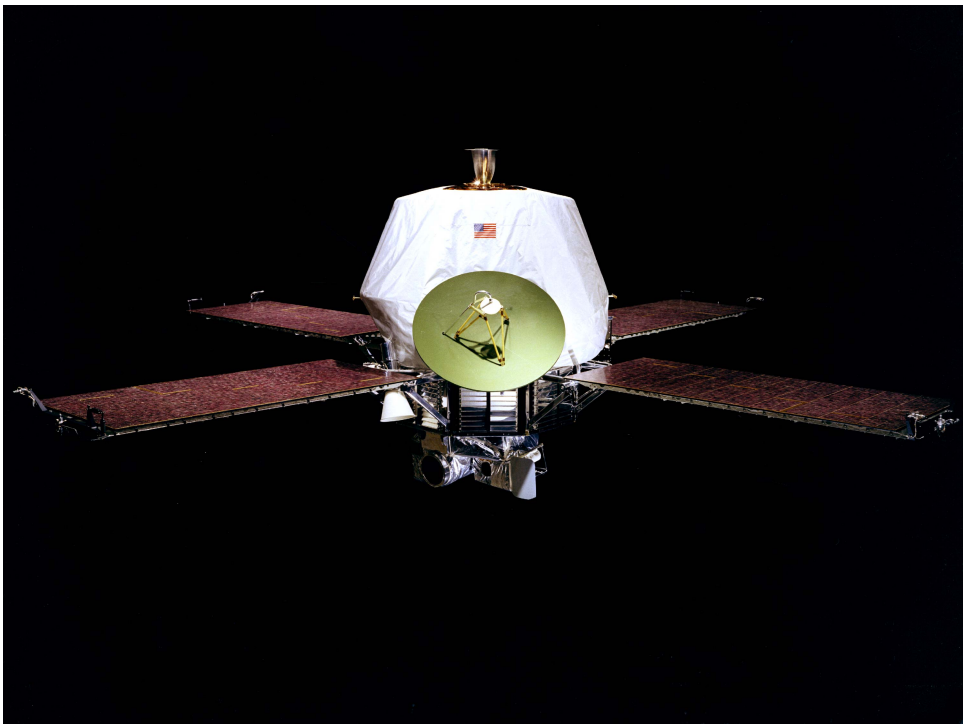
Mariner 9

The first space vehicle to orbit another planet.
A brief history of early space, imaging and data transmission sciences.



The spacecraft showing composite white micro-meteorite shield and reflective blanket covering the fuel tanks.

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Date : December 2017

Preface.

This is a history of a space mission. I have written it to be accessible by the lay reader and have mostly avoided the use of complex mathematics. It will however be useful background, possibly inspiration, for those who wish to delve into the deeper scientific aspects. There is a historical background to this story where a number of developing technologies converged toward the end of the 1950's to make space travel viable. Firstly, although rockety was being experimented with during the 1930's it was not until the V2 of WW2 that an engine of suitable power was available to boost a sensible load to a sensible velocity and altitude. This V2 technology was captured, analyzed and developed by the USA, Soviet Union and the UK post war. It then took around 8 years to develop lighter weight and more powerful engines. If you look up a picture of a V2's engine you will see that it appears to be rather agricultural and heavy compared to what was to come. The drive for development came from the cold war, therefore booster technology was essentially an adaptation of ICBM (inter-continental ballistic missile) technology. Secondly, in 1947 Bardein, Brattain and Shockley came up with a workable transistor which would eventually replace all the glass vacuum valve tubes then in use. The significance of this was that the predominant glass vacuum valve tube was not suitable for rocketry because the vibration would very likely damage it, indeed prior to this all large aeroplanes carried a wireless operator whose job amongst others was to frequently replace faulty valves in the radio and radar system. The V2 used a three axis gyro system for flight control which maintained directional stability, once in flight the direction could not be changed. Thirdly, as a result of improved electronics we can also say that radar technology had sufficiently improved to enable better tracking and reception of signals from vehicles in and above the Earths orbit, as was demonstrated by Sputnik in October 1957. These improved electronics also faciltated the development of on board missile guidance systems, with improved gyros and new accelerometer technology which in turn allowed for better telemetry to send system data back to Earth for analysis. Hence faults might be detected, understood and improvements made. This new technology also enabled the development of flight ready scientific sensors. Finally the work on developing new types of rocket fuel had largely been done allowing choices of engine types with differing fuels for into orbit and then on to deeper space operations. We tend to look at the big things like boosters which are very impressive, but one might also argue that because the improved electronics were initially developed in the USA, to a far higher level than anywhere else, this enabled them to more rapidly develop their space programme. In particular, the extensive use of digital computers to run thousands of orbit simulations and model flight characteristics was important.

The Mariner programme ran for over 15 years (in fact the derived Voyagers are still running). Across NASA, JPL plus all the aerospace and computing industries this single mission required the involvement of many thousands of people. This text focuses solely on some of the technology used to enable the mission. The spacecraft carried many scientific sensors of which each performed valuable service, however rather than focus on the scientific results I have chosen to focus on the infrastructure and various sub-systems that were all required to inter-operate in order to make it all happen. To

this end I have included a reasonably technical section that describes the functioning of the radio telescopes of the Deep Space Network (DSN), without which such missions would have been impossible and worthless. Who would want to spend $200m in the very early 1960's without being able to get any useful information sent back to Earth?

The Mariner programme ran concurrently with many other NASA projects including : Ranger, Surveyor, Mercury, Gemini and Apollo, this is in addition to all the evolving telecommunications satellites and all those required by the military. They were all competing for funding and resources and had to be scheduled according to favourable planetary alignments and the availability of the DSN and launch facilities. By 1971 space was getting very busy. For those who wish to know more about the resulting planetary science and to view some excellent graphics I would refer you to "SP-337 The New Mars " which can be found at website: `ntrs.nasa.gov`.

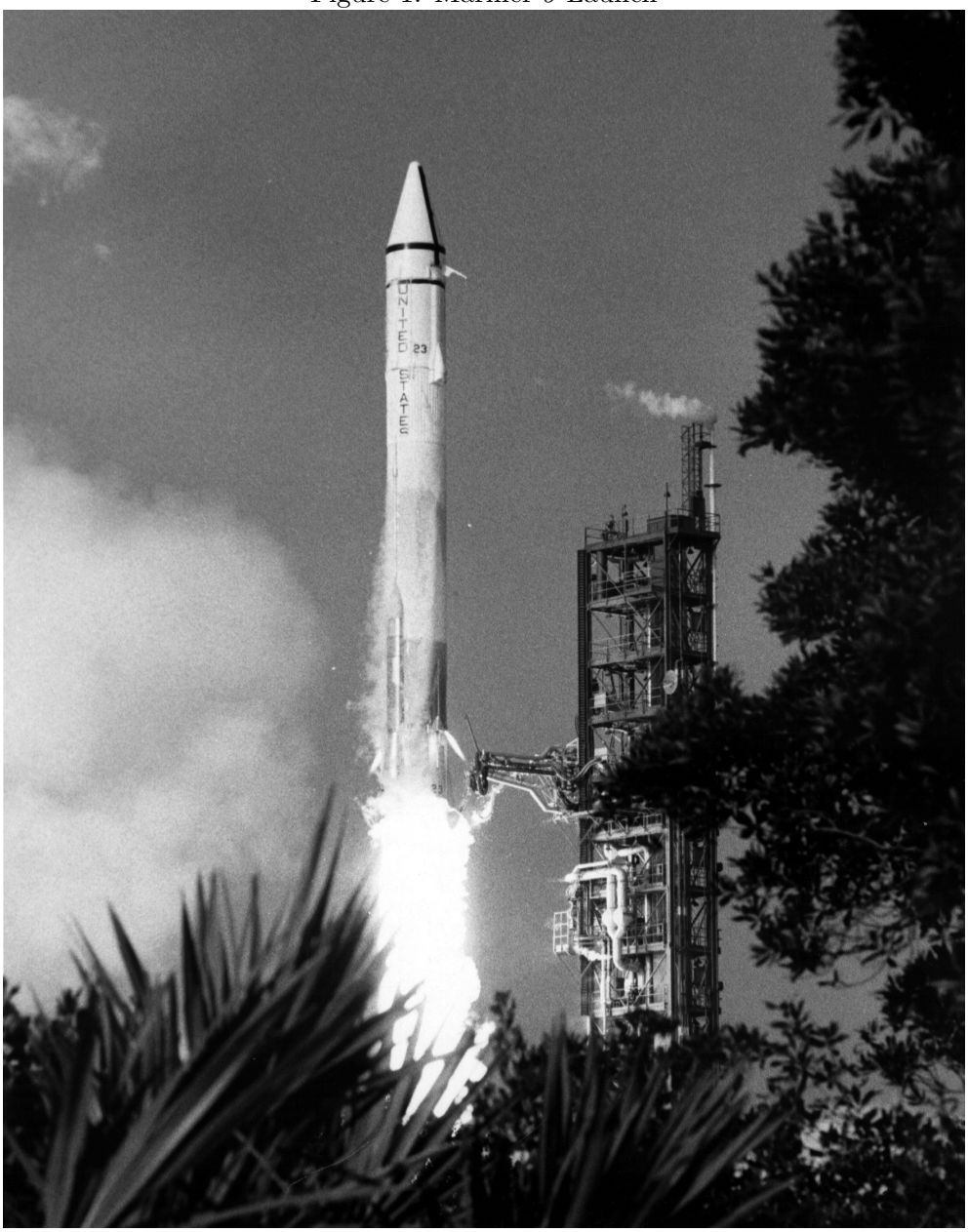
All photos and many diagrams used within, plus most of the technical detail are from NASA and JPL archives, I also used Wikipedia as a gateway point for my research. Some of the relevant text was also extracted from NASA source materials, such as mission reports, these are shown in italics. In some cases the original text has been adapted for better legibility.

Many of the specifically useful source texts from `https://ntrs.nasa.gov` which I referred to are listed in the Bibliography (Appendix 1.). We have to thank NASA and it's associates for allowing this information and images into the public domain, at the time of the missions, specifically leading up to them, some of the technical information would have been very highly classified.

This document was prepared using LaTeX. Some none NASA sourced diagrams were created with ViaCAD.

`http://rayhillwrites.com`

Figure 1: Mariner 9 Launch



An Atlas-Centaur vehicle carrying the Mariner 9 spacecraft lifted off to Mars at 6:23 p.m. EDT, May 30, 1971, from Cape Kennedy's Launch Complex 36B. The spacecraft is scheduled to orbit Mars following a six-month journey that will span 287 million miles. Mariner will map a major portion of the planet, studying selected areas to observe composition, density, pressure and temperature of the atmosphere and the temperature and composition of the surface. Two on-board television cameras will return highly detailed pictures of the Red Planet. (NASA KSC-71P-0354)

Contents

1 Overview

The Mariners were a family of spacecraft designed during 1958 and 1959 for planetary investigations within the solar system. The family was effectively a common octagonal chassis structure with some solar panels, propulsion and guidance to which could be attached a variety of sensors and communications equipment. Some of the inevitable failures must be ascribed to the fact that there was a lot of new technology continually being used and despite extensive ground testing even a fault in a tiny electronic component may have been catastrophic. The era of micro-electronics had barely come into widespread use when these spacecraft were originally designed, very basic transistor radios had only come to market in 1954. Mariner 1 was an attempt to get to Venus in 1962, however, the Atlas booster failed. Later that year Marina 2 flew by Venus at a range of about 21,000 miles. Marina 3, launched in 1964, was the first attempt to reach Mars. Although the launch was successful, the spacecraft's aerodynamic launch shroud failed to separate. Later in 1964 Mariner 4 successfully completed it's planned flypast of Mars at a closest altitude of 6,118 miles and transferred the first ever pictures (22 in total) of the surface of Mars and also delivered much useful scientific data throughout and even beyond its planned mission timescale. Subsequently Mariner 5 went to Venus in 1967, then the Mariner 6 and 7 1969 missions completed further flybys of Mars at a closest point of around 3,000 miles. Using much improved camera and telecommunications techniques were able to send back much better images and far more of them than Mariner 4. As the capabilities improved NASA began to build on successes in getting close to the planets and the Moon. The next logical step was to test getting into orbits, which would require a better propulsion system, more fuel and more automation when the spacecraft was on the "dark side", hidden from Earth communications. Therefore the Mariner 8 and 9 missions for the next Mars opposition opportunity in 1971 were re-designed to be the first spacecraft to orbit a planet (the moon was "old hat" by then) and to map the surface. Mariner 8 failed due to a problem in the gyros on the Centaur stage, so the designated 9 mission was rapidly re-profiled to accommodate key tasks from 8 and was designated for a 90 day mission once in Martian (or Aerocentric) orbit. There was also an element of competition with the Russians, who also wished to take advantaged of this specific Mars/Earth opposition window and in fact had three Mars missions that year, two of which achieved orbit and one deployed a lander capsule which failed. Over a three to four month period a total of 60 images were gathered from orbit. Although the Russians launched Mars-2 11 days before Mariner 9 it was the USA that arrived in orbit 13 days before, presumably as a result of finding a better orbital trajectory. In terms of the "space race" we should bear in mind that the USA could launch from Florida at around 20 degN whereas Baikonur (Star City) in Kazakhstan is around

45 degN, this meant that the USA were able to benefit from a better velocity vector that is inherent in the rotation of Earth at point of launch, this translates to less fuel or better payloads at launch. In many ways Marina 9 was the culmination of the first decade of local planetary exploration. Mariner 10 was launched in 1973 and carried out a flyby of Venus and then used gravity assist to reach Mercury performing a flyby, close enough to enable using the gravity assist of Mercury to then loop itself around the Sun. Mercury completed two solar orbits whilst the spacecraft completed one and was able to intercept for a second flyby of Mercury. This test of gravity assist would have been very useful for the later far more complex Voyager 1 and 2 missions to Saturn, Jupiter and beyond which were launched in 1977. The Voyagers were orginally designated Mariners 11 and 12, if you look below the 3m parabolic antenna on a photo of a Voyager you will see the octagonal frame of the Mariner family.

Any launch into space whether it is to a planet or an Earth orbit has a designated mission and requires a very high level of remote control and monitoring of systems status to ensure that the mission objectives are achieved. This document specifically deals with some key technologies that Mariner 9 utilised to become the first spacecraft to orbit another planet and to return high quality images that mapped about 80% of the surface. A key planning part of the mission was the initial orbit design and subsequent monitoring of a continuous stream of telemetry sent back from the spacecraft from which doppler information could be used to assess its position and velocity vectors. This telemetry contained a vast amount of information from all key components such as temperatures, pressures and voltages. This required a lot of highly complex telecommunitions equipment which was constantly being improved, so it is useful to understand how that worked. I have therefore included a section which covers one site in the DSN (deep space network), which is the large parabolic reflector telescope at Goldstone. After launch, smaller sites at Woomera and Johannesburgh were also used to keep contact with the craft whilst the Earth rotated with its daily spin around it's polar axis. Wherever possible engineering and science information was processed in real time on a multi-vendor computer suite at SFOF (spaceflight operations facility JPL, Pasadena). Johannesburgh, Woomera and other sites all had a local data capture and storage facility and could present a near realtime view of the health of all spacecraft systems, which was also passed by data links to SFOF, hence there were early alarms of any readings that were outside expected boundaries. All Data was also stored locally upon receipt and whenever real-time processing was not available. Whilst the flight was in progress there was another huge mission going on the background, which was the processing of all the science data on many separate computer systems, often with copies of data being being moved around on magnetic tape.

Mariner 9 was designated to orbit Mars so, unlike earlier flybys, had to perform an additional engine burn to reduce its velocity so it could "fall" into a Martian orbit. This meant using a much better engine and much more fuel, so was almost double the mass of the previous Mariners. The system stack retained the Atlas booster but the Agena booster of Mariner 4 had been long replaced by the far more powerful Centaur, which could not only boost the additional mass but also offer a far broader choice of ΔV's (velocity changes) and hence a wider choice of trajectory options in case the orbit plan

needed to be adjusted because of delays.

The key points that will be covered within this document are as follows:

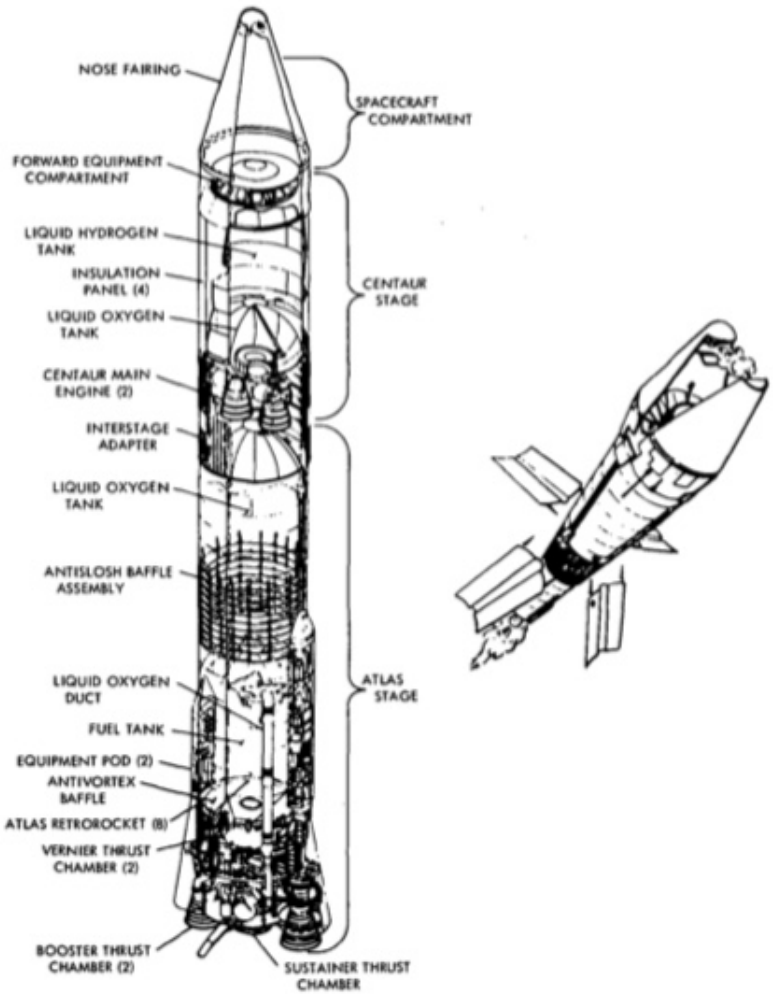
- The Launch.
- Tracking and Data Acquistion.
- Spacecraft systems.
- Navigation.
- The flight plan.
- Data Transmission.
- Camera Technology and Image Enhancement.

