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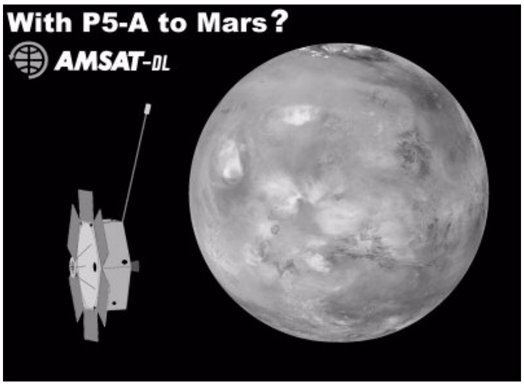
To Mars with P5-A



After the launch of AMSAT-OSCAR-40 the question for the satellite builders of AMSAT has arisen which future projects to pursue.

Although not all systems on AO-40 operate as hoped for, there will be those interested in building a smaller P-3 satellite as a replacement for AO-40 on the VHF/UHF bands. At the same time plans have reached a concrete stage considering the feasibility of a planetary mission by AMSAT. With this article I would like to open a discussion with our members about the future of AMSAT's projects.

Background

During the construction phase of P3-D it became clear that the P3-D structure is suitable to achieve a very high propulsion capabilitywith the built-in rocket propulsion systems (delta v) and that this is sufficient to carry out inter- planetary missions. AMSAT-DL had a so-called ` kick off' meeting in Marburg in 1996 for a satellite (P5-A) which could be put into an elliptical orbit around Mars. In the first instance it was decided to explore the interest for such a mission. The project proposal found a surprisingly large resonance, and concrete proposals for such a project were drafted. Because of the delays in the P3-D program further progress had to wait, and only after the launch of AO-40 the earlier work couldbe resumed. After the reactions which I have received this year, the interest for such a mission has, in the meantime, become greater, especially with the younger people.

At the 1996 meeting it became clear that there was a large demand for a relay station around Mars; there are many Mars missions planned, all of which have the same problem, to bring their data back to the earth. Actually the radio links to the earth are quite difficult, and amateur radio could play a key role, in that it could offer a service with a satellite in orbit around Mars, similar to the beginnings in amateur radio.

Such a mission requires money which can certainly not be raised from AMSAT or from amateur radio. However, since such a project finds interest for many reasons from other institutions, and the ability to finance such a mission appears to be a smaller problem, if it remains in the scope of customary AMSAT projects.

The Challenges

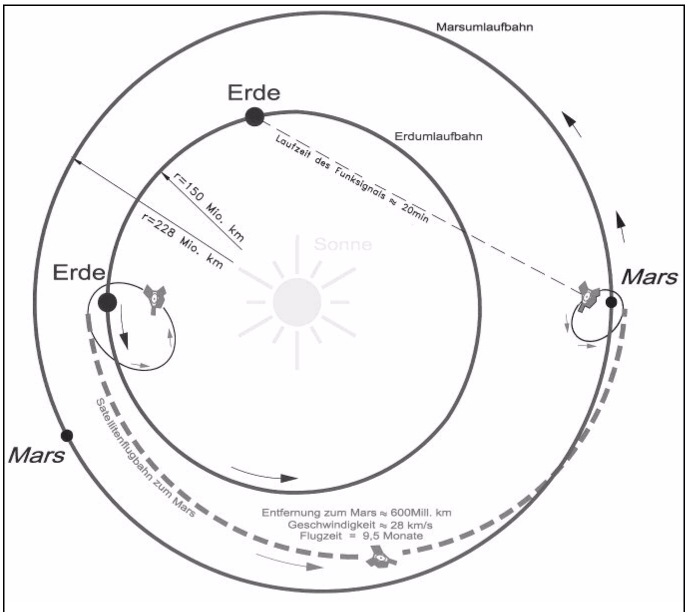
A mission to Mars presents insignificantly larger demands than the construction of P3-D; the space vehicle is actually less complex than P3-D. The challenges lie, rather, in the mastery of the communication links and the flight path. The distance between the earth and Mars comes to between 70 million km and 325 million km, and Mars is thereby about 2,000 to 10,000 times as far as our P3-D satellite. In light of that, the radio link is up to 80 dB (!!!) more difficult. One could say that radio amateurs could not seriously hope to carry out such a mission. I would like to demonstrate that this is not necessarily so.

At these gigantic distances to Mars the space vehicle must be navigated precisely even to arrive at Mars. A distance measuring technique was developed for the P3-D project, which could play a key role with only little additional development.

As already mentioned, the satellite itself does not have to be more complicated. But after it became known that AMSAT was thinking about such a mission, a number of suggestions were received, as to what additional interesting things could be done which such a satellite going to Mars. Certainly it would pay to include such experiments in the plans.

In the first part of this presentation I would like to illustrate more fully the flight path to Mars and to discuss the requirements for operationally reliable radio links from and to the satellite . In a second part I will discuss a space vehicle proposal, which is derived from P3-D, and also present some different experiment suggestions.

