kg)	Power Total (W)	Power Subtotal (W)
Communications:	23.82		242.00	
Transponders (6)		11.40		240.00
Antenna (1)		4.55		
Combiner (2)		1.82		
Receiver (2)		1.50		2.00
Coupler (2)		1.82		
Waveguide (6)		2.73		
Power System:	52.55		10.00	
NiCd Batteries[1]		18.50		
Solar Array (2)		20.00		
Control Electronics		4.50		
Array Drive Sys.		9.55		10.00
Attitude Control:	9.8		14.00	
Momentum Wheels (2)[2]		3.30		1.50
Sun Sensors (2)[2]		1.00		2.50
Earth Sensors (2)[2]		1.50		5.00
Gyro Assemblies (2)[2]		3.00		4.00
Torque Rods (2)		1.00		1.00
Command and Control:	8.78		32.00	
Computer (1)		5.00		8.00
Receiver (1)		0.48		1.00
Transmitter (1)		0.30		20.00
Antenna (1)		1.00		
Cabling		2.00		
Propulsion:	101.60		20.00	
AKM Dead Weight (1)		18.20		
Thrusters (8)		4.10		
Tankage		10.80		
Plumbing		7.00		
Pressurizing Sys.		3.00		
Fuel (Hydrazine)		58.50		
Thermal:	7.00		30.00	
Mechanical & Structure:	50.00			
TOTAL:	253.55	TOTAL:	348.00	
LV Mass to GSO:	254.50	ARRAY O/P:	420.00	
Margin:	+0.95	Margin:	+72.00	

NOTES: 1) Approximately 4 kg could be saved here if NiH_2 batteries were used.
2) Quantities given are for a redundant system. If nonredundant then 4.4 kg. could be saved.

Attitude Determination and Control Subsystem

The selection of an attitude control configuration will have major design impacts on the satellite's subsystems; most notably the power and thermal control subsystems. The final preliminary analysis came down to a selection between a spin stabilized configuration versus a three axis stable configuration. These choices are not surprising since the vast majority of geostationary communication satellites built to date have either used a spin stabilized configuration (the classic Hughes design), or they have used a three axis stable configuration (very common among the GE/RCA or Ford Aerospace designs). The principal driver in the final selection process proved to be the amount of mass that can be conserved in the power system if flat sun tracking solar arrays could be used. Analysis and study indicated that the overall satellite mass could be lighter if a three axis stabilized configuration was selected[4].

The system proposed would be a momentum bias system using a single pitch axis momentum wheel, a single Earth sensor assembly for determining pitch and roll attitude, and magnetic torquer bars for primary active control about the roll axis. Sun sensors and gyro units would also be used during transfer orbit operations. Thrusters are used for active control about the pitch axis to support momentum wheel unloading operations and to provide east/west station keeping capabilities. Thrusters for north/south station keeping can also be used for roll/yaw backup control in case of magnetic torquer failure. Such an attitude control system has flown on several communication satellites and has proven to be quite reliable[14,15,16,17,18].

Electrical Power System

The electrical power subsystem must be capable of supplying and regulating 420 watts of electrical power for a 7 year mission lifetime. To accomplish this, the satellite is equipped with two sun tracking solar panels that measure 40 by 72 inches each (deployed). These arrays use currently existing solar cell technology and have been sized to provide the needed 420 watts of power at the mission's 7 year end-of-life. The power system is regulated to 28 volts by a shunt regulator while the spacecraft is in sunlight, and uses a boost converter to regulate the line voltage during battery operations. This approach results in a well regulated 28 volt power bus, which simplifies the design and increases the efficiency of the various individual converters operating from the bus. The power conditioner for the transponders is a step down converter, operating at high frequency, which conditions and regulates the 6 volt power for the transponders.

The battery system uses nickel-cadmium batteries to supply power during solar eclipse. The battery system was sized for a 200 watt eclipse load to minimize battery weight. This means that during eclipse the communication payload operates using only two transponders. Since eclipse season in GSO is about 90 days per year, and even then maximum eclipse is only 1.2 hours per day, this was considered a reasonable trade-off in order to keep satellite mass down. Nickel-Hydrogen batteries would be an ideal alternative, but would have a significant cost impact on the electrical power system.

Propulsion Subsystem

The propulsion subsystem's principal tasks are to perform the thrust firing for final orbit insertion, provide periodic orbit (station keeping) control maneuvers throughout the satellite's operational lifetime, and support attitude control operations. The "Lone Star" conceptual satellite incorporates both a solid system and a monopropellant hydrazine system to accomplish these tasks. A small solid rocket motor is ideally suited for use as "Lone Star's" apogee kick motor (AKM). The AKM is an integral part of "Lone Star" and when fired, provides the ΔV necessary to insert the satellite into its final geostationary orbit. The unit selected for this conceptual design is a Morton Thiokol Star 24 solid rocket motor. The performance of this motor makes it an ideal choice for inserting "Lone Star" into its final GSO position from a 5° inclination transfer orbit. This motor provides an average thrust of 4,825 pounds at an effective specific impulse of 282.3 seconds. The total loaded motor weighs 481 pounds and contains 440 pounds of propellant. The associated dead weight with the motor is approximately 40 pounds[19].