This document is a sample precis of well over 5,000 pages of various mission reports which are available to view at `https://ntrs.nasa.gov/`. Using a keyword search of "Mariner 1971" will yield a decent sample list of source material. Since Mariner 9 was a logical evolution of an existing design using some carried forward hardware, I also found that it was often necessary to look at the Mariner 7 and 8 specifications which will be found in a "Mariner 1969" search. It is well worthwhile spending at least some time scanning a few of these documents which offer a very clear picture of the amount of detail and complexity that is required for unmanned spaceflights. In order to to pre-empt any confusion it is worthwhile being aware that internally prior to launch the Mariners were referred to as Mariner A to L. The post launch reports referred to Mariners 1 to 10; whilst some of the early design documentation talk about Mariners K and L they were later renamed as Voyagers 1 and 2. For any budding astronomers reading, a search on NASA's ephemerides site `https://ssd.jpl.nasa.gov/?ephemerides` offers all sorts of interesting details about planetary motions, distances and orbital velocities. The more adventurous might even use it to to plan their own theoretical space missions. One example of my own use may be found here `https://www.amazon.com/More-Spaceflight-Theories-frontier-Humans-ebook/dp/B06VW1WQ37/ref=sr_1_2?ie=UTF8&qid=1511863991&sr=8-2&keywords=raymond+hill+mars`.

2 The Launch

The following diagram shows the complete system at launch, the image to the right shows the Centaur/Mariner after separation from the Atlas. This also illustrates the point at which the Centaur and spacecraft fairings are being ejected, just before the Centaur begins its burn.

Figure 2.1: Mariner 9 Launch Stack



The following image (from JPL Technical Memorandum 33-523, Vol. I), shows the launch sequence summary after countdown at the zero point of release. Some points here are worthy of explanation: BECO means booster engine cutoff, which are the two motors either side of the bottom of the stack, these are ejected to lose weight. SECO is the sustainer engine cutoff, the Atlas has a single sustainer engine between the two outboard booster motors which is also ignited at lift-off. The two outer booster engines were gimballed; which allowed for yaw, pitch, and roll control. After BECO and ejection the only engine running was the single sustainer, this engine was also gimballed, however with a single engine, roll control is not possible so two smaller vernier engines were designed in to provide roll control (and some thrust) after the booster engines had been jettisoned, so if you see a film of an Atlas launching with exhaust plumes out of either side that is the verniers. VECO/SECO would occur just before Atlas separation at which point the Atlas fired retro rockets to slow it down for a fall into the Atlantic, thus avoiding it to coast and cause debris to fall over inhabited areas.

At the time of SECO the remaining system has effectively attained orbit, there was a 12 second coast period before Centaur main engine start. You might note (below) that the interval between Centaur engine start and MECO (main engine cutoff for the Centaur) stage after which the required orbital velocity and altitude has been attained is 452 seconds. This is a critical pre-planned burn time in which the spacecraft is accelerated beyond escape velocity toward the chosen planetary orbit, which to clarify is often an elliptical solar (heliocentric) orbit which take it to a date with Mars at a planned time and place.

The documentation refers to tracking by AFETR, which is air force eastern test range and is in practice a series of tracking stations for launches out of the Kennedy Space Centre in Florida. Some of these may be land, mobile or even on ships at sea.

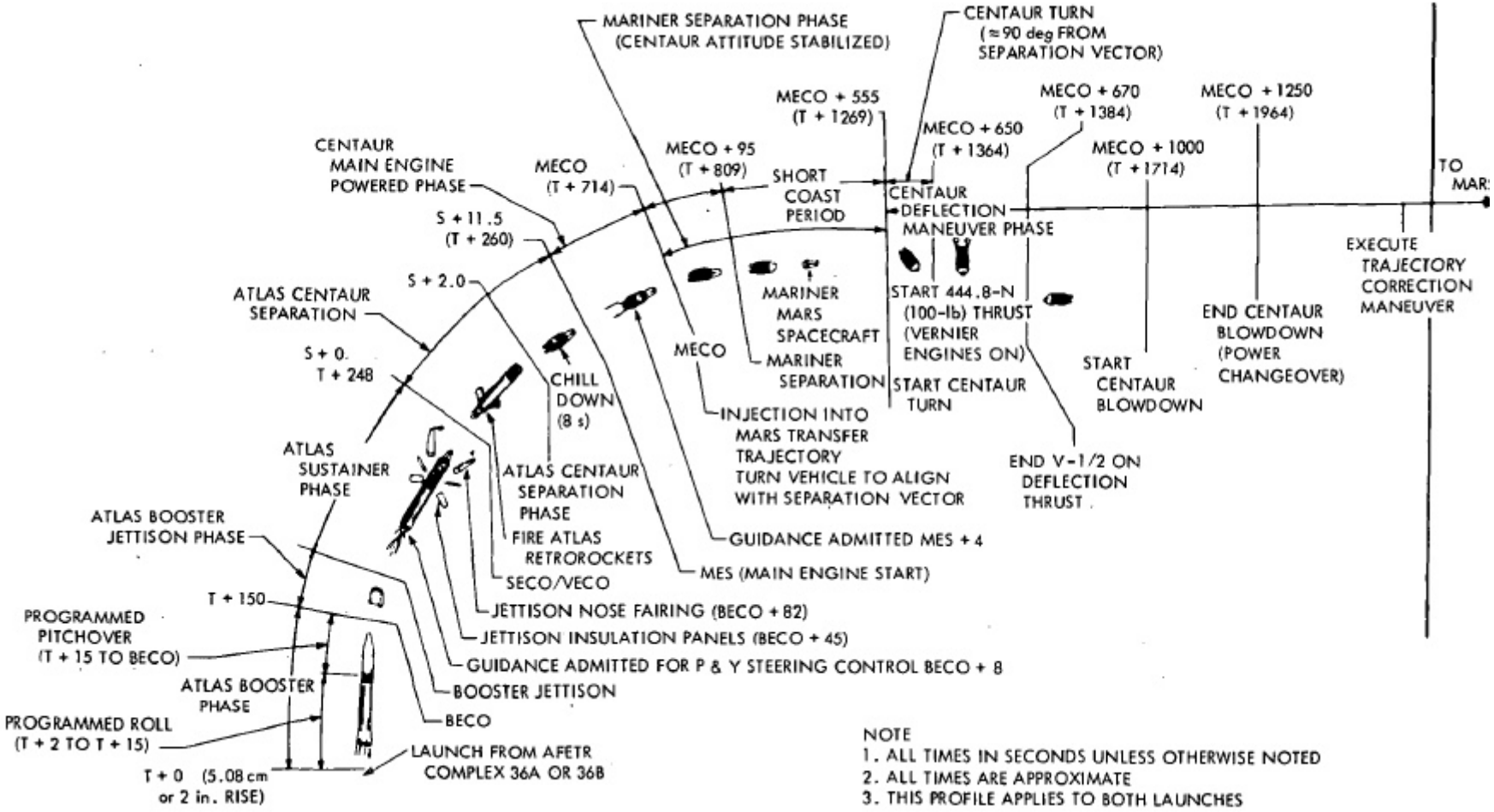
Figure 2.2: Mariner 9 Launch Data

Table 56. Mariner 9 summary of observed mark events

Mark Event No.	Mark Event	Nominal Time L+sec	MIL USB Observed GMT (L+sec)	Bermuda Observed GMT (L+sec)	AFETR Observed GMT (L+sec)
-	Liftoff 5.08-cm (2-in.) motion				2223.04.463
1	Atlas BECO	147.86	22:25:35.5 (151.04)		2225:35.55 (151.09)
2	Atlas Booster Engine Jettison	150.97	22:25:38.5 (154.04)		2225:38.5 (154.04)
3	Centaur Insulation Panel Jettison	192.87	22:26:20.5 (196.04)		
4	Nose Fairing Jettison	234.87	22:27:02.6 (238.14)		
5	SECO	250.89	22:27:07.5 (243.04)		2227:08.4 (243.94)
6	Atlas/Centaur Separation	252.89	22:27:10.5 (246.04)		2227:11.6 (247.14)
7	Centaur Main Engine Start	262.39	22:27:20.9 (256.44)	22:27:21.2 (256.74)	
8	Centaur MECO	714.68		2234:46.9 (702.44)	2234:47.1 (702.64)
9	Separate S/C	809.68		2236.22.6 (798.14)	
10	Reorient Centaur to Deflection Vector	1269.68		22:44:03.6 (1259.14)	2244:02.75 (1258.29)
11	Start Blowdown	1714.68		22:51:27.1 (1702.63)	
12	End Blowdown	1964.68		22:55:37.5 (1953.04)	
13	Power Changeover	1964.68		22:55:38.1 (1953.64)	

The following typical launch profile image would have applied to both Mariners 8 and 9, it is also representative of a fairly standard procedure for any launch into orbit from a variety of booster systems.

Figure 2.3: Mariner 9 Launch Profile



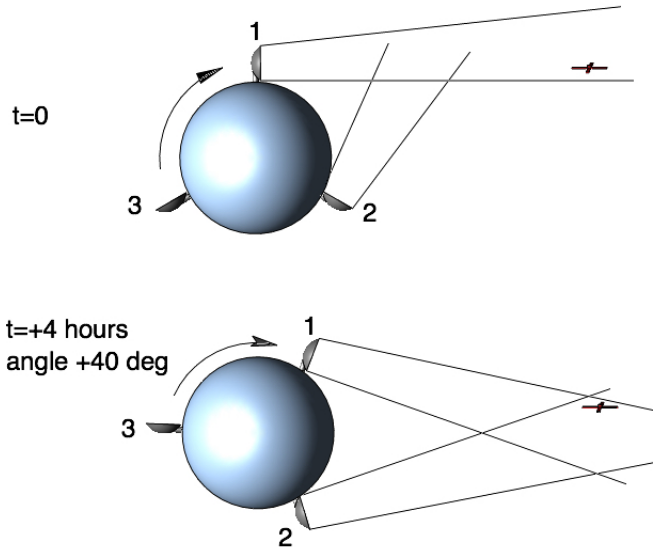
3 Tracking and Data Acquisition

One of the often unseen but most mission critical background processes of spaceflight is that of TDA (Tracking and Data Acquisition), without availability of effective capability the whole mission might be pointless.

To control and collect data from a spacecraft during a mission lasting often over a year or much longer, or as in the case of the Voyager spacecraft maybe even four decades, requires a tracking system that can be in direct contact for as much of the time as possible. Since the Earth rotates, antennas will often be on the dark side so a network of antennas around the Earth is a necessity, as is shown in the diagram below, each of these may be horizontally rotated and elevated vertically in step with the continuous change in spacecraft and Earth's relative positions.

The figure below shows three Earth base stations separated by 120°. As the Earth rotates (clockwise) different stations in turn each have vision of the spacecraft. The situation below shows that it is possible for two stations to have overlap, hence duplicate data may simultaneously be received by two stations on Earth, this data will later need to be de-duplicated. All data is collected on the computer system at each site and is processed at JPL's Space Flight Operations Facility (SFOF) in Pasadena. By the time of arrival in 1972 higher data rates at the 64metre Goldstone facility made it possible to collect and display images in near realtime.

Figure 3.1: Base station separation

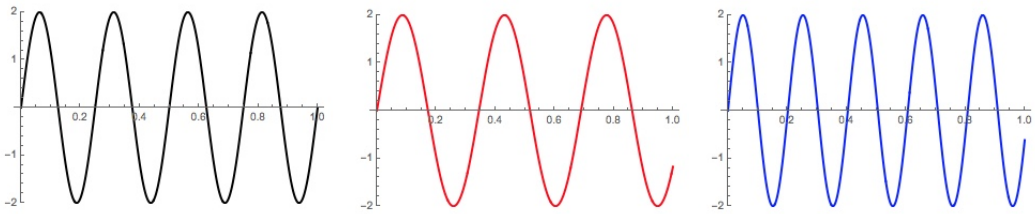


On the Earth we may have a pair of parabolic reflector antennas directly pointing at each other over a fixed distance so the phases between wave cycles once tuned are theoretically always in step. In space travel these reflectors are constantly in relative motion. Firstly a reflector fixed to the Earth is rotating with the Earth on a daily basis and secondly the Earth is also rotating about the Sun in it's annual orbit. Even if we had the impossible situation of a spaceship in a fixed position in space relative to the Sun, for missions of nearly a year there would always be periods when the Earth was behind the Sun so we would lose sight of the spacecraft. There is a further complication of a spacecraft travelling either away from or towards the Earth, some of that relative velocity may well be caused by being in orbit around another body, e.g. Mars or the Moon. These factors give rise to the doppler effect where the wavelength is either apparantly stretched or contracted depending on whether the relative bodies are approaching or receding. On Earth we often hear this (Doppler effect) as the change in pitch of police sirens getting closer or further away. In astronomy it is known as red or blue shift where the wavelength of light from Stars or Galaxies appears to be altered, hence the colour appears to change with relative motion.

The three diagrams below illustrate the change in wavelength for transmitters and receivers in motion. On the left the black graph represents transmission between fixed points, in this graph four cycles are shown, the red graph shows the effect of a velocity change that increases distances, the time base is increasing hence fewer cycles , 3 in this case so the signal is red-shifted (red has wavelength of around 700nm). In the final case of decreasing distance the blue graph shows five cycles so is blue shifted (blue has a wavelength of 400nm). Since transmitters and recievers are tuned to frequency, a change in relative velocity will alter the frequency therefore the system will go out of tune. It is possible to build in an auto re-tune facility, which gets used extensively throughout such a mission. Signals travel through space at the speed of light ($2.998 \times 10^8 ms^{-1}$) wherease

the relative velocity of a spacecraft is likely to be not much more than $10^5 ms^{-1}$. The frequency shift is therefore likely to be in the order of plus or minus $n \times 10^3$. Once in orbit depending on the position, the spacecraft may, from the relative perspecive of Earth, appear to be either approaching or receding. Since Mariner 9 orbited Mars every twelve hours there would be perceived frequency changes at approximaly 6 hourly intervals to which would also need to be added the effects of the continual differing velocities and distance between the orbits of Earth and Mars.

Figure 3.2: Wavelengths at different velocities



The following example calculations illustrate the kind of results which we might suppose to see.

In free space an electro-magnetic wave of frequency 2.3Ghz travels at the speed of light $c = 2.998 \times 10^8 ms^{-1}$ so we might choose to calculate the wavelength as follows:

$$\lambda = \frac{c}{f}$$

$$\lambda = \frac{2.998 \times 10^8}{2.3 \times 10^9} = 0.130348m$$

The value above, 13cm, falls into the centimetric range of wavelengths, as commonly used for earlier (second generation) military radars.

A fixed antenna on Earth has an equatorial velocity of $v_{eq} = 456ms^{-1}$, meanwhile a spaceship in Martian orbit may have an average velocity of about $3,500ms^{-1}$. If both are going away from each other in their respective orbits then the combined velocity is $v = 3,956ms^{-1}$. If the (static) wavelength we are using to communicate is 2.3Ghz , then we can calculate the frequency change due to respective motion at this given point in time as :

$$\lambda = \frac{c - v}{f}$$

$$\lambda = \frac{2.998 \times 10^8 - 3,956}{2.3 \times 10^9} = 0.130346m$$

So the actual frequency drop is very small, however that over simplified example excluded the magnitude of the varying velocity of a vector of the differences between Earth and

Mars orbit's, which depending on relative positions could be as much as $18,800ms^{-1}$, so in an extreme case we might be looking at something like :

$$\lambda = \frac{2.998 \times 10^8 - 3,956 - 18,800}{2.3 \times 10^9} = 0.130337m$$

The percentage differences above are actually quite small but nonetheless need accounting for. You saw that it was chosen to use a frequency of 2.3Ghz, why? The immediate reason is that data is encoded within wave cycles so the more frequent the wave cycles, the more data can be sent in any given time interval. The other reason is that we need to choose frequencies that are immune from known noise and losses due to absorption, for example if the signal goes through oxygen or water molecules in the Earth's atmosphere it may be attenuated by them at certain frequencies. This chosen gigahertz frequency is referred to as being within the S band, later missions can also work at X band which is much faster, however at the time of Mariner 9 design the signal processing technology was still under development (and was tested on Mariner 10 in 1973) to operate reliably at that speed. The other prime reason for that choice is that it needs to be compatible with technology at both ends. Whilst it may be relatively easy to physically change to a faster chip on a spacecraft under construction, the workload on updating all the Earth stations may be considerably larger, particularly with any operational restrictions due to over lapping multiple project timescales (e.g. Apollo missions) and also restrictions on other planned mission windows due to planetary alignment opportunities. A further factor is that of change control, every time something is changed some risk of failure is inevitably introduced. Although I was not there, it is safe to suppose that any system changes other than those essential for safety or continuance of operation (repairs) would have been highly frowned on during the over-lapping timescale of the ongoing manned Moon landings.

Mars has an average orbital radios of 1.524AU (Astronomical Unit $1.496 \times 10^{11}m$) and Earth that of 1AU. An AU is simply the average distance frome the centre of the Earth to the centre of the Sun. Therefore the closest they ever come together is about 0.524AU and the furthest 2.524AU, apart from this explaining why windows of opportunity for flights are limited it also shows the extremes of distance at which communications must be maintained with the following one way signal propagation delays.

The extremes of signal propagation delay are:

$$t_{nearest} = \frac{0.524 \times 1.496 \times 10^{11}}{2.998 \times 10^8} = 261.5seconds(4.3mins)$$

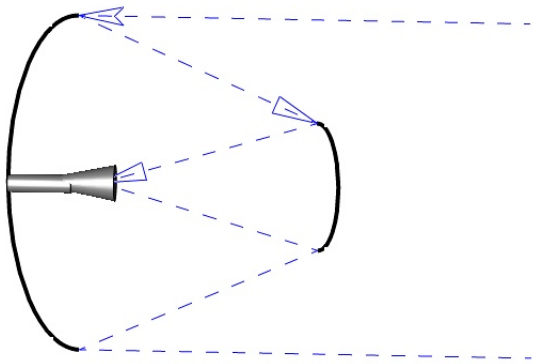
$$t_{furthest} = \frac{2.524 \times 1.496 \times 10^{11}}{2.998 \times 10^8} = 1,2259seconds(21mins)$$

Now seems to be an appropriate time to understand a bit more detail about those mission critical ground stations on Earth.

As mentioned earlier the antennas are actually parabolic reflectors of the Cassegrain (dual mirror) design. At the time of the mission the DSN had a 64 metre one in Goldstone

(USA) plus two 26metre ones in Jonannesburgh (South Africa) and Woomera(Australia), these three sites were able to provide full 360^0 cover. Sites that were out of vision could thus be re-assigned for up to 8 hours to other missions. The parabolic reflector design creates a signal path with a relatively narrow beam, but with very high gain. The narrowness of the beam means that target acquistion needs to be very precise and they are built on very sturdy structures. The narrowness of the beam also minimises signal attenuation and can also be pointed so as to avoid external noise which may corrupt or weaken the signal. The principles discussed below also apply in a very similar way to your satellite TV reception and for the very few that have a radiotelescope, to astronomy.