The Flight Path to Mars



Geometry of Earth and Mars orbits and trajectory

At the P5-A discussions until now, we came away with the conclusion that only Arianne 5 could be considered as a viable and cost effective launch possibility; P5-A as secondary payload means that we can achieve only a geo-stationary transfer orbit (GTO) and we have no influence on the precise launch time and date. As mission goal we have assumed that we would like to achieve an elliptical orbit around Mars without special requirements. Such an orbit is possible to achieve with the least energy expenditure.

The earth flies in a near circular orbit in a distance of 150 million km around the sun; Mars‘ path is somewhat more elliptical and on average about 220 million km from the sun — both orbits are only a little bit inclined to each other.
As is well known one modifies the orbit of a satellite around earth by first entering an intermediate elliptical orbit, which touches both the original and the desired orbit. Hohmann has shown in the 1930‘s that this method is optimal (with the least driving energy required), if the changes are not too extreme.

In a flight to Mars one proceeds exactly the same way, except that here the central body now is the sun. One leaves the orbit of the earth around the sun by increasing the orbital velocity. The resultant ellipse touches the orbit of Mars at its farthest point from the sun after about 9 months of flight time. At this point one accelerates the satellite once more and thus matches the speed of the satellite to that of the Mars-orbit around the sun. With these maneuvres two complications have to be considered:

1. When one arrives in the Mars orbit,Mars must also be there.
2. At the departure from the earth orbit the spacecraft is actually in an elliptical orbit, and upon the arrival at Mars one wants also to enter in an orbital orbit

The consequence of the first is that one cannot fly from the earth at any time, but one must wait until the flight time for the Mars orbit and the Mars position fit together. This geometry is ` right‘ about every two years and leads
to the fact that one has a 'departure window‘ every two years in which a take off from the earth-orbit is possible. Because the Mars orbit is somewhat elliptical and is somewhat inclined relative to the earth‘s orbit, these windows are not all of equal value. This means that there are windows, which require more or less propulsion energy. Unfortunately the windows in the years 2005 and 2007 are especially poor, and therefore, a mission in these years can transport only distinctly smaller payloads to Mars.

If one calculates the velocity changes of the satellite, which are required to both leave the earth‘s orbit and to reach the Mars orbit, one arrives at values of 2.5 to 3.0 km/s for both maneuvers. With a naive look at the situation one could conclude that a propulsion requirementr of about 5 to 6 km/s is required. With a satellite of the P3-D type one can actually only achieve 2.5 km/s, and such a flight wouod seem impossible.

Actually we are already in an orbit earth. With a closer examination it turns out that the law of energy conservation of physics comes to our aid: if one accelerates the satellite at perigee, only about 1 km/s is needed to leave the earth‘s orbit and to •take along" about 3 km/s. This is the result of the high velocity at perigee and of the fact that the energy is a square function of the velocity.

The same trick can be used at the arrival at Mars: one flies in an orbit that passes close to Mars, and at this point one reduces the velocity of the satellite to enter into an elliptical orbit around Mars. It turns out that the P3-D satellite has enough propulsion power for these maneuvers.

Unfortunately, the problem of the flight path is not yet fully solved. If one wants to leave the earth orbit, the ellipse must have the right orientation so that the departure velocity points into the right direction. The orientation of the GTO-ellipse as given to the satellite by the Ariane launch is a result of the exact launch time and date. Thus we cannot influence it to a significant extent, furthermore the orientation changes continually due to perturbations amd the movement of the earth around the sun..

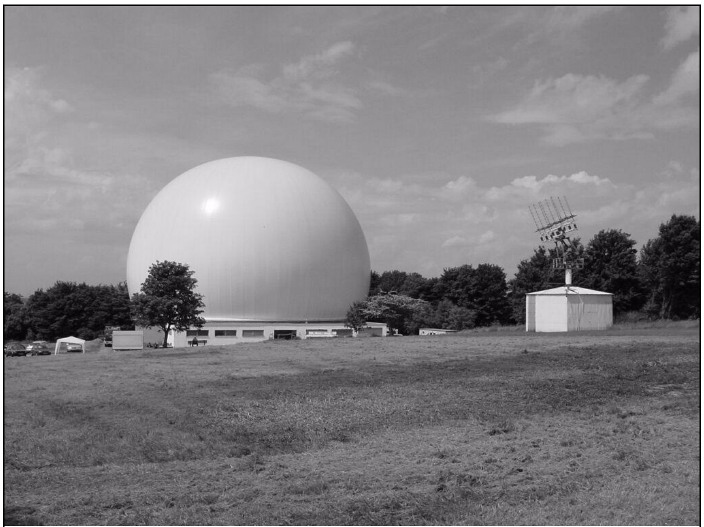
In the past years this problem has been investigated to some extend,because other missions would also like to use the GTO as a cheap launch option. Penzo has found a solution to this problem. It is based on one or more moon flyby‘s to modify the orbit orientation and to achieve the required ` Epoch Synchronization‘. This approach In practice means that it would be possible to wait up to several months on the GTO for the right departure window. Little additional propulsion energy is required for thes maneuvers.

