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DESIGN OF AN ELECTRICALLY-PROPELLED ASTEROID RENDEZVOUS PROBE LAUNCHED AS A SECONDARY PAYLOAD

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Abstract

This paper summarises a design study undertaken as a final year project for the author's B.Eng. in Aerospace Systems Engineering. A spacecraft design is outlined for a vehicle to perform a rendezvous with a Near Earth Asteroid after being launched as a secondary payload into Geosynchronous Transfer Orbit. The 380 kg (dry) spacecraft would use a bipropellant chemical propulsion system to manoeuvre in, and escape from, Earth orbit. Interplanetary manoeuvring would be accomplished with an ion propulsion system.

Various mission and system design aspects are described.

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Introduction

The objective of the study was to determine what kind of mission could be accomplished with a modest spacecraft using electric propulsion launched as a secondary payload. In particular it examined a Near Earth Asteroid Rendezvous (NEAR) mission. Using a variety of advanced technologies and techniques, a highly sophisticated and capable spacecraft can be constructed of a size suitable for launch as a secondary payload.

This paper summarises the results of the study report (ref.1) which is 155 pages long. There is insufficient space here to repeat the details of calculation methods and other background information, so if applying these results to other missions, take care!

As an exercise, the mission was based around a number of components and technologies under development or manufactured in the UK.

Launch

The Ariane launch vehicle was selected as a baseline for a number of reasons: it is European, and therefore perhaps the least politically-sensitive of available boosters; also it is launched frequently (about 10 times/year), and has a long history of collaboration with secondary payloads (AMSAT OSCAR 10,13; UoSATs 3,4; Microsats; Viking etc.)

The spacecraft is designed as a 'satellite porteur' (ref.2) - built around the 1920mm/937mm launch adaptor (just as the AMSAT Phase 4 spacecraft.) A cylindrical volume allocation (2.26m dia by 1m high) was assumed. No fixed launch mass limit was taken, as this would depend on the primary passenger on the launcher. A dry mass in the region 300-350 kg was aimed for, making the wet mass around 700-800 kg. This mass can be accommodated by uprating the Ariane booster, which is available in a number of versions with GTO payload masses from 1900kg to 4200kg. (see figures 1,2,3)

Mission - Earth Orbit Phase

The mission analysis is effectively divided into two main parts: the Earth orbit phase and the interplanetary cruise. Initially, the Earth orbit phase was regarded as a formality, the intention being to escape immediately using a solid motor into a circular heliocentric orbit of radius 1 AU and use the electric propulsion system from there. However, when the effects of using a slightly oversized motor were investigated, it was found that great improvements in V-infinity (the hyperbolic departure velocity) can be obtained by slightly increasing the magnitude of the escape burn due to a non-linearity in the celestial mechanics (see figure 4.)

However, for the excess V-infinity to be useful, it must be in the correct direction. This necessitates manoeuvring in Earth orbit - hence requiring a restartable propulsion system (i.e. not a solid.) Additionally, since the launch date is specified by the primary passenger on the launch vehicle, the spacecraft may have to wait in earth orbit for some months before departure. GTO is a most unfavourable orbit in which to spend such a period - aerodynamic drag, torques and heating, and most importantly, the high radiation dose all degrade the mission performance. Therefore the craft manoeuvres into an expanded orbit (say 89000x500 km) for the phasing period as suggested in (ref.3.) This orbit reduces the radiation dose considerably: it also breaks the escape burn into two smaller burns so that a smaller motor can be used. The perigee is raised slightly for safety but kept fairly low to maximise the V-infinity performance of the escape burn. An inclination change manoeuvre would be performed just after entering the waiting orbit to control apsidal precession and nodal regression of the orbit. Inclination would be adjusted again a few orbits before departure to orient the departure direction.

The Earth orbit manoeuvres are summarised in figure 5.

Mission - Deep Space Phase

A number of potential targets were examined, selected by eye from the TRIAD (Tucson Revised Index of Asteroid Data) file (ref.4) and other asteroid mission studies (refs.5,6,7). Delta-V requirements were evaluated assuming Hohmann transfers (not strictly correct given that some of the manoeuvres are conducted with a low-thrust propulsion system, but accurate enough for the purpose of this study.)

Thus two velocity changes, delta-V-one and -two, are required for the mission. These can be traded off against each other by varying the fraction of the inclination change that is performed at rendezvous (parameter alpha.) The delta-V requirements for some of the easier targets are plotted in figure 6 with the effect of varying alpha shown; actual figures for alpha=0.7 are given in table 1.

The delta-V-one requirement is met by the hyperbolic departure velocity, plus a velocity increment from the ion propulsion system if necessary just after departure; delta-V-two is met by the ion propulsion system in deep space.

The nominal targets selected are mildly inclined (less than 10 degrees to the ecliptic), and have aphelia 1.8AU or closer. The three principal targets are Eros, Anteros and Bacchus.

The capabilities of a combined chemical-electric and a chemical-only spacecraft of the same launch mass are shown in figure 7. By comparing figures 6 and 7, it is seen that a chemical-only vehicle is only just capable of performing a rendezvous mission with the easiest-reached target, 1982 HR. A chemical-electric vehicle, as proposed here, has a much larger performance envelope and allows far more flexibility in target selection. Given the launch as a secondary payload, this flexibility is vital.

An all-electric vehicle would be capable (at first inspection) of performing missions of even wider scope. However, in this case, the thruster lifetime, rather than propellant supply, becomes the limiting parameter. Escape from GTO using electric propulsion would be a nightmare from an attitude control point of view, would incur substantial radiation damage, and would take over one year (using the two UK-10 thrusters with 2kW of available power.) Further, since a slow, spiral escape leaves the craft with a hyperbolic departure velocity of zero, the actual propellant mass saving for a typical mission (with delta-v-one equal to about 4 km/s) will be very small.

Accordingly, the combined chemical/electric mission is retained.

Configuration

The configuration drivers are principally the volumetric constraint imposed by the launch vehicle, and the cruise pointing requirements of the ion thrusters, solar arrays and communications antenna. A number of possible configurations were tried but rejected on the grounds that they required complicated mechanisms.