Figure 3.3: Optical trace of a Cassegrain parabolic antenna



In the above diagram 2.3GHz radio signals, coming in from the spaceship, are collected by the large dish and reflected to the smaller dish at the centre of the antenna, from this the more focussed signal is again reflected toward the collector. Transmission works in reverse but at a different frequency of 2.1GHz. This has a direct similarity to an optical Cassesgrain telescope which is limited to collecting light signals across the visible range of the electro-magnetic spectrum.

The area of an antenna (Goldstone 64m diameter) is calculated as follows:

$$A = \frac{\pi \times D^2}{4}$$

$$= \frac{\pi \times 64^2}{4} = 3,217m^2$$

The gain of an antenna (Goldstone, operating at $\lambda = 0.130348m$) is calculated as follows:

$$G = \frac{4 \times \pi}{\lambda^2} \times \epsilon \times A$$

Substituting for A from the first equation reduces the above to:

$$G = (\frac{\pi \times D}{\lambda})^2 \times \epsilon$$

In the above ϵ represents the antenna's efficiency which for this type is around 58%.

$$G = (\frac{\pi \times 64}{(0.130348})^2 \times 0.58 = 1,655,517$$

$$log_{10}1,655,517 = 61.4dB$$

I will admit, for simplification and inability to find a precise value, to prior adjustment of the ϵ value so that it matches Goldstone's published gain performance at that time.

The beamwidth of a parabolic antenna:

$$\psi = \frac{70 \times \lambda}{D}$$

$$\psi = \frac{70 \times 0.130348}{64} = 0.142^0$$

This represents an angle of the transmit cone (or boresight) of around 10 minutes, something like trying to focus a rifle sight onto a tennis ball 10 miles or so away. However the signal sent from the spacecraft had a much broader boresight ranging from about 130 Low Gain Antenna (LGA) to 9 High Gain Antenna (HGA) degrees, depending which antenna it was using, the wider one from the low gain antenna had much lower data rates which was good enough for lower volume engineering and doppler telemetry data but the narrower beam high gain antenna was used for higher volume image data transmission.

If we assume Goldstone is fitted with a transmitter of 100kWatt power, then the radiosity or intensity at the point the signal leaves the parabola is simply power divided by the previously calculated antenna area :

$$J_e = \frac{100,000}{3,217} = 31.085Watts/m^{-2}$$

The beamwidth of this antenna was calculated above as 0.142^0, so if the signals travel to Mars at it's closest point to Earth (0.524AU) then the signal area at point of reception will cover an area of :

$$Signal_{beamradiusatMars} = (0.524AU \times 1.496 \times 10^{11})tan(\frac{0.142}{2}) = 5.57 \times 10^9 m$$

$$Signal_{beamareaatMars} = \pi(5.57 \times 10^9)^2 = 9.76 \times 10^{19} m^2$$

From above a little extra calculation would show that, in order to acquire the expected signal frequency, the antenna at Earth needs to scan an area that has a diameter of about one tenth the distance from Earth to the Sun. When it detects the frequency that the spacecraft is transmitting from it can then adjust its alignment until the signal strenth is strongest, so is centred, or locked onto the spacecraft. Since both transmitter and receiver are in constant relative motion, the ground station is continually making small adjustments, i.e. tracking the spacecraft.At this point each can phase lock with the others signal and create open transmission channels.

The radiosity/intensity at the spaceship is therefore:

$$J_e = \frac{100,000}{9.76 \times 10^{19}} = 1.02 \times 10^{-15} Wm^{-2}$$

This is collected by the spaceship's antenna (It has three) but near Mars, for image download, it will use the parabolic high gain antenna (HGA) with a diameter of 1.02m, this antenna will therefore collect radiosity across its area as follows:

$$A = \frac{\pi \times 1.02^2}{4} = 0.817m^2$$

Above assumes the transmitter and receiver are precisely aligned in which case the received radiosity is a mere:

$$Power = 0.817 \times 1.02 \times 10^{-15} = 8.33 \times 10^{-16} Wm^{-2}$$

That is a huge loss, in magnitude of of 83.3 quadrillionths (10^{-15}), so the challenge now is to see how or even if that can be used sensibly and then to turn the problem around to what Earth is receiving from the spacecraft. The calculations that follow are when Earth and Mars are at 0.524AU apart which is d=78.3904 billion km, frequency used is $\lambda = 0.130348m$. We can now thow away all of the previous math, which I put in to show the underlying physics. It is now more convenient to work in decibels where power is described as a loss as it gets weaker or a gain as it gets stronger.

We start with the concept of free space loss, which is how much a signal weakens without interference from things in the way, such as an atmosphere, or affected by other electromagnetic radiation. Free space path loss:

$$FSPL = (\frac{4\pi d}{\lambda})^2$$

Since $\lambda = \frac{f}{c}$:

$$FSPL = (\frac{4\pi dc}{f})^2$$

Free space path loss in decibels:

$$FSPL = 20log_{10}(d) + 20log_{10}(f) + 20log_{10}(\frac{4\pi d}{c})dB$$

$$FSPL = 20log_{10}(78,390,400,000) + 20log_{10}(2.3) + 32.44 = -257.56dB$$

minus because it is a loss The constant value of 32.44 that was introduced is dependant on units used. Here we were using Gigahertz as the dimension of frequency and metres for distance , the dimension of the speed of light is 2.998×10^8, in metres. So $20log_{10}\left(\frac{4\pi \times d}{c}\right)$ required the inclusion of 10^9 to account for the Gigahertz, so the full expression is evaluated as $20log_{10}\left(\frac{4 \times \pi \times d \times 10^9}{2.998 \times 10^8}\right) = 32.44$. The constant is different for each combintion of units used, e.g Khz and metres uses a constant of -147.55 and GHz and kM requires the use of 92.45.

Path loss accounting for antenna gain at each end:

$$PL = (20log_{10}(d) + 20log_{10}(f) + 32.44) + G_{tx} + G_{rx}dB$$

$$PL = -257.56 + 61 + 42.6 = -153.96dB$$

This calculated path loss meets the documented threshold value for the spacecraft's receiver equipment, note that the gain on the receiver is the sum of the sending antenna, modulation and transmitter equipment as specified within Mariner 9 documentation.

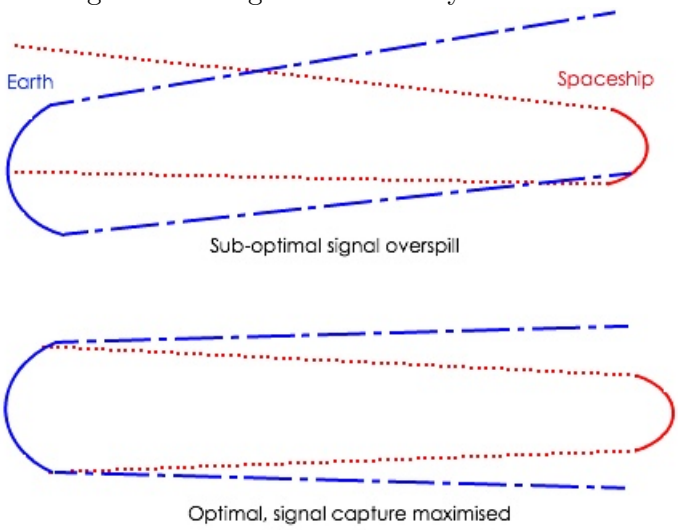
The above calculations only required the knowledge of distance, frequency, the speed of light and 4π. Other than using a published antenna gain values it said nothing about the capability of systems at either end so can be said to be valid for either way of transmission, for this particular antenna gain pairing. The calculated values of -257.56 and -153.96 are in similar magnitudes to table 2 page 238 of the Mars 1969 (Mariner 7) write up in JPL technical report 32-1460 vol 1, which uses a larger distance of 97billion Km. Although the figures here are probably not particularly accurate this serves as a reasonable point of reference for a worked example, specifically since wherever possible technology and parts from the 1969 mission were carried forward and sometimes upgraded for the 1971 missions.

Table 3.1: Comparisom of Goldstone and Spacecraft high gain antenna characteristics

Parameter	Goldstone	Spacecraft HGA
Diameter	64m	1.02m
Beamwidth	0.142^0	8.945^0
Transmitter Power	100kW	18.2W
Transmitter Gain	61 dB	42.6 dB
Free Space Loss	-257.56 dB	-257.56 dB
Signal Energy	-153.96 dB	-153.96 dB

This is a typical result that may be obtained at a given distance, in reality the HGA was used near Mars and whilst in orbit. Depending on noise factors this enabled far higher data rates (16.2kbps) for image transmission. When the spacecraft was aligned with Woomera or Johannesburgh, with smaller antennas, the data rate and characteristics would be far inferior(8kbps or less). To maximise the signal quality precise alignment would be essential.

Figure 3.4: Alignment accuracy is critical.



The top picture shows that the spaceship will get a fair signal from Earth , however the signal that Earth receives from the spaceship is considerably dimished, the lower picture unsurpisingly is the ideal.

The Goldstone station weighs over 8,000 tons with a lot of that weight being the sturdy base and machinery that enables precise alignment changes to be made, in a predictable target following fashion. Your TV satellite at home is fixed to point to a satellie in a high geo-stationary orbit (always fixed relative to it's given position above Earth).

Figure 3.5: Goldstone radio telescope

The spacecraft used TWT (travelling wave tube) technology as an amplifier for transmission. The 64m telescope at Goldstone was equipped with a ruby TWM (travelling wave maser) which is super-cooled to 5^0K by liquid Helium. The super cooling was essential to reduce noise within the system to a point where it would not corrupt the extremely weak signal received from the spacecraft.

Much of the technology used was carried over from the 1969 Mariner 9 mission, from which I have taken a lot of the telecommunications information. This approach would have alleviated any design change development and also saved money. Technical Memorandum 33-535 states the following changes.

A. Phase-Lock-Loop Receiver The loop gain was increased by a factor of 10 to reduce loop phase error with frequency offsets. This eliminates requirements for retuning the uplink during the orbital phase, where doppler shifts on the uplink approach 40 kHz over one station pass.

B. S-Band Antenna Coupler and Medium-Gain Antenna The medium-gain antenna (MGA) was added to the system using a passive 6-dB directional coupler in the low-gain antenna (LGA) circuit. The MGA was mounted to provide telemetry coverage and two-way doppler during the orbit insertion and orbit trim phases of the mission. Although the MGA has a bore sight gain of approximately 14 dBi, when installed with the 6-dB coupler the effective peak gain was approximately the same as the LGA.

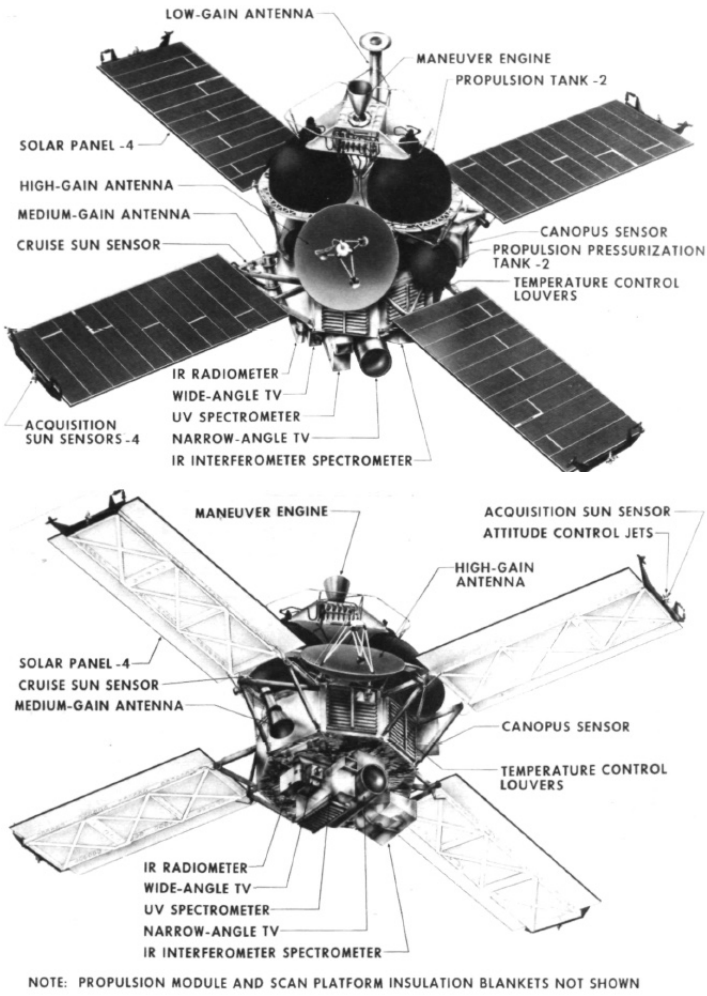
C. High-Gain Antenna Mariner 1971 uses the same high-gain antenna (HGA) used on Mariner 1969, a 1.02-meter circular parabola. However, rather than being fixed in position as on Mariner 1969, the antenna was used in two positions. The first position was used preinsertion and for the first 60 days of orbit operations (assuming a November 14, 1971, orbit insertion). The second position was used for the remainder of orbit operations and provides an effective gain greater than the LGA until approximately 140 days in orbit (April 1, 1972).

The low-gain antenna, which is a shortened version of the Mariner 1969 LGA is always used for receiving the uplink signal and for transmitting (downlink) during the early cruise phase and for maneuvers. The HGA is used exclusively for transmitting during the late cruise and orbital phases.

Finally for this section, I would draw your attention to the fact that the prior planning for this mission went into the level of detail so they could predict telecommunications performance after 140 days in Martian orbit, meaing around 320 days since launch. They used a program called TPAP (Telecommunications Prediction and Analysis Program) which enabled them to monitor actual performance against expected or even required performance through the whole mission. This program was a major revision and a combination of CP2M and CMPM that were used for the 1969 Mariners.

4 Spacecraft systems

Figure 4.1: Spacecraft layout



Whilst some of the below was covered in the previous section, this extract does gives some further insight into the detail of the antenna configuration.

The radio frequency subsystem uses three S-band antennas. The two-position high-gain

antenna, a 101.6cm parabolic reflector with a right-hand circularly polarized feed, is used for transmitting before and during orbit. The antenna operates at the frequency of 2, 295 5 MHz. The antenna can be oriented to a second position during the orbital period to maximize communication time in orbit. The fixed low-gain antenna, also circularly polarized, is mounted on the sunward side of the spacecraft and has a hemispherical pattern approximately symmetrical about the roll axis. It is used to receive and to transmit when the high-gain antenna cannot be used and provides forward hemispheric coverage. It operates in the frequencies of $2,115 \pm 5MHz$ *and* $2,295 \pm 5MHz$*. The medium-gain antenna is a right-hand circularly polarized radiator and provides coverage during the orbit insertion maneuver. The antenna is coupled to the low-gain antenna and operates in the same frequency range.*

The high-gain antenna structure consists of a reflector and a feed support truss. The reflector is an aluminum honeycomb parabola with a circular perimeter 101.6cm in diameter. The feed is supported at the focus of the parabola by a fiberglass truss. The antenna is a two-position, pyrotechnically activated device that allows optimum pointing of the antenna toward the Earth during the pre-orbit and orbital periods. The medium-gain antenna structure consists of a 10.16cm diameter circular wave guide approximately 30.48cm long with a frustrum-shaped reflector approximately 24.13cm in diameter, mounted at its extremity with the flared end unsupported and oriented toward Earth during spacecraft orbit insertion. The low-gain antenna structure is composed of a circular wave guide approximately 10.16 cm diameter and 144.78cm long with a frustrum-shaped reflector mounted at the extremity. It is supported vertically at the base by a bracket above Bay VI on the upper octagonal ring and is supported laterally by two truss members running between the low-gain antenna and the engine thrust structure

From the same document is a description of the solar cell assembly.

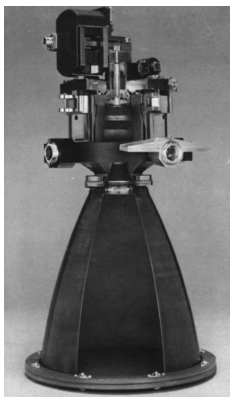
Four solar panels for mounting of solar cells provide a total area of approximately $7.71m^2$*. Each panel is 214cm long by 90.17cm wide. The cell surface substrate'is a single skin on transverse corrugations supported by two cross-braced longitudinal spars. The panels are supported during launch in a 15 deg-from-vertical position. Each panel is attached at the hinge points to panel support outriggers and is supported laterally at the tips by a pair of boost dampers running between adjacent panels. The panels are opened after spacecraft separation by pin-pullers at one end of each boost damper pair and are deployed approximately 105 deg by a deployment mechanism. After deployment, the panels are latched in a plane normal to the spacecraft roll axis by engaging the attached damper mechanism. Primary spacecraft power is provided by the four photovoltaic solar panels. The panels convert solar energy to electrical energy when the sensitive surfaces are facing the Sun. Each panel is divided into six separate sections, each wired to deliver the rated solar panel voltage. The total panel area is* $7.71m^2$ *and was able to collect 800W of energy at Earth and around 450-500W at Mars.* These of course also kept the batteries in charge for maintaining system operations when the spacecraft was on the dark side of the planet.

Reading the source documents and text extracts can become quite repetitive, particularly

as sections are frequently copied across many documents. The key common point about the spacecraft is the octagonal chassis of the whole family, I see that as really representing the baseline for it's continued development. In fact the chassis design set the boundary size conditions for all other equipment that the contractors had to install. The process of continual adaptation and upgrade of many components throughout the series massively improved the system capabilities from 1962 to 1977. The diagrams at the beginning of this section offer a good visualisation, however the following high level overview of major components may also help. To inside each side panel was fitted the key electronic subsystems and externally the thermal control management panels plus of course the supporting structures for the solar panels and antennas. To the top of the chassis (that points toward the Earth) was fitted an additional frame containing the fuel tanks and the propulsion unit. The top is mostly covered in a micro-meteorite shield and heat reflective blanket, as per the cover picture. The bottom of the spacecraft is largely fitted with most of the scientific instruments and the scan platform which adjusts their orientation plus some additional blanketing.