In summary it can be stated that it is possible to fly to Mars with a 2.5 km/s delta-v and to enter there into an elliptical orbit. To arrive at this point the mission would last about one year.

The Communications Links

Since a flight to Mars is basically possible with a satellite of the P3-D type from the standpoint of the propulsion technology, the question arises, which problems must be solved in practice to make such a mission a reality. The great distances and the related problems of the communications links turn out to be one central problem to accomplish such a flight.

On earht the link loss with direct sight of most radio communication systemms are of minor importance . With the great distances to Mars though, a careful analysis is required to construct reliable communication links.. Thus first a small detour into the fundamentals of the radio link design.



The radome of the 20 m dish at the IUZ Bochum

Two formulas are important for the analysis of the radio links: the link losses between two isotropic antennas and the gain of antennas as a function of sicze and frequency. The link-loss between two isotropic antennas over the distance E is expressed as:

dii(dB) = 22 + 20 log (E/lambda)
(where lambda is the wavelength)

This formula is just another statement of fact that:

* the transmitter power is distributed on a spherical surface of 4 Pi E2 and
* the receiving isotropic antenna has an effective area of lambda2/4Pi and
* the power received and going to the receiver is ther fraction of the two areas.

The gain of the parabolic antennas with a diameter of D is approximately

G(dB) = 7 + 20Log(E/lamba)
The actual path loss from Tx antenna connector to Rx antenna connector thus becomes:

D(dB) = dii— Gtx - Grx

(The other misc. losses can be neglected in this order of magnitude study or can be assumed as 3 to 5 dB.)

Several important results arise from these simple formulas for three typical cases:

1. If one uses isotropic antennas or those with constant gain at both ends, the formula for D indicates that with reduced wavelength the link loss increase with each doubling of the frequency by 6 dB. In this case one should select the lowest possible frequency.

2. If an antenna of constant gain is used at one end (at transmitter or receiver), but the other antenna is a parabolic antenna of constant diameter, the antenna gain rises by 6 dB with the doubling of frequency and thus compensates the effect of frequency in the formula for D. In this case the path loss is independent of frequency.

3. If parabolic antennas of constant size are used at both ends, the gain increase appears at both antennas with rising frequency; this means that the net link-loss is reduced by 6 dB with each doubling of the frequency.


The IUZ Cassegrain dish reflector, useful up to 24 GHz

For the mission to Mars case 2 and 3 are the most important ones. Later it will be shown that very high antenna gains are required on the spacecraft, and thus one should provide on the satellite the largest possible parabolic antenna. The P3-D structure permits to give the upper side of the structure a parabolic shape . This antenna then can have a diameter of 2 m. On the ground one would likewise use a parabolic antenna, so that case 3would be the normal situation for the mission - thus the highest possible frequencies should be used for the radio links.

For the design of the mission, however, the following case is more critical: if the the spacecraft, for whatever reason, cannot be pointed towards earth, omni antennas or antennas with small constant gain must be used on the spacecraft to maintain communications.. It is essential that command operations remain feasible and contact with the spacecaft is not lost under these circumstances. For this case the link-loss is independent of the frequency (case 2 above).

In the above elementary analysis the choice of frequencies has been driven only by link and antenna performance. In the real world, other factors need to be taken into consideration. Actually there are four additional considerations:

1. The effective noise temperature at the receiver. Technology of the devices for receiver inputs has become so good that the effective noise temperature of the sky determines the actual attainable receiver sensitivity. In the frequency range of 1 GHz to about 12 GHz the sky temperature has a flat minimum, and this would be the preferred frequency range from a noise performance point of view.

2. The power and efficiency of the transmitter. The efficiency falls off with increasing frequency with semiconductor ransmitters. At higher frequencies, however, it may make sense to use travelling wave tube transmitters in the spacecraft which have good efficiency to above 12 GHz. But also this point argues for not to use frequencies above 12 GHz.

3. With increasing frequency the beam-diameter of the antenna decreases. A 2 m antenna on the satellite at 12 GHz has a beamwidth of about 1° and this is about the limit which comfortably ca be handles to the P3-D attitude control system.

4. Frequency stability. Control of frequency stability above 12 GHz becomes problematic , especially at lower data rates.

To sum up, it is fair to state: if the frequency is limited to 1 to 12 GHz, the effective link performnace is indeed driven mainly by choice of frequency and antenna size and the other factors have only a small influence on over all performance.

After this excursion into the fundamentals of the radio link design we can now approach our concrete problem, namely to examine how the radio links will perform to a space vehicle to Mars.

So first the frequency choices have to be made. Because of the above considerations the following bands are natural choices: 10.5 GHz, 2450 MHz and, for completness sake 435 MHz; as ` worst case‘ a distance of 375 million km will be assumed. Additionally, antenna gains for antennas with a diameter of 2 meters and 20 meters will listed and the total loss is calculated or several practical cases. (Table 1.)

These loss values by themself do not tell us much yet as what performance to expect. We are really interested in what data rates can be transmitted. For this, further assumptions are needed, and one must distinguish between uplink (to the satellite) and downlink (from the satellite).