The configuration that finally evolved has a body-fixed payload, chemical motor and communications antenna. The vehicle is an octagonal prism, sized to fit the envelope in figure 2. A cruciform solar array is deployed by unfolding four four-panel 'wings' from the sides of the probe. These arrays are body-fixed and do not articulate about any rotating joints. (In some respects, the probe resembles the Mariner Mars probes.) The only mechanisms used are booms for two of the instruments, gimbals for the ion thrusters, and one-shot mechanisms to deploy the solar arrays and antenna feed.

Chemical Propulsion

Initially the concept of the mission was to use the electric propulsion system to the full. Since a spiral escape from GTO is not easy, it was initially proposed to use a solid motor to escape from the Earth. However, when the escape manoeuvre was considered in more detail, bearing in mind that the launch date and orbit parameters would be dictated by the primary passenger, a more flexible propulsion system, capable of multiple burns, was required. This necessitated a storable bipropellant system: this comprises a 500N motor, (the Leros 1 engine manufactured by Royal Ordnance) with four titanium 100 litre tanks with capillary propellant management devices (manufactured in the Uk by Dowty, under licence from Marietta. The Leros motor uses mixed oxides of nitrogen and ordinary hydrazine (rather than the more usual monomethyl hydrazine.) thus the hydrazine can also be used for the attitude control thrusters in a monopropellant mode without having to carry separate tankage and pressurant.

The nozzle of the 500N motor projects from the centre of the 937mm adaptor ring on the nominal anti-sun face of the spacecraft. The volume allocation for the nozzle would have to be negotiated with the launch authority. The tankage is mounted inside the adaptor ring, such that the chemical propulsion system (except for the attitude control thrusters) can be kept separate from the rest of the vehicle until final integration. The modular construction of the probe is shown in figure 8.

Analysis indicates that for the spacecraft mass considered here, 500 N is adequate thrust to provide sufficient delta-v over the permitted thrust arc. Lower thrust levels would extend the required arc to the point where pointing losses during the spin-stabilised burn would degrade the mission performance unacceptably.

Electric Propulsion

The electric propulsion system is built around two UK-10 xenon ion thrusters (ref.8). In order to eliminate disturbance torques generated by thruster misalignment and movement of the spacecraft centre of mass during the mission the two thrusters are mounted on gimbals which allow the thrust vector of each to be tilted +/- 4.2 degrees. Additionally thrust vectoring over a wider range (+/- 20 degrees) is possible about one axis by differentially throttling the two thrusters - this allows optimal thrusting while maintaining earth-pointing. This configuration also allows at least a minimal mission to be accomplished even if one of the thrusters fails.

The thrusters are run on Xenon stored near its critical point: the 50kg propellant capacity requires 100 litres of storage: unfortunately the tanks used for the bipropellant do not have the pressure rating required (and indeed another tank of this size would be difficult to fit in the available volume) so the Xenon would be stored in two or four smaller tanks.

The ion thrusters are run with an acceleration voltage of 940V - this gives a specific impulse of 3160 seconds. The thrusters can be throttled over a range 10-25 mN each, with corresponding power consumption of 275-600 W. Some of this power is dissipated in the thruster power conditioning units, which have an area on the anti-sun face of the craft to radiate away this waste heat.

Power

The power demands of the spacecraft are somewhat higher than usual for its size - this is clearly due to the power required by the ion propulsion system. To simplify construction (and to avoid potential elasticity problems - c.f. Hubble Space Telescope) the solar arrays are of the rigid foldout type. This also allows the possibility of withstanding shocks due to landing on the asteroid surface if this is to be attempted.

The four arrays are each four panels long, each panel equal in area to one of the eight sides of the spacecraft. Taking a packing fraction of 0.8, this gives room for 20400 2x2cm cells. Gallium Arsenide cells were selected for their higher efficiency and more importantly for their higher radiation tolerance.

The arrays provide 1940W of power at BOL and 1AU: allowing for pointing conditioning losses and a 10% margin, this leaves 1400W, permitting both ion thrusters to be run at full power continuously. At 1.8 AU, taken to be the design maximum solar distance, the available power is about 475W - sufficient to run one ion thruster at low power and run the other spacecraft systems.

In Earth orbit, the arrays are folded against the sides of the craft, with only the outermost panel exposed to the sun. Depending on solar aspect angle, up to 140W can be generated. However, power requirements in Earth orbit are low.

When the power storage requirements (about 450 W-hours) are considered, driven mainly by eclipse duration in orbit about the earth or the asteroid, it is found that Nickel-Cadmium batteries are uncomfortably massive, so Nickel-Hydrogen cells are selected instead. The cells are arranged in two batteries of twelve cells. Each battery has a DC-DC converter to boost the battery voltage (nominally 14V) to the bus voltage (28V). Should any cell fail, it can be switched out of circuit and the battery will continue to function: the lower battery voltage can be accommodated by the DC-DC converter. This combination of redundancy (2 batteries) and the graceful degradation of the batteries themselves offers extremely high reliability.

Since substantial excess power is developed by the arrays if the ion thrusters are not firing, a shunt dissipator is required: this is situated on the anti-sun face of the spacecraft.

Communications

The targets examined for the mission have aphelia up to 1.8 Astronomical Units, so in principle, the Earth could be up to 2.8 AU distant from the spacecraft. However, this would make the line-of-sight pass very close to the sun resulting in a poor link. An arbitrary limit on the sightline-to-sun angle in the ecliptic plane of 20 degrees was set: this fixes the maximum design communications range at 2.5 AU.

The primary communications link uses a 1.65m parabolic S-band antenna, sited inside the 1920mm adaptor ring on the nominal sunside of the craft. A feed is mounted on an arm which swings into position (using a one-shot mechanism) after the escape burn. During the main cruise phase, the sun-spacecraft-earth phase angle is found to be less than about 30 degrees, so that the craft can be held Earth-pointing with a 10% or lesss cosine loss in solar power generation.

Using a 10W transmitter and convolutional signal encoding, the link can support 400 bits per second downlink at a distance of 2.5 AU. Link performance is increased at shorter ranges, higher transmitter power and with more elaborate coding. The nominal link budget is shown in table 2.

During the Earth orbit phase an omnidirectional antenna is used.