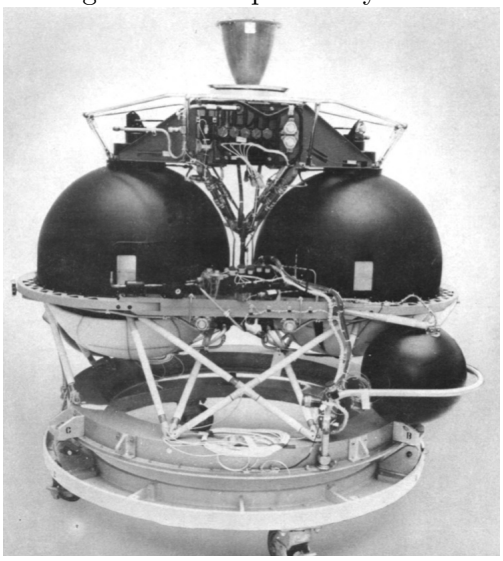
The final engine used was a modification of a design used in the Minuteman (ICBM) programme. This engine provided 1,334 Newtons of thrust and was restartable. It was planned to need to perform one burn to correct the trajectory after leaving Earth, one burn to adjust the trajectory for insertion into Martian orbit and then at least two orbit trim burns. The fuel mix used was Nitrogen tetroxide and monomethyl hydrazine (MMH) at a ratio of 1.57:1 , this mix is hypergolic which means it ignites on mixing and does not require the additional complication of an igniter system. The fuel mass carried would allow for a total of $1,650ms^{-1}$ velocity changes over the whole mission. The fuel tanks were adapted from the manned Gemini system.

Figure 4.2: Motor assembly



The above is an illustration of the combustion chamber and nozzle.

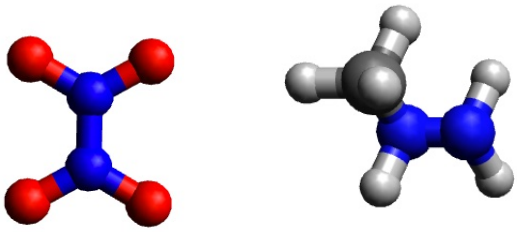
Figure 4.3: Propulsion system



The previous image is the complete propulsion system assembly with the engine and fuel tanks installed within their frame. Note that the third, smaller, tank is for the Nitrogen that is used to pressurize the fuel tanks.

The fuel used is nitrogen tetroxide and monomethylhydrazine.

Figure 4.4: Fuel: Left: Nitrogen tetroxide N_2O_4 Right: MMH $CH_3(NH)NH_2$



Apart from the propulsion system tanks the spacecraft also had smaller tankage within the octagon which was storage for the cold gas attitude adjusters that were fitted to the tips of the solar arrays.

*The data storage subsystem (DSS) was a completely new design (all digital, reel to reel) derived from earlier laboratory development efforts. This design incorporated selectable playback speeds of 16, 8,4,2, and 1 kilobits per second (kbps) with an 8-track capability using 2 tracks at a time. High packing density provided a total storage capability of 180 Mbits on the 168-m (550-ft) tape. Data was recorded at 132 kbps. Each playback rate was controlled to a pre-recorded speed (frequency). In this case, little or no design or hardware inheritance was realized from previous flight programs.*This system could store 32 pictures which were sent immediately transmission was available.

The central computer and sequencer (CC&S) design was changed primarily by the increase in memory to 512 words over the 128 words used previously. This provided the flexibility required for orbital operations to set up automatic sequences for repetitive orbital work. Lesser changes were incorporated to provide improved operations between computer and sequencer, better checking of stored information, additional systems requirements of accelerometer control and autopilot conditioning, etc.

The attitude control (A/C) subsystem underwent major changes to adapt to the orbital requirements. The attitude control electronics (ACE) were new to accommodate the logic changes and the new autopilot. The inertial reference unit (IRU) was redesigned to include an accelerometer to control the firing duration of the propulsion subsystem rocket engine and electronic integrators to provide both position and rate information separately from the gyros. The gyros were of a modified MM'69 design. The rocket engine autopilot gimbal actuators were new. There were considerable changes in the Canopus tracker (C/T) electronics. The Sun sensors were re-packaged to accommodate the configuration changes. The gas system was similar, with only minor modifications, to that of the MM'69 spacecraft.

The data automation subsystem (DAS) was a completely new logic design to accommodate the new instrument payload and mission requirements of MM'71. The integrated circuit logic family and the packaging techniques used were inherited from the Mariner Venus 67 and Mariner Mars 1969 DAS.

The radio subsystem carried over from MM'69 had a troubled operational history. Several key problems required correction and many lesser problems existed. A great deal of emphasis was placed on establishing a clear understanding of the problems and then deciding which ones required correction and how. A major change was made in the exciter, where a design used in Apollo was incorporated. Another change incorporated a new traveling-wave tube (TWT) in the power amplifier. Many other changes minor in nature but providing significant improvements in performance were carried out. The inheritance factor remained high, however, because a great deal of the complexity and RF idiosyncrasies were well understood or problem characteristics were reasonably established, permitting a rigorous analysis and test program to be established.

The purpose of the mission was, of course, to carry out some science and within the constraints of overall spacecraft mass the following equipment was installed.

- a) Two television (TV) cameras.
- b) An infrared radiometer (IRR).
- c) An infrared interferometer spectrometer (IRIS).
- d) An ultraviolet spectrometer (UVS).

In addition, using the systems already described, the following scientific activities could also be carried out.

- a) Celestial mechanics.
- b) S-band occultation.

The camera systems are discussed in a later section, however the following background information is useful to include at this point..

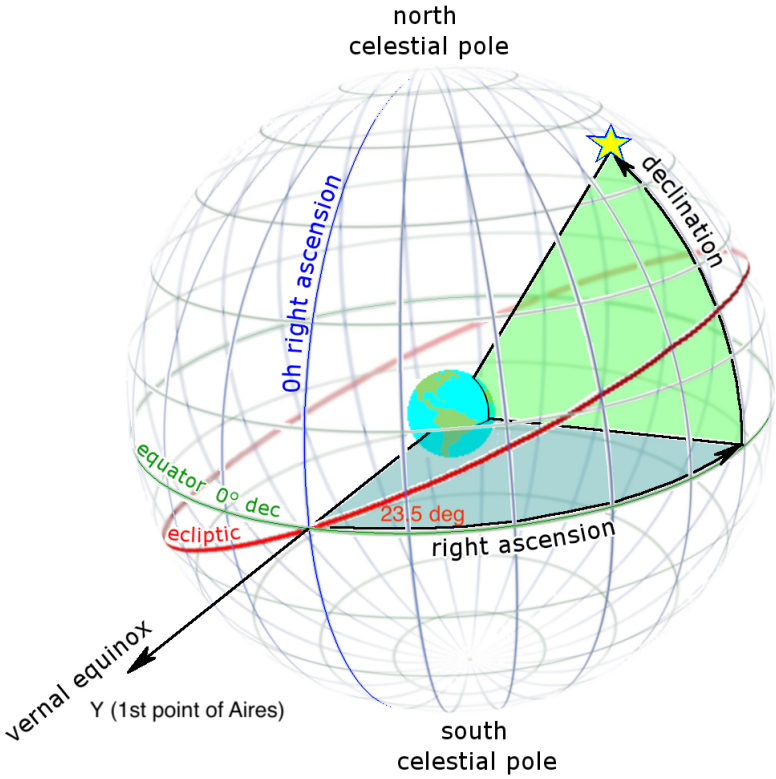
Another subsystem which underwent extensive change was the television (TV) subsystem. This subsystem employed two cameras, and much of the circuitry, optics, vidicons, etc., could be carried forward. However, the Mariner Mars 1969 subsystem had noise problems, required a great deal of processing of both analog and digital signals into usable video, had less dynamic range, and was not as adaptable as considered necessary to cope with possible variations of planet surface conditions for the orbiter. Therefore, an all digital system was developed with eight selectable filters in the wide-angle camera, automatic and commandable shutter speeds and picture sequencing, and reduced effects from aging and temperature variations. The functions of centralized timing and control were removed from the TV subsystem and transferred to the data automation subsystem. Optics were retained. Again, the experience factor with components, circuits, and functions provided a significant inheritance factor, minimized developmental costs and risk, and provided a high-performance TV subsystem.

5 Navigation

The first 1964 mission to Mars required use of stellar navigation. The basic technique was to launch the spacecraft into orbit and then align it facing the Sun, apart from the immediate need to set one navigation point this was also necessary to maximise the Sun's energy onto the solar panel arrays. A second fix was then required and the second brightest star, Canopus (313 light years away) was chosen as a reference point because, being in the Southern ecliptic hemisphere it would never be occluded by a planet in the ecliptic plane. A star tracker was mounted onto the spacecraft at a precise angle so that by rolling the spacecraft about its axis it would eventually detect and home into Canopus. Often spacecraft in orbit are spin stabilized, however the requirement to maintain a constant fix on the Sun and Canopus meant that these craft had to be stabilised in three axes and attitude maintained precisely. In space, short of a collision that effectively alters the stability and velocity vector, any changes in position are likely to only to come from the spacecraft itself or to a far smaller extent from asymmetric solar wind pressure on the solar panels. Once the spacecraft had made its stellar fixes the gyro system could be switched off and the flight control system continue with reference to Sun and Canopus fixes. At the same time the telemetry would be relaying back a variety of navigation, attitude deviations and situational hardware reports to the DSN. Any required variations to attitude could be adjusted by the automatic use of cold gas jets at the tips of each solar array, which were under the control of the FCS (Flight Control System). These systems had been honed to near perfection during the 1969 missions in order to correct difficulties that had been encountered during the (experimental for this sub-system) 1964 mission. A further essential reason to have a fixed attitude was to ensure that the antenna (of the three available) in use was optimally pointed to ensure a good quality of bi-directional telemetry and control to and from the Earth.

The diagram below shows how coordinates in space are referenced. The common reference is to the first point of Aires at the Vernal Equinox whose position in space does apparently change, but over many decades. It is called the first point of Aries because it (γ, gamma, Arietis, in Aries) was so defined by Hipparcus in 130BC. Since then our Sun has moved so the vernal equinox now points to λ Piscium in Pisces, which is 106 light years away. This current reference point dates to year 2000 January 1st at 12:00, despite this change the coordinate is still referred to as the first point of Aries. The next update to this reference point will require that all planetary and star chart information currently in use will become obsolete and will need re-generation.

Figure 5.1: Space coordinates.



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The best description of the ecliptic I have found is from `http://hyperphysics.phy-astr.gsu.edu/hbase/eclip.html`, as follows. *If the sun's path is observed from the Earth's reference frame, it appears to move around the Earth in a path which is tilted with respect to the spin axis at 23.5deg. This path is called the ecliptic. It tells us that the Earth's spin axis is tilted with respect to the plane of the Earth's solar orbit by 23.5deg. Observations show that the other planets, with the exception of Pluto, also orbit the sun in essentially the same plane. The ecliptic plane then contains most of the objects which are orbiting the sun. This suggests that the formation process of the solar system resulted in a disk of material out of which formed the sun and the planets. The 23.5deg tilt of the Earth's spin axis gives the seasonal variations in the amount of sunlight received at the surface".* More generally I personally like to think of the ecliptic plane as a sort of an average disc on which the planets orbit the Sun, although Pluto is a bit naughty.

The image above shows an object in space (a star), with its coordinates shown as right ascension and declination, Canopus has a negative declination so would be seen down toward the South Celestial Pole as decribed below.

The celestial cordinates of Canopus are Right ascension: 6 hours 23 minutes 57.1 seconds and Declination: -52 degrees 41 minutes 45 seconds, which is negative so is our perception of South. Right ascension is the angle (measured against the time of Earths rotation) Eastward away from the vernal equinox where a point on the equator crosses the ecliptic plane or towards the first point of Aires where the Sun crosses the equator at the March equinox. The declination is the angle above or below the plane of the Earth's equator. Since Canopus is 313 light years away the apparent motion relative to any body orbiting in the solar system is miniscule although over a far longer period there is detectable apparent motion between our sun "Sol" and Canopus.

For spaceflight the brightest object is the Sun, it will be a very long time before we travel out to where other stars become dominant! This means that it is the strongest point of reference for an axis fix, accordingly spacecraft are fitted with a Sun sensor which offers a fix against its direction of travel (along its velocity vector), this also keeps the solar arrays collecting maximum photons. The system adopted by Mariner 9 comprised three types of module. Firstly there were four Acquisition sun sensors mounted at the tip of each solar array these would generate a signal to the FCS (flight control system) which could automatically adjust the attitude of the spacecraft until the signals had equal maginitudes, or at least fell within acceptable tolerances. The attitide of the spacecraft was adusted by a pair of six nozzle cold gas thrusters known as reaction control assemblies (RCA) which were mounted at opposite ends of a pair of solar arrays. Next there is a cruise mode sun sensor which has a narrower field of view and provides finer pitch and yaw signals, then there is a Sun Gate which is mounted on a solar array outrigger and indicates a Sun acquired state to the FCS. During attitude setup the inertial reference unit is active and comprises a three axis gyro system plus an accelerometer, positional information is then generated by integrating gyro rate changes. Previous Mariners had vane adjusters in the exhaust of the propulsion engine, Mariner 9 required a more powerful engine to slow down to achieve a Martian orbit and was fitted with a gimbal system.

At spacecraft separation from the launch vehicle, a signal from the pyrotechnic subsystem places the ACS in the Sun acquisition mode. In the Sun acquisition mode, the acquisition Sun sensors and the IRU rate signals cause the -Z axis of the spacecraft to be yawed into Sun alignment. When the Sunline falls within the FOV (field of view) of the Sun gate, the Sun gate circuitry issues a signal that identifies a Sun-acquired condition. The acquisition Sun sensor inputs to the RCA electronics are then disabled, and the cruise Sun sensor inputs are used exclusively. Meanwhile, the roll control channel, driven by the roll rate signal from the IRU, reduces the roll rates to within the rate deadband.

Figure 5.2: Spacecraft coordinate system.

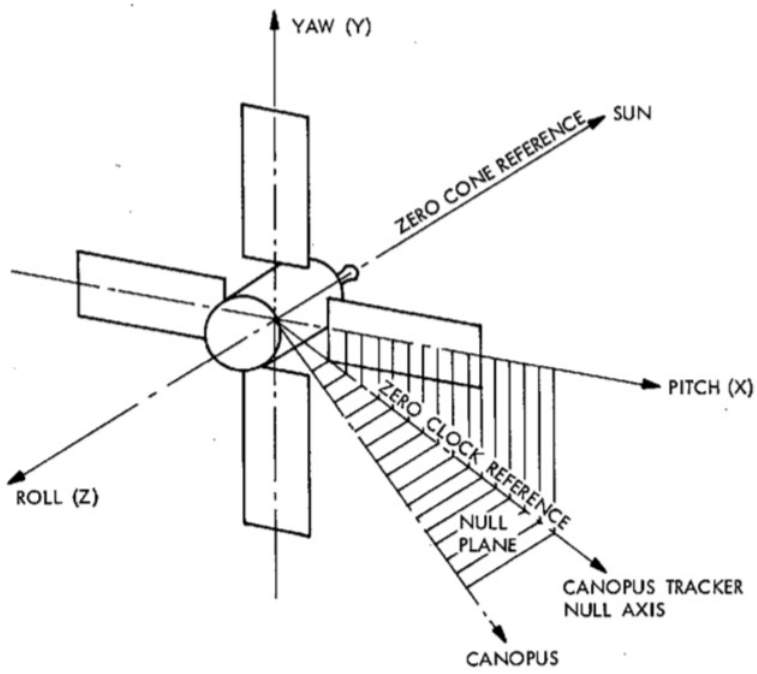


Figure 5.3: Inertial sensors.

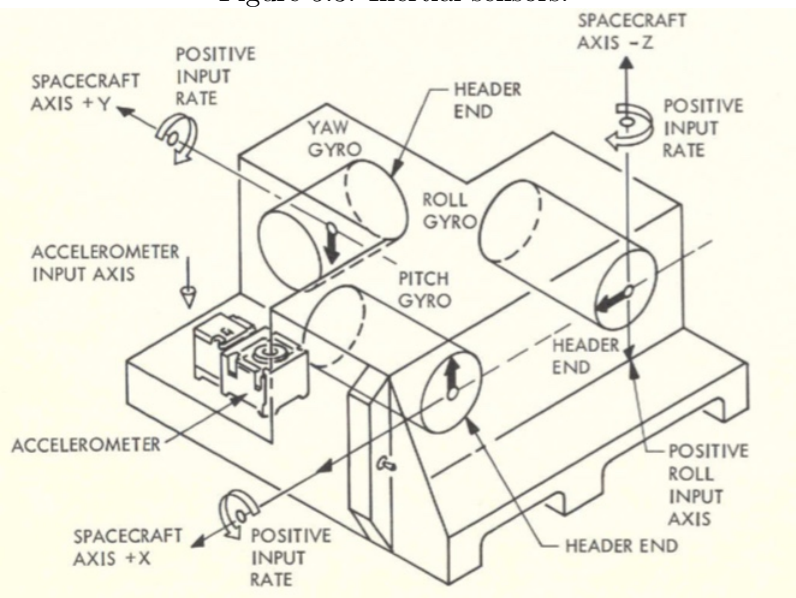
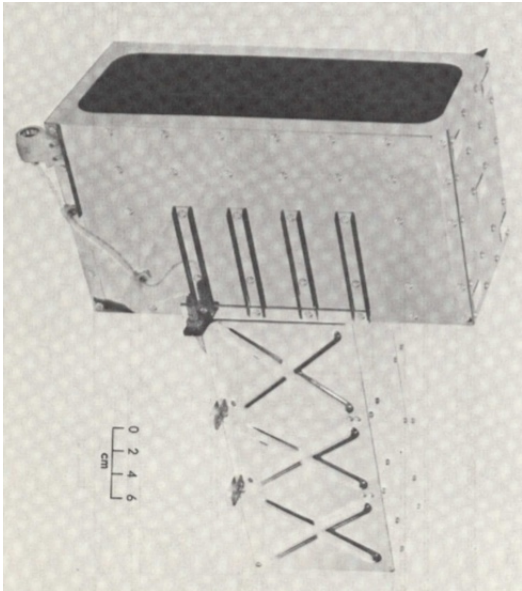
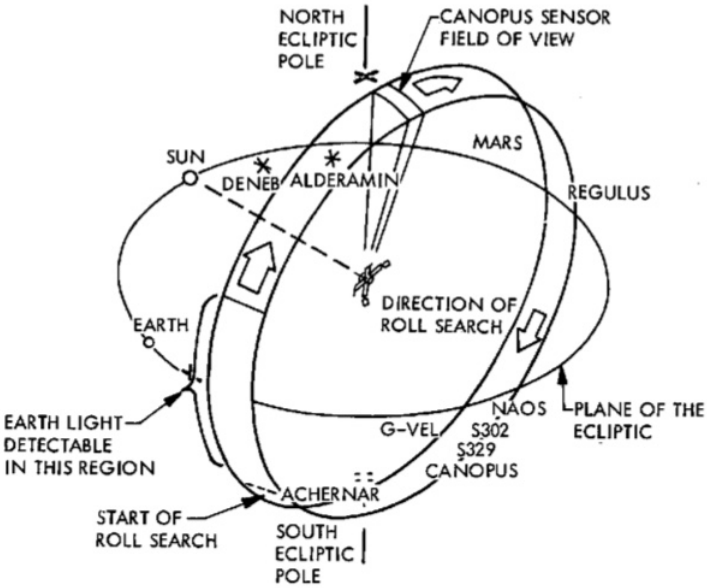


Figure 5.4: Canopus tracker.