The following assumptions for the downlink are made:

1. The transmitter power is 100 Watts.
2. Coded PSK is used requiring about 5 dB Eb/N.
3. The noise temperature of the receiver is 100K.

These values are quite conservative, but since the various smaller losses are ignored, they are a good basis for planning. With these figures it turns out that a maximum loss of 223.6 dB is permissible for 1 bit/s.

Table 1

	10500 MHz	245 MHz	435 MHz
d (375 million km)	284.4 dB	271.7 dB	256.7 dB
g (2m)	43.9 dB	31.3 dB	16.2 dB
g (20m)	63.9 dB	51.3 dB	36.2 dB
s (2m on Sat, 20m Ground)	176.6 dB	189.1 dB	204.3 dB
s (2m on Sat, 2m Ground)	196.6 dB	209.1 dB	224.3 dB
s (5 dBi on Sat, 20m Ground)	215.5 dB	215.5 dB	215.5 dB

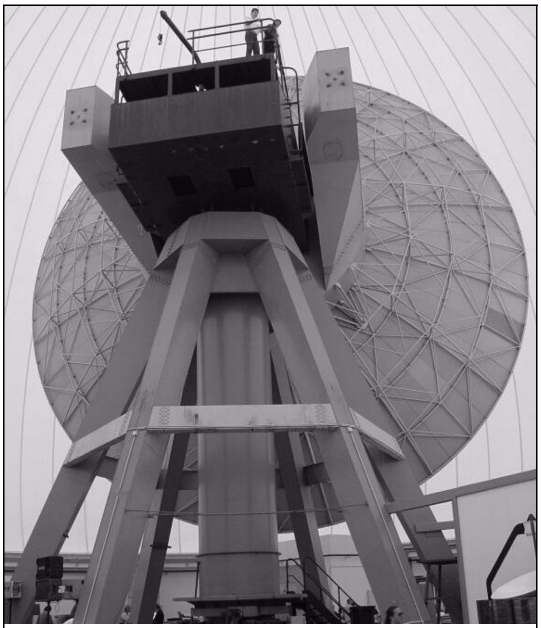
From Table 1 it immediately follows that in the case of the 5 dBi antenna on the satellite and the 20 meter ground antenna there is only 8 dB margin; that means that under these conditions a data rate of 6 bits/s can be achieved at best. The selected frequency does matter in this case. This is the emergency situation if the attitude control does not point the 2m antenna towards earth.

With the 2m antenna on the satellite and a 20m ground antenna, the situation looks much better: There is a margin of 47 dB, which supports a data rate up to 50,000 bit/s on 10.5 GHz. In fact, this margin permits the reception of up to 500 bit/s with only a 2m antenna on the ground.

However, these figures are only valid for a 10.5 GHz downlink; at 2.45 GHz the data rate with the 20m antenna sinks to 2800 bit/s and on 435 MHz to 85 bit/s. This means that on these frequencies the 2m ground antenna delivers only data rates of 28 bit/s (2.45 GHz) and less than 1 bit/s (435 MHz) . This demonstrates the large advantage which is provided by using 10.45 GHz as downlink.

2.45 GHz would be the preferred choice for the uplink (primarily the command channel) — a different band from the downlink also would eliminate interference between the links and the critical case without attitude control is not frequency dependent, anyway. The transmitter power can be assumed to be 1 kW (synchronized microwave oven magnetron)- the o-ther assumption are similar as those for the downlink. Losses up to 233.6 dB are permissible for data rates of 1 bit/s.

For the emergency condition (loss of attitude control) up to 80 bit/s for command operation are possible with the 20m ground antenna. It actually would make sense to provide for larger safety margins for this case and reduce the data rate to about 10 bit/s.


The dish with tower and engine rooms

With a2m antenna on the ground the uplink data rate would fall below 1 bit/s, a data rate that is operationally barely ac-ceptable. Thus, a 20m antenna is definitely needed for the execution of the P5-A mission, if only to provide a reasonable mission safety margin. If the attitude control points the satellite to earth, 100 to 200 bit/s can be transmitted by 2m antennas; this means that for routine command operation a 2m ground antenna is suf-ficient.

Because of the need to have a large antenna available for the mission, AMSAT-DL in 1997 signed a sui-table agreement with the management of the Bochum 20m antenna. This antenna, which was built in the 1960‘s, currently isbeing refurbished for the newer requirements. The agreement states that the anteanna will be available on priority basis for the P5-A mission after the launch. Because of the difficult links to Mars it seems desirable to make his 20m antenna and also other antennas with a diameter of 2 to 3 m acces-sible tover the internet in order to make P5-A available to a large number of users. AMSAT would prefer to have the command operations remote from the sie of the antennas, This presents the challenge for radio a-mateurs to become proficient in the operational and technical requiremts posed by the mission.. A large part of the technology and also some of the operational techniques can be developed and tested on AO-40.

Summarizing, it can be stated that the radio operations of a Mars mission still are wothin reach of amateur radio, achievable, especially if the Internet is inlcuded. But for emergency situations a large antenna must be available.

In the second part of this article the key problems building the P5-A satellite will be discussed and also so-me interesting experiments for this satellite will be presented.