Immediately after Earth departure, the sun-probe-earth angle is found to be much greater than 30 degrees: the spacecraft is then sun-pointed and communications continue with the omni antenna. The omni antenna is capable of supporting 400 bps until the Earth-probe distance exceeds 0.1 AU, by which time the phase angle has reduced to a point where the high-gain dish can be used without incurring excessive solar array mispointing losses.

Attitude Control

During the Earth orbit phase, the spacecraft is spin-stabilised. Even with the solar arrays folded up against the sides of the craft, the moments of inertia are suited for spin stabilisation, which serves to hold the attitude rigid during motor firings.

Attitude determination is by sun and earth sensors in this phase. In the event of a failure of any of these components, some additional attitude information could be provided by the star sensors (if the spin rate, probably 10-15 rpm, is low enough for the sensors to cope) and a magnetometer carried as payload. Spin rate adjustments and slew manoeuvres are performed by monopropellant hydrazine thrusters.

After departure, the craft is despun, the arrays are deployed and sun-lock achieved. Thereafter the craft is three-axis stabilised.

The dominant torque in the cruise phase turns out to be the ion thrusters. While the hydrazine requirement to combat this torque is not excessive (although large) the number of thruster firings would exceed the rated life of the thrusters, so gimbals have to be used to orient the ion propulsion thrust vector to eliminate disturbance torques. The gimbals are based around an existing antenna pointing mechanism (by BADG) with a mass of 4.2 kg and a power consumption of 8W (ref.9). Remember that gimballing will only be necessary when the ion thrusters are firing, when there will be plenty of power available.

The use of reaction wheels was investigated: these would eliminate propellant consumption during limit-cycling in cruise and slewing for imaging (the payload is body-fixed) at the target asteroid. Whether a reaction wheel suite is less massive than the propellant otherwise required depends on the number of slews required (the limit-cycling fuel requirement is very low) - break-even occurs at about 2000 slews. A figure of 2500 slews was taken, making wheels slightly better in mass terms: however, this area needs more careful analysis and whether the 5kg mass saving is worth the additional complexity is debatable. Nominally, then, no wheels are carried.

Attitude determination during cruise and rendezvous is accomplished with star sensors (based on CCD cameras- ref.10) and sun sensors. Additional information, in the event of sensor blinding, is provided by attitude references which could be based on a number of low-mass technologies (e.g. gas gyros, fibre-optic gyros or solid-state gyros.)

It is acknowledged that the attitude control of a vehicle on this sort of mission is indeed complex and will demand substantial on-board processing capability. It is conjectured that the development effort involved would be considerable.

Payload

While the main aim of the study was to see what can be achieved with available technology, clearly the mission in practice will be driven to a large extent by scientific objectives. These are to determine

1. Global characteristics: shape, size, mass distribution, rotation axis and period
2. Surface morphology: regolith structure and depth, cratering record etc.
3. Elemental and Mineral Composition
4. Dust and Plasma environment; magnetic field if any

A survey of previous asteroid mission proposals (refs. 6,11,12,13,14) was undertaken to see what kind of instruments are carried to meet these objectives. The instrument payload selected as baseline resembles those of other missions:

1. Optical cameras based on CCD imagers, probably fairly rudimentary by planetary exploration standards. The exact design of the optics was beyond the scope of this study and would depend on the final mission plan. Note that while providing highly important science data from orbit about the asteroid, the cameras would also be used to assist in the navigation of the

spacecraft just prior to rendezvous. The cameras are fixed to the side of the spacecraft, which is slewed round to aim them.

2. Gamma Ray Spectrometer - to identify the elemental composition of the asteroid: resolution is of the order of the spacecraft-asteroid distance. The instrument is deployed at the end of a boom to reduce contamination of the measurements by the spacecraft material itself.

3. Dust Detector - to investigate the dust particle ditribution about the asteroid. The detectors could use any spare 'real estate' on the spacecraft surface.

4. Magnetometer - to investigate any possible remnant magnetism. This instrument too would be mounted at the end of a boom to minimise contamination.

5. Imaging Infrared Spectrometer - to identify surface mineral distribution. This instrument is optional, subject to available mass, power and data budgets. Like the cameras, the instrument is body fixed.

6. Secondary Ion Mass Spectrometer - to study surface mineral and elemental composition. An experiment of this type was flown (but not used before the craft was lost) on the Soviet Phobos 2 mission: an ion beam is used to sputter material from the asteroid surface for analysis on-board in a mass spectrometer. Here an ion thruster would be used to generate the primary ion beam (ref.15). This instrument, too, is optional.

The science objectives met by the various instruments are summarised in table 3.

Although international collaboration is attractive, most of these instruments could be sourced in the UK. In particular, the teams at Kent (Dust Detectors) and Imperial College, London (Magnetometers) have excellent reputations.

Groundstation

Overall programme costs can soar if missions are not managed correctly. It is proposed that most spacecraft operations be conducted autonomously, and that only one groundstation is used. Use of ESTRACK or DSN is not considered in this report (except in contingencies) as they would be hideously expensive and, in the anticipated timeframe of the mission - the late 1990s - these facilities will be in great demand for the many planetary and solar-terrestrial physics programmes taking place.

A single, dedicated groundstation would be expensive to construct from scratch. However, there is a 10m antenna and a control facility at the Rutherford Appleton Laboratory in Oxfordshire (figure 9): these have been unused since the US/Dutch/UK IRAS mission. Although the dish is old, and construction work has severed the cables connecting the dish to the control room, the station could be restored to operation for about £2 million.

Conclusions

A viable asteroid rendezvous mission can be conducted with a spacecraft launched as a secondary payload. In the case examined here, a combined chemical/electric propulsion system is required to provide adequate performance: for other missions or launch orbits either all-chemical or all-electric vehicles may be better.

While the vehicle described is indeed complex and uses many new technologies, no 'magic' is required. Unlike a number of other proposals for planetary missions using small electrically-propelled vehicles (refs.3,16) the technologies and components suggested are in manufacture (or at least on the workbench) rather than being simply on paper.