The Canopus star tracker had a maximum field of view of 36.8 deg along the flight axis with a 10.8 deg view along the roll axis.

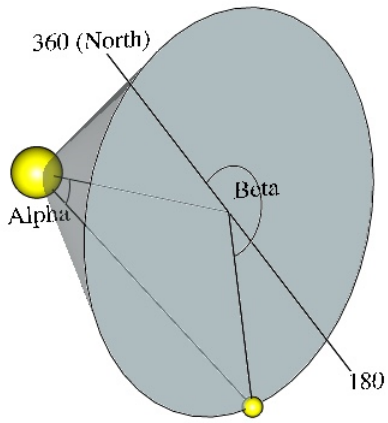
Figure 5.5: Canopus roll search.



This section from the final project report nicely describes the process of Canopus acquisition. *The fourth "hours" scan produced a 7B command, turning on the Canopus sensor. Since Sun acquisition, while the spacecraft was fully stabilized in pitch and yaw*

(with reference to the Sun), it was drifting without a reference for the roll axis. Application of power to the Canopus sensor caused it to immediately search for Canopus, the roll reference star. On the first roll, Achernar was acquired instead of Canopus, as expected. The second ground command sent to the spacecraft since the flight began, a DS-21, was transmitted to disacquire Achernar and continue the search for Canopus. The next star acquired was Canopus, and the spacecraft was fully stabilized in 3 axes at 02:25:10. Three min and 36 s later, the 3-min timer turned the gyros off.

Figure 5.6: Clock and cone coordinate system.



The diagram above refers to a cone with apex at the sun, in the yaw and pitch axis its flight axis will drift away form the centre of the cone by a given number of degrees (Alpha), the cone angle. The purpose of the FCS is to maintain the craft as close to zero as possible so that solar cells get maximmum exposure and also that the antenna are aligned with Earths DSN. In the roll axis a Canopus tracker will instruct the spacecraft to roll until it gets a strong lock on Canopus (angle Beta), this angle is called the clock angle. This tracker will be fixed onto the craft so as to be aligned for the correct declination. In previous missions it is believed that bright debris from the spacecrafts engine may have confused the trackers so this (time limited) factor has been negated out. We might also visualise a cone with Mars at the apex and another one with Canopus as a reference. Indeed the instrument platform had a Mars gate from which a theoretical cone and clock would be used to control the position of the sensor platform.

6 The Flight Plan

A flight plan involving thousands of orbital calculations was devised in order to meet the 1971 Mars/Earth opposition opportunity. Marina 9 was launched from Earth by an Atlas booster on May 30, 1971. It was then flown out of Earth orbit by a smaller Centaur booster into a hyperbolic escape orbit which would give it a trajectory to join a heliocentric (Sun centred) orbit. This heliocentric orbit was precisely planned and timed so that it would intersect closely with the natural path of Mars. At a point where the gravity of Mars was greater than that of the Sun's gravitational SOI (Sphere of Influence) the spacecraft would fall into a hyperbolic Martian orbit for an "impact" somewhere previously designated within the Martian SOI. At the designated arrival point the spacecraft would be rotated and rolled so its engine was pointing toward the position required to enter a planned Martian orbit and ignited for a chosen to time to adjust velocity into that which would be required to enter the orbit. If the velocity was too slow the craft would fall into Mars, too fast and it might evolve into another flyby mission. Providing telemetry was available to Earth some corrective instruction for a further engine burn might be made to correct any error. Since the orbit of Mars is around 12 hours there would probably be far less than 6 hours available to design and send the correction to the spacecraft and activate it as soon as it emerged from the dark side of Mars, i.e. back into communication with Earth. If you want to view fuller details of orbit planning then JPL's then "Technical Memorandum 33-100" will give a flavour. Nowadays a computer might be able to receive an up to date position vector, calculate the orbital situation, evaluate required changes and send the commands back in not much more time than the bi-directional signal propogation delay, however by that time I will conclude that the spacecraft may well be fuel constrained so at best may only be able to make a slightly higher orbit in order to prolong its useful life.

This following section shows the ephemeris data for Mars that would have been used for orbital calculations. This is open to public use at `https://ssd.jpl.nasa.gov/horizons.cgi` It is important to set the observation point, since the majority of the journey was a heliocentric orbit from Earth to Mars I have chosen the Sun. The small variation in the delta values, illustrate the ellipticity of Mars orbit around the Sun.

Figure 6.1: Launch day

```
*****************************************************************************************************
 Date__(UT)__HR:MN     R.A._(ICRF/J2000.0)_DEC  APmag  S-brt            delta      deldot    S-O-T    S-T-O
*****************************************************************************************************
$$SOE
 1971-May-30 00:00     18 15 30.02 -24 40 32.7   0.09   3.88 1.44781853324368  -2.0033030   0.0000   0.0048
 1971-May-31 00:00     18 18 02.77 -24 40 20.4   0.08   3.88 1.44666459741033  -1.9926137   0.0000   0.0048
$$EOE
*****************************************************************************************************
```

Figure 6.2: Orbit arrival

```
*****************************************************************************************************
 Date__(UT)__HR:MN     R.A._(ICRF/J2000.0)_DEC  APmag  S-brt            delta      deldot    S-O-T    S-T-O
*****************************************************************************************************
$$SOE
 1971-Nov-13 00:00     01 03 45.73 +05 42 18.8  -0.02   3.82 1.41091641428961   1.4801843   0.0000   0.0050
 1971-Nov-14 00:00     01 05 59.73 +05 57 09.9  -0.02   3.82 1.41177651135860   1.4982406   0.0000   0.0050
$$EOE
*****************************************************************************************************
```

Column meaning: Prior to 1962, times are UT1. Dates thereafter are UTC. Any 'b' symbol in the 1st-column denotes a B.C. date. First-column blank (" ") denotes an A.D. date. Calendar dates prior to 1582-Oct-15 are in the Julian calendar system.Later calendar dates are in the Gregorian system. Time tags refer to the same instant throughout the solar system, regardless of where the observer is located. For example, if an observation from the surface of another body has an output time-tag of 12:31:00 UTC, an Earth-based time-scale, it refers to the instant on that body simultaneous to 12:31:00 UTC on Earth.

The Barycentric Dynamical Time scale (TDB) is used internally as defined by the planetary equations of motion. Conversion between TDB and the selected non-uniform UT output time-scale has not been determined for UTC times after the next July or January 1st. The last known leap-second is used as a constant over future intervals.

R.A. (ICRF/J2000.0) DEC = J2000.0 astrometric right ascension and declination of target center. Adjusted for light-time. Units: HMS (HH MM SS.ff) and DMS (DD MM SS.f)

APmag S-brt =Target's approximate apparent visual magnitude and surface brightness. For planets and satellites, values are available only for solar phase angles in the range generally visible from Earth. This is to avoid extrapolation of models beyond their valid (data-based) limits. Units: MAGNITUDE and VISUAL MAGNITUDES PER SQUARE ARCSECOND

delta deldot = Range ("delta") and range-rate ("delta-dot") of target center with respect to the observer at the instant light seen by the observer at print-time would have left the target center (print-time minus down-leg light-time); the distance traveled by a light ray emanating from the center of the target and recorded by the observer at print-time. "deldot" is a projection of the velocity vector along this ray, the light-time-corrected line-of-sight from the coordinate center, and indicates relative motion. A positive "deldot"

means the target center is moving away from the observer (coordinate center). A negative "deldot" means the target center is moving toward the observer. Units: AU and KM/S

S-O-T /r = Sun-Observer-Target angle; target's apparent SOLAR ELONGATION seen from the observer location at print-time. Angular units: DEGREES

The '/r' column indicates the target's apparent position relative to the Sun in the observer's sky, as described below:

For an observing location on the surface of a rotating body (considering its rotational sense): /T indicates target TRAILS Sun (evening sky; rises and sets AFTER Sun) /L indicates target LEADS Sun (morning sky; rises and sets BEFORE Sun)

For an observing point NOT on a rotating body (such as a spacecraft), the "leading" and "trailing" condition is defined by the observer's heliocentric orbital motion: if continuing in the observer's current direction of heliocentric motion would encounter the target's apparent longitude first, followed by the Sun's, the target LEADS the Sun as seen by the observer. If the Sun's apparent longitude would be encountered first, followed by the target's, the target TRAILS the Sun.

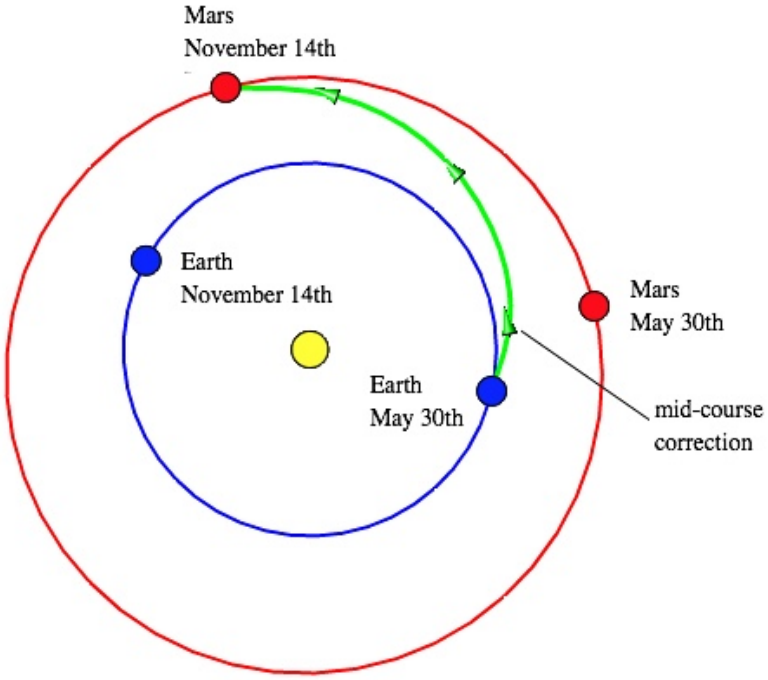
NOTE: The S-O-T solar elongation angle is numerically the minimum separation angle of the Sun and target in the sky in any direction. It does NOT indicate the amount of separation in the leading or trailing directions, which are defined in the equator of a spherical coordinate system.

S-T-O = "S-T-O" is the Sun-¿Target-¿Observer angle; the interior vertex angle at target center formed by a vector to the apparent center of the Sun at reflection time on the target and the apparent vector to the observer at print-time. Slightly different from true PHASE ANGLE (requestable separately) at the few arcsecond level in that it includes stellar aberration on the down-leg from target to observer. Units: DEGREES

Computations by ... Solar System Dynamics Group, Horizons On-Line Ephemeris System, 4800 Oak Grove Drive, Jet Propulsion Laboratory Pasadena, CA 91109 USA Information: http://ssd.jpl.nasa.gov , Connect : telnet://ssd.jpl.nasa.gov:6775 (via browser), telnet ssd.jpl.nasa.gov 6775 (via command-line), Author : Jon.D.Giorgini@jpl.nasa.gov

The scale of the diagram below does not allow for showing the hyperbolic exit and entry orbits from Earth to Mars.

Figure 6.3: The actual flight path showing planetary positions at launch and arrival.



Two trajectory correction maneuvers (TCM) were planned for Mariner 9 during its cruise to Mars. These maneuvers would eliminate an intentional target bias (required by planetary quarantine considerations) and correct for any launch vehicle-induced trajectory errors. Establishment of an accurate trajectory, time of arrival at Mars and target point, would allow insertion into the desired Mars orbit with minimum use of propulsion fuel. Final preparations for the first trajectory correction maneuver began shortly after the successful Mariner 9 launch on May 30, 1971. Successive orbit determination calculations were run, and a maneuver strategy was developed. As additional tracking data were obtained from the Earth-based antennas, knowledge of Mariner 9's trajectory became increasingly accurate. Maneuver strategy studies led to the best procedure for turning the spacecraft to the desired orientation for firing the engine. From the Sun- and Canopus-stabilized orientation, it was decided to first roll the spacecraft 141 deg (counterclockwise as viewed from the Sun) about an axis through the rocket nozzle and then turn the spacecraft about its yaw axis -45 deg (counterclockwise as viewed from above, looking down on the spacecraft toward the star Canopus).

These turns would orient Mariner 9 with its rocket pointing almost toward Earth. After engine firing, the spacecraft would be returned to its previous three-axis stabilization by reversing the order and direction of the turns. On June 3, 1971, operations planned for the Mariner 9 first trajectory correction maneuver were checked on the proof test model at the JPL Air Force Eastern Test Range facility. Beginning at 19:30:00 GMT on the same day, a series of six coded commands (CC-4) were transmitted to Mariner 9, followed by thirteen CC-1 and CC-2 pairs on one-minute centers starting at 19:36:00,

to load the maneuver parameters into the fixed central computer and sequencer (CCS). Next, the maneuver-enabled direct command (DC-14) was sent, followed by a DC-33 to put the CCS in the tandem standby mode and a DC-29 to disable the divide-by-32 network in the accelerometer circuitry. The tandem standby mode was a necessary condition for executing a "tandem" maneuver. The spacecraft was then placed in the roll inertial mode prior to propulsion subsystem pressurization to avoid the possibility of loss of Canopus due to bright particles released by the pressurization impulse. DC-65 was transmitted at 21:17:25 to fire pyrotechnic valves in the propulsion subsystem. With these valves open, both oxidizer and fuel storage tanks were pressurized, forcing propellants through the lines to the main engine valve. The impulse about the yaw axis was quickly damped out; no bright particles were observed. Canopus reference was restored by DC-19 at 21:48:00. The first trajectory correction maneuver was executed on June 4, 1971. The spacecraft CCS loads were further refined by two CC-4's and twelve CC-1 and CC-2 pairs, respectively. A time-critical DC-52 transmitted at 22:19:04 started the on-board maneuver routine; then gyros were turned on.

The following table indicates the programmed and actual values of the maneuver:

Parameter	Programmed value	Actual value
Roll turn magnitude, deg	-140.806	-140.717
Yaw turn magnitude, deg	-44.725	-44.828
Roll turn time, s	777	777
Yaw turn time, s	247	247
ΔV impaired to spacecraft ms^{-1}	6.731	6.723
Accelerometer pulse count	223	223

Nominal performance occurred during the spacecraft roll and yaw turns.

At 00:22:00 GMT, June 5, 1971, the main engine valve was opened, and the hypergolic propellants, nitrogen tetroxide (oxidizer) and monomethyl hydrazine (fuel), burned for 5.1s until the main engine valve was automatically closed. Spacecraft yaw and roll unwind was accomplished. At 00:48:44. After a short roll search, Mariner 9 reacquired the Sun and Canopus celestial references. The gyros were turned off 3 min and 36 s later. Tracking data indicated that the first trajectory correction maneuver was extremely accurate and the orbit determination computations on June 14, 1972 showed:

Parameter	Targeted	Achieved	Error
ΔTCA (time of closest approach)	19 h 06 min 36secs	19 h 04 min 28 secs	02 min 08 s
ΔB (B-plane target point correction (vector target point error), km	24,948	24,869	140 (vector error)

Tracking data and orbit determination computations performed in September and October 1971 showed that the first TCM was sufficiently accurate to justify cancellation of

the second TCM. The Mars orbit insertion (MOI) maneuver (Ref. 56) would occur on November 13, 1971 (November 14, 1971 GMT). The purpose of this maneuver was to decrease the spacecraft speed so that the Mars gravity field would capture the spacecraft in an orbit whose parameters were:

Table 6.1: MOI (Mars Orbit insertion)

Orbit parameters	MOI (as of 11/15/71)
Period	12.5 h
Periapsis	1300 km
Inclination (to Mars equator)	65 deg

The orbit insertion maneuver also would take place over Goldstone because the high-gain antenna would be pointing off the Earth and the engineering telemetry would have to be played back over the medium-gain antenna and received over the 64-m antenna at Goldstone. The motor burn would begin about 28 min prior to closest elliptic approach. Total burn time would be approximately 16 min and the time to reacquire Canopus and initial doppler data would be approximately 2 h. The orbit insertion maneuver would be a planar transfer from the hyperbolic orbit to the elliptic orbit. This meant that there would be no change made in the inclination at orbit insertion. Due to the required rotation of the periapsis of the elliptic orbit, the orbit insertion would not be a minimum energy transfer.

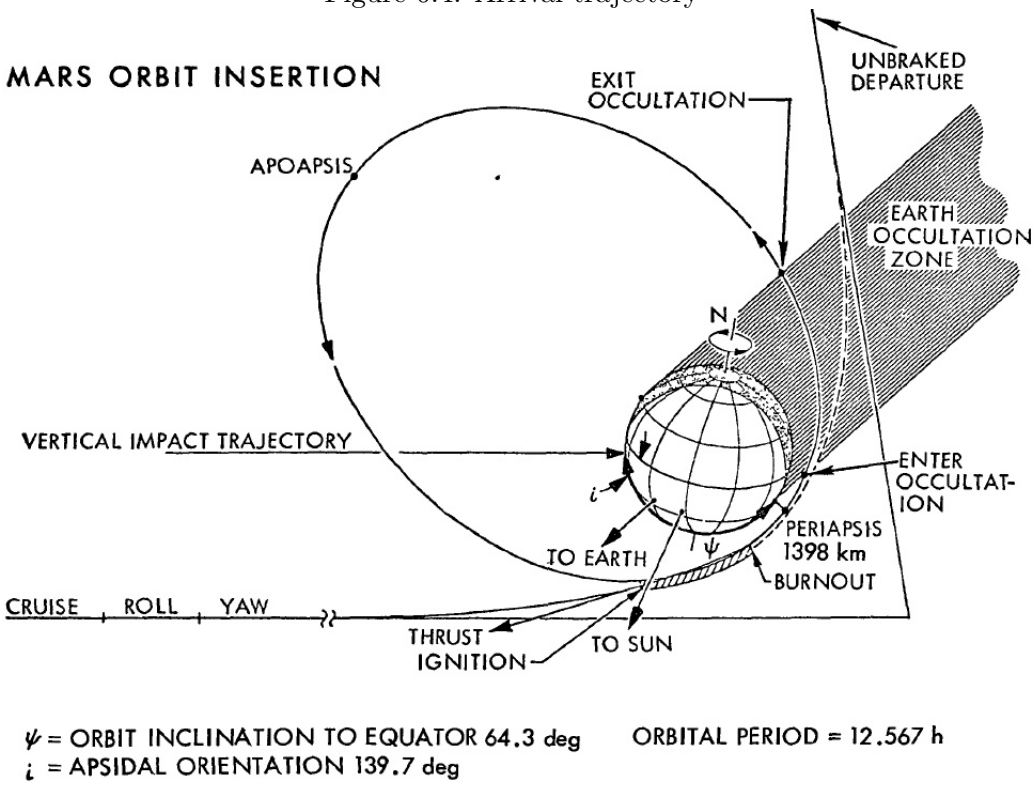
The above text was taken directly from NASA Technical Report 32-1550 Volume 1 P.57 onwards (Mariner Mars 1971 Project Final Report), we need to look to Volume 3 (P.46) to get orbit entry details.