So far it was shown that a spacecraft based on P3-D can be modified so that a flight into an orbit around Mars becomes possible. Now we will examine the details of how the spacecraft needs to be modified to be suitable for such a mission.

We already noticed that two requirements esentially drive satellite design:

1. The propulsion capacity of the satellite has to be in the order of 2.5 km/s to be able to reach the ne-cessary orbits.

2. The radio links are quite difficult, consequently an antenna with a diameter of about 2 m on the spacecraft is necessary. This also has consequences, naturally, on the attitude control.

The Propulsion Requirement

The necessary propulsion performance is needed for the most part at two places: During the departure from the earth‘s orbit and during entering into the Mars orbit. These maneuvers have to be carried out during the perigees of the respective elliptical orbits, and they must be of a very short duration. This means that only chemical propulsion can deliver the required thrust. This suggestst to use the same 400-N propulsion system which has already been flown on P3 satellites several times.

The satellite propulsion capability Vs, results from the propulsion system‘s characteristics (Isp, which is the effective exit velocity of the rocket motor, also called the specific impulse) and from the fraction of the fuel mass (mf) to the satellite‘s dry mass (md).

Vs = Isp * ln((mf+md)/md) [1]

Take-off weight mf + md = 650 kg, Isp = 3.03 km/s:

Drive Performance Vs:	Dry Mass md:	Fuel Weight mf
2.25 km/s	309.3 kg	340.7 kg
2.50 km/s	284.8 kg	365.2 kg
3.00 km/s	241.5 kg	408.5 kg

For planning purposes a propulsion requirement of 2.5 km/s can assumed; the precise value (including the reserves for navigation inaccuracies) depends on the launch year and the details of the the various required maneuvers. Currently there are studies underway to determine the necessary range; actually the necessary value might increase by up to 20% or be reduced by up to 10%. The 400-N motor has an Isp of 3.03 km/s. According to formula [1] this fixes the fuel mass if a maximum take-off weight of 650 kg as in the case of AO-40 is assumed:

The 400-N propulsion system uses MON1 (N2O4) as oxidizer and MMH as fuel. The density of MON1 is 1.445 g/cm3, and that of MMH is 0.8788 g/cm3. The ratio of the densities is 1.665. For this reason the engine has been adjusted to operate with this mixture ratio;, thus tanks of equal size can be used for both propellants.

Both propellants together have an average density of 1.1619; thus for the above fuel quantities tanks of a total of 294, 315 and 352 1 are necessary. AO-40 has six tanks of each 49 1 (294 1 total), which means that those tanks would only be adequate to achieve 2.5 km/s. It therefore makes sense to plan for somewhat larger tanks to be able to deal with more difficult propulsion situations. Moreover, the tanks were specially fabricated for us in Russia; currently it is not clear whether more tanks can be gotten from this source. In any event it must be investigated if even larger tanks can be accomodated in the sapacecraft. In this case four rather than six tanks could turn out sufficient ; the freed-up volume could then be used for subsatellites (see below).

From the figures above it is clear that the mass of the fuel and the resultant volume for the tanks constitute a very important element in the design of the satellite.

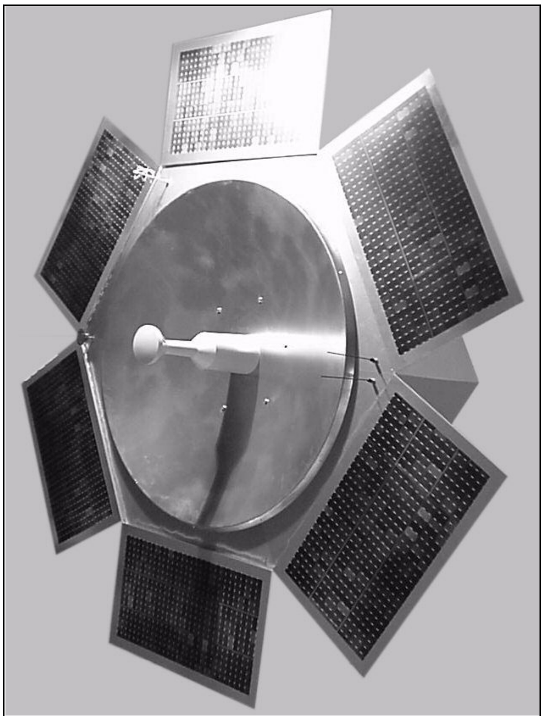
The Antenna and the Satellite‘s Performance

The diameter of the P3-D structure‘s permits the use of the upper face of the satellite for a parabolic antenna with an approximate diameter of 2 m. The design will be influenced primarily by the requirement to keep the structural height of the satellite as low as possible in order to use the available SBS structure (Fig. 3). The SBS was flight-qualified for the launch of AO-40. A new flight qualification would be required for any changes, which would result in appreciable effort and cost.