While programme costs are notoriously difficult to estimate, launch as a secondary payload more than halves the price of the launch. While most components will have flown at least once before the mission takes place, the technologies are still relatively new and favourable terms could probably be negotiated. The really difficult part to estimate is the cost of operations, software development and systems integration. If the programme is managed appropriately, with universities undertaking much of the work, overall programme costs could be kept down to the point where even the UK might be able to afford the mission.

Acknowledgements

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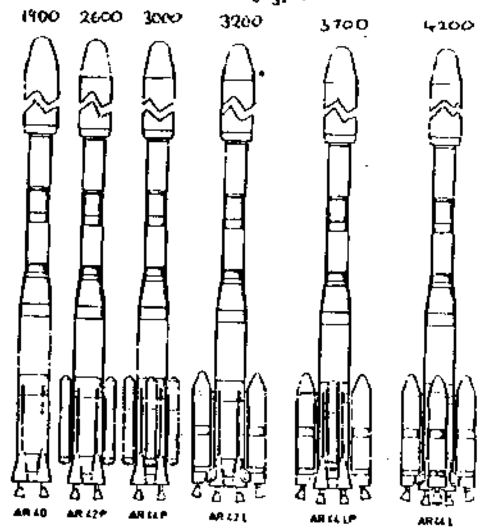
FIG : 1
THE ARIANE-4 FAMILY OF LAUNCHERS



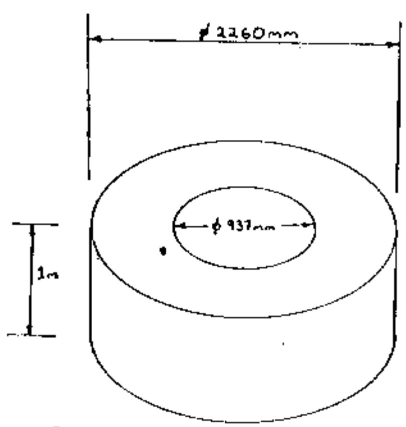


FIG 2
NOMINAL LAUNCH ENVELOPE

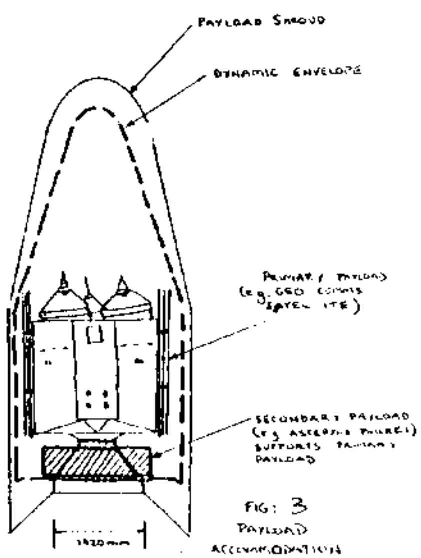
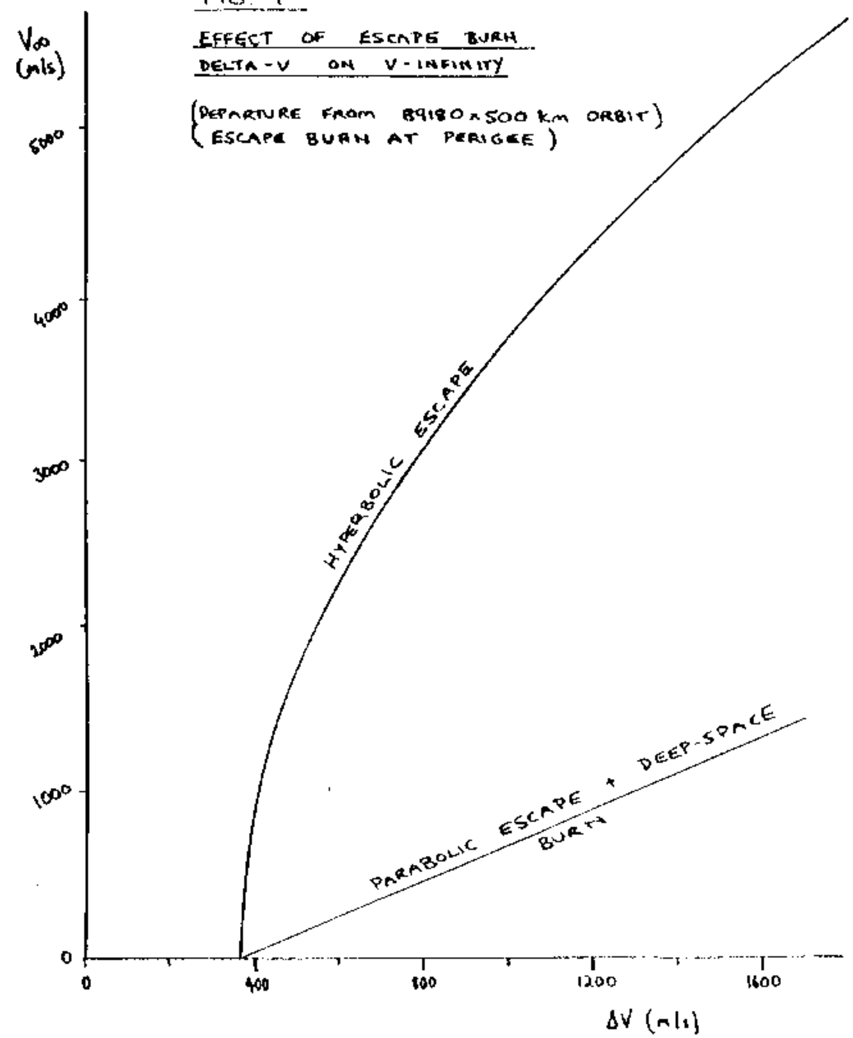


FIG: 3
PAYLOAD ACCOMMODATION

FIG. 4
EFFECT OF ESCAPE BURN DELTA-V ON V-INFINITY
(DEPARTURE FROM 89180 x 500 km ORBIT)
(ESCAPE BURN AT PERIGEE)