Continued tracking and orbit determination up to a few hours prior to orbit insertion showed significant changes in miss distance, which would affect the design orbit parameters unless compensated for by an adjustment in the orbit insertion maneuver. After weighing the alternative of carrying out the designed orbit insertion maneuver or altering the program for the maneuver, it was decided to do the former. The spacecraft was inserted successfully into orbit by a 15-min motor burn. The maneuver was carried out on schedule, with motor ignition occurring at 0:17:39 GMT on November 14, 1971. The resulting orbit had a period of 12 h, 34 min, 1 s, a periapsis altitude of 1398 km, an apsidal rotation angle of 139.7 deg, and an orbital inclination of 64.4 deg. The orbital period placed the spacecraft at the ideal time and place to execute an orbit trim maneuver near the fifth periapsis passage or after four complete revolutions about Mars.

It is interesting to briefly summarise and comment on the above sequence of events. At arrival near Mars on November 13 1971, the spacecraft was firmly under the influence of Martian gravity. At 2.14 p.m. (PST) the autopilot switched on. Three minutes later ground control issued four DC-27 commands for a manouevre sequence. At 2.44 p.m. the pitch and yaw gyros were started for about an hour before use to warm up and stabilise the inertial reference unit. At 3.52 p.m. a roll turn was began and a minute later the system switched to the low gain antenna. At 3.56 the roll turn was stopped, this roll took four minutes so the rate was very slow, presumably to avoid overruns.

At 4.04 the yaw turn began, from 4.06 to 4.13 there was a telemetry blackout until the medium gain antenna aligned toward Earth, at 4.16 the yaw turn stopped. The spacecraft was now in position to slow down to enter a Martian orbit so at 4.24 a sixteen minute engine burn took place to get to the required velocity to adjust into an elliptical Martian orbit. From 4.41 to 4.46 the previous yaw manouevre was reversed, during which time, at 4.53, the disappeared behind Mars (Occultation). From 4.57 to 5.01 the previous roll manouevre was reversed and at 5.22 p.m. occultation ended and spacecraft was pointing back to Earth with the low gain antenna. At 6.24 the high gain antenna was activated and transmission of images back to Earth began, which included images of Phobos and Deimos plus images of Mars also taken during the arrival phase. After the initial commands were sent the entry into orbit was totally automated, therefore nobody knew whether it was a success until occultation ended and doppler ranging and engineering data telemetry back to Earth resumed.

Figure 6.4: Arrival trajectory



Top right above, "unbraked departure" shows the departure asymptote that the spacecraft would have taken if the engine had failed to operate. If the engine had operated for too long at best there would have a been a wrong orbit, throwing out of plan the imaging schedule, at worse the spacecraft might have impacted the planet. The inital orbital data is recorded in the tabel below.

Table 6.2: MOI (Mars Orbit insertion)

Orbit parameters	MOI (as of 11/15/71)
Period	12.5 h
Periapsis	1398 km (869miles)
Orbit Inclination to Mars equator i	64.4 deg

For orbit trim manoevres the following specifications are required.

Orbit trim maneuver (OTM) 1 was designed primarily to adjust the orbit period from the 12 h, 34 min obtained after orbit insertion down to the 11-h, 58-min, 48-s period needed to synchronize periapsis passages with Goldstone zenith. The OTM was accomplished as planned, with motor ignition occurring at 2:37:53 GMT on November 16, imparting a velocity increment of 15.25 m/s

Table 6.3: Orbit Trim Maneuvers

Orbit parameters	OTM 1	OTM 2
Period	11hr 58mins (mean)	11hr 59min 28s (mean)
Periapsis	1387 km (861 miles)	1650km (1025 miles)
Orbit Inclination to Mars equator i	64.4 deg	64.4 deg

After arrival in orbit the science phase immediately began but it was soon discovered that much of Mars was covered in a huge dust storm, so a more limited investigation was put in place. Although the dust storm obscured the surface, it did give valuable insights into other factors such as atmospheric circulation. It was not not until the second part of December that the dust storms appeared to abate and the bulk of the orbital science and imaging phase of the mission could resume, for which the OTM 2 burn was activated on December 30th which allowed for better synchronisation with the 64m antenna at Goldstone for high speed scientific download (images).

7 Data transmission

By 1971 radio transmission had evolved to a point where higher data rates over longer distances could be achieved. Part of this was due to improvements in the DSN. Whereas the images from Mariner 4 were sent at 8.33 bits per second, I emphasise this is correct, 6 years later the higher power at both ends meant less signal attenuation and a much improved signal to noise ratio (SNR), which meant that less corruption of the signal was likely. Furthermore significant advances in data encoding technologies had also been made, this meant that error correction became possible and the system could be operated with higher tolerance towards SNR aberrations. I should also point out that whilst this mission was underway design work was already proceeding toward taking this a step further for the planned 1977 missions to Saturn and Jupiter. Mariner 10, in 1973, would be testing some of the X-band capability that would be required to communicate effectively at the much further distance of the outer planets.

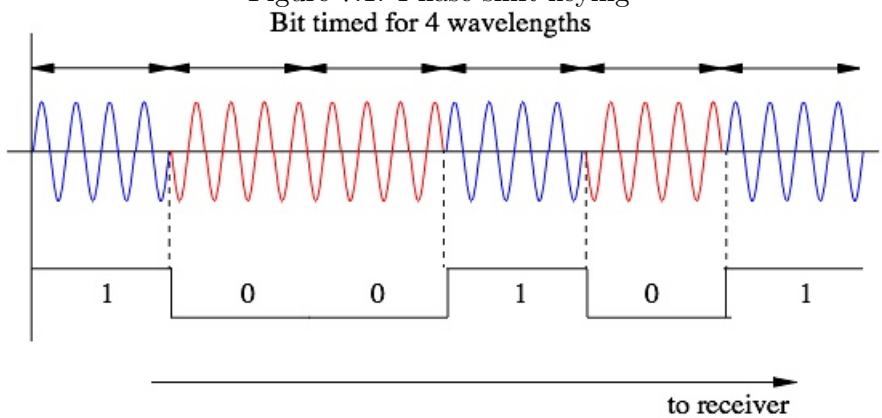
Unlike a modern communication protocol (e.g TCP) there could be no synchronisation or re-transmission. Signals were sent asynchronously in each direction (at different frequencies). In the case of images it is not required to always have 100% accuracy for every pixel, an unusual pixel can be compared to its neighbours and an area trend established to calculate the probabibilty of interpolating a better fit value. However for some of the engineering telemetry it would be far more critical to not have any errors accrue during transmission which may be interpreted as perhaps the spacecraft spinning or failure in a major system. Each image contains a number of pixels which are represented as a data number (DN) value in the range of of 0 to 511. If for example a pixel was sent as decimal 36 that is represented in 9 bits as binary value 000100100, if the most significant bit (the highest value bit) were corrupted this may be received as 000000100 which is decimal 4. If on analysis the adjacent pixels were all found to have data values in the range of say 30 to 40 it is realistic to assume that 4 is an error and to select a suitably interpolated value in the range of 30 to 40 as a substitute. For geographic images this is acceptable but the question does arise as to what area does a pixel represent at long range? Maybe 100 square metres, or 4 square kilometers, it all depends on the camera being used and the distance. In either case it will not show you where the Martian anti-alien defence system is sited! OK that last part is a bit fatuous but it would not show details such as if there were ever any Martian built pyramids. On the other hand if 36 was an accepted pressure level for a fuel tank and mission control got a value of 4 then alarms might well trigger to instigate a mission failure condition. This top is covered in more detail further on.

By implication if one bit could be encoded per wave cycle at 2.3GHz we might think, at

least in theory, that we can send 2.3 gigabits per second. Alternatively a timed sequence over several wave cycles might give better clarity and be used to represent a bit. This system used phase shift keyed modulation so every state change from binary zero to one was shown by a phase change. A problem also arises that maybe the computer cannot work that fast. At the time of writing (2017) an 8th general Intel Core processor can operate at clock frequencies of up to 4.7GHz whilst the CCS system of Mariner 9 was clocked at 2.4KHz, i.e. a million times slower. Digging a bit deeper it may also be found that the radio system exciter multiplies the frequency and phase by a factor of 120 for transmission through S-band, so can only be receiving ALL input at no more than 19.2MHz from the modulator. To this we have to also consider that uplink channels are from 2110-2120Mhz and comprise 29 channels (5-33) each of approximately 400KHz and the downlink channels are from 2290-2300Mhz of which 27 channels (1-27)are defined each of approximately 350KHz average, the precise frequency for each channel does vary to some extent through the available 10MHz frequency spectrum. These frequencies are registered and published in the Master International Frequency Register, which is itself part of the ITU (International Telecommunications Union, a department on the U.N.) in Geneva. The highest possible data rate for this mission was 16.2kbps, which will be described shortly.

Following is an example of a hypothetical bi-phase communication system. Bit timing is performed every six cycles, bit values are dependant on the phase (shown shifted by 180 degrees plus colour changes for the wave at dotted lines). The first bit value received is to the right so the reciever has received 101001, if the system is configured to process 6 bit words we have to decide which is the MSB (most significant bit) if the first bit received is MSB then the value of the word is 37, if the MSB received is last then the value would be 41. The vertical dotted lines show points where the phase is shifted (this ought to only happen every 4 cycles). We can determine the bit time simply by $t = frequency/bitcycles$. So as an example for a 350Khz channel where a bit is timed across 6 waves the bit timing is $1.7 \times 10^{-5} secs$ and the bit rate would be 58kbps and if the bit sampling rate was 24 cycles the data rate would be 14.6kbps. Later communications system would offer options for quadrature phase shifting plus some early data compression techniques.

Figure 7.1: Phase shift keying



The science package sent data to the Data Acquisition Sub-system where it processed and formatted into frames for use by the Flight Telemetry Sub-system (FTS) from which it was modulated and send to the Radio Frequency Sub-system (RFS) and then to the Travelling Wave Tube amplifier and onto the antenna. A further consideration here is that the link may well consist of more than one channel, which are all multiplexed together. For the DSN it would be far more convenient to process sub-channels for science (at least four types), radar ranging, engineering telemetry etc so it can be isolated by type at reception and passed immediately over to appropriate processing systems. From documentation that I have read that the 1969,71 and 73 Mariners used two downlink channels: one, for science and one for engineering and control it would therefore be necessary to interleave all the different types and source of data within their respective channels. The engineering channel had 94 measurement which contained a selection of analogue and digital word measures from various sub-systems and components within. Analogue voltages were converted to 7 bit NRZI prior to processing. This could be sent at either 8.33 or 33.33 bits per sec. The science channel was used to transmit all data from science systems which were present in digital formats of differing types. Realtime science data could be sent at 50 bits per sec, block-coded realtime science data at either 16 or 8kbps, depending on antenna gain. Whenever antennas were occulted from Earth or for other factors all data was writen to tape which would later be played back at 16,8,4,2 or 1Kbps, again depending upon antenna gain, SNR ratio and the DSS stations in view at Earth. The highest data rates were only possible to Goldstone at that particular time.

To compensate for errors found in the reception of data sent, the data was encoded using an error-correcting code (ECC). For this it was decided to use a Hadamard code (a 1st order Reed-Muller type). This code is often notated as follows [32, 6, 16], the first value in brackets is the bit length of the full coded block, the second the bit length of the original data and the third length is the minimum Hamming distance, so assuming that both transmitter and receiver are operating identically we send 32 bits instead of 6. The math' behind this requires use of a multiplication of 6 by 6 and 32 by 6 matrices and dot product vector calculation of the received messsage with each row. This method may detect up to 15 errors of which up to 7 may be fixed. The choice and evolution of method is often taken after detailed statistical analysis of the probable bit error rate and whether that is acceptable. If transmitting a voice message, picture or a movie for immediate human consumption, if one bit in 100 is wrong it will not make that much difference. If however transmitting an instruction, which might be for example a location coordinate or a boost burn time command to a spaceship then accuracy is extremely important, so repetitive transmission may be used so as to ensure that enough data samples agree before invoking any action, if they don't then a request signal can be sent to repeat.

Each image comprised 700 scan lines and 832 bits per line, which is 582,400 pixels per image. Each pixel had a DN in the range of 0 to 511 which required 9 bits (2^9) 9 bits so each image comprised at least 5,241,600 bits to which would also need to be added additional control information which at its basic level would be image identifier and line number. Now bear in mind that the coding process took six bit sections and

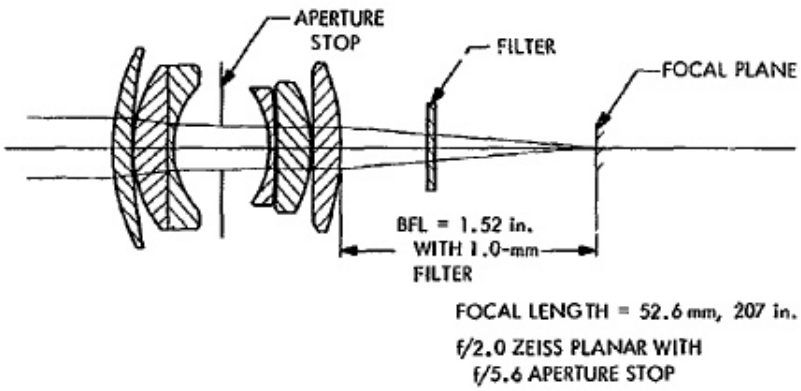
"stretched" them to 32 the image size became at least 27,955,200 bits.The maximum available (coded) data rate using the High Gain Antenna was 16,200bps (16.2kbps) which implies a transmission time for each image as 1,725 secs (29 mins), compare that with around 8 hours for a far inferior Mariner 4 image. Not bad progress in just 7 years! Without continued improvement in transmission capability later journeys to Jupiter, Saturn and Uranus may have been pointless. A few years later with the use of X-band, data rates of 100Kbps became available and were tested on Mariner 10. it is interesting that a lot of customer available earth bound data transmission services trailed a bit behind what NASA and JPL were achieving, I wonder whether the science (with budget) that NASA were developing aided the telecommunications industry or was it simply synergy across different disciplines?

8 Cameras and Image post processing

The spacecraft carried two cameras, one provided narrow angle and the other wide angle coverage.

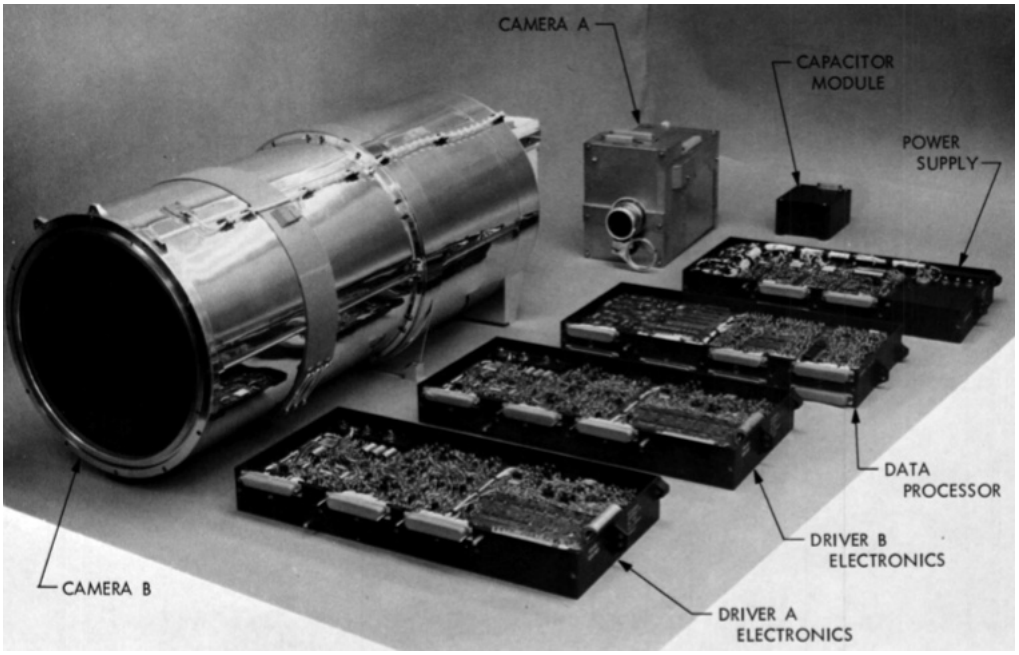
Television camera A. (a) Rectangular 11 by 14-deg wide-angle field of view. (b) Exposure time controlled either by on-board logic (DAS exposure algorithm) or ground command. (c) Eight-position filter wheel with the filter cycled automatically through even position 2 through 8, or set by ground command.(d) Each picture composed of 700 lines, each having 832 picture elements pixels) per line. The brightness of each element was encoded to a 9 bit resolution.(e) Data recorded in DSS for delayed playback. Also, selected data transmitted in real time in selected video format of 16.2 kilobits per sec.

Figure 8.1: Wide angle camera



Television camera B. (a) Rectangular 1.1 by 1.4-deg narrowangle field of view. (b) As per A. (c) Single fixed filter. (d) & (e) As per A.

Figure 8.2: The camera system



Camera B is in fact a small Cassegrain telescope system with an imaging backbody. Those who wish to delve into the fuller details should refer to: Technical Memorandum 33-505 "Development and Testing of the Television Instrument for the Mariner Mars 1971 Spacecraft"

If you displayed the raw data received you might have an image that looks like a) below which is not immediately useful. Every dot/pixel has a corresponding DN value (0 to 511) which can be "numerically image processed" to yield a very useful image, see b) in the next Figure. So the question is how do we get from a) to b)? Before that question is explored, prior to launch a lot of work was carried out creating calibration images, using the cameras that would actually fly that could deal with expected images received at different optical intensities and at different view angles and orientation. Furthermore the calibration could identify any optical flaws and characteristics so that algorithms could be built to adjust image areas which were known to have errors, a bit like using an electronic pair of spectacles to clarify and enhance an image. In actual practice the optical aspects of the system offered more accuracy than the videcon recording system, this was known from previous missions.