Basically a flat form of the antenna is desirable, because then the least amount of the satellite‘s volume is lost due to the concave format. A parabolic antenna becomes flatter as the focal length becomes larger. However, then the feed system is located at a greater distance from the reflector, increasing the total height of the satellite. An improvement of this situation can be achieved by using a hyperbolic secondary reflector (Cassegrain-antenna). This reflector is arranged between the focal point and the main antenna shortening of the overall height. The shorter the construction, though, the increased shadowing from the subreflector will reduce the gain of the antenna.

Besides these basic considerations the very practical question has to be addressed, namely how to build the antenna with its parabolic shape. One possibility would be to install a commercialy antenna made of aluminum thus forming the upper surface of the antenna. In this case, the internal vertical walls of the satellite must be shaped such that they can be riveted to the antenna. This solution is limited to commercially available antennas. It would be good luck if an antenna with just the required diameter and the necessary focal length could be bought which is also would be especially lightweight.

Alternatively the antenna could be built from sheet metal segments. Then the main problem becomes how to create the required shape with the necessary precision. The same problem also arises in building the auxiliary reflector, although this structure could be made out of moldable composite material (fiberglass, for instance). The construction of the antenna presents a chalenge, which needs special attention, in any event.



A first model for a P5-A spacecraft

An antenna with a 2-m diameter has a beamwidth of about 1° at 10.5 GHz; this means that the spacecraft must be pointed towards earth with an accuracy of about 0.1°. At this point I won‘t discuss how to achieve this degree of accuracy on the satellite nor how to maintain it. I would rather like to review the impact of this pointing on the rest of the operations of the satellite. In fact for his question it is decisive of whether spin-stabilisation or 3-axis control is being used.

The orbit geometry in Figure 2 shows that the relative alignment between the spacecraft and the earth will not change significantly during the flight to Mars. Actualy we will exploit this characteristic for the electrical power generation (see further below).

The attitude of the satellite ultimately will be stabilzed by a gyro-effect; either the complete satellite is set into rotation or reaction wheels are used to store the angular momentum. Attitude thus is fixed in inertial space. The orbit around the sun, however, reults that the direction to earth changes. The attitude of the satellite thus needs to be corrected daily by approximately 1°.

This is quite unwelcome with a spin-stabilized satellite; for a directional accuracy of 0.1 approximately 10 daily corrections are required. Since there is no magnetic field between the earth and Mars, it means that a small rocket motor has to expel mass for this purpose.

Because of this spin-stabilisation of the entire satellite turns out to be less attractive than one might naively assume. On the other hand AMSAT has developed reaction wheels for P3-D, which could be used for its mission. Using this type of attitude control there is always one axis (the direction of the sum of the angular momentum of the wheels) about which the stabilization is quite unstable and must therefore continuously be controlled. Considering the Mars mission geometry this axis would be selected the one perpendicular to the ecliptic; this would turn the disadvantage of the instability into an asset. Since the control would require a continous correction in the East/West direction, it could accomodate the the daily geometry changes by the same token without having to expend mass. In addition this geometry would allow linear polarization for the communications links; this would simplify the ground installations.

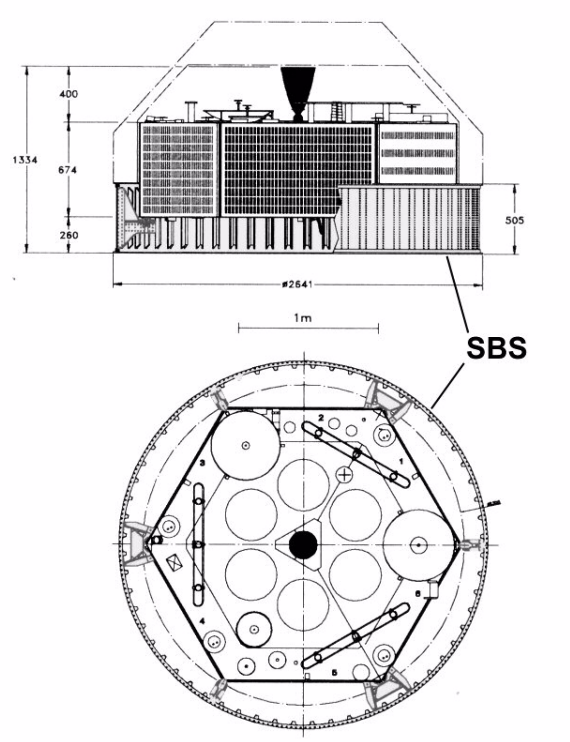
A big advantage of the three-axis control would be the ability to temporarily use different orientations of the spacecraft for experiments on Mars. The disadvantages of three-axis control is it's inherent complexity and possibly lower reliability; also the spacecraft will be heavier.

And during the 400-N engine propulsion phase spin stabilization would be needed, anyway. Thus the use of the spin stabilization cannot be completely avoided during some parts of the mission.

Electrical Power Generation and Thermal Considerations

The radio links require that the orientation of the satellite‘s antenna be toward the earth most of the time as well as during the flight to Mars and in its orbit. This also fixes the geometry with regard to the sun. Since the orbit of the spacecraft is outside of the earth orbit around the sun, most of the time the direction to the sun never deviates more than 45 from the direction of the earth. This suggests to let the solar generator look in the same direction as the earth, and avoids a separate orientation of the generator. Of course this is associated with a loss like on AO-40; ideally the sun would shine perpendicularly onto the generator. But the complexity of a steering the generator to look at the sun simply can be avoided by providing a somewhat larger generator to produce the same amount of power.