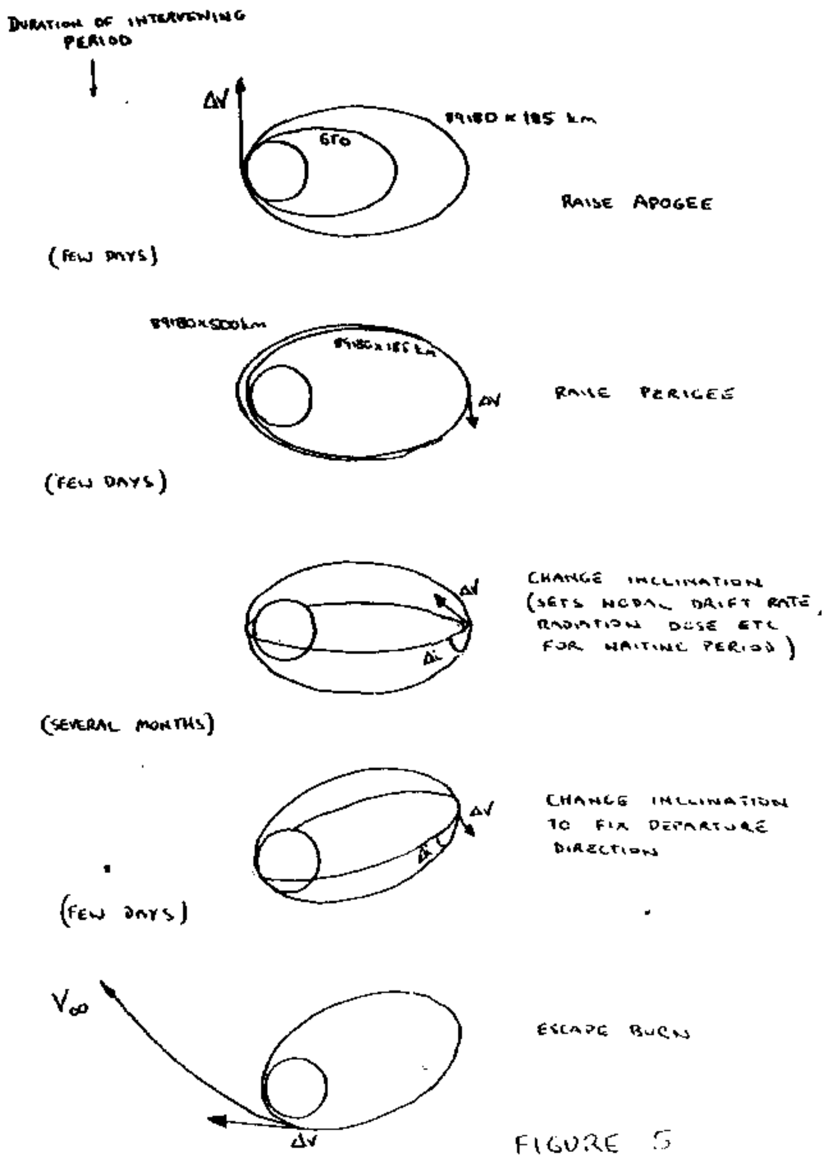


FIGURE 5

SCHEMATIC OF EARTH-ORBIT

TABLE I DELTA-V REQUIREMENTS FOR RENDEZVOUS (alpha=0.7)

No.	NAME	A	E	I	Peri	Aph	dV1	dV2	dVtot
1235	SCHORRIA	1.910	0.154	25.0	1.62	2.20	6129	6199	12328
1685	TORO	1.368	0.436	9.4	0.77	1.96	4939	2371	7310
1747	WRIGHT	1.709	0.110	21.4	1.52	1.90	5174	5857	11031
1139	ATAMI	1.947	0.255	13.1	1.45	2.44	6022	3263	9286
1627	IVAR	1.860	0.397	8.4	1.12	2.60	6199	1583	7782
1919	CLEMENCE	1.936	0.095	19.3	1.75	2.12	5493	5539	11032
1862	APOLLO	1.470	0.560	6.4	0.65	2.29	5598	2504	8102
1566	ICARUS	1.077	0.827	22.9	0.19	1.97	12082	8955	21037
2063	BACCHUS	1.077	0.349	9.4	0.70	1.45	3342	3307	6649
2100	RA-SHALOM	0.832	0.437	15.8	0.47	1.20	4500	6817	11318
1943	ANTEROS	1.430	0.256	8.7	1.06	1.80	4250	2075	6325
2061	ANZA	2.260	0.537	3.7	1.05	3.47	7383	528	7910
2608	SENECA	2.480	0.586	15.6	1.03	3.93	8409	1845	10254
433	EROS	1.460	0.223	10.8	1.13	1.79	4321	2705	7026
887	ALINDA	2.500	0.555	9.2	1.11	3.89	7983	1208	9191
1865	CERBERUS	1.080	0.467	16.1	0.58	1.58	5137	4913	10051
2102	TANTALUS	1.290	0.298	64.0	0.91	1.67	12436	14740	27176
1863	ANTINOUS	2.260	0.606	28.4	0.89	3.63	9599	3426	13025
2062	ATEN	0.966	0.183	18.9	0.79	1.14	3745	6068	9813
1221	AMOR	1.920	0.434	11.9	1.09	2.75	6659	2006	8665
2101	ADONIS	1.875	0.764	1.4	0.44	3.31	7156	3213	10368
1620	GEOGRAPHOS	1.245	0.335	13.3	0.83	1.66	4390	3286	7676
	1937 UB	1.639	0.624	6.2	0.62	2.66	6341	2454	8795
	1950 DA	1.683	0.502	12.2	0.84	2.53	6415	2163	8577
	1980 AA	1.891	0.444	4.2	1.05	2.73	6314	732	7047
	1954 XA	0.777	0.345	3.9	0.51	1.05	1044	5362	6407
	1984 QA	0.989	0.468	9.9	0.53	1.45	3691	4826	8518
	1982 FT	1.774	0.284	20.4	1.27	2.28	6229	4354	10583
	1982 XB	1.837	0.447	3.9	1.02	2.66	6179	653	6832
	1982 HR	1.210	0.323	2.7	0.82	1.60	3315	1428	4743
	1983 TF2	1.343	0.387	7.8	0.82	1.86	4492	1999	6491
	1980 PA	1.926	0.459	2.2	1.04	2.81	6418	403	6821
996	HILARITAS	3.102	0.126	0.7	2.71	3.49	7373	4276	11649
1103	SEQUOIA	1.934	0.094	17.9	1.75	2.12	5410	5265	10675
1224	FANTASIA	2.304	0.199	7.9	1.85	2.76	6401	3358	9759
1225	ARIANE	2.233	0.075	3.1	2.07	2.40	5631	3818	9449
4	VESTA	2.360	0.089	7.1	2.15	2.57	6021	4132	10153
	MERCURY	0.388	0.206	7.0	0.31	0.47	6389	12499	18888
	VENUS	0.726	0.007	3.4	0.72	0.73	2493	3094	5587
	MARS	1.590	0.093	1.9	1.44	1.74	3794	2267	6061

A - semi-major axis (AU)
E - eccentricity
I - inclination to ecliptic (deg)
Peri - Perihelion (AU)
Aph - Aphelion (AU)
dV1 - Earth Departure delta v (m/s)
dV2 - Rendezvous delta v (m/s)
dVtot - Total delta v (m/s)

70% of inclination change pe
in deep space

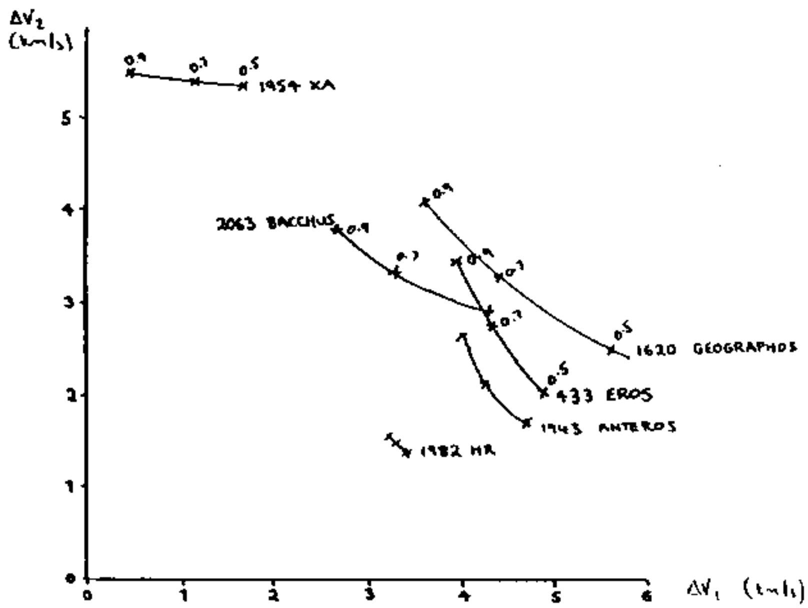


FIG 6 : ASTEROID RENDEZVOUS
ΔV REQUIREMENTS

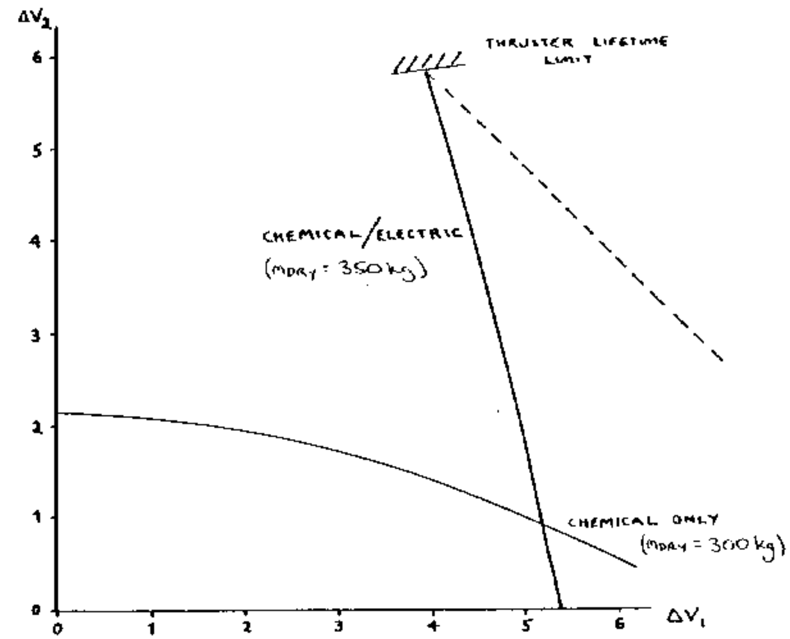
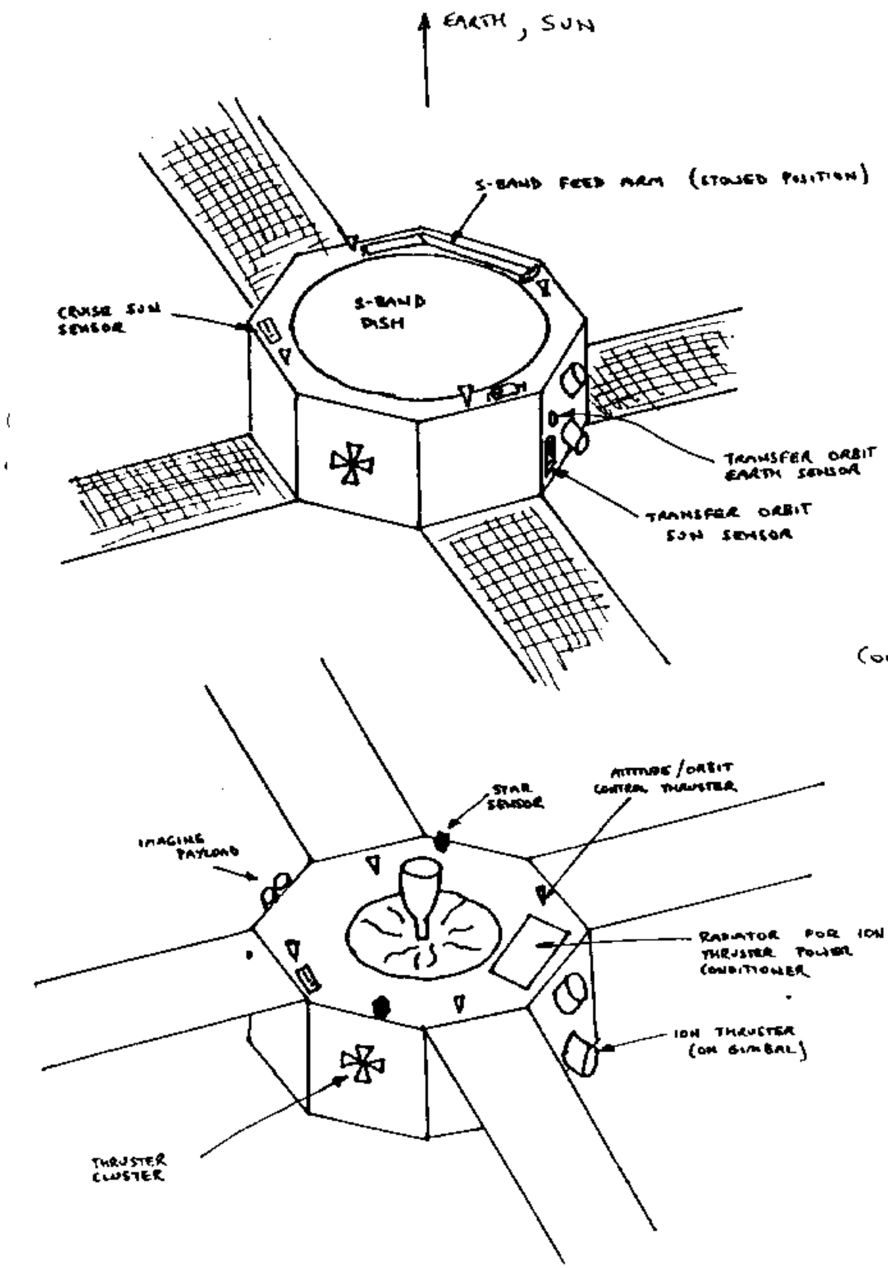


FIG 7 : MAXIMUM PROPULSION CAPABILITY
(LAUNCH MASS 700 KG)



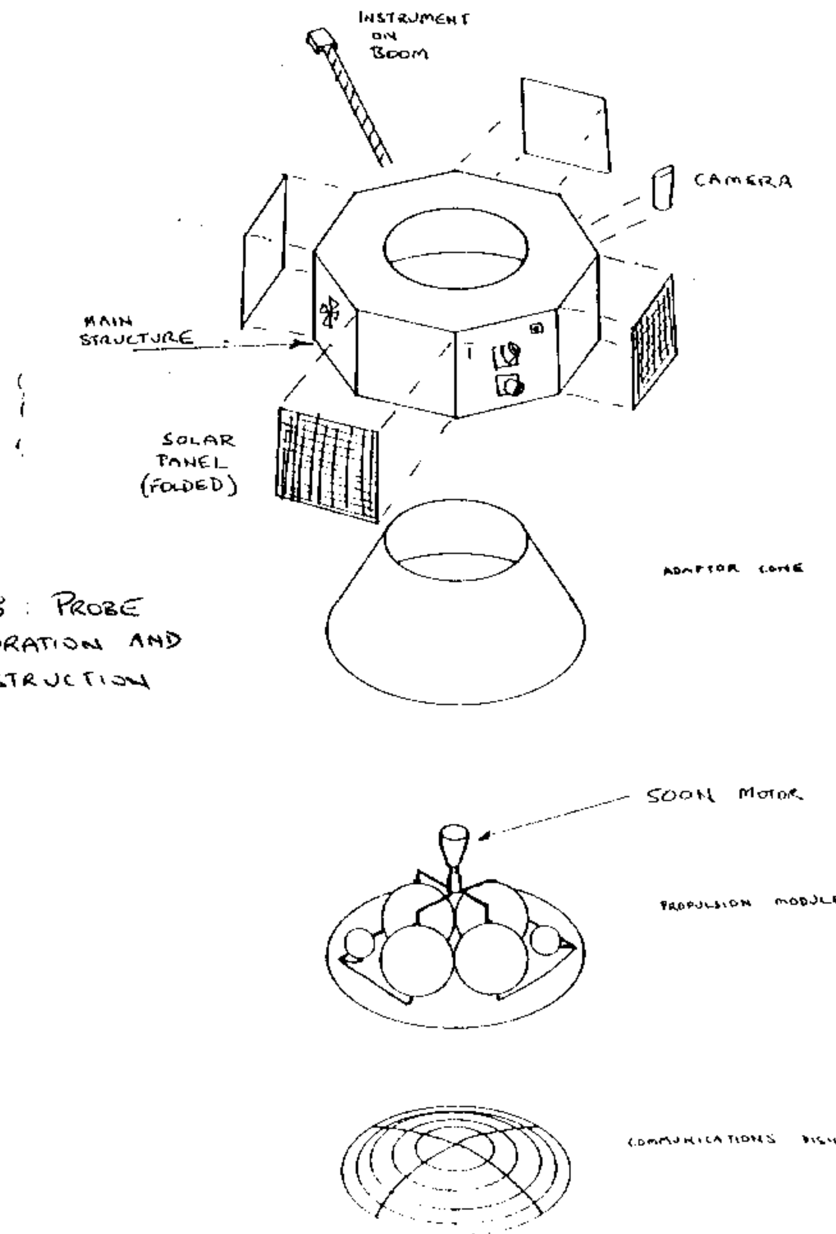


FIG 8 : PROBE CONFIGURATION AND CONSTRUCTION

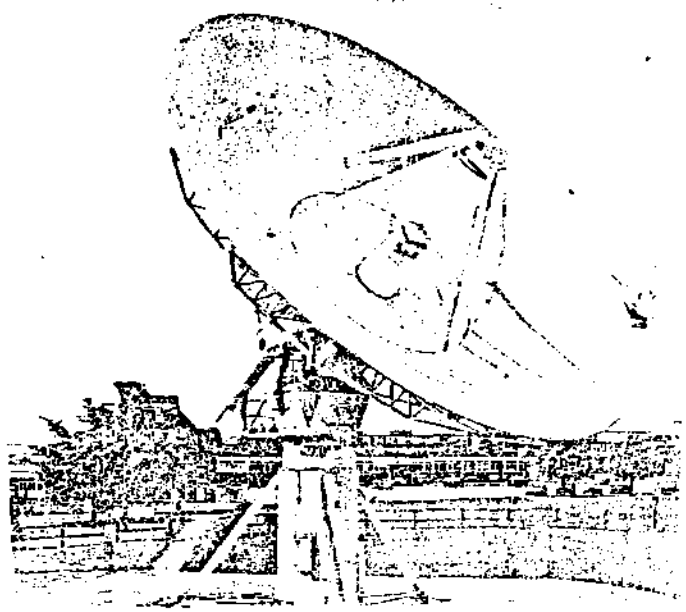


FIG 9

THE EK-IRAS 12m TRACKING ANTENNA AT RUTHERFORD APPLETON LABORATORY, CHILTON

TABLE 3

ASTEROID RENDEZVOUS PROBE - TELEMETRY LINK BUDGET

Tx Power	10 W	10.00 dBW
Circuit Loss		2.00 dB
Modulation Loss		0.10 dB
Tx Antenna Gain	1.65 m dia	26.91 dBi
Pointing Loss		3.00 dB
EIRP		33.81 dBW
Space Loss	2.5 AU	229.89 dB
Atmospheric Loss		0.30 dB
Boltzmann's Const		228.60 dBJ/K
Effective G/T	IRAS Groundstation	5.29 dBK
Pointing Loss	(see below)	0.50 dB
Demod Loss		3.50 dB
C/NO		33.51 dBHz
Bit Rate	399 bps	26.01 dBHz
Required Eb/NO	(see below)	4.50 dB
Required C/NO		30.51 dBHz
Margin		3.00 dB

Notes:
Frequency 2 Ghz
Antenna Efficiency 0.65
Eb/NO for 10E-5 Bit Error rate (Convolutional Code)
Demod Loss includes Polarization Loss

TABLE 3 - PAYLOAD SCIENTIFIC OBJECTIVES

	IMA	MAG	DUS	GRS	SIM	IIS
Global Characterisation	***	+++				
Surface Properties	***					***
Material Composition	+++			***	***	***
Dust/Plasma Environment		***	***		PPP	
Cruise Science		+++	+++	+++	PPP	

IMA - Imaging System
MAG - Magnetometer
DUS - Dust Detector
GRS - Gamma Ray Spectrometer
SIM - Secondary Ion Mass Spectrometer
IIS - Imaging Infrared Spectrometer

* - Prime objective
+ - Secondary objective

(P - when used in passive mode - ion beam off)

APPENDIX

Asteroid Rendezvous Mass Budget (Provisional)

System	Component	No. Off	Unit Mass	Mass
Chemical propulsion				
	Bipropellant Motor	1	4	4.0
	Propellant Tanks	4	6	24.0
	Pyro Valves	2	0.25	0.5
	Isolation Valves	4	0.5	2.0
	Latch Valves	2	0.5	1.0
	Filters	2	0.2	0.4
	Non Return Valves	2	0.2	0.4
	Regulator	1	0.5	0.5
	Pressure Transducers	6	0.2	1.2
	Pressurant Tanks			1.5
	Pressurant			1.5
	Pipework (Titanium)			1.5
	CHEMICAL PROPULSION TOTAL			38.5
Electric Propulsion				
	15 Ion Thrusters	2	1.1	2.2
	PCUS	2	8	16.0
	Pipework, Latch Valve			1.2
	Cabling			1.0
	Mounting Structure	2	1	2.0
	FMMS	2	2.8	5.6
	Tanks	5	4	20.0
	ELECTRIC PROPULSION TOTAL			48.0
Primary Power				
	Base panels	16	1.3	20.8
	Hinges	16	0.6	9.6
	Hold Down Devices	4	1.2	4.8
	Cable, Wiring			16.0
	PRIMARY POWER TOTAL			51.[illegible]
Secondary power				
	Batteries	2	13.5	27.0
	Battery Charge Regs	2	0.8	1.6
	Power Conditioner	1	2	2.0
	Harness			4.0
	Shunt Dissipator			0.5
	SECONDARY POWER TOTAL			35.1

Communications	Omni Antenna	2	0.5	1.0
	S-Band Dish			8.0
	S-Band Feed and Support			4.0
	RF Harness			2.0
	Transmitters	2	1.8	3.6
	Receivers	2	1.8	3.6
	COMMUNICATIONS TOTAL			22.2
Thermal Control	Heaters			1.0
	Thermal Blankets and Coatings			6.0
	THERMAL TOTAL			7.0
Attitude Control	Earth Sensors	2	0.6	1.2
	Sun Sensors	4	0.3	1.2
	Star Sensors	2	1.5	3.0
	Gyros			6.0
	Accelerometers			1.0
	Computer			2.0
	Nutation Damper			0.7
	Hydrazine Thrusters	16	0.25	4.0
	Pipework			1.5
	Ion Thruster Gimbals	2	4.2	8.4
	Gimbal Electronics			2.0
	ATTITUDE CONTROL TOTAL			31.0
TT&C/OBDH	Data Bus			1.0
	Mass Memory			6.0
	Data Compression Units			2.0
	Telemetry Encoders			1.0
	Processors			4.0
	Latches, Relays			5.0
	TT&C/OBDH TOTAL			19.0
Structure	Main Central Cone			48.0
	Additional Structure			54.0
	STRUCTURE TOTAL			102.0

Payload	Wide Angle Camera	2	1.8	3.6
	Narrow Angle Camera	2	1.9	3.8
	Magnetometer (including boom)			5.0
	Dust Detector			6.0
	Gamma-ray Spectrometer (including boom)			6.0
	PAYLOAD TOTAL			24.4
	DRY MASS TOTAL			378.4
	Attitude Control Propellant			40.0
	Hydrazine for Manoeuvring			8.0
	Xenon for Ion Thrusters			54.5
	ESCAPE MASS			472.9
	Bipropellant			306.6
	LAUNCH MASS			779.5

Delta-V budget

	Isp(s)	Delta-V (m/s)	
Biprop	306	1500	(Earth Orbit Manoeuvring and Escape Burn)
Hydrazine	195	40	(Near-Asteroid Manoeuvring)
Xenon	3162	3800	(Deep-Space Manoeuvring)