Figure 8.3: Image before and after enhancement

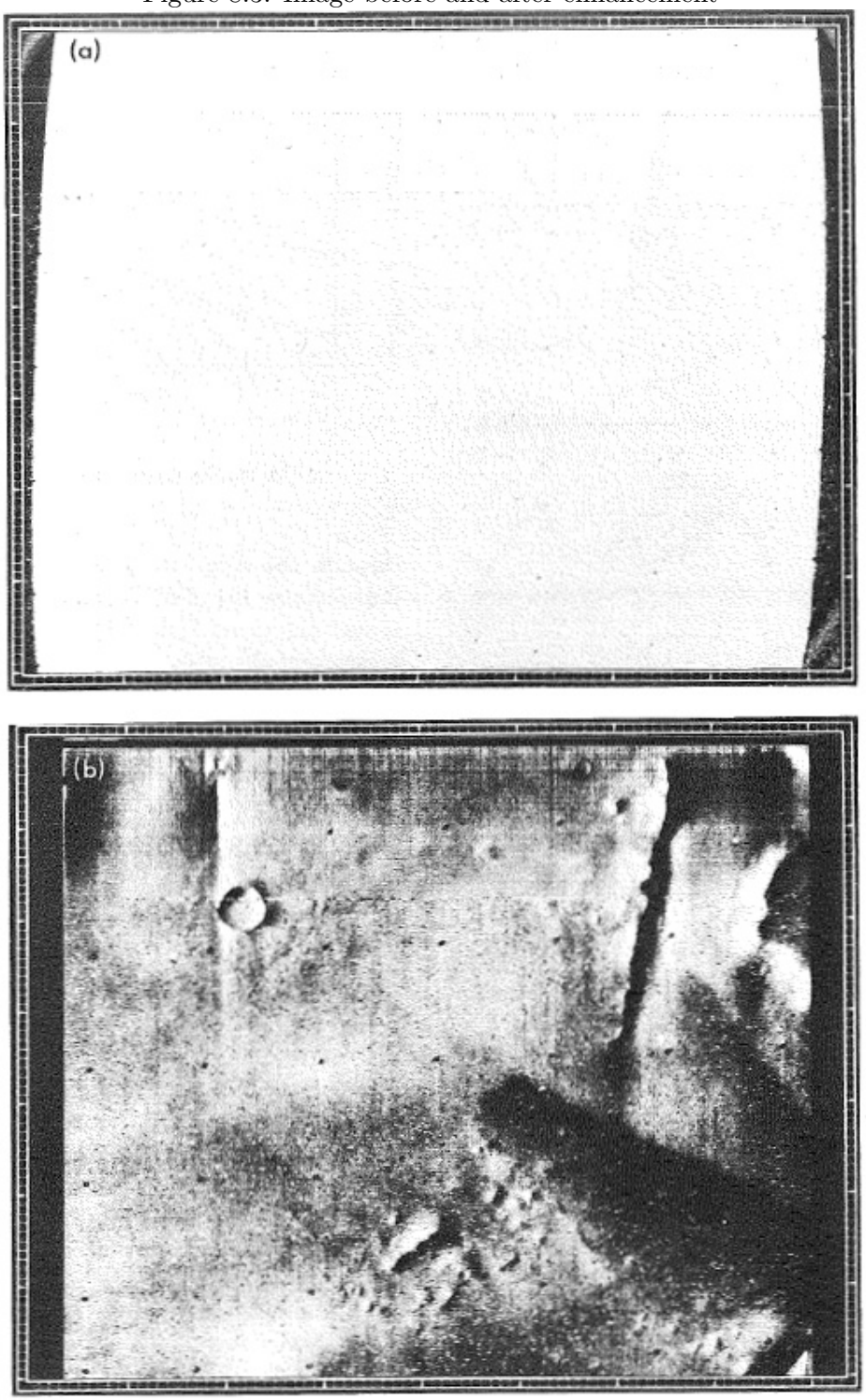


Fig. 33. Typical TV picture received at low signal-to-noise ratio after SNR and PN bit errors were less constrained (first 25% of picture was received at 2.5 dB SNR, last 75% at 0.5): (a) "Raw" picture before enhancement, (b) Enhanced picture

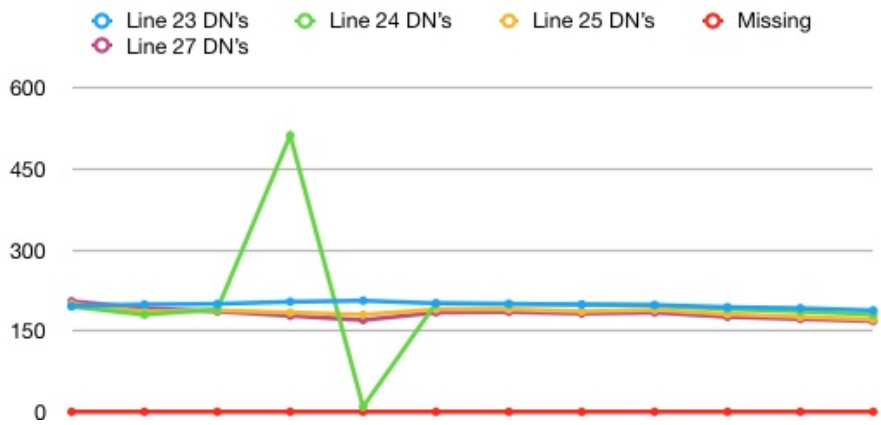
The data upon arrival was nown as the EDR (Experiment Data Record) which contained the raw numerical values of pixels (DN) on a 9-bit (0 to 511) scale for the 832 by 700 line matrix of intensities transmitted from the Mariner 9 cameras. For preservation this was immediately written to magnetic tape, thus allowing re-processing should a computer fail. From this point the following types of processes we carried out.

1. correct missing lines and any bit errors. Errors in the data stream were possible due to signal to noise variations, these might be bursty in nature and could corrupt whole lines of an image. There were also spikes in pixels that are higher in value (lighter) or lower in value (darker) than the surrounding pixels because of bit errors in the higher-order bits of the 9-bit code that represents the numerical value of that pixel. Where a line was invalid it was possible to interpolate values between previous and later lines, similarly a spike could be dealt with by comparing with adjacent pixels and also those adjacent in previous and next lines.

The following table shows a section of an image from line 23 to 27 and pixels 47 to 58. It is clear that line 24 has problems which are highly visible in the green line on the graph below, also line 26 is missing, we just have a string of zeroes, in red on the graph.

Figure 8.4: EDR before correction

Pixel No	47	48	49	50	51	52	53	54	55	56	57	58
Line Number	23											
DN	196	199	200	204	206	201	200	199	198	194	192	188
Line Number	24											
DN	195	180	190	512	9	201	200	199	197	190	186	180
Line Number	25											
DN	200	186	188	184	180	190	190	186	189	181	176	171
Missing	0	0	0	0	0	0	0	0	0	0	0	0
Line Number	27											
DN	205	192	186	178	170	184	185	182	184	176	172	168



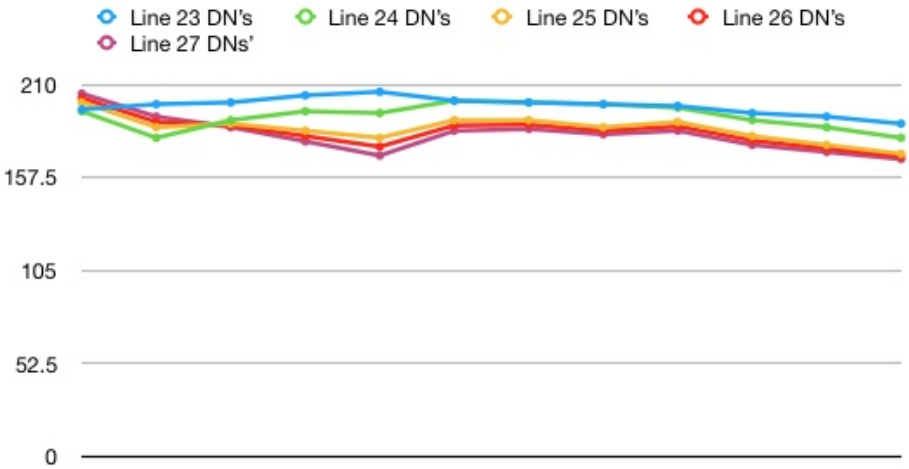
To fix these errors on line 24 (pixels 50 and 51) I simply took an avarage of the best adjacent values, both horizontally and vertically and then averaged these results, rounding to the nearest whole number. To interpolate the missing line 26 I also simply took the average of the corresponding vertical pixel values in lines 25 and 27, the results are below. The method I have used may not be precisely what was used over 40 years ago but the method does illustrate the type of repair and adaption that can be made. Now for a word of caution I have adapted and interpolated values; over the wide angle camera I have actually averaged out an area of around 50 square Km. I may have missed a town, on the other hand a value of 512 at the extreme is more likely to be a simple bit error. If that image had been through the narrow angle telescopic camera the value of 9 I replaced might have covered an area of maybe just 100 square metres. So did I just obliterate a Martian Pyramid or a large sink hole? This whole process is one of statistics and probablities, one way of increasing confidence is by comparing and overlaying many images from the same site.

Figure 8.5: Interpolated values

Horizontal correction				196	196							
Vertical Correction				194	193							
Average Correction				**195**	**194**							
Missing Line Fix	**203**	**189**	**187**	**181**	**175**	**187**	**188**	**184**	**187**	**179**	**174**	**170**

Using the repaired results the RDR record is created as per below.

Figure 8.6: EDR after correction to an RDR



The graph tells us one thing immediately, that we have values ranging from around 160 to 200, so that's just 40 out a possible 512 values, i.e. less than 10% of possible values.

What this means is that any difference in contrast will be very hard to detect by the human eye, as illustrated by the raw image earlier. It would be nice if we could stretch that range of 40 to increase contrast, so we get a better visualisation of the topology. Thankfully using the quite simple equation below can help us quite a bit.

$DN_{stretch} = 511\left[\frac{DN_{RDR}-low}{high-low}\right]$ To process this one must choose values for high and low that find a good stretch solution and is done by trail and error. In the extreme high the results will give some values higher than 512, or in the extreme low less than zero,each would mean losing clarity out of the RDR. The eventual values that I chose was high=220, low=160, there may be a better range, but this is useful to appreciate the concept.

Figure 8.7: Stretching a RDR

Pixel No	47	48	49	50	51	52	53	54	55	56	57	58
Line Number	23											
DN	196	199	200	204	206	201	200	199	198	194	192	188
DN Stretched	307	332	341	375	392	349	341	332	324	290	273	238
Line Number	24											
DN	195	180	190	195	194	201	200	199	197	190	186	180
DN Stretched	298	170	256	298	290	349	341	332	315	256	221	170
Line Number	25											
DN	200	186	188	184	180	190	190	186	189	181	176	171
DN Stretched	341	221	238	204	170	256	256	221	247	179	136	94
Line Number	26											
DN	203	189	187	181	175	187	188	184	187	179	174	170
DN Stretched	362	247	230	179	128	230	234	204	226	158	119	81
Line Number	27											
DN	205	192	186	178	170	184	185	182	184	176	172	168
DN Stretched	383	273	221	153	85	204	213	187	204	136	102	68

Immediately the graph shows us a revised DN range from about 80 up to 390, which offers us a lot more optical variation, around 60% of the possible contrast range when these numbers are imaged. The above sequence of events created a Reduced Data Record (RDR) of each image (there were around 7,000 to be processed). This could be stored on tape and processed locally or by external agencies (e.g. Univerities) who were looking at the science. From the RDR further processes such as high pass filtering and any additional useful contrast stretching could be performed.

2. removal any geometric distortion, normalise an image to remove the effects of known

errors in the equipment. To further facilitate this a grid of Reseau Marks were laid over the faceplates of the television cameras that could be used as geometric reference points. These were scaled so that the marks at the edge were gradually closer to compensate for the predicted optical distortion. The reseau marks created a shadow on the image DN=0, which would need correcting out as per the previous method.

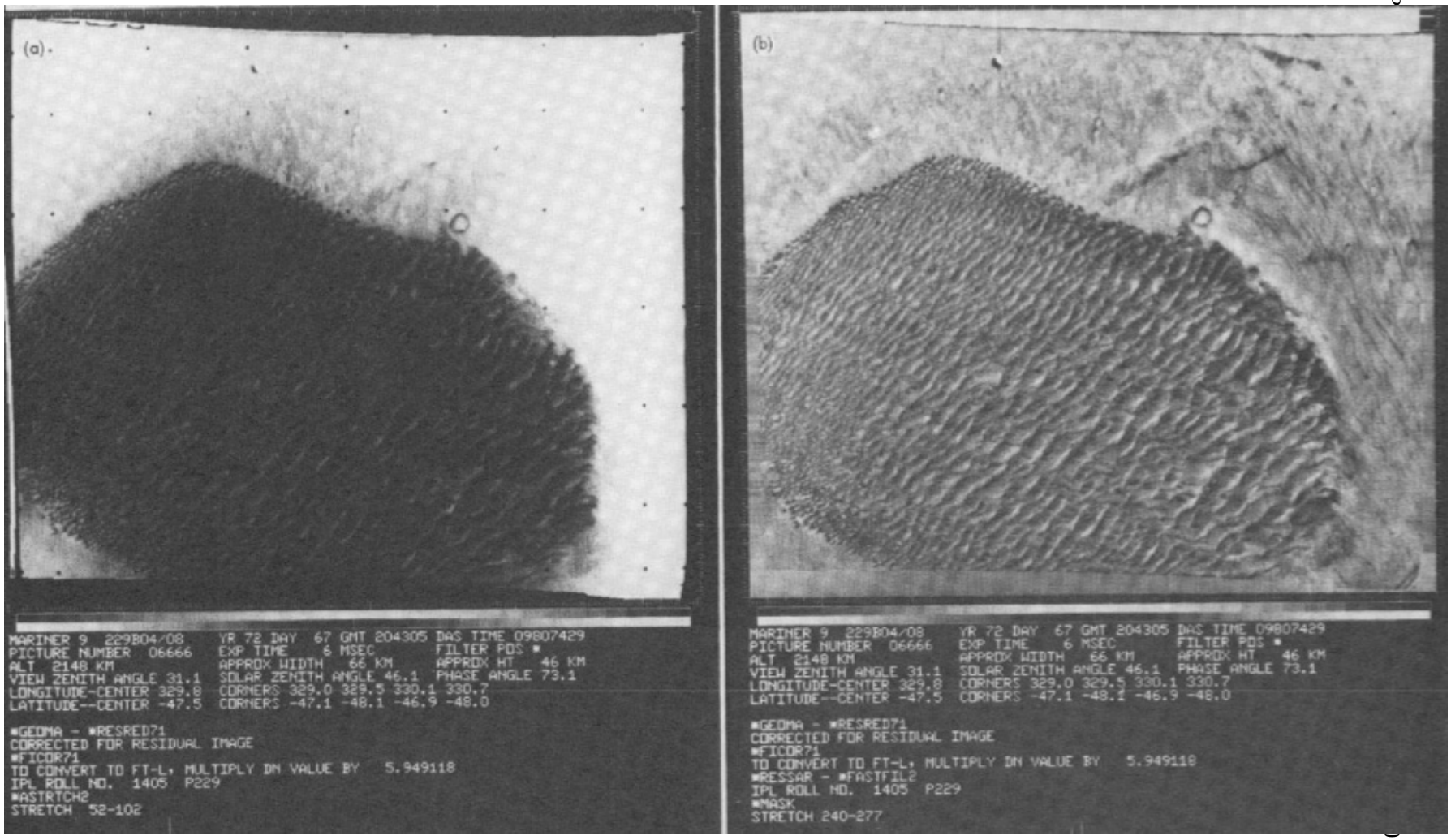
3. residual image reduction, it is possible that with a vidicon previous illumination data has not fully reset, leaving a "ghost image", so the new data is superimposed above leading to spuriously increased lunimosity values, another form of spike. On a wide angle camera image this might be wrongly interpreted as a crater. Many images were often taken of the same region, so there was some overlap to adjacent images, it was possible to compare and verify any errors.

4. reduction of photometric distortion. The process of removing photometric and geometric distortions from television data using camera calibration information.

Each RDR has a unique header and contains spaceship id (Mariner 9, Camera ID) Year 72, Day number, time GMT, Picture Number, Exposure Time, Filter Position, Altitude, Image Width, Image Height, View Zenith Angle, Solar Zenith Angle, Phase Angle, Logitude Center and Corner info, Latitude Centre and Corner info. Plus any processing history info for that record. At an altitude of 1,659Km the image of the wide angle might typically cover an area of 398 by 340Km, for this camera a pixel therefore represents an area with sides of magnitude around 500metres. By contrast the area covered by the narrow camera, from an altitude of 2,140Km, may be in the order of 70 by 50Km therefore each pixel may represent something around 70 metres per side. The obliqueness of the camera angle needs to be considered and the altitude varies considerably with the ellipticity of the orbit. The diagram below, from some latet post processing, to re-align for discovered polar changes, shows clearly the verying obliqueness of the wide angle camera labelled with even numbers, the fact the size dimishes is evidence of increasing orbital altitude. The narrow angle camera shots are shown by a number of much smaller odd numbered squares. The axes of the diagram are longitude and latitide, from the longitude shown on the x axis we can infer that this is for nearly half an orbit, so since the orbital period was about 12 hours this is for a sequence approaching 6 hours.

The diagram below shows another example of the effects of contrast stretch on the same image. To the left the stretch has taken place in a range of 52-102 and to the right from 240-277. The detail within the notation below the images is also very interesting.

Figure 8.8: Effects of contrast stretching



As the camera angles changed through orbit the image was rarely a proper rectangle, or consistent size. In order to merge all the images to eventually produce a full planetary map a lot of normalisation and stretching to consistent shapes to map into latitute and longitude coordinates was required. To aid this a virtual "planetary net" was used, into which a mosaic of separate images were cut, overlaid and processed.