The picture of the model (Fig. 1) shows a possible realisation of this concept: during the launch six panels are located parallel to the six exterior walls of the satellite and thereby take up little room. After the launch these six panels will be swung upward and thus look in the same direction as the parabolic antenna.



The SBS to carry a P3/5 satellite on top of an ARIANE 5

The solar constant in earth orbit (the amount of ligh-power per area) is 1.4 kW/m2, and it goes down to 0.6 kW/m2 reaching the Mars-orbit. However, this does not mean that only 40% of the power from the solar generator can be obtained. Actually also the temperature of the solar generator goes down. This increases the power production efficiency, and about 50 — 60% electrical output can be expected in the Mars-orbit.

If panels of equal size of AO-40 were used, there would be a total area of 3.78 m2 available. At a area efficiency of 10% (including battery losses, over all) it means that in the earth orbit with perpendicular illumination 500 W are produced. With an illumination offpointing 60°, we still would get 250 W. On Mars the available power would then be greater than 250 W or 125 W, respecively. With present planning this just would be adequate for the operations including running a 50 W transmitter.

Managing the temperature of the spacecraft is more difficult because of the reduced sun‘s radiation in orbit around Mars. The temperature of the space vehicle represents a balance between the radiated power (infra-red) and the incoming radiation from the sun (light). The power from the sun depends only on the distance from the sun, not on the temperature of the space vehicle. In comparison, the infra-red radiation is a function of the temperature and follows the Stefan-Boltzmann law:

P(radiated) = e * A * Sigma * T 4 where

e = Infra-red emission coefficient of the surface (wavelenght about 10 micro-m)
A = Area of the Satellite
Sigma = 5.67*10-8 W m-2 K-4
T = Absolute temperature in Kelvin

By selecting a (the absorption coefficient for visible light of the surface) and e more or less any desired temperature of the spacecraft can be set for constant solar input. Unfortunately, the incoming solar input (sunshine) changes very considerably from the earth's orbit to the Mars orbit, and consequently the temperature of the satellite changes. If the temperature in earth's orbit is set at 40C, for instance, the temperature at the Mars orbit will drop to -20C.

This range is too large to be operationallly useful. Arrangements thuis must be made which change either the surface characteristics or its geometry such that the temperature remains more or less constant.

There are several methods to this end. For instance, shielding films can be provided on the surface which greatly reduce the amount of sunlight heating the spacecraft in the earth orbit. Later these shields can be discarded while in transit thus increasing the temperature. Also dedicated mechanical arrangements can be provided which thermostatically (bi-metal) expose the visible surface in smaller or larger zones and thereby regulate the temperature. Heat pipes were also discussed which operate with gas ballast and would become inactive below a certain temperature.

In the course of developing such a spacecraft, studies have to show which of these methods or which combinations are simplest and most reliable, and thus which are best suited for this mission. It would be sufficient if the temperature could be elevated by 30° C in a one-time event, since during the flight to Mars the temperature will constantbe be getting lower as mars is approached. It is obvious from these explanations that during the entire development phase of the satellite the temperature situation must be kept in mind, since it represents a central problem for such a mission.

More Special Requirement of the Mission

Aside from the discussed problems there are several requirements of the basic mission that must also be carefully addressed. In comparison to P3-D these are mainly:

1. Navigation

The spacecraft must be guided accurately to within a few hundred kilometers of Mars, so that entry into the Mars orbit becomes feasible. Currently it not clear how this can best be implemented, since AMSAT has neither the experience nor the ground equipment available, which NASA has. On the other hand sufficiently precise position determinations are possible by ranging measurements, so the problem appears to be soluble. In the vicinity of Mars an optical camera can additionally be used.

2. Attitude Determination

On P3-D/AO-40 optical sun and earth sensors are used. For the flight to Mars the sun can also be used as one direction reference. To determine the attitude, another direction needs to be determined, which probably makes a star sensor necessary. In the past a bright star (Canopus) often has been used which is not far from the South Pole of the sky and thereby a faciltates the orientation in regard to the ecliptic.

3. Attitude Control

Independent of whether reaction wheels are used or if the spacecraft is spin-stabilized, it will be necessary apply torques to the spacecraft to effect the attitude control. In the past we used the earth's magnet field for this. In interplanetary space there is no sufficiently strong field so that a small rocket is needed for this purpose. which naturally needs expendable material (fuel). Thus the expected torque requirement for the execution of the mission must therefore be carefully estimated during the planning of this subsystem. Possibly a part of the required torque can be generated by light pressure, a technique that has been used by AMSAT before.

4. The Communication System

One of the function of P5-A should be that other payloads, e.g. on the ground of Mars, could send their data to P5-A . Our satellite would then relay the data to earth, which could then be received by radio amateurs, processed accordingly and then eventually relayed and made generally accessible. This implies that not only the radio technology of the P3 formats has to be supported, but also that P5-A can handle the communication formats as they are used by ESA or by NASA, for example. Actually the necessary technology is already largely available within the RUDAK payloads: many communication formats there can be sopported just by the right software.

Sub-Satellites

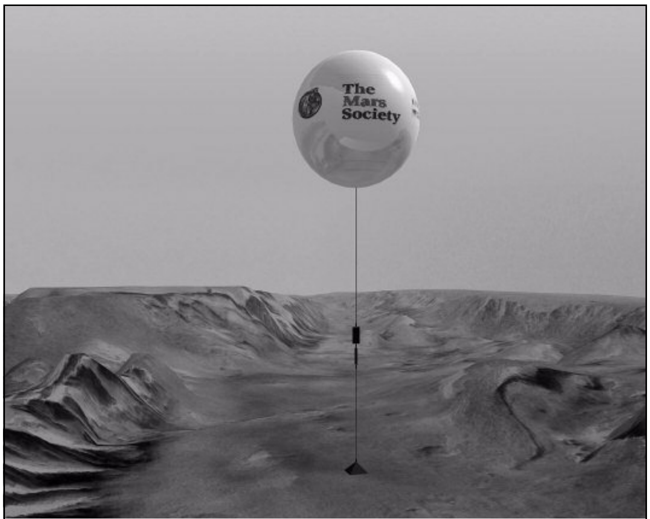
AMSAT has received several suggestions stipulating that P5-A should take along one or more sub-satellites to Mars. These sub-satellites would be separated in the Mars orbit and perform specialized tasks. Two of the proposal I would like to illustrate here in greater detail.

One of the main questions of Mars research is where the water has gone, which it is fairly certain existed on Mars during an earlier time. Today we don‘t know whether there are large quantities of water still in the ground or whether the water has totally disappeared. This question does not only have a big importance for planetary physics; if water is still accessible, it would make human exploration of Mars much easier; water can be used to generate breathable oxygen as well as for the production of rocket fuel for the return flight.

If water is still in the ground in some locations, a certain vapor content should be measurable in the air at these locations. A technique called •radio occultation• has already previously been used to perform such measurement. If two satellites are in contact with each other, the time it takes for the radio singals to make the roundtrip can be measured relatively easily. If the radio signal propagates through the Mars atmosphere, the radio wave is slowed down somewhat when passing through a volume of gas (exactly like the velocityof light which is lower in gas than in a vacuum). With two satellites in different orbits there are always occurances in which Mars gets in between the satellites and interrupts the link. Before this takes place the radio signal passes through ever lower layers of the atmosphere, until it finally reaches the ground and is shadowed. If the propagation time is measured during this process, an exact profile of the atmosphere at that point can be constructed at which the radio wave touches the Mars ground. Thereby the water content results in a clearly measurable effect. With multiple satellites this process naturally occurs more frequently, and observing many of these occultations, the water content on many points on Mars can be determined. From these points a rough map of Mars with its water deposits can be generated.

These measurements are technically very easy to perform; merely sub-satellites with coherent transponders are needed, which can be built at little expense. The necessary sub-satellites can be very small, for instance, cubes with an side length of about 10 cm (so-called Pico-Sats), These are especially well suited to be carried along a mission such as P5-A.

We received anotehr proposal from the German Mars Society. In this case a somewhat larger sub-satellite should have to be carried along. This satellite would be ejected in a Mars‘s orbit. After the separation and at the right location a small solid propellant rocket would be ignited which makes this satellite enter the Mars atmosphere. A heat shield would decelerate the satellite, and at about 10 km altitude a balloon would unfold and be inflated. This balloon will then fly around Mars a few times taken along by the high velocity winds of the upper Mars atmosphere and send pictures and other data of this flight to P5-A.



Proposal of Marsballoon by Marssociety Germany

This brief description clearly shows that this project is technically more demanding and difficult than the radio occultation satellites. Further work is needed to find out whether this project is feasible, and whether it is compatible with the technical parameters of P5-A.

How to Proceed?

Meanwhile, several studies have been started to clarify the addressed technical questions and to definitively work out the requirements and their impact on the satellite design. We expect that during the next six months a reference or so a reference design will emerge which then can serve as a basis for the P5-A project.

Simultaneously it will become clear which persons and which organizations want to collaborate on this mission. Closely tied in with this is also the question which organizations want to finance such a project; it is clear that amateur radio and the AMSAT members cannot fund such a project. Depending on these results and the corresponding interest of the members, AMSAT-DL will then have to decide how much weight the P5-A mission will receive for the future of AMSAT-DL.

But even now it is already very apparent that P5-A presents an interesting challenge for a large number of young people, and that AMSAT-DL can already note a gain of highly motivated people and new talent created by this project. Communication technology and amateur radio‘s role in society have changed dramatically in the past ten years. The planetary space beyond earth‘s orbit is perhaps exactly the chalenge that AMSAT needs on its path into the future to put it right on the cutting edge of technological development and space travel.

(translation: John J. Bubbers, W1GYD, post-editing: Karl Meinzer, DJ4ZC)