The monopropellant system for the conceptualized "Lone Star" satellite will be used to perform orbital station keeping manuevers and provide support for the attitude control system. The system is configured to use a total of eight hydrazine thrusters that are grouped into four pairs. Two pair (four of the thrusters) are 5 pound thrust units that perform north/south station keeping maneuvers and provide backup roll and yaw control for the attitude control system's magnetic torquer bars. The remaining two pair of thrusters are lower thrust units, .1 to .2 pounds, and are used to perform east/west station keeping and to provide pitch control torque for momentum wheel unloading. The propulsion system operates in a blowdown mode that uses a pressurized inert gas, typically nitrogen or helium, to provide the feed pressure for the hydrazine propellant[4,18].

The hydrazine requirements for the propulsion system can be estimated based upon the amount of ΔV that must be supplied for both station keeping and attitude control purposes. The vast majority of this propellant will be consumed supporting station keeping functions and performing the initial orbital trim corrections necessary after AKM firing. If the orbital trim requirement is estimated to be 10% of the ΔV provided by the AKM, which works out to be 162 m/s, then the amount of hydrazine required by this maneuver will be 40 pounds. The station keeping

ΔV requirements are estimated to be 50 m/s per year, over the course of a 7 year mission this will be 350 m/s which will require another 76 pounds of hydrazine[20]. The propellant requirements for attitude control maneuvering are estimated to be about 50 m/s for a 7 year operational life, which will require an additional 12.5 ponds of hydrazine[4]. Therefore, the total propellant mass requirement for the satellite will be 128.5 pounds of hydrazine.

Tracking, Telemetry, and Control Subsystem

The tracking, telemetry, and control subsystem (TT&CS) combines those elements that allow for satellite control, knowledge of health and performance, and ground station knowledge of satellite orbit and attitude positioning. For "Lone Star" this system would consist of a small onboard computer that would handle the control and switching of the satellite's systems. Interfaced to the computer would be an S-band transmitter and receiver that would allow commands to be up-linked from the ground and satellite telemetry to be down-linked to the ground. The S-band communication link uses an omni-directional antenna system, which allows for communications link to and from the satellite during transfer orbit operations.

Thermal Management and Control Subsystem

The conceptualized "Lone Star" satellite will rely on passive thermal management techniques. The exception would be the use of strip heaters in critical areas to maintain temperatures during solar eclipse. The system must manage the waste heat dissipated within the satellite (about 300 watts) and the incident solar flux onto the satellite. The satellite's three axis stabilized configuration allows for the north and south structural panels to be used for heat radiators. These two panels can provide a total area of approximately 3 square meters that could be used for heat rejection. In addition, the nadir (Earth) pointing panel can be used to reject some heat. The east and west faces of the satellite are blanketed-off with multi-layered insulation.

Mechanical and Structural Subsystem

"Lone Star's" structural system must provide the necessary integrity for the expected loads, while consuming a minimal amount of the available mass resources. This design is achieved by using high strength-to-weight and stiffness-to-weight materials. The early concept considers two honeycomb panels to serve as the north and south structural panels of the satellite. The Earth face of the satellite would be an additional panel made from either honeycomb or machined metal. The communications equipment can be mounted internally on the Earth facing panel, while the K_u band antenna and attitude sensors are mounted on its exterior surface. Additional subsystem elements and the solar array drives are mounted on the north/south panel. These three panels are interconnected and reinforced by an internal truss frame that would support the AKM and propulsion storage tanks.

Optional Subsystems

The initial directives for the "Lone Star" satellite included provisions for exploring new technologies for the purpose of advancing the technical capabilities of Texas industries and campuses. As part of this challange, two subsystems are being considered as additions to the subsystems normally found in a communications satellite. These two subsystems are:

(1) A small scientific package to take advantage of the GSO position. One application for this package would be to provide an early warning of solar storms that sometimes block radio transmissions and also cause electrical black-outs over many parts of North America.

(2) A small package to provide two-way communications and one or more beacons on frequencies available to the radio amateurs. This unit could be used to provide additional emergency communications during disasters where radio amateurs have in many instances been the first and only group to communicate with remote sites.

These sub-subsystems could be made of almost any complexity; however, it is believed that both could be accommodated in a minimum form within a mass budget as small as 10 kg and with a power allowance of 20 watts.

It is expected that the scientific package would be the joint responsibility of the universities, and the amateur package would be provided by AMSAT.

CONCLUSION

A study was made to determine the feasibility of developing a Texas designed, fabricated, and launched communication satellite. A preliminary design was developed that was the result of discussions with state government, university, and industry representatives. After establishing several broad guidelines and reviewing communication link budgets, orbital parameters, and satellite subsystems, a 250 kg., 6 transponder, 3-axis stabilized spacecraft was conceived. The bandwidth of each transponder is 144 MHz for a total of 432 MHz for a set of 3 transponders. Three transponders will be horizontally polarized and three vertically polarized. This design would meet the communication requirements established during the workshop sessions.

In conclusion, it appears that the design developed by this study is technically feasible. A cost review is being made of this design to help the state determine if a satellite should be considered as part of a Texas-wide telecommunication system.

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