Figure 8.9: Obliqueness differences during part of an orbit

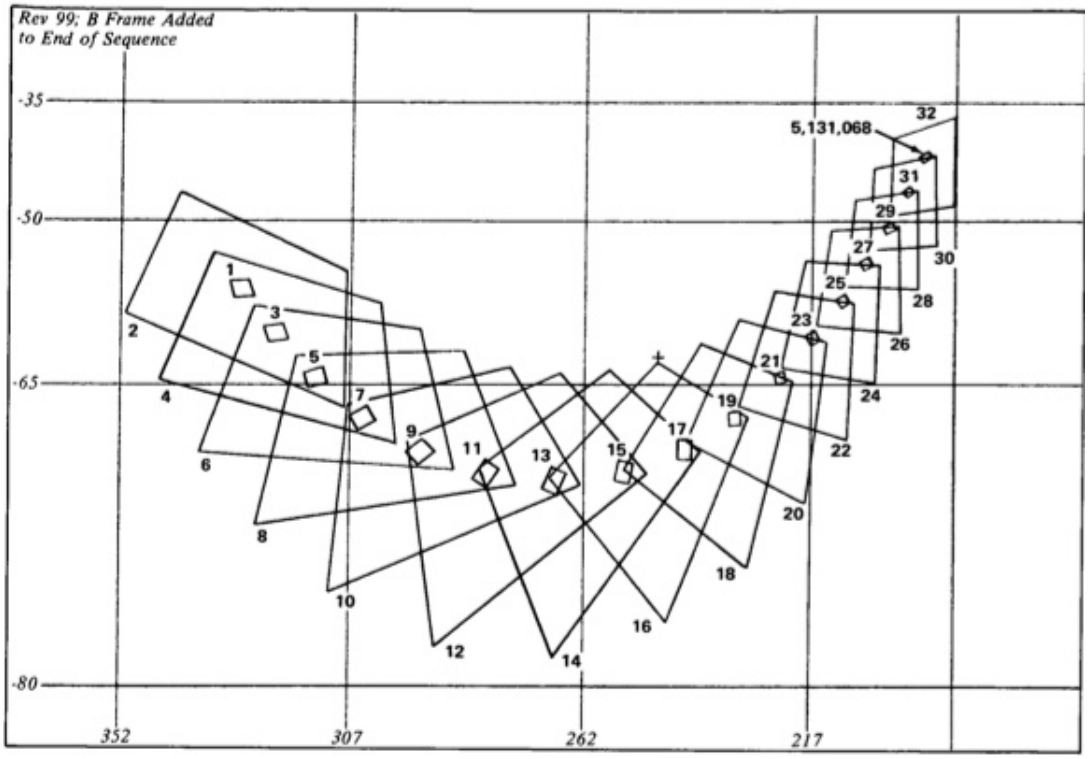
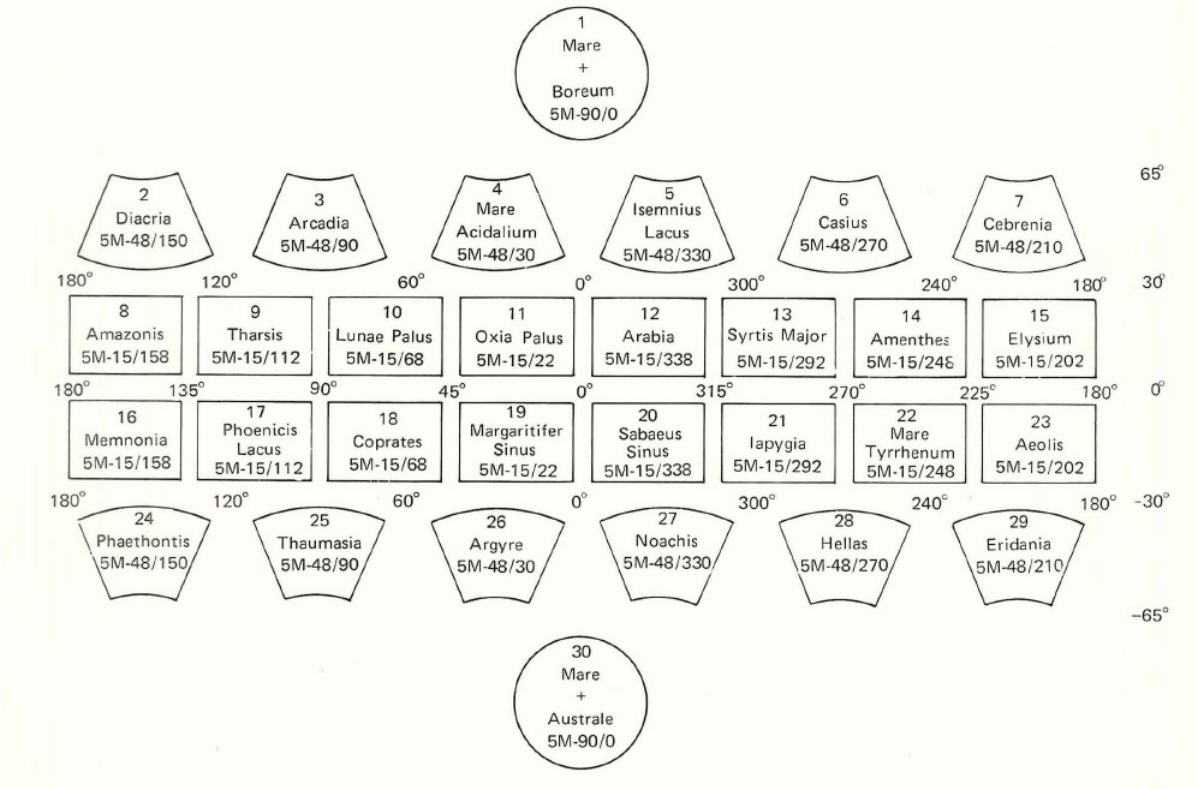
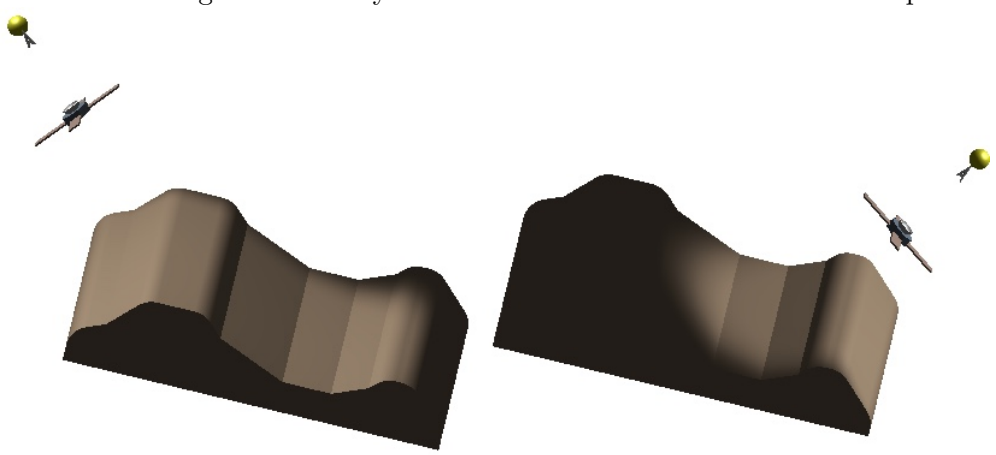


Figure 8.10: Idea of a net from which a full planetary map was created



The diagram below illustrates the effect of imaging the same terrain from different camera angles and time as the relative position of the Sun changes. The resultant DNs from each image would be vastly different.

Figure 8.11: Shading and intensity variations from different camera & Sun positions



Putting together all 7,300 images to piece together a map of Mars was a major project in its own right. This short work of mine cannot even begin to do this topic justice. The more serious reader should look at "Technical Memorandum 33-585 Volume I Experiment Design and Picture Data James A. Cutts" which will be found on NASA's ntrs server. Many areas were designated for more intensive imaging as possible Viking lander sites, so it was essential that the topographical view could be as accurate as possible. Volume IV of the project final report (32-1550) offers a very full set of images plus coverage of the complex mathematics that were used to build up topographical details, as illustrated below.

Use of techniques called Analytic Photogrammetry eventually enabled very impressive contour maps to be built as below, all from a set of data numbers!

Figure 8.12: Contour map

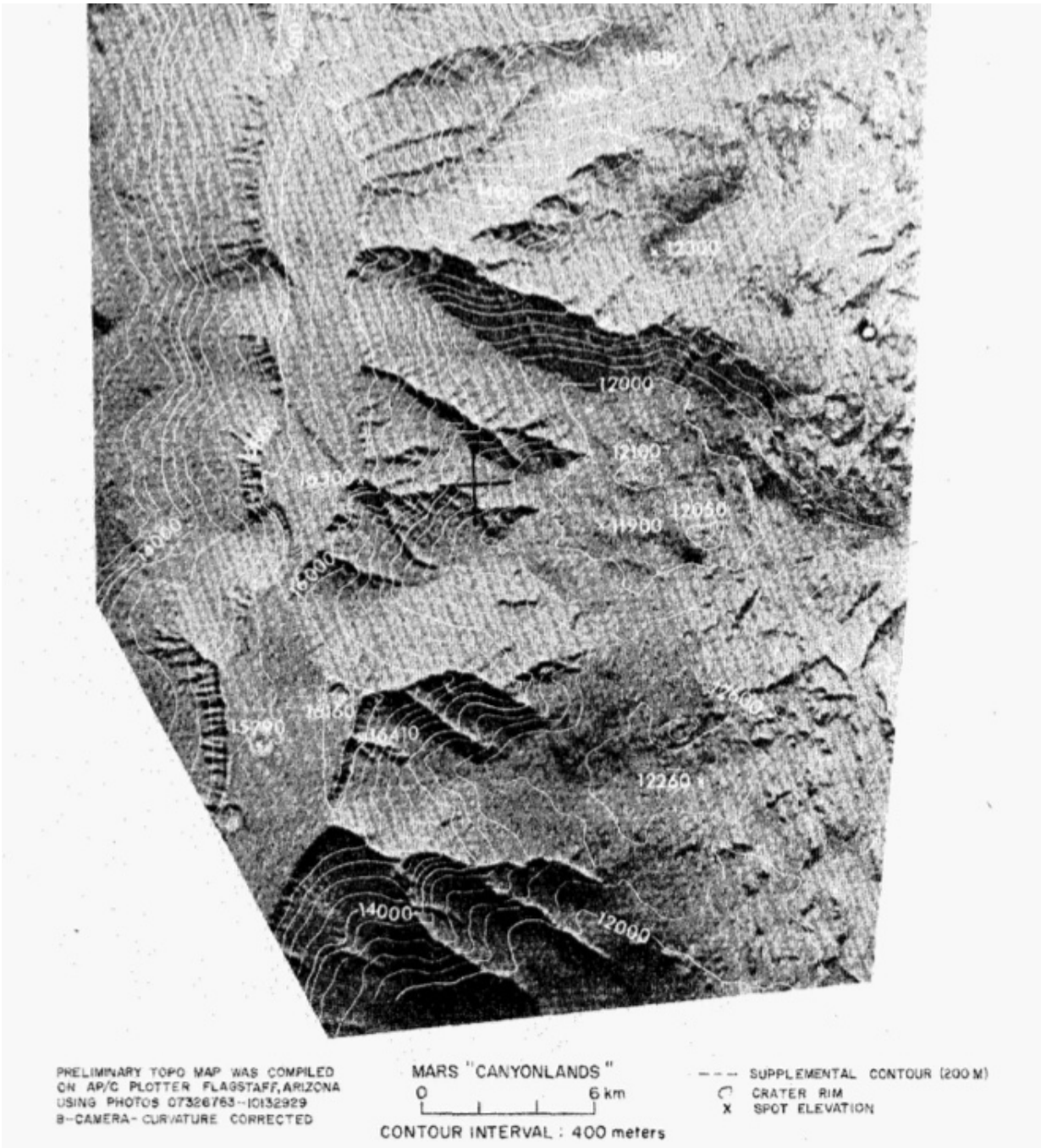


Fig. XXXXI-4. Photo map of canyonland (original scale, 1:100,000) constructed from narrow-angle pictures MTVS 4191-42, DAS 07326763 and MTVS 4273-11, DAS 10132929. Photo base was rectified picture MTVS 4191-42, DAS 07326763.

This highly processed image was published by the University of Texas around 1973. It is about a quarter of the full image and represents approximately a quarter of a million square miles

Figure 8.13: Coprates Canyon

9 Summary

The end of the mission is described in volume IV of the project final report (32-1550) as follows: *Near the conclusion of end of track on October 25, there was a critically low gas supply in the spacecraft. Therefore, for the first time, the Mariner 9 downlink was turned off by ground command. This precautionary procedure was taken so that, if the spacecraft had a malfunction and could not be commanded, the Mariner 9 radio signal would not interfere with future programs. Early on October 26, Mariner 9 responded to a command to turn the radio signal back on. The plans were revised so only two recorder tracks of science data would be played back, instead of four tracks, to conserve gas. The spacecraft radio was again commanded off in preparation for the high gain antenna maneuver on October 27. Early on October 27, commands were again sent to turn the Mariner 9 transmitter on; the spacecraft responded. Telemetry analysis showed that the spacecraft was extremely low on gas, and the limit cycle was not normal. The one remaining gas bottle was below regulation pressure. A meeting was held to determine whether or not the high gain antenna maneuver should be attempted. There were, at this time, 15 pictures on the tape. The decision was to proceed with the maneuver. The spacecraft performed a zero roll turn as programmed. When it was time to exercise the yaw turn, the spacecraft took approximately 2 min to gain near normal speed for the turn. However, during the activity, the gas was exhausted, and the turn could not be stopped. The spacecraft continued to yaw as the low gain antenna approached the predicted null, and the last data from the Mariner 9 spacecraft were received at 17:41:10 GMT. Procedures were initiated immediately to turn the spacecraft off. The spacecraft then entered Earth occultation, which prevented further attempts to enter commands. After the spacecraft exited occultation, it was observed that the spacecraft signal was being momentarily locked every 51 min, which was in close agreement with the tumble rate prediction of attitude-control personnel. With that knowledge, a timed uplink sweep was performed to acquire the spacecraft and to obtain a command lock while the low gain antenna was pointed toward Earth. A series of commands was sent to turn the transmitter off; the last command was sent at 22:10:00 GMT. The last signal from Mariner 9 was received at 22:31:00 GMT. The end of mission operations and the end of a very successful Mariner 9 mission, was declared at 23:41:00 GMT on October 27, 1972, at the end of track by DSS 14. Mariner 9 operated for a total of 515 days, 19 hr, and 18 min.* An interesting statistic, is that before the spacecraft powered down it had transmitted 5.03×10^{10} bits of information of which over 80% was from the camera system, in today's language that is about 50.3 Gigabits or a bit more than 6 Gigabytes which would all fit quite comfortably on your smartphone.

Inevitably in a mission of this complexity, that required using many leading edge components, there were many problems. However overall the huge success cannot be negated. Apart from providing a workable topological map of Mars and being the first spacecraft to orbit another planet it also provided some test results which would pave the way for the Voyager missions to Jupiter, Saturn and beyond and also the next generation of Mars exploration vehicles, the Viking landers, whose sites were chosen based on the imagery from the mission.

For people who wish to know more about the planet itself Volume V of the Mariner Mars 1971 Project Final Report 32-1550 is a good starting point. This offers a comprehensive overview of the following experimental disciplines: Celestial Mechanics , S-Band Occultation, Infrared Radiometry, Infrared Spectroscopy, Ultraviolet Spectrometer, Television, Surface Properties, Volatiles and Atmospheric Phenomena.

I hope you have found this mission precis of interest. If you want to know more about the fundamentals of space technology, which is of a moderate scientific and mathematical nature then I do have some e-books, details follow, which are available on Amazon.

Please click on the required image, if you do wish to see the detail, you will need to navigate from this UK site to your national Amazon site, they are free to Amazon Prime subscribers.

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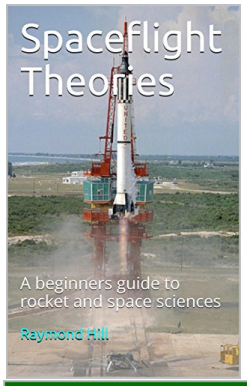

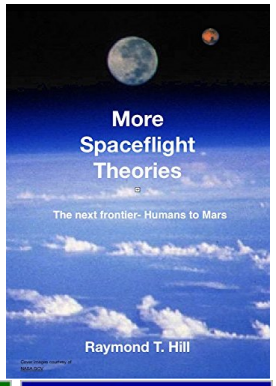

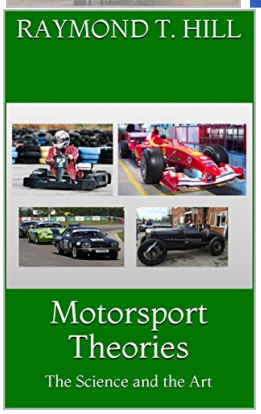

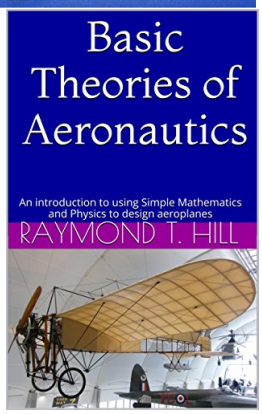

10 Appendix 1 - Bibliography

Some specifically useful source texts from `https://ntrs.nasa.gov` which I used were:

1. NASA Mariner Mars 1971 Project Final Report : 32-1550 , there are 5 volumes.

2. NASA Mariner-Mars 1964 Final Project Report SP-139

3. JPL Technical Memorandum 33-523

4. Mariner Mars 1971 Orbiter Study Report Technical Document 610-31 1 February 1968.

5. Technical Memorandum 33-681 Mariner Mars 1971 Attitude Control Subsystem.

6. Space Programs Summary 37-66, Vol. II The Deep Space Network For the Period September 1 to October 31, 1970

7. Technical Memorandum 33-503 Development and Testing of the S-Band Antenna Subsystem for the Mariner Mars Spacecraft

8. JPL Publication 95-20 The Evolution Of Technology In The Deep Space Network

9. Technical Memorandum 33-505 Development and Testing of the Television Instrument for the Mariner Mars 1971 Spacecraft

10. Technical Report 32-1460 Volume 1 Mariner Mars 1969 Final Project Report

11. Technical Memorandum 33-535 Telecommunications System Design for the Mariner Mars 1971 Spacecraft

12. SP-337 The New Mars, the discoveries of Mariner 9 (this includes an excellent summary of the scientific results)

13. Technical Memorandum 33-574 Development and In-Fliqht Performance of the Mariner 9 Spacecraft Propulsion System

14. Technical Memorandum 33-552 Development and Testing of the Propulsion Subsystem for the Mariner Mars 1971 Spacecraft

15. Technical Memorandum 33-100 Volume 4 Part C Earth-Mars Trajectories, 1971

16. Technical Memorandum 33-628 A User's Guide to the Mariner 9 Television Reduced Data Record

Other useful sources:

1. Satellite Communications by Dennis Roddy

2. `www.radio-electronics